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MASTER
PRELIMINARY OPERATIONAL HAZARDS
SUMMARY REPORT
FOR THE
TASK 2 THERMOELECTRIC GENERATOR

DECEMBER, 1959

Prepared by:
George P. Dix, Jr.

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FOREWORD

This report is an operational hazards report prepared for the Task 2 Thermoelectric Generator under contract AT(30-3)217.

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SUMMARY

This report describes the operational hazards associated with the use of an isotope-fueled auxiliary power unit for a satellite mission.

Chapter I, Introduction, explains how the report deals only with the effects on the thermoelectric generator of various types of missile aborts, and includes a cursory discussion of probabilities of missile failure.

Chapter II describes the generator design, the properties of the various fuel forms investigated, the characteristics of the fuel capsules, and the provisions for biological shielding. The chemical, physical, thermal, nuclear, and radiobiological properties of the radiocerium fuel are set forth in detail to serve as a reference throughout the report. Some emphasis is given to fuel capsule design and materials, since their optimization determines to some extent the fate of the fuel.

Chapter III describes the integration of the generators into a typical missile system. The location of the generators with respect to the propellant load, launch complex, and satellite are discussed.

Chapter IV describes the transport of the fuel cores from the point of fabrication to the point of launch. Various modes of shipment are analyzed with respect to safety, economics, and feasibility. Air shipment, the most feasible form of transport, is described in detail, and certain shipping hazards are investigated.

Chapter V describes the ground support system for preparing the nuclear APU for flight. This system provides cooling, monitoring, and shielding of the fuel capsule during its transfer from the shipping cask into the generator, during ground handling, and during installation in the missile.

Chapter VI describes the definition of aborted missions. Aborted missions are broadly categorized into launch failures, ascent failures, and final stage failures.

Chapter VII describes the forces and anomalous transient environment imposed on the Task 2 generators by aborted missions. The forces are either thermal, chemical, or mechanical.

Chapter VIII presents the ultimate fate of the radiocerium fuel following all aborted missions. Three charts summarize the various failures, and locations of the fuel following failures are plotted on a polar projection.

Chapter IX is a summary of past and future hazards test programs. It is shown that, where theoretical methods are not readily applicable in problems encompassing complex variables, experimental programs may resolve certain hazards empirically.

I. INTRODUCTION

This report has been prepared for the U.S. Atomic Energy Commission to define and analytically resolve the potential operational hazards involved in the use of the Task 2 thermoelectric generator as an auxiliary power unit in a typical satellite mission. No attempt is made in the body of the report to define the probabilities of failure of the missile system. Rather, the report defines the mode of failure, forces imposed, and the immediate and ultimate fate of the radiocerium fuel of the generator for 38 specific types of aborted missions. Therefore, a somewhat pessimistic analysis is presented here which precludes a successful mission and considers only aborted missions.

There are no accurate reliability statistics available for a typical Atlas-Hustler missile system. However, preliminary statistics based on all missile firings to date have been compiled in connection with the Dyna-Soar Program. These estimates indicate that aborted missions are distributed as follows:

<u>Flight Phase</u>	<u>Aborts per Thousand</u>
Launch	21
Boost Phase I	75
Boost Phase II	42

On a successful mission utilizing the Task 2 generator, the radiocerium fuel will decay in space, so only a very minor amount of radiocerium will be released to the stratosphere. As a consequence, ultimate fallout from a successful mission will be orders of magnitude less than the fallout from an aborted mission resulting in fuel burnup.

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II. DESCRIPTION OF TASK 2 THERMOELECTRIC GENERATOR

Two Task 2 thermoelectric generators are utilized to power certain electrical equipment in a Weapons System similar to 117L. The generators supply 250 electrical watts for a period of one year. Each generator unit has an electrical power output of 125 watts at 28 volts and 4.46 amperes. Figure 1 shows the operating characteristics of a single generator. A single generator produces 6500 thermal watts initially from the decay heat of the radioisotope core fueled with 0.88 megacuries of Cerium-144. Of the gross initial thermal output, 4000 watts are radiated to the outside by the thermal shutter, 540 watts are lost through the insulation, and 1960 watts are utilized for electrical power production through the thermoelectric elements at a materials efficiency of 6.38 percent. During the one year life span, the thermal shutter is gradually closed to compensate for radioisotope decay. In this way, a constant power output is maintained. Figure 2 shows the energy balance of a Task 2 generator.

A. GENERATOR DESIGN

1. Complete Unit

The outside configuration of the generator is cylindrical with hemispherical ends and is 34.1 inches long by 24.0 inches in diameter. The unit weighs approximately 196 pounds during operation and 4196 pounds with the mercury shield present during prelaunch preparations. As shown in Fig. 3, the unit consists of a fuel core and core support structure, radiation shields, and an outer shell. The fuel core and core support structure are discussed in more detail elsewhere in the text.

There are two shields, an instrument shield and a biological shield. The instrument shield consists of a tungsten block placed directly above the fuel capsule to provide shadow shielding of sensitive components in front of the unit. This block is 3.75 inches in diameter by 3.0 inches long and is at a temperature of 1535° F. The biological shield is 4000 pounds of mercury, which fills the internal volume of the unit. The mercury is pumped into the generator for prelaunch ground handling and is drained five minutes before launch.

The outer shell of the generator consists of a sandwich of aluminum outer skin 0.051 inch thick, a meat of MIN-K-1301 insulation 1.5 inches thick, 554 lead telluride thermoelectric elements (277 pairs), and an inner skin of 316 stainless steel 0.063 inch thick. The outer skin is at 335° F and the inner skin is at 1025° F during operation. The thermoelectric elements operate at a hot junction temperature of 1000° F and at a cold junction temperature of 350° F. The thermal shutter is located on the bottom of the unit and is activated by a servomechanism which is dual controlled, primarily by the hot junction temperature and, secondarily, by the power output of the unit.

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Fig. 1. Task 2 Generator Operating Characteristics

Fig. 2. Energy Characteristics

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Fig. 3. Thermoelectric Generator

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2. Core Support Structure

The core support structure and the internal frame structure of the generator are important factors in the overall integrity of the generator. As shown in Fig. 3, the core support structure consists of an upper core positioning ring, a lower core support ring, and four guide struts between the two rings, all of 316 stainless steel. Above the core support structure is a 316 stainless steel guide sleeve, a hold-down positioning rod attached to the loading hatch, and the loading hatch, which is bolted down after insertion of the fuel core and tungsten shield.

The internal frame structure and its relationship to the core support structure are shown in Fig. 4. This structure consists of a top and bottom load ring, four hollow trusses connecting the core support structure and load rings, four lateral lift points attached to the upper core support ring, and four secondary lift points attached to the upper core support ring. Therefore, a rigid frame structure with eight mounting points is provided for loading the unit into the vehicle and attaching it to the bottom frame ring of the final stage adapter. The internal frame structure is made of 316 stainless steel.

B. FUEL

1. Fuel Form

Ceric oxide, cerous oxide, cerous fluoride, and cerium metal chemical forms have been considered for the cerium-144 radioisotope fuel to be used in the generator. Each unit contains 0.88 megacuries of Cerium-144 (3440 grams) which provides thermal power through radioisotope decay of Cerium-144 and its daughter Praseodymium-144 to Neodymium-144. The ceric oxide fuel form was first considered because of its stability at elevated temperatures, power density, and ability to be separated and processed. However, as the concept of post-orbital re-entry of fuel cores evolved to one of fuel burnup in the upper atmosphere on re-entry, a less stable, lower melting point material became more feasible. Fuel forms considered for burnup are ceric oxide with metal additives, cerous oxide, cerous fluoride, and cerium metal. These were selected for their power density and lower melting and boiling points relative to pure ceric oxide.

The oxide and fluoride forms are pelletized into right cylindrical pellets by compacting under 25 tons per square inch pressure and sintering at elevated temperatures. Cerium metal, however, is cast into a hollow fuel block as a single cylindrical mass. Tables 1 through 5 show the properties of Cerium-144 and its various fuel forms.

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Fig. 4. Generator Truss Support Structure

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TABLE 1

Chemical Properties of Cerium and Compounds

<u>Chemical Form</u>	<u>Chemical Series</u>	<u>Solubility</u>			<u>Stability in Air</u>
		Cold H ₂ O	Hot H ₂ O	Other	
Cerium metal (Ce)	Rare earth	Insoluble	Insoluble	Soluble in dilute acids, insoluble alcohol	Pyrophoric
Ceric oxide (CeO ₂)		Insoluble	Insoluble	Soluble in H ₂ SO ₄ , HNO ₃ , insoluble dilute acids	Stable at high temperatures in air
Cerous oxide (Ce ₂ O ₃)		Soluble, forms	Ce(OH) ₃	Soluble H ₂ SO ₄ insoluble HCl	Oxidation
Cerous fluoride (CeF ₃)		Insoluble	Insoluble		Oxidation (slow)

Preparation

Ce	Electrolytic reduction Ca bomb reduction
CeO ₂	Decomposition of cerous oxalate
Ce ₂ O ₃	CeO ₂ reduction
CeF ₃	HF treatment of cerous nitrate

2. Chemical Properties

Cerium and its daughters Praseodymium and Neodymium are members of the rare earth series of elements. Radiocerium is separated from fission products by solution with nitric acid, and precipitation as iodate, arsenate, and oxalate. Table 1 presents the chemical properties of cerium and its applicable compounds.

Cerium (Ce).- Cerium metal is insoluble in hot and cold water and is soluble in dilute acids and insoluble in alcohol. It is prepared by reducing cerous chloride by calcium in graphite at 575° C, or in a calcium oxide lined steel bomb according to the following reaction:

Cerium metal ignites in air at 150 to 180° C and burns vigorously in halogen vapors above 200° C. One reaction of cerium metal in air is

Ceric oxide (CeO₂).- Ceric oxide is insoluble in hot and cold water. It is soluble in sulphuric and nitric acids and insoluble in dilute acid solutions. Ceric oxide is stable in an oxidizing atmosphere, but reduces to cerous oxide in contact with certain metals or in a reducing atmosphere. CeO₂ is prepared by thermal decomposition of ceric oxalate or nitrate at 650° C in air. Ce-144 O₂ will evolve oxygen according to the following radiochemical reaction:

Cerous oxide (Ce₂O₃).- Cerous oxide will react slowly with water to form cerous hydroxide according to the following reaction:

Cerous oxide is prepared by thermal decomposition of cerous salts in hydrogen and by reducing ceric oxide in hydrogen.

Cerous fluoride (Ce F₃).- Cerous fluoride is insoluble in hot and cold water. It is relatively stable in air but does undergo slow oxidation at elevated temperatures. It has a high vapor pressure at elevated temperatures. Cerous fluoride is prepared by wet chemical reaction of a cerous salt and hydrofluoric acid.

3. Physical Properties*

The physical properties of cerium, ceric oxide, cerous oxide, and cerous fluoride are presented below and in Table 2.

Cerium (Ce).- Cerium is a grey metal which has an atomic weight of 140.13. It has a melting point of 804° C, which is low relative to its compounds. Its approximate boiling point is 2500° C. Cerium has a density of 6.78. It occurs in two crystalline forms, α cerium which is face centered cubic and β cerium which is hexagonal.

Ceric oxide (CeO₂).- Ceric oxide is a yellow-white compound which has a molecular weight of 172.13. It has a melting point of 2600° C and a boiling point of about 3800° C. It has a density of 7.3 and a cerium density of about 5.8 grams per cubic centimeter. Its crystalline structure is face centered cubic. The bulk density of cerium pellets from the ORNL process is about 5.6 gm/cc.

Cerous oxide (Ce₂O₃).- Cerous oxide is a grey-green compound which has a molecular weight of 328.26. It has a melting point of 1690° C and a boiling point of about 3300° C. It has a density of 6.8 and a cerium density of about 5.8 gm/cc. The crystalline structure of cerous oxide is hexagonal.

Cerous fluoride (CeF₃).- Cerous fluoride is a white-buff compound with a molecular weight of 197.13. It has a melting point of 1430° C and a boiling point of 2300° C. Its compound density is 5.8, and it has a cerium density of approximately 4.1 gm/cc. The crystalline structure of cerous fluoride is hexagonal.

TABLE 2

Physical Properties of Cerium and its Compounds

	<u>Ce</u>	<u>CeO₂</u>	<u>Ce₂O₃</u>	<u>CeF₃</u>
Molecular weight	140.13	172.13	328.26	197.13
Color	Grey-metallic	White-yellow	Grey-green	White-buff
Melting point (°C)	804	2600	1690	1430
Boiling point (°C)	2500	3800	3300	2300
Density (gm/cc)	6.78	7.3**	6.8	5.8
Cerium density (gm/cc)	6.78	5.8	5.8	4.1
Crystalline structure	Face centered cubic hexagonal	Face centered cubic	Hexagonal	Hexagonal

* H. Barr

** Bulk density of CeO₂ sintered pellets is 5.6 to 6.4 gm/cc

4. Thermal Properties

The inherent thermal power of Cerium-144 is of particular interest because of its influence upon the physical and chemical properties of radiocerium compounds. Table 3 lists the thermal properties of Cerium-144 and its compounds. The values shown are theoretical; actual values are significantly less and depend upon a number of variable factors such as decay time, purity, additives, and other cerium isotopes contained. Figure 5 shows the vapor pressure of cerium compounds as a function of temperature.

TABLE 3

Thermal Properties of Cerium-144 and its Compounds

	Form			
	<u>Ce</u>	<u>CeO₂</u>	<u>Ce₂O₃</u>	<u>CeF₃</u>
Thermal conductivity (cal/sec-cm-°C)	0.026	0.01	--	--
Specific heat (cal/gm-°C)	0.0423	0.1	--	--
Thermal coefficient of expansion (cm/ cm-°C)	5 x 10 ⁻⁶	12.1 x 10 ⁻⁶	--	--
Cerium (wt %)	100	81	85	71
Theoretical specific activity (curies/gm)	3.1 x 10 ³	2.5 x 10 ³	2.6 x 10 ³	2.2 x 10 ³
Specific activity- ORNL pellets (curies/ gm)	--	256 (approx- imately)	260 (approx- imately)	220 (approx- imately)
Theoretical power density (watts/gm)	24.5	19.8	20.8	17.4
Specific power ORNL pellets (watts/gm)	--	1.9*	2.08	1.74
Power activity ORNL (watts/curie)	0.0074	0.0074	0.0074	0.0074

* ORNL estimated value 1.9 watts/gm corresponds to 12.5 watts/cc for processed CeO₂ based upon a bulk density of 6.4 gm/cc.

Fig. 5. Vapor Pressure of Cerium Compounds

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Cerium (Ce-144).- Cerium has a thermal conductivity of 0.026 cal/sec-cm-°C and a thermal coefficient of expansion of 5×10^{-6} cm/°C. For pure Cerium-144 metal at zero decay time, the specific activity is 3.1 kilocuries/gm. The specific power value for this material is 24.5 watts/gm. This is equivalent to the measured value of 0.0074 watts/curie.

Ceric oxide (Ce-144 O₂).- Ceric oxide has a thermal conductivity of 0.01 cal/sec-cm-°C, a specific heat of 0.1 cal/gm-°C, and a thermal coefficient of expansion of 12.1×10^{-6} cm/°C. Pure ceric oxide is 81% cerium by weight. For pure Ce-144 O₂ at zero decay time, the specific activity is 2.5 kilocuries/gm. The corresponding specific power value is 19.8 watts/gm.

The specific activity for ORNL-processed ceric oxide is approximately 256 curies/gm. This corresponds to a net specific power of 1.9 watts/gm. One might note that these values differ from the values for pure Ce-144 O₂ by a factor of 10. There are several reasons for this. First, the processed material contains a number of impurities, such as Nd, La, Yb, and other isotopes of cerium such as Ce-140, Ce-141, and Ce-142, which contribute approximately 15% dilution by weight. Second, the cerium is allowed to decay in the chemical, handling, and fuel fabrication processing. Third, additives are contained in the fuel pellets to enhance their physical properties.

Cerous oxide (Ce-144₂ O₃).- Cerous oxide is 85% by weight cerium. Pure Ce-144₂ O₃ has a specific activity of 2.6 kilocuries/gm at zero decay time. This corresponds to a specific power of 20.8 watts/gm.

It might be expected that the processed material would have values less by a factor of 10. This would correspond to 260 curies/gm and 2.08 watts/gm.

Cerous fluoride (Ce-144 F₃).- Cerous fluoride is 71% by weight cerium. Pure Ce-144 F₃ has a calculated specific activity of 2.2 kilocuries/gm at zero decay time. This corresponds to a specific power of 17.4 watts/gm.

It is expected that the processed material would have a source strength of 220 curies/gm and a specific power of 1.74 watts/gm.

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5. Nuclear Properties

Cerium-144 is a fission product which is separated from reactor wastes. Some of the radiation and resultant thermal power associated with radiocerium originates in the beta decay of Cerium-144 to its daughter Praseodymium-144, but most of the thermal power is derived from the decay of Praseodymium-144 to Neodymium-144. Aside from the decay gamma radiation emitted by radiocerium, a large portion of the photon radiation emitted is in the form of X-radiation originating from the slowing down of energetic beta decay particles. Table 4 lists the nuclear properties of Cerium-144. Figure 6 shows the decay scheme of Cerium-144.

TABLE 4

Nuclear Properties of Cerium-144 and its Daughters

	<u>Radionuclide</u>		
	<u>Ce-144</u>	<u>Pr-144</u>	<u>Nd-144</u>
Half-life	285 days	17.5 min	5×10^{15} yr
Mode of decay	β	β	α
Specific activity curies/gm*	3.2×10^3	7.5×10^7	4.9×10^{-13}
Energy of decay particles (Mev)	β , 0.327	β_x 3.01	α 1.9
	β_2 0.258	β_y 2.30	
	β_3 0.16	β_z 0.79	
Abundance (%)	75.0 (β_1)	98 (β_x)	100
	~ 5.0 (β_2)	~ 1.3 (β_y)	
	~ 20.0 (β_3)	~ 1.0 (β_z)	
Gamma energy (Mev)	γ_1 0.054	γ_x 0.69	No
	γ_2 0.080	γ_y 1.50	
	γ_3 0.134	γ_z 2.18	
Relative abundance (%)	3.0 (10%) γ_1	1.6 γ_x	--
	3.0 (10%) γ_2	0.25 γ_y	
	7.5 (15%) γ_3	0.8 γ_z	
Other radiation	X from	X from	--
	Bremsstrahlung	Bremsstrahlung	

*Pure isotope with 0 decay

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Fig. 6. Decay Scheme of Cerium 144

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Genetic prelationship.- Cerium-144 has the following genetic relationships:

Nuclear properties.- Cerium-144 has a half-life of 285 days and decays through beta emission to Praseodymium-144. The Cerium-144 beta energies in Mev are 0.30 (75%), 0.25 (~5%), and 0.16 (20%). The decay gamma radiation varies from 0.012 to 0.134 Mev.

Praseodymium-144 has a half-life of 17.5 minutes and decays through beta emission to Neodymium-144. The Praseodymium-144 beta energies in Mev are 3.0 (98%), 2.3 (~1.3%), and 0.79 (1.6%). It is readily apparent that the 3 Mev beta of Praseodymium-144 is the major source of thermal power and high energy bremsstrahlung from the radiocerium fuel. The decay gamma radiation varies from 2.18 to 0.694 Mev.

Neodymium-144 has a half-life of 1.5×10^{15} years and decays through the emission of a 1.5 Mev alpha particle. Because of its long half life, Neodymium-144 is considered stable.

Bremsstrahlung.- The spectral distribution of decay gamma radiation from Ce-144 to Pr-144 is well known; however, the spectrum for X-radiation from bremsstrahlung is somewhat obscure. The bremsstrahlung covers a broad range of energies, with a maximum energy approaching the maximum energy of the impinging beta particles and an average energy which is much lower than the maximum energy. The bremsstrahlung contributes a large portion of the photon radiation emitted by radiocerium fuel compounds and is an important factor in the design of shielding.

6. Radiobiological Properties

The radiobiological properties of Cerium-144 and its daughters are shown in Table 5. The table combines the properties of Cerium-144, Praseodymium-144 and Neodymium-144. The critical organs for soluble radiocerium are the large lower intestine of the gastrointestinal tract, the skeleton, and the liver. The critical organs for insoluble radiocerium are the lung and lower large intestine. The maximum permissible total body burden is 20 microcuries. The total internal radiation dose from 1 microcurie of Cerium-Praseodymium-144 inhaled is 1160 millirem.

TABLE 5

Radiobiological Properties of Cerium-144 and its Daughters

Radionuclide and Type of Decay	Form	Organ of Reference	MPC* Total Body (μc)	<u>Maximum Permissible Concentration</u>			
				For 40 hour Week		For 168 hour Week	
				Water ($\mu\text{c}/\text{cc}$)	Air ($\mu\text{c}/\text{cc}$)	Water ($\mu\text{c}/\text{cc}$)	Air ($\mu\text{c}/\text{cc}$)
Ce-144 (β , γ)	Soluble	GI (LLI)**	--	3×10^{-4}	8×10^{-8}	10^{-4}	3×10^{-8}
Pr-144 (β , γ)		Bone	5	0.2	10^{-8}	0.08	3×10^{-9}
Nd-144 (α)		Liver	6	0.3	10^{-8}	0.1	4×10^{-9}
		Kidney	10	0.5	2×10^{-8}	0.2	7×10^{-9}
		Total body	20	0.7	3×10^{-8}	0.3	10^{-8}
	Insoluble	Lung	--	--	6×10^{-9}	--	2×10^{-9}
		GI (LLI)	--	3×10^{-4}	6×10^{-8}	10^{-4}	2×10^{-8}
	<u>Biological Half-Life</u>		<u>Effective Half-Life</u>	<u>Internal Dose*** (mrem/μc)</u>			
	500 days		180 days	1160			

* For occupational exposure

** GI = gastrointestinal tract, LLI = large lower intestine

*** Total dose from a single inhalation

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C. FUEL CAPSULE (CORE)

The isotope fuel capsule or core will be fueled with 0.88 megacuries of Cerium-144. Three specific core configurations have been considered, each dictated by either the consequences of the potential failures of the missile system utilized to place the weapons system into orbit, the desired post-orbit fate of fuel capsules, or by the specific radiocerium fuel form to be employed. However, each core considered has approximately the same external dimensions, source strength, and thermal characteristics. The initial design concept was of a fuel core which would withstand pre-orbital injection failures, withstand aerodynamic heating upon re-entry following post orbital decay, and survive impact upon the Earth's surface at terminal velocity. A molybdenum multitubular fuel capsule coated with ablating material was designed to satisfy this initial requirement.

The present design concept is of a fuel core which will withstand any credible missile failure, but will burn up at high altitude upon re-entry. Three fuel capsule designs have been considered for the present requirement: multitubular, cerium metal, and cluster fuel capsules of high nickel alloys. Though a certain element of flexibility is presented by alternate designs and material selections, it appears that the present requirements can be achieved by optimization of the capsule design, radiocerium fuel form, and core materials.

1. Design

Multitubular fuel capsule.- The multitubular core is a single right cylindrical metal block with seven tube channels drilled through the block parallel to its axis. The top and bottom of each tube is fitted with a 0.49-inch thick threaded cap. The external dimensions of the core are 11 inches in length and 3.75 inches in diameter. The fuel tubes are loaded with radiocerium pellets and will contain 0.1257 megacuries per tube. The screw caps are emplaced and welded at the chamfer. Figure 7a shows the multitubular fuel capsule configuration.

Cerium metal fuel capsule.- This capsule is designed for cerium metal fuel and is essentially a hollow metal block when assembled. Initially, the block is in two halves which are welded together at the mid-plane. The internal volume of the block is 9.75 inches in diameter and 11 inches long. The cerium metal fuel is poured into the block in the molten state and the 0.49-inch thick screw cap is seated and welded at the chamfer. Figure 7b shows the cerium metal fuel capsule.

Cluster fuel capsule.- This core consists of a cluster of four discrete fuel tubes banded together. Each fuel tube is 11 inches long and 1.55 inches in diameter and contains 0.22 megacuries of Cerium-144. Each tube is sealed with a 0.49-inch thick screw cap on one end. The four

Fig. 7. Fuel Capsule Configurations

tubes are banded into a cluster by stainless steel bands to produce a single fuel capsule 3.75 inches in diameter and 11 inches long. The relative merits of this configuration are discussed in Chapter VII. Figure 7c shows the cluster fuel capsule design.

2. Core Materials

As in the selection of a fuel form and core design for the generator, the selection of core materials is dependent upon the re-entry concept. For the initial concept of intact re-entry and impact of the core, these requirements were imposed upon core materials: (1) impact and shock resistance at elevated temperatures; (2) corrosion resistance in sea water; (3) oxidation resistance at high temperature; and (4) high melting point. Molybdenum was selected for the initial requirement contingent upon the development of an oxidation resistant protective coating.

For the present concept of an intact core following aborted missions and burnup in the upper atmosphere upon re-entry, the following requirements are imposed upon core materials: (1) impact resistance at elevated temperatures; (2) corrosion resistance in sea water; and (3) low melting and vaporization temperatures. Stainless steel type 316, alloy S 816, Inconel X, and Hastelloy B are being considered for the present requirement.

The properties of the core materials considered in the course of the Task 2 program are presented below. Figure 8 shows the ultimate tensile strength of these materials as a function of temperature. Table 6 lists the properties of these materials.

Molybdenum.- Molybdenum is a high temperature material which melts at 4748°F and boils at $10,121^{\circ}\text{F}$. It is alloyed with 0.5% titanium which is also a relatively high temperature metal (melting point 3272°F). It oxidizes rapidly when exposed to air at high temperatures. At the core operating surface temperature of 1680°F molybdenum has an ultimate tensile strength of about 75,000 psi. It has a specific heat of 0.07 and relatively good thermal conductivity (65 Btu/ft-hr- $^{\circ}\text{F}$). Molybdenum will resist sea water corrosion, but is tarnished initially.

Since molybdenum oxidizes at elevated temperatures when exposed to air, an oxidation-resistant coating was developed. The Colmonoy 5 alloy coating is flame sprayed on the molybdenum and is an alloy of 79% nickel, 12% chromium, 4% silicon, 4% iron, 2% boron, and 0.5% carbon. Colmonoy melts at 1950°F .

In addition to the protective coating, a layer of graphite was provided as a final coating to resist aerodynamic heating effects.

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Fig. 8. Ultimate Tensile Strength of Various Fuel Core Materials as a Function of Temperature

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TABLE 6
Properties of Core Materials

Composition (wt %)*	<u>Molybdenum</u>	<u>Hastelloy B</u>	<u>S 816</u>	<u>Inconel X</u>	<u>SS 316</u>
	Mo(99.5),Ti(0.5)	Ni(61),Cr(1),Fe(5.5), Mo(28),Co(2.5), Si(1),Mn(1)	Ni(20),Cr(20),Fe(3), Mo(3.7),W(4),Co(42), Cu(4)	Ni(73),Cr(15),Fe(7), Ti(2.5),Cb(1), Al(0.9)	Ni(12),Cr(17),Fe(65), Mo(2.5),Mn(2),Si(1)
Melting point (°F)	4748	2408 to 2462	2350 to 2450	2540 to 2600	2500 to 2550
Density (gm/cc)	10.2	9.2	8.6	8.3	7.9
Specific heat (Btu/lb/°F)	0.07	0.09	0.12	0.10 to 0.11	0.12
Corrosion ex- posure time (months)	--	72	--	32	13
Quiescent sea pitting--avg (inches)	--	0.023	--	0.063 (max)	0.073
Water corrosion rate (inches/ year)	--	0.003	--	> 0.002	--

* Approximate

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Hastelloy B.- Hastelloy B is characterized by a high nickel and molybdenum content. It melts at 2408 to 2462° F and has a specific heat of 0.09 Btu/lb/°F. It is particularly resistant to the corrosive effects of hydrochloric acid. At the core surface operating temperature of 1680° F Hastelloy B has an ultimate tensile strength of about 40,000 psi. In a quiescent sea water environment, the maximum pitting for 72 months exposure time is 0.033 inches. The estimated corrosion rate is 0.003 inches per year.

S 816.- S 816 is characterized by its high cobalt, nickel, and chromium content and has a density of 8.6 gm/cc. It has a melting point of 2350 to 2450° F and a specific heat of 0.12 Btu/lb/°F. At the core operating temperature, it has an ultimate tensile strength of about 38,000 psi. Sea water corrosion data on S 816 is meagre, though it might compare with Hastelloy B in that respect.

Inconel X.- Inconel X is primarily nickel and has a density of 8.3 gm/cc. It has a melting point of 2540 to 2600° F, a thermal conductivity of 7.8 Btu/ft-hr-°F, and a specific heat of 0.11 Btu/lb/°F. At the core operating temperature of 1680° F, Inconel X has an ultimate tensile strength of about 20,000 psi. It has a maximum pitting in quiescent sea water of 0.063 inches for 32 months exposure. Its corrosion rate in flowing sea water is estimated at less than 0.002 inches per year.

SS 316.- Type 316 stainless steel contains iron, nickel, and chromium as major constituents and has a density of 7.9 gm/cc. It has a melting point of 2500 to 2550° F and a specific heat of 0.12 Btu/lb/°F. At the operating temperature of the core, SS 316 has an ultimate tensile strength of about 19,000 psi. When exposed to quiescent sea water for 13 months, SS 316 exhibited a maximum pitting of 0.105 inches and an average pitting of 0.073 inches.

3. Thermal Characteristics*

The thermal characteristics of the multitubular fuel capsule have been determined by theoretical and empirical means. The cerium metal and cluster fuel capsules will, in general, exhibit the same external thermal characteristics as the multitubular capsule. The thermal energy output from the radiocerium within any capsule design at any given time is the same; this value is initially 6500 thermal watts.

The operational temperature distribution within the multitubular fuel capsule is, therefore, dependent upon the thermal energy output of the fuel and the conductivity of the core material. For example, the molybdenum capsule has a centerline temperature of about 1750° F and a

* T. S. Bustard

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surface temperature of 1680° F initially. An Inconel X capsule has a centerline temperature of about 1880° F and a surface temperature of about 1680° F. The greater centerline temperature of the Inconel X core over that of the molybdenum core is directly attributed to the difference in the thermal conductivity of the two materials. (Molybdenum = $65 \text{ Btu/ft-hr-}^{\circ}\text{F}$, Inconel X = $7.8 \text{ Btu/ft-hr-}^{\circ}\text{F}$.)

D. BIOLOGICAL SHIELD*

During ground handling each Task 2 unit is filled with approximately 4000 pounds of mercury. This shield is drained at launch and refilled in case of a significant hold in the countdown. The discussion of the biological shield includes the dose rates from a bare unshielded source, the dose rates with the shield in place, and the doses during the field transfer operation.

1. Bare Core Source

The radiation dose rates are calculated for 2 cores at zero decay (1.7 megacuries), for one core at zero decay or two cores after one half-life (0.88 megacuries), and for one core after one half-life (0.44 megacuries). The dose rates from a 1.7 megacurie source (2 cores) are 5×10^4 roentgen/hour at 3 feet, and 46 roentgen/hour at 100 feet. The dose rates from a 0.88 megacurie core source are 2×10^4 roentgen/hour at 3 feet and 22 roentgen/hour at 100 feet. The dose rates from a 0.44 megacurie core source are 10^4 roentgen/hour at 3 feet and 11 roentgen/hour at 100 feet. Figure 9 shows the dose rate as a function of distance from the three bare core sources, neglecting air attenuation and including self-absorption.

2. Shielded Generator

Figures 10, 11 and 12 and Table 7 show the dose rates at various points around a generator with the mercury shield in place. The dose rates vary from point to point depending upon the geometry of the unit. The dose rates at the surface of the unit vary from 11 to 750 milliroentgen/hour. At 1 meter from the surface of the unit, the dose rates vary from 2 to 95 milliroentgen/hour.

* A. M. Spamer

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Fig. 10. Location of Dose Points in Vertical Plane Through Generator

All dimensions given in inches

Fig. 12. Dose Rates on Surface of Shielded Generator

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TABLE 7

Dose Rates at Various Distances from Center
of Generator with Shield Filled with Mercury

Dose Point	0	1	2	3	4	5
θ	0°	15°	30°	45°	60°	75°
Radius	Dose Rate mr/hr					
(on Surface)	116	55	35	48	210	600
2.5 ft	28	15	9	10	29	106
1 meter	15	8.8	5	5.5	16	62
5.0 ft	6.2	3.6	2	2.2	6.7	26
10.0 ft	1.5	0.88	0.5	0.5	1.7	6.6
25.0 ft	0.23	0.14	0.077	0.083	0.27	1.1
100.0 ft	0.014	0.0086	0.0048	0.0051	0.017	0.066

Dose Point	6	7	8	9	10	11	12
θ	90°	105°	120°	135°	150°	165°	180°
Radius	Dose Rate mr/hr						
(on Surface)	750	600	230	74	31	15	11
2.5 ft	160	106	34	13	7	4.3	3.4
1 meter	95	62	19	7	4	2.3	2
5.0 ft	42	26	8	2.8	1.5	0.95	0.77
10.0 ft	10	6.6	2	0.64	0.34	0.22	0.18
25.0 ft	2	1.1	0.31	0.10	0.052	0.035	0.029
100.0 ft	0.1	0.066	0.02	0.006	0.003	0.002	0.002

NOTE: Locations of dose points in relation to generator are shown in Fig. 10.

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3. Fuel Transfer

The shipping and field loading cask is designed for a maximum dose rate at three feet of 10 milliroentgen/hour. Figure 13 shows the total directional doses received at 3 feet during the fuel transfer operation which lasts for 15 to 20 seconds. Significant directional doses are received in specific areas such as around the sliding plug at the bottom of the cask, around the collar shield, and below the collar shield. Though the doses go as high as 5 roentgen, they are of a radial beam type geometry and do not represent steradian doses. Therefore, these doses cannot be interpreted as total body doses. Precautions will be taken to minimize personnel doses from this source of direct radiation.

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Fig. 13. Total Doses in Exposed Areas During Fuel Transfer

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III. INTEGRATION OF GENERATORS AND TYPICAL MISSILE*

This section describes the integration of the Task 2 thermoelectric generators into a typical missile. The Task 2 module, consisting of two generators, is mated to the final stage at ground level and the complete assembly is hoisted into position atop the booster as described in Chapter V. Figure 14 shows the overall configuration of the vehicle containing the generators in position atop the booster.

A. LOCATION OF GENERATORS

The generators are located next to the final stage rocket exhaust nozzle in the aft portion of the satellite vehicle. The two units are located 180 degrees apart as shown in Fig. 15.

1. Structural Connections

The two units are attached to the lower frame in the satellite vehicle section of the missile. A ring has been welded to the generators for pickup attachments to the frame. Each boiler has two columns 180° apart connecting the boilers to a fitting located on the frame. A minimum number of fasteners are used for rapid and easy installation of the units to the frame. Built into the tubing lines of the mercury biological fill and drain system are quick disconnects that are remotely operated at the time of launching.

2. Spatial Factors

In the aft section of the satellite vehicle with the thermoelectric generators are several components necessary for the successful operation of the final stage. The generators are positioned so that there is no effect from the following components: exhaust nozzle of rocket engine, ullage rockets, and gas control nozzles. There is no interference with the helium spheres when projected out or the ullage rockets which tumble away from the vehicle as they exhaust their fuel.

3. Location Relative to Fuel Tanks

Figure 15 shows the relationship of the generators with respect to the fuel of the missile. The units are located directly above the liquid oxygen tank of the first stage. The propellants of the final stage, unsymmetrical di-methyl hydrazine (UDMH) and inhibited red fuming nitric acid (IRFNA), are located approximately 40 inches above the generators.

* V. Kelly

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Fig. 14. Typical Vehicle Configuration

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Fig. 15. Aft Section of Typical Satellite Vehicle

Located under one generator unit is the liquid oxygen boil duct of the booster. The oxidizer of the final stage flows through a duct past the units to the turbine exhaust where it is heated prior to entry in the combustion chamber.

4. Location Relative to Launch Complex

The generators are approximately 70 feet over the launching pad when installed in the missile. The gantry can be remotely removed to a position 200 feet from the launching pad. When the gantry is removed the umbilical tower is positioned to service the missile. The launch operations building is located about 1000 feet from the launch pad as shown in Fig. 16.

B. MISSILE PROPELLANT LOAD

The propellants for the booster and sustainer rocket engines are RP-1 and liquid oxygen. The combined weight of the booster-sustainer propellant load is 260,000 pounds, of which 83,000 pounds is RP-1 and the remainder, 177,000 pounds, is liquid oxygen.

The final stage utilizes the Bell Hustler rocket engine which is fueled with IRFNA as the oxidizing agent and UDMH as the fuel. The total fuel load, 6570 pounds, consists of 1850 pounds of UDMH fuel and 4720 pounds of IRFNA oxidizing agent.

C. ORBITAL ORIENTATION OF TYPICAL SATELLITE

Figure 17 shows the orbital orientation of a typical satellite. The generators are exposed in the aft portion of the satellite vehicle on either side of the final stage engine nozzle.

Fig. 16. Sketch of Typical Launch Complex at VAFB

Fig. 17. Orbital Orientation of Typical Satellite Vehicle

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IV. TRANSPORT OF GENERATORS

The shipment of the APU to the launch site is divided into two specific operations: (1) shipment of the radioactive fuel capsules and (2) shipment of nonradioactive hardware, such as the empty generators, field handling equipment, and auxiliary components. These capsules and generator components will be assembled at the launch site as described in Chapter V. The second operation, shipment of nonradioactive hardware from Baltimore to the launch site, is governed by conventional shipping procedures and does not warrant further discussion. The point of origin for shipment of fuel capsules is Oak Ridge National Laboratory (ORNL), located near Knoxville, Tennessee, and the destination would be Vandenberg AFB, near Santa Maria, California. The following discussion includes a description of the transport cask, the mode of shipment, shipping procedures, and shipping hazards.

A. TRANSPORT CASK*

A shipping cask for each fuel capsule is designed to (1) provide shielding of the fuel; (2) provide adequate cooling of the fuel capsule; and (3) provide containment of the capsule during credible transport accidents. The shipping cask is shown in Fig. 18.

1. Shielding

The shipping cask is designed for a dose rate of 10 milliroentgen/hour or less at three feet from the surface. This is achieved by using a 13-inch thickness of lead around the fuel capsule. Dose rates obtained during the field transfer operation from the shipping cask are discussed in Chapter II, Section E.

2. Cooling of Cask

Cooling of the fuel block and cask is provided by three means: (1) by natural convection around the cooling fins on the outside of the cask; (2) by woods metal surrounding the fuel capsule, and (3) through a cooling coil located around the fuel capsule container of the cask. Water can be circulated through the cooling coil during transportation in case of an anomalous temperature rise of the fuel capsule.

3. Cask Design and Containment

The lead is encased in a steel shell fitted with cooling fins and is supported by tension straps. The center of the cask contains a mounting bracket which supports the fuel capsule receptacle. The bottom of

* E. J. Reiley et al

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Fig. 18. Shipping Cask

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the cask is fitted with a loading ram and coupling devices and the top is fitted with a slide plug for capsule loading. The remainder of the inner volume is filled with lead. The function of the cask components is described in Chapter V. Each cask when loaded weighs approximately 15,000 pounds.

B. MODE OF TRANSPORT

Three means of transporting the radioactive fuel capsules from ORNL to Vandenberg AFB were considered. A comparative table is presented for air, truck, and rail transport over the required distance of 2100 statute air miles which shows both the time in shipment and percent decay of the fuel for each mode of transport.

	<u>Time in Transport (days)</u>	<u>Decay (%)</u>
Air	1	0.24
Truck	5	1.21
Rail	8	1.82

The following safety statistics have been compiled from Sandia Corporation and ICC data for commercial carriers:

	<u>Chances of Accident per 1000 miles*</u>
Air (commercial cargo)	1/5550
Truck	1/230
Rail	1/3300

The serious accident statistics compiled by H. A. Knapp of the AEC are as follows:

	<u>Fraction of Accidents Resulting in a Serious Accident</u>	<u>Chance of a Serious Accident per 1000 miles**</u>
Air (commercial cargo)	1/8	1/42,000
Truck	1/500	1/115,000
Rail (freight)	1/25	1/82,300

* Based on involvement of different carriers that might be used for transporting hazardous materials within the U.S. for an average haul between 2 points 1000 miles apart.

** Based on the overall chance of a serious accident for various methods of transporting an assumed hazardous cargo between 2 points 1000 miles apart.

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According to R. V. Batie of the AEC, there were two shipments from NRTS that involved megacurie quantities from July 1956, to July 1957. These were unescorted off-site shipments.

The following is a discussion of the relative merits of each mode of shipment.

1. Air Shipment

Air transport of fuel capsules is the preferable mode of transport for several reasons. First, the saving in transport time will minimize radioactive decay of the fuel (0.24%). Second, air shipment will be controlled by the U.S. Air Force due to the nature and weight of the shipment. Third, air transport will minimize human contact with the shipment in transit. Fourth, although the probability of a serious accident for air transport is higher than other modes of transport, based upon commercial cargo shipments, the probability for a single accidental event is lower.

2. Truck Shipment

Truck shipment of fuel capsules from ORNL to Vandenberg AFB would take approximately 5 days. In transit decay would amount to 1.21% or a loss of approximately 20,000 curies. Such a shipment might entail security personnel, certain possible local restrictions, selective routing, and delay due to weather and road conditions. The frequency of truck accidents is higher than that for air or rail for a single event, but the frequency of serious accidents is lower than the other two modes of transportation.

3. Rail Shipment

Rail shipment of fuel cores would originate at either ORNL or Knoxville, and would take approximately 8 days from the point of origin to Vandenberg AFB. In transit decay would amount to 1.82% or a loss of approximately 32,000 curies. For frequency of accidents and severe accidents rail shipment is intermediate between air and truck shipment.

C. SHIPPING PROCEDURES

It is assumed that the fuel will be shipped by air to Vandenberg AFB along a route similar to that shown in Fig. 19. Air shipment would require that the shipping casks be transported by truck from ORNL to Knoxville Airport, a distance of about 40 miles. At Knoxville Airport, the cask skids would be loaded on a C133A aircraft which has a payload of 46,800 pounds with a 3,500 nautical mile range. The C133A would fly nonstop 2100 miles to Vandenberg and land at the military airport near the launch complex, where the casks would be unloaded for truck shipment to the launch pad.

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Fig. 19. Air Shipping Route ORNL to VAFB

The fuel cores are designed for impact at their terminal velocity of 500 feet per second, so that they would withstand impact conditions resulting from an inflight accident. Any conceivable ground accident would impose less mechanical forces than those imposed during terminal velocity impact.

3. Combined Effects

Ground transport accidents between ORNL and Knoxville Airport and VAFB Airport and the launch site would be under the control of trained ORNL and AFBMD personnel. The containment provided by the lead casks would not be breached by any conceivable ground accident to expose the fuel cores.

Air accidents might conceivably expose the fuel cores under extreme conditions. However, burial is likely and would provide attenuation of direct radiation. Release of Cerium-144 to the atmosphere does not appear possible since the cores are designed to withstand terminal velocity impact and the thermal forces of a propellant fire.

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V. GROUND HANDLING AND VEHICLE INSTALLATION*

The ground support system for preparing the nuclear APU for flight has been designed in order to eliminate the requirement for a hot cell at the launch complex. This system provides cooling, monitoring, and shielding of the fuel capsule during its transfer from the shipping cask into the generator, during ground handling of the fueled generator, and during installation in the missile.

The following distinct operations are performed:

- (1) Preparation for transfer
- (2) Transfer of the fuel capsule to the generator
- (3) Mating of the generator and the final stage
- (4) Mating of the final stage and the booster.

A. PREPARATION FOR TRANSFER

The shipping casks containing the fuel capsules will arrive at their destination and will be skidded from the transport vehicle to a flat bed trailer for movement to the launch complex. At the launch complex each cask will be hoisted upon a field transfer structure, inverted, and positioned by a portable crane. Simultaneously, the dolly containing the cold generators will be positioned in the transfer structure beneath the cask.

B. TRANSFER OF THE FUEL CAPSULE TO GENERATOR

When both the shipping cask and generator are in position, the collar shield will be connected between the cask and generator as shown in Fig. 20. The collar shield maintains shield continuity between the lead cask and generator. The collar shield is a tank of single unit construction oriented with the generator and dolly but supported by the transfer structure. The collar shield and generator are filled with mercury.

An additional retention shield shell is provided around the biological shield of the generators. The retention shell functions to retain any mercury inadvertently released from the generator shield. Consequently, if the generator shield is breached, the level of mercury in the retention shield will rise proportionally to the decrease in the level of the generator shield. The retention shell is present throughout the ground handling operation but is removed when the generators are mated to the final stage vehicle.

* J. Reilly et al

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Fig. 20. Transfer of Fuel Core to Generator

The shipping cask is fitted with two devices to facilitate transfer, as shown in Fig. 21. First, the bottom of the cask is fitted with a sliding section which is manually driven by a screw jack to clear the fuel capsule for passage into the generator. Second, the core is attached to a hand operated ram rod fitted with a remote connect and disconnect mechanism. The ram rod is actuated to insert and seat the core in the generator as shown in Fig. 22. The core is locked in place and the ram rod disconnected and withdrawn. Next, the tungsten block is inserted and the loading hatch on the top of the generator is seated and secured. This completes the transfer operation.

C. MATING OF GENERATORS TO FINAL STAGE

When the generators are fueled on the dolly, the dolly is moved beneath the final stage mating structure as shown in Fig. 23. Since the generator is designed for tension loads, it must be loaded into the final stage vehicle in a vertical position. Once the generators are positioned on either side of the engine nozzle, they are jacked upward into the final stage fittings. The bolts are attached and the generators are secured to the missile support ring. The mercury shield within the generators is maintained throughout this operation.

D. MATING OF FINAL STAGE TO BOOSTER

When the final stage system is checked out, the final stage is hoisted to its position atop the Atlas booster. During this operation the generator cooling and shield systems are functioning. The generator auxiliary systems are functional from the umbilical tower from the time the gantry is removed up to launch. At $t = -300$ seconds the mercury is purged from the shields, and the quick disconnect of the service lines executed at $t = 0$. Should a hold in the prescribed launch procedure occur, the mercury shield is pumped into the generators.

E. AUXILIARY SUPPORT EQUIPMENT

Certain auxiliary support equipment must be provided to facilitate field handling, core transfer, mating of the final stage and generators, and generator servicing on the booster. There are essentially two auxiliary systems: (1) the handling, transfer, and mating system and (2) the umbilical tower system.

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Fig. 21. Shipping Cask (inverted during fuel transfer)

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Fig. 22. Internal View of Fuel Transfer Operation

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Fig. 23. Mating of APU to Satellite Vehicle

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1. Handling, Transfer, and Mating System

A servicing module will accompany the generators to provide cooling of the cores, a mercury shield system, radiation monitoring and control of temperature and mercury levels. Figure 24 is a schematic of this system which specifically provides:

- (1) Cask cooling of core prior to transfer (water).
- (2) Generator cooling of core (water).
- (3) Compressed gas for pumping mercury (nitrogen).
- (4) A mercury fill and drain system for the collar shield and generators (mercury).
- (5) Gamma radiation monitors on the cask, collar shield and generators.

2. Umbilical Tower System

This system becomes operative when the final stage is prepared for final installation atop the Atlas booster. It is shown schematically in Fig. 25. It specifically includes:

- (1) Core cooling (water).
- (2) Compressed gas for pumping in and purging out the shield (nitrogen).
- (3) A shield fill and drain system (mercury).
- (4) Radiation monitors.

The above services are provided with quick disconnect lines to expedite prelaunch operations. Heavy components are located on the pad with service lines through the umbilical tower to the generators.

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Fig. 24. Schematic Diagram-Ground Servicing and Cooling Vehicle Fluid Systems

Fig. 25. Launch Pad Servicing and Cooling Schematic

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VI. DEFINITION OF ABORTED MISSIONS

The definition of the cause of, effects from, and anomalous transient environment imposed upon the generators by aborted missions must be set forth through theory, experimentation, and past observation of similarly aborted missions. Once these events are defined, the fate of the generators can be more closely predicted. Furthermore, the result of such an accident in terms of its fate in the atmosphere, hydrosphere, lithosphere and biosphere can be more readily understood.

Aborted missions or missile system failures can be logically divided into broad categories: A. Launch Failures: B. Ascent Failures; and C. Final Stage Failures.

Tables 8, 9, and 10 in Chapter VIII summarize the specific aborted missions treated.

A. DEFINITION OF LAUNCH FAILURES

Launch failures are broadly defined as those failures which occur either on or directly above the launch pad. These failures occur within an altitude range of 0 to 1000 feet and within a time range of several minutes before launch ($t = 0$) to 15 seconds after launch. Three types of launch failures were considered.

- (1) Propellant Tank Failure
- (2) Propulsion Failure
- (3) Guidance Failure.

These are illustrated in Fig. 26.

1. Propellant Tank Failure

The propellant tank failure results from either overpressurization or spontaneous structural failure of the propellant tanks. Two failure mechanisms are possible. First, there is a possibility that failure of the bottom of the liquid oxygen tank of the booster and rupture of the top of the RP-1 tank could cause mixing of propellants within the missile without immediate side seam failure of the tanks. Second, the most probable failure is side seam failure of the tanks and resultant mixing of the propellants on the launch pad.

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Fig. 26. Examples of Typical Launch Failures

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Once mixed, liquid oxygen and RP-1 present a considerable potential source of energy. This mixture can be ignited from a variety of sources, such as flame from the engine exhaust, electrical sources, sparks from impact of components, concussion from falling objects, or external thermal sources. It is probable that such an accident would occur at 0 altitude.

The energy yield from a propellant tank failure depends upon a number of factors which are functions of the failure and ignition mechanisms. For example, if ignition of the mixture is not immediate, a portion of the liquid oxygen will leave the mixture as vapor, the RP-1 will soak into the launch pad, and some of the mixture will be lost through the flame bucket of the launch pad. Also, the increased uniformity of mixing of the propellants will increase the energy yield. On the other hand, if ignition is immediate with the arrival of the falling propellants, the fire will be essentially gravity fed until it eventually reaches the unexpended propellants still in the tanks as the missile collapses.

A final aspect must also be considered. In observed failures of several multistaged missiles, two distinct fireballs could be observed; these resulted from ignition of the first stage and consequent ignition of the final stage. The WS-117L has similar characteristics in that the burning of the first stage will present a source of ignition for the final stage Hustler engine fueled with IRFNA and UDMH. Furthermore, the latter fuels are hypergolic (will spontaneously ignite when mixed), which indicates that mere impact of the fuel tanks will present an additional source of energy. However, the ignition of both stages simultaneously is not likely since collapse of the missile and exposure of the final stage presents a chronological sequence of events.

It is significant that the propellant tank failure accident postulated can occur before launch ($t = 0$), since the missile is fueled prior to that time. It can probably also occur up to $t = 15$ seconds after launch.

2. Propulsion Failure

Propulsion failures during the launch phase defined can occur from $t = 0$ to $t = 15$ seconds. Failures would be caused by restricted fuel flow, incomplete combustion, or other malfunctions of the booster engine during lift-off on the vertical launch trajectory. This failure results in fallback of the missile or missile debris on or in proximity of the launch pad, since the missile lift-off trajectory is normal to the center of the launch pad up to 1000 feet altitude. Two specific types of accidents are presented: (1) fallback of the missile on the pad and igni-

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tion of the fuel load there or (2) mid-air ignition of the fuel load. The former is perhaps the most probable of the two accidents at lower altitudes and has the greatest consequence in terms of energy released. Mid-air ignition of propellants yields less energy due to poor mixing, diffusion of propellants, and selective separation of missile components and propellants by gravity. Of course, for either accident less energy is available as the missile approaches 1000 feet altitude due to fuel consumption.

The propulsion failure of the booster engine will lead to the ignition of the Hustler engine due to the hypergolic nature of its fuel. However, the propulsion failure will make spatial separation of the booster fuel fire and the Hustler fuel fire more probable in the tank failure accident described.

3. Guidance Failure

Significant guidance failures or perturbations of the typical missile from the prescribed launch trajectory from several hundred to 1000 feet altitude would necessitate destruction of the missile by the launch operations officer. The method of destruction employed is a ground or vehicle self-destruct signal which detonates charges causing fuel tank rupture and ultimately complete missile destruction. In this case, missile components would fall in proximity to the launch pad. This would probably cause a mid-air ignition of the Hustler propellants. The total energy yield of this accident would approximate that of the propulsion failure.

B. DEFINITION OF ASCENT FAILURES

Ascent failures are broadly defined as failures occurring on the ascent trajectory from about 1000 feet altitude (Pitchover) to 1,570,000 feet altitude (Final Stage Ignition). These failures can occur within a time range of 15 seconds to 585 seconds after launch ($t = 0$). Failures can, of course, occur at any point along this trajectory, but six specific types of failures have been selected that are representative of the six critical points on the ascent trajectory.

The following specific types of ascent failures were considered:

- (1) Failure at pitchover
- (2) Failure at maximum dynamic pressure
- (3) Failure at booster separation

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- (4) Failure at sustainer burnout
- (5) Failure at vernier burnout
- (6) Failure at final stage ignition.

These ascent failures are illustrated in Fig. 27.

An analysis of the above failures includes the initial condition at failure, apogee of the missile from the point of failure, and the down range impact point of the missile. Throughout most of the ascent phase range safety destruct capability is present in case of deviation of the missile from the prescribed azimuth.

1. Failure at Pitchover

Pitchover is the point at which the pitch program places the missile in a ballistic turn. At this point, the missile is tilted from its vertical launch trajectory and programmed into the arcuate ascent trajectory. The initial conditions at pitchover are an altitude of about 1000 feet, an attitude of 90° , a velocity of about 200 feet per second, and a range of 0 nautical miles. If the pitch program is not initiated or if a propulsion failure occurs at that point, the missile or its debris will subsequently impact at a range of 0 nautical miles. Pitchover occurs at $t = 15$ seconds.

2. Failure at Maximum Dynamic Pressure

The point where the maximum dynamic pressure is applied to the missile on the ascent trajectory occurs at $t = 60$ seconds. At this point, the angle of attack of the missile is reduced to zero to start the zero lift trajectory. The initial conditions at this point are an altitude of 38,000 feet, an attitude of 54° , a velocity of 957 feet per second, and a range of about 0.1 nautical mile. After failure at this point the missile would coast to an apogee of 41,000 feet 0.5 nautical mile down range where its velocity becomes essentially zero at an attitude of 0° . From apogee the missile will impact about 1 nautical mile down range.

3. Failure at Booster Separation

Booster burnout and separation occur at the end of the zero lift trajectory at the beginning of the sustainer pitch program. This occurs at $t = 147$ seconds. At this point, the skirt and booster engines are separated and tumble away from the vehicle. The initial conditions at this point are an altitude of 250,000 feet, a velocity of 10,890 feet per second, an attitude of 23° , and a range of 60 nautical miles. On-trajectory propulsion failure at this point would result in the missile coasting to an apogee of 586,000 feet at a range of 320 nautical miles. The missile would ultimately impact 640 nautical miles down range.

Fig. 27. Conditions of Ascent Abort at Various Points on Preinjection Trajectory

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4. Failure at Sustainer Burnout

Sustainer burnout occurs at $t = 260$ seconds at the end of the sustainer pitch program. The initial conditions at sustainer burnout are an altitude of 720,000 feet, an attitude of 14.8° , a velocity of 17,000 feet per second, and a range of 340 nautical miles. From this point, the missile would coast to an apogee of 1,417,000 feet at a range of 1145 nautical miles. The missile would ultimately impact 2290 nautical miles down range.

5. Failure at Vernier Burnout

Vernier burnout occurs at $t = 289$ seconds, where the Atlas separates from the final stage vehicle and tumbles away. This occurs at the beginning of the coast phase. At this point, the initial conditions are a velocity of 17,000 feet per second, an altitude of 850,000 feet, an attitude of 13.6° , and a range of 420 nautical miles. Since the vehicle is designed to coast from this point to orbital altitude, it will coast to an apogee of 1,584,000 feet at a range of 1200 nautical miles. Ultimate impact from apogee will be 2400 miles down range.

6. Failure at Final Stage Ignition

Final stage ignition occurs at $t = 585$ seconds at the end of the coast phase. The initial conditions at this point are a velocity of 17,781 feet per second, an attitude of 2.5° , and an altitude of 1,570,000 feet. If the final stage fails to ignite, the final stage will attain an apogee of 1,584,000 feet, 1200 nautical miles down range. The final stage would ultimately impact 2500 nautical miles down range.

C. DEFINITION OF FINAL STAGE FAILURES*

Final stage failures are broadly defined as those failures which occur during orbital injection of the satellite vehicle. These failures occur from 585 seconds after launch to 691 seconds after launch at altitudes from 1,570,000 feet to 1,584,000 feet during the final stage boost phase. Though there is a broad spectrum of points to consider, three distinct types of failures have been analyzed. During this phase the vehicle is beyond the capability of the range safety destruct system.

The following specific types of final stage failures were considered:

* R. Oehrli, W. Hagis, C. Milewski

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- (1) Propulsion failures
- (2) Vertical thrust misalignment (pitch)
- (3) Horizontal thrust misalignment (yaw).

1. Propulsion Failures

Propulsion failures considered third stage engine failure after various burning times. The effect of third stage engine failure is to assign to the third stage vehicle various velocities short of orbital velocity. However, once the final stage is ignited, it is probable that the Hustler engine will burn to completion for orbital injection.

$$\Delta V = V_o - V_{pf}$$

where

V_o = orbital velocity

V_{pf} = velocity due to propulsion failure

Velocity increments of 400, 600, 2000, 4000 and 6000 feet per second were considered.

The following parameters were employed for the final stage:*

W_F , fuel weight = 6640 pounds

I , specific impulse = 266.43 pounds/pound/second

t , burning time = 106.4 seconds

Initial flight path angle = 2°

Initial altitude = 1,584,000 feet

Initial velocity = 17,781 feet per second

Initial longitude = $121.67^\circ W$

Initial latitude = $16.37^\circ N$

Orbit = polar

* Estimated from available final stage data.

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2. Vertical Thrust Misalignment (pitch)

The final stage was analyzed for pitch misalignments at final stage burnout velocities minus various velocity increments (ΔV). For various increments, the center line of thrust was instantaneously tilted up from the flight path at angles of 5° , 45° , and 90° .

The following cases were analyzed:

<u>ΔV</u>	<u>$\gamma_p(o)$</u>	<u>$\gamma_y(o)$</u>
600	5, 45, 90	0
2000	5, 45, 90	0
4000	5, 45, 90	0
6000	5, 45, 90	0

The deviations considered were the most conservative conditions that could be imposed, through 45° and 90° deviations are not within the realm of credibility. Actually, even with total failure of the final stage guidance system, the vehicle will tend to stay on its prescribed trajectory.

3. Horizontal Thrust Misalignment (yaw)

Lateral deviations from the prescribed trajectory were considered using various velocity increments (ΔV). The following cases were considered:

<u>ΔV</u>	<u>$\gamma_p(o)$</u>	<u>$\gamma_y(o)$</u>
600	0	$\pm 5, 45, 90$
2000	0	$\pm 5, 45, 90$
4000	0	$\pm 5, 45, 90$
6000	0	$\pm 5, 45, 90$

Again 45° and 90° deviations are not credible because of the tendency of the vehicle to follow the prescribed trajectory after a guidance failure, hence the results of this analysis are quite conservative.

Figure 28 shows the method of analytical procedure employed for final stage failures. Figure 29 shows the end points for various thrust-cutoffs or thrust tilts during final stage operation.

Fig. 28. Stages of Analytical Procedure

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Fig. 29. Elevation Showing End Points for Various Thrust Cut-offs or Thrust Tilts During Third-Stage Operation--Standard Core

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VII. DEFINITION OF FORCES IMPOSED ON GENERATORS BY ABORT

The forces imposed on the Task 2 generators by the aborted missions described in Chapter VI are discussed here. Each broad type of failure (launch failure, ascent failure, and final stage failure) and the transient forces which result are considered. The forces imposed are categorized as thermal, mechanical, or chemical.

A. LAUNCH FAILURE FORCES

1. Thermal Forces

The thermal forces imposed on the generator by launch failures originate from the ignition or detonation of the propellants of the first and final stages of the missile. The thermal forces can be applied to the generator on the launch pad or in mid-air. Maximum forces are presented.

Launch pad fire (fallback).- As the Atlas propellants are spilled to the launch pad and ignited following a tank rupture, the missile will disintegrate vertically into the conflagration. After several seconds, both the Task 2 generator and the Hustler propellant tanks will enter the initial fireball. Almost immediately the Hustler propellants will ignite adding a second fireball. The generator will be located either in or in proximity to this environment.

The first fireball will be several hundred feet in diameter and will rise at a velocity of 15 to 20 feet per second. After about five seconds, the liquid oxygen is expended and the RP-1 and metal fragments will burn on the ground. After the center of the first fireball rises about 30 feet, the second fireball will form near the pad and will have about the same characteristics as the first except for a smaller diameter. The launch pad fire chronology is illustrated in Fig. 30.

From Fig. 30, it is apparent that the Task 2 generator might be exposed to the following temperature environment:

<u>Origin</u>	<u>Temperature (°F)</u>	<u>Duration (sec)</u>
Fireball (Atlas and Hustler)	6000	Several micro-seconds
Pad fire (fuel and metals)	4500	4
Pad fire	4500 to 3000	3
Pad fire	3000 to 2000	490
Pad fire	2000 to 1500	400
Pad fire	1500 to 100	300
Total duration		1200

Time - Time After Launch

Fig. 30. Chronology of Launch Pad Fire

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It is difficult to predict the ability of the Task 2 generator to withstand this environment. However, a rough approximation of events can be made. It appears that the first seven seconds of exposure above 3000° F will seriously damage or completely consume the outer aluminum skin, Min-K insulation and inner stainless steel shell of the biological shield. This will expose the core support structure containing the core, at its operating temperature of 1680° F, to a temperature environment of 2000 to 3000° F for 490 seconds. This is the critical period for the core, since stainless steel and superalloy cores have melting points of 2350 to 2600° F.

However, many variables will influence the heat input into the fuel core. First, of all the heat available in the region of the fire where the core is located, only a portion will reach the core. Second, the ability of the core to radiate the internal heat from the radiocerium fuel and the additional heat input from the propellant fire will be a factor. The core will also lose some heat by conduction and convection. Third, it is conceivable that the generator would be ejected from the region of conflagration by the structural failure of the missile or by an ejection device incorporated in the final stage. However, the above temperature environment from a launch pad accident is a very conservative one in that it presents the maximum temperature history.

Past tests of typical fuel cores have indicated that the fireball environment is of such short duration that Colmonoy (melting point 1950° F) coatings on molybdenum test specimens did not melt when exposed to the fireball. In December 1959, tests will be conducted in which Task 2 fuel cores will be exposed to a fireball and sustained metal and fuel fire.

Mid-air launch fire.- The mid-air ignition or detonation of the Atlas and Hustler propellants at 100 to 1000 feet altitude would be characteristic of post-lift-off propulsion or guidance failures. The thermal forces presented by this type of accident would be of lesser magnitude and shorter duration than the thermal forces presented by the launch pad failure described above. The detonation of propellants in mid-air would subject the generator to two successive fireballs each having a peak temperature of 6000° F and a duration of several microseconds. During this event, mechanical forces (gravity, drag, shock overpressures) would separate the generator from the fire in several seconds. The fire would rapidly diffuse and dissipate. Since the generator would not be confined in the thermal environment and the thermal environment is of short duration, the fuel core would not sustain thermal damage.

2. Mechanical Forces

The mechanical forces applied to the Task 2 generator are applied either as shock loads or impact loads.

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Shock loads.- Recent work by Smalley and Anderson of STL has yielded a formula for the expected overpressures from a liquid oxygen RP-1 explosion.

$$P = \frac{593}{Z^3} - \frac{28.8}{Z^2} + \frac{20.7}{Z}$$

where

$$Z = \frac{R}{W^{1/4}}$$

P = average side on overpressure (psi)

R = distance from explosion (feet)

W = total weight of propellants (pounds)

Though this and similar formulae do not specifically apply for the distances considered here, they serve as a rough approximation of overpressures. For the Atlas the following computations can be made:

R = distance of generator from explosion = 35 feet

W = 260,000 pounds

Z = 1.54

P = 166 psi = pressure on Task 2 generator from explosion initiating at junction of Atlas fuel tanks.

The formula from the NDRC TNT curve might also be used:

$$P = \frac{8240}{\left(\frac{R}{Y^{1/3}}\right)^3} - \frac{166.67}{\left(\frac{R}{Y^{1/3}}\right)^2} + \frac{49.77}{\left(\frac{R}{Y^{1/3}}\right)}$$

where

Y = weight of TNT (pounds)

R = distance from explosion (feet)

P = side on overpressure (psi)

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To derive Y, however, the yield value of RP-1 and liquid oxygen in terms of pounds of TNT must be considered. For a 20% yield of liquid oxygen and RP-1

$$Y = (0.2) (260,000) = 42,000 \text{ pounds}$$

$$R = 35 \text{ feet}$$

$$\frac{R}{Y^{1/3}} \approx 1$$

$$P = 8240 - 166.67 + 49.77$$

$$P = 8100 \text{ psi}$$

These values are well within the impact strength of encapsulating materials.

Applying similar computation to the Hustler propellants yields 3531 psi from the STL equation. However, recent data on the energy yield of UDMH and IRFMA indicates that more thermal energy is released than mechanical energy (shock waves).

The shock overpressures from propellant detonations were somewhat overestimated in the earlier investigations of shock wave effects upon Task 2 type generators. Since the core materials employed have ultimate tensile strengths in the 20,000 to 40,000 psi range, they should withstand the shock loads from missile fuel detonations. However, this problem will be further investigated in February 1960, to obtain more definitive shock overpressure measurements in the distance ranges under consideration (5 to 35 feet).

Impact loads.- Impact velocities from the free fall of the Task 2 generator fuel cores vary from 62 feet per second for free fall from the top of the Atlas to about 250 feet per second for free fall from 1000 feet altitude. The latter impact velocity would apply an impact load of about 5000 psi upon a fuel core. The cores are designed to withstand impact velocities on the order of 500 feet per second, so that the loads applied during a launch failure are not excessive. Core impact temperatures would be less than 1680° F.

Impingement of objects accelerated by a detonation of propellants is another type of impact load that might be applied during a launch failure. One might consider an accelerated object traveling at an initial velocity of 2000 feet per second striking the biological shield. The force applied is a function of the density of the impinging object and its velocity ($P = \rho V^2$).

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For example, a lead slug traveling at 2000 feet per second would exert a local pressure of 600,000 psi on the biological shield and penetrate it. During penetration the missile would probably reduce its velocity to about 500 feet per second. Impact upon a fuel core at this velocity would apply a local pressure of about 30,000 psi on the core. This would penetrate the core a fraction of an inch causing a local dent in the impact area, but would not cause significant structural failure of the core. The cylindrical geometry of the core also presents a deflecting side surface for impinging objects.

3. Chemical Forces

Relatively little importance is to be placed upon the chemical forces present during a launch failure for two reasons. First, the fuel core is protected from direct exposure by the biological shield and core support structure. Second, the chemical agents present will react with one another or vaporize so that exposure will be of limited duration.

Of the reagents available, liquid oxygen, RP-1, UDMH, and IRFNA, the IRFNA could cause a small but insignificant amount of corrosion of exposed core materials. The alloys considered are categorized as slightly resistant to fuming nitric acid. The alloys considered are also characterized by their oxidation resistance at elevated temperatures.

B. ASCENT FAILURE FORCES

1. Thermal Forces

The thermal forces imposed upon the Task 2 generators by ascent failures have two distinct origins. There are forces imposed by propellant detonations similar to the mid-air launch fire described above. There are also forces imposed by aerodynamic heating from re-entry of the generators through the atmosphere following a failure.

Mid-air ascent fire.- The mid-air ignition or detonation of Hustler propellants remains as a potential event throughout the ascent phase of the missile because the Hustler engine is dormant until the end of the ascent phase. This would subject the generator to a fireball which would be constant in magnitude and intensity regardless of altitude. The fireball would present a peak temperature of 6000° F for several microseconds. During and immediately after this event, mechanical forces (gravity, drag, shock overpressures), would separate the generator from the diffusive fire in several seconds. This event would not cause thermal damage to the fuel core.

The ignition of the Atlas propellants also remains as a potential event in the earlier phases of ascent. However, after sustainer burn-out the Atlas fuel load would be expended, eliminating a propellant deto-

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nation. Following booster burnout, the Atlas propellant load would be greatly expended, minimizing a detonation. Therefore, prior to sustainer burnout, an Atlas propellant detonation could occur which would vary in magnitude and duration as a function of altitude (or fuel expended). This would probably cause the subsequent detonation of the Hustler engine.

Successive detonations would be possible up to 720,000 feet altitude which, when combined, would subject the Task 2 generators to successive fireballs. This combination of events would not thermally damage the fuel cores as has been demonstrated earlier.

Aerodynamic heating*.-

Cases B.1. and B.2.**

For the six ascent failures under consideration, significant aerodynamic heating would not be encountered on the first two cases, pitch-over (Chapter VI, B.1.), and at the point of maximum dynamic pressure (B.2.).

Case B.3.

Failure at booster separation would cause the generators to re-enter the earth's atmosphere from an apogee of 586,000 feet. The resultant heat input is relatively low and would produce a maximum stagnation temperature of about 2000° F upon the outer shell of the generator. This would not be sufficient to melt the outer shell and expose the core. Therefore, integrity of the core would be maintained. The core impacts at a temperature of 1600° F.

Case B.4.

Failure at sustainer burnout subjects the generator to the re-entry heating environment shown in Fig. 31. The heating rate is above 100 Btu/ft² sec for a period of 20 seconds with a peak heating rate of 231 Btu/ft² sec. The aluminum skin of the final stage and the aluminum and stainless steel shells of the generator biological shield melt, exposing the core at the latter part of the heating cycle. The temperature rises to about 2000° F, but the core does not melt. The core impacts at a temperature of about 1600° F, intact.

Case B.5.

Failure at vernier burnout subjects the generator to a re-entry heating cycle similar to but slightly more severe than that imposed by sustainer burnout abort. The heating rate is above 100 Btu/ft²-sec for

* R. Oehrli, W. Hagis, C. Milewski

** Data on the cases referred to are summarized in Chapter VIII.

Fig. 31. Heating Rate vs Time, Ascent Failures B.4. and B.5.

20 seconds, with a peak heating rate of $238 \text{ Btu/ft}^2 \text{ sec}$. The aluminum skin of the final stage and aluminum and stainless steel shells of the generator biological shield melt, exposing the fuel core at the latter part of the heating cycle. The core rises in temperature but does not melt, providing for intact impact. The re-entry heating environment is shown in Fig. 31. The temperature at impact is about 1600° F .

Case B.6.

Failure of final stage ignition imposes a slightly more severe re-entry heating environment than that described above. This case is intermediate between (1) failure at vernier burnout and (2) final stage failure from Hustler engine cutoff yielding a 6000 feet per second velocity increment short of the planned orbital velocity (Case 5e of final stage failures). The probable re-entry heating effect is melting of the aluminum and stainless steel shells of the generator biological shield, and melting of the outer surface of the core. The core would probably impact with its outer walls somewhat reduced in thickness and at a temperature of about 1600° F .

2. Mechanical Forces

The mechanical forces applied to radioisotope fuel cores as a consequence of ascent failures are applied as shock loads or impact loads.

Shock loads.- The maximum shock load applied during any propellant detonation is described in section A.2. above. Mid-air detonations on the ascent phase would always be less in magnitude than those on the launch pad for several reasons. First, the Atlas propellants are expended at sustainer burnout leaving the Hustler propellants at that point as the only source of detonation. Second, the propagation of shock waves is less at higher altitudes due to decreased air density. Third, the propellants are not confined, reducing the mechanical (or shock) energy release.

Therefore, the shock loads imposed by Atlas propellant detonations would not exceed about 8100 psi during pitchover, maximum dynamic pressure, and booster separation aborts (Cases B.1., B.2., and B.3.). There would be no Atlas propellants available as a source of shock waves during sustainer burnout, vernier burnout, and final stage ignition aborts (Cases B.4., B.5., and B.6.).

The shock loads imposed by Hustler propellant detonations would probably not exceed 3500 psi during any of the ascent aborts considered.

Therefore, it is concluded that fuel cores will not sustain significant damage from shock loads applied during ascent failures.

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Impact loads.- Impact loads from accelerated fragments generated by propellant detonations have been described. The mechanical integrity of fuel cores is maintained during this event.

However, impact upon the Earth's surface following ascent abort (Case B.6.) does result in marginal core integrity*. Figure 32 shows the theoretical impact pressure as a function of impact velocity for cores impacting upon rigid target media. These are the maximum impact pressures that could be encountered, since they represent a rigid target. Actually, impact from ascent aborts would not occur upon rigid media. For all ascent aborts considered, the impact media are nonrigid, i.e., either water or unconsolidated soil. Impact temperatures are also considered because of their bearing upon the impact strength of core materials.

Cases B.1. and B.2.

Following pitchover and maximum dynamic pressure aborts, fuel cores would impact either close to the launch pad (Case B.1.), or 0.1 mile down range on VAFB. In this area of the base the surface geologic regime consists primarily of unconsolidated Quaternary beds consisting of dune sand, alluvium, and terrace sediments. These impact media are nonrigid. Hence, at core impact velocities of 250 feet per second (B.1.) and 500 feet per second (B.2.) at an impact temperature of 1600° F, the cores would not fail. They would ultimately come to rest 4 to 8 feet beneath the surface intact. These are maximum impact conditions based upon the loss of the outer shell of the generator and subsequent free fall of fuel cores.

Cases B.3, B.4., and B.5.

Following booster separation abort (B.3.) the fuel cores would impact in the Pacific Ocean 640 miles down range at a temperature of about 1600° F and at a maximum terminal velocity of 500 feet per second. Following sustainer burnout abort (B.4.) and vernier burnout abort (B.5.), the same impact conditions would be imposed. However, in the latter two cases, impact would occur at respective down range distances of 2290 and 2400 nautical miles out in the South Pacific Ocean. These aborts do not result in the loss of integrity of fuel cores.

Case B.6.

Following the failure of the Hustler to fire, the fuel core would be subject to a re-entry environment which would reduce the core wall thickness slightly. The core would impact at 500 feet per second at 2000° F, approximately 2500 miles down range. Due to the reduction in wall thickness and reduction of impact strength from heat, marginal core integrity might result.

* Core integrity is defined as the ability of the core to retain 100% of the radiocerium immediately after impact.

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Fig. 32. Impact Pressure as a Function of Core Impact Velocity (impact on rigid target media)

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3. Chemical Forces

Cases B.1. and B.2.

Significant chemical forces are not imposed upon cores following pitchover and maximum dynamic pressure aborts. In these cases, cores would be buried at a depth of 4 to 8 feet in unconsolidated soil for a short finite time period. Minor quantities of reducing agents such as dilute sulfuric acid could be introduced to the cores through the percolation of meteoric water. However, as all of the core alloys considered are resistant to acid attack and the exposure time is short, no significant chemical attrition is anticipated.

Cases B.3., B.4., and B.5.

These cases of abort would introduce intact cores into a saline water environment. Assuming corrosion would advance at a rate of 0.003 inches per year, the sea water corrosion rate for Hastelloy B, it would take approximately 50 years to penetrate the wall of a multitubular fuel capsule to expose the radiocerium. This time represents 64 half-lives of Cerium-144.

Case B.6.

Failure of the final stage engine at ignition would also introduce cores to a saline water environment. Assuming that the wall thickness of the fuel capsule is reduced by a factor of 2 by aerodynamic heating, it would take approximately 25 years (32 half-lives Ce-144) to penetrate the wall of a multitubular fuel capsule.

C. FINAL STAGE FAILURE FORCES

1. Thermal Forces

The thermal forces imposed upon the Task 2 generators by final stage failures have two distinct origins. These are the thermal energy liberated by Hustler propellant detonations or hypergolic reactions, which is minor, and the aerodynamic heating of generators by re-entry to the earth's atmosphere. Aerodynamic heating is quite significant in all of the 29 cases of final stage failure considered.

Mid-air fire.- The detonation of the total Hustler propellant load (6570 pounds), as shown previously, would yield a fireball of 6000° F for microseconds duration and would expose the generators to temperatures above 4500° F for several seconds. This would not thermally damage fuel cores. Of the final stage failures considered, the following velocity increments would correspond to the following unexpended fuel loads available for detonation.

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<u>ΔV (fps)</u>	<u>Propellant Expended (%)</u>	<u>Propellant Remaining (%)</u>
400	97	3
600	95	5
2000	82	18
4000	55	45
6000	20	80

Obviously, any case considered would yield less thermal energy than the cases of mid-air Hustler propellant detonation discussed previously. Hence, this effect can be neglected.

Aerodynamic heating*.- In the final stage failures analyzed the overall effects of Hustler engine detonation were precluded.

C.1. Propulsion Failures

Of the propulsion failures of the final stage considered, all cases (5a-e) will result in extreme aerodynamic heating effects upon fuel cores. In Cases 5a, b, and c cores are completely consumed by aerodynamic heating at altitudes varying from 234,700 feet to 121,250 feet. Figure 33 shows the aerodynamic data as a function of range from third stage ignition for Case 5a. Figure 34 shows the disintegration of the generator and fuel cores by aerodynamic heating for Case 5a.

The latter two cases of Hustler propulsion failure (5d, e) will result in partial burnup of fuel cores. In Case 5d the fuel core is 70% burned up and impacts at a temperature of 2379° F. In Case 5e the fuel core is 28% burned up and impacts at a temperature of 2169° F. Figure 35 shows the aerodynamic data as a function of range for Case 5d. Figure 36 shows the disintegration of the generator and core by aerodynamic heating.

C.2. Vertical Thrust Misalignment

In Cases 1a-c of vertical thrust misalignment, Cases 1a and b result in orbital injection. Case 1c results in 100% burnup of the core down to an altitude of 172,000 feet.

* R. Oehrli, W. Hagis, C. Milewski

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Fig. 33. Trajectory 5A Aerodynamic Data as Function of Range from Third-Stage Ignition

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Fig. 34. Size of Re-Entry Body and Average Heating Rate vs Range

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Fig. 35. Trajectory 5D Aerodynamic Data as Function of Range from Third-Stage Ignition

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Fig. 36. Size of Re-Entry Body and Average Heating Rate vs Range

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In Cases 2a-c of vertical thrust misalignment, a complete spectrum of results is presented: Case 2a goes into orbit, Case 2b burns up at 163,300 feet altitude, and Case 2c impacts intact at a temperature of 2195° F.

In Cases 3a-c, Case 3a goes into orbit, Case 3b burns up completely at an altitude of 118,200 feet, and Case 3c impacts intact at a temperature of 1688° F. Figure 37 shows the aerodynamic heating rate as a function of time for Case 3c.

In Cases 4a-c, Case 4a goes into orbit, Case 4b is 90% consumed by aerodynamic heating, and the core remnants impact at a temperature of 2460° F. Case 4c impacts intact at a temperature of 2098° F.

Summarizing the 12 final stage vertical misalignment failures, 5 result in orbital injection without any aerodynamic heating; 3 result in 100% burnup between 118,200 and 172,000 feet altitude; 3 result in minor aerodynamic heating but impact intact; and 1 results in 90% burn-up before impact.

C.3. Horizontal Thrust Misalignment

Twelve cases of horizontal thrust misalignment of the final stage were considered.

Of Cases 6a-c, Cases 6a and 6b result in orbital injection and Case 6c results in 100% burnup of the fuel core at 185,000 feet altitude.

In Cases 7a-c, Case 7a results in orbital injection. Case 7b results in 100% core burnup at 178,400 feet altitude. Case 7c results in 100% core burnup at an altitude of 105,000 feet.

In Cases 8a-c, Case 8a results in orbital injection. Case 8b results in 100% core burnup at 169,200 feet altitude. Case 8c results in the fuel cores being 80% consumed by aerodynamic heating. The core remnants impact at a temperature of 2267° F.

Cases 9a-c give a complete spectrum of results. Case 9a results in orbital injection. Case 9b results in 100% core burnup at an altitude of 134,960 feet. Figure 38 shows the heating rate with respect to time for Case 9b. In Case 9c, the core is 44% consumed by aerodynamic heating at impact, and core remnants impact at 2253° F.

In summary, for the 12 cases of horizontal thrust misalignment considered, 5 cases result in orbital injection; 5 cases result in 100% core burnup between 105,000 and 185,000 feet altitude; and two are partially consumed by aerodynamic heating prior to impact.

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Fig. 37. Aerodynamic Heating Rate as a Function of Time, Case 3c.

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Fig. 38. Aerodynamic Heating Rate vs Time, Case 9b.

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2. Mechanical Forces

The mechanical forces considered are primarily those generated by impact of complete fuel cores or core remnants upon the earth's surface following re-entry to the earth's atmosphere. Of the 29 cases of final stage failure considered, only 3 cases (all resulting from vertical thrust misalignment) result in impact of complete cores. Five cases result in partial burnup of cores prior to impact.

Impact of complete cores.- Cases 2c, 3c, and 4c of final stage vertical thrust misalignment failures result in impact of complete cores. Case 2c impacts in the South Pacific Ocean near the Antarctic circle at a temperature of 2195° F with an impact velocity of 500 feet per second. The impact strength of the fuel core would be quite low (ultimate tensile strength = $< 10,000$ psi) so that somewhat marginal core integrity would result.

Case 3c impacts in the South Pacific Ocean at a velocity of 500 feet per second at a temperature of 1688° F. Since the impact temperature is low, the core impact strength would be relatively good (ultimate tensile strength = $< 20,000$ psi). Therefore, core integrity would be maintained.

Case 4c impacts in the South Pacific Ocean at a velocity of 500 feet per second at a temperature of 2098° F. At this temperature the impact strength of the core would be low (ultimate tensile strength = $< 10,000$ psi). This would present a marginal case of core integrity.

Impact of core remnants.- Cases 4b, 5d, 5e, 8c, and 9c result in the impact of core remnants in the South Pacific Ocean or on the Antarctic Continent.

Case 4b impacts at the South Pole with 90% of the core consumed by aerodynamic heating and the remaining 10% molten. Since impact occurs in Antarctica probably upon a rigid ice target with the core remnant in a plastic condition (impact temperature 2460° F), the core material will be scattered by impact at a velocity of 500 feet per second.

In Cases 5d and 5e core remnants (30% and 72% of the cores) will impact south of the Equator in the South Pacific Ocean at a velocity of about 500 feet per second. The impact temperatures are 2379° F for 5d and 2169° F for 5e which would result in plastic deformation and marine dissemination of the fuel and core alloys on impact.

In Cases 8c and 9c the core remnants (20% and 56% of the complete cores) will impact in the South Pacific Ocean at about 500 feet per second. The impact temperatures are 2267° F for 8c and 2253° F for 9c. This would result in marine dissemination of the fuel and core alloys.

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3. Chemical Forces

Following impact from final stage failures the salt water attack upon complete fuel cores or remnants of cores would cause, in some cases, corrosion of fuel materials.

Only in one case (4b) does a core remnant impact upon ice. In this case, only 10% of the core would be present at impact, representing an unhomogeneous mass of radiocerium fuel and core alloy. However, since this remnant is emitting thermal energy, it would tend to melt the ice and move downward into the ice at a finite decreasing rate under the force of gravity.

In Cases 5d, 5e, 8c, and 9c, core remnants would sink to the floor of the Pacific Ocean and undergo corrosion and possibly solution. The chemical admixture presented by core remnants and the reactions of free radicals with this admixture are quite complex. These phenomena and also the marine ecological path of radiocerium are beyond the scope of this report, but the problems posed suggest future investigations in this area.

In Cases 2c, 3c and 4c cores would be subject to corrosion by sea water. The corrosion rate of 0.003 inches per year would lead to exposure of fuel from a multitubular fuel core in approximately 50 years (64 half-lives of Ce-144).

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VIII. FATE OF RADIOCERIUM FUEL

This section presents a summary of the three general types of missile failures discussed. The fate of the radiocerium fuel after all types of aborted missions is predicted. Figure 39 is a map showing the location of the radiocerium fuel following abort.

A. LAUNCH FAILURES

Table 8 summarizes launch failures, failure forces, and the fate of the radiocerium fuel for typical launch failures. In almost all cases core integrity is maintained. A very remote possibility of a minor release exists when a cerium metal fuel core is exposed to a sustained propellant fire. In all cases of launch abort, fuel cores impact within the controlled areas of VAFB. Here fuel cores can be retrieved for safe disposal*

Further examination of contamination by radiocerium from a launch failure is warranted. The critical time interval when the core is exposed to a sustained propellant fire is the 490 second period when the fire temperature is between 2000 to 3000° F. During this period, core alloys (melting points 2350 to 2600° F) could conceivably melt and release fuel pellets to this environment. The mere exposure of pellets does not constitute a contamination hazard unless the fuel materials vaporize or become mechanically powdered and are carried by convective air currents. Only two fuel forms, cerium metal and cerous fluoride, might vaporize. The remaining fuel forms have melting points above 4000° F. Since the remaining fuel forms have been initially pressed under 25 tons per square inch pressure and sintered at elevated temperatures, it is unlikely that they would become powdered mechanically and dispersed as aerosols.

B. ASCENT FAILURES

Table 9 summarizes the six ascent failures discussed. In the first two cases (B1, 2) the cores impact upon the launch base maintaining core integrity so that safe disposal can be achieved.

In Cases B3, 4, and 5, cores impact down range in the Pacific Ocean and sink to the bottom intact. At this point, salt water corrosion attacks the core alloys at a maximum corrosion rate of 0.003 inch per

* A study has been conducted by the Cornell Aeronautical Laboratory for the Air Force Special Weapons Center, investigating various methods of retrieving fuel cores following launch abort.

Fig. 39. Polar Projections Showing Abort Areas and Orbital Traces
(zone of 75% fallout shown)

TABLE 8
Summary of Launch Failures (A.1, 2., 3.)

Vehicle Failure Considered	Time T O at Launch (sec)	Altitude (ft)	Cause	Nature of Energy Release	Fate of Missile After Failure	Forces Presented by Abort Considered			Immediate Fate of Fuel Capsule		Ultimate Fate of Radiocerium Fuel
						Thermal	Mechanical	Chemical	Capsule Integrity Maintained	Location After Abort	
A. Launch Failures (Summary)	-2000 to 15	0 to 1000	1. Propellant tank 2. Propulsion 3. Guidance	Atlas and Hustler propellants ignited in air or on ground	1. Fallback 2. Midair disintegration 3. Self-destruct 4. Command destruct	Fallback fire on pad--2 successive fireballs and sustained pad fire or Midair fireballs of short duration	1. Shock loads from propellant detonations 2. Impact loads from: Free fall impact, Impact with accelerated objects	Minor IRFNA attack possible	Except when capsule is immersed in sustained fallback fire	On VAFB in missile debris or exposed on ground	Retrieved for safe disposal, minor loss of fuel is possible if cerium metal or fluoride is used
1. Propellant Tank Failure	-2000 to 15	0 to 1000 (usually occurs at lower altitudes)	1. Seam rupture 2. Over-pressurization	Atlas and Hustler propellants ignited Confined or nonconfined release	Fallback or midair disintegration	1. Fallback Fire Temperature (°F) Duration (sec) 6000 10 ⁻⁵ 4500 4 4500 - 3000 3 3000 - 2000 490 2000 - 1500 400 1500 - 100 300 2. Midair Fire 6000 10 ⁻⁵ 4500 2	1. Shock loads: Atlas and Hustler propellant detonate Maximum overpressures LOX-RP-1 166 to 8100 psi UDMH-IRFNA 3531 psi 2. Impact loads: Accelerated objects 2000 fps Impact on ground-- 250 fps at impact temperature-- 1680° F.	Minor attack by IRFNA possible	(Fuel leak on pad is remote possibility during sustained propellant fire if capsule exposed to fire)	On launch pad in missile debris On ground at VAFB either in missile debris or exposed on/or near ground surface	Retrieved for safe disposal (decontamination and minor loss of fuel is possible if cerium metal or fluoride is used
2. Propulsion Failure	0 to 15	0 to 1000 (occurs at or after lift-off)	1. Fuel flow restriction 2. Incomplete combustion	Atlas and Hustler propellants ignited Confined or nonconfined release	Fallback or midair disintegration	Same as above, except less rigorous thermal environment for failures well after lift-off	Same as above, except less rigorous shock overpressures for failures well after lift-off	Same as above	Same as above	Same as above	Retrieved for safe disposal (decontamination and minor loss of fuel is possible if cerium metal or fluoride is used
3. Guidance Failure	5 to 15	100 to 1000	1. Electrical malfunction 2. Mechanical malfunction	Atlas and Hustler propellants ignited Nonconfined release	Mechanical self- or command destructs Midair disintegration	Midair Fire Temperature (°F) Duration (sec) 6000 10 ⁻⁵ 4500 2	Same as A-1 except much lower shock overpressures from propellant detonations	No significant chemical forces	Yes	Near launch pad in missile debris On ground at VAFB either in missile debris or exposed near ground surface	Retrieved for safe disposal on government waste burial ground

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TABLE 9

Summary of Ascent Failures (B.1, 2., 3., 4., 5., 6.)

Vehicle Failure Considered	Time T = 0 at Launch (sec)	Cause	Initial Conditions				Apogee		Impact Range (naut mi)	Forces Imposed During Abort			Fate of Immediate Fuel Core		Ultimate Fate of Radiocerium
			Altitude (ft)	Attitude (deg)	Velocity (fps)	Range (naut mi)	Altitude (ft)	Range (naut mi)		Thermal	Mechanical	Chemical	Location After Abort	Integrity Maintained	
B. Ascent Failures* (Summary)	15 to 585	1. Propulsion 2. Guidance 3. Structural 4. Improper separation of stages	1000 to 1,570,000	90 to 2.5	200 to 17,780	0 to --	1000 to 1,584,000	0 to 1200	0 to 2500	Midair propellant fire and aerodynamic heatings	Propellant detonations (shock) Accelerated fragments and impact on earth	Very minor attack by soil acids Sea water corrosion at 0.003 in./yr	Burial on base or burial at sea 0 to 2500 naut mi down range	Yes, with marginal Case (B-6)	1. Safe disposal on government waste burial ground 2. Disposal in Pacific Ocean 3. Possible marine release for marginal Case (B-6)
1. Failure at Pitchover	15	1. Guidance 2. Propulsion 3. Structural	1000	90	200	0	1000	0	0	Midair propellant fire (Atlas and Hustler) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating negligible	Shock loads (Atlas and Hustler) shock overpressures <8100 psi Atlas <3500 psi Hustler Impact loads 1. Accelerated fragments 2. Impact on soil at 250 fps Impact temperature 1600° F	Minor attack by soil acids	Core and generator debris on base near launch pad	Yes	Disposal on government waste burial ground
2. Failure at Maximum Dynamic Pressure	60	1. Structural 2. Guidance 3. Propulsion	38,000	54	950	0.1	41,000	0.5	1.0	Midair propellant (Atlas and Hustler) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating negligible	Shock loads (Atlas and Hustler) shock overpressures <8100 psi Atlas <3500 psi Hustler Impact loads 1. Accelerated fragments 2. Impact on soil at <500 fps Impact temperature 1600° F	Minor attack by soil acids	Core buried 6 to 8 feet 1 mile down range from launch pad	Yes	Disposal on government waste burial ground
3. Failure at Booster Separation	147	1. Propulsion (sustainer) 2. Guidance 3. Booster does not separate properly	250,000	23	10,890	60	586,000	320	640	Midair propellant fire (Hustler D) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating Skin temperature 2000° F (maximum, low heat input)	Shock loads (Hustler and Atlas) shock overpressures <1000 psi Atlas <3500 psi Hustler Impact loads 1. Accelerated fragments 2. Impact on water at 500 fps Impact temperature 1600° F	Sea water corrosion 0.003 in./yr	Core on bottom of Pacific 640 miles down range near Guadalupe Island	Yes	Containment of cerium on floor of Pacific for 50 years. Release of 10 ⁻¹³ curies thereafter
* After abort trajectories by D. B. Cross and R. Oehrli															

* After abort trajectories by D. B. Cross and R. Oehrli

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TABLE 9 (continued)

Vehicle Failure Considered	Time T O at Launch (sec)	Cause	Altitude (ft)	Initial Conditions		Range (naut mi)	Apogee		Impact Range (naut mi)	Forces Imposed During Abort			Fate of Immediate Fuel Core		Ultimate Fate of Radiocerium
				Attitude (deg)	Velocity (fps)		Altitude (ft)	Range (naut mi)		Thermal	Mechanical	Chemical	Location After Abort	Integrity Maintained	
4. Failure at Sustainer Burnout	260	1. Guidance	720,000	14.8	17,000	340	1,417,000	1145	2290	Midair propellant fire (Hustler only) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating Skin stagnation heating rate of 100 Btu/ft ² sec for 20 seconds Peak heating rate 231 Btu/ft ² sec	Shock loads (Hustler) shock overpressures < 3500 psi Overpressures minimized by rarified atmosphere Impact loads 1. Accelerated fragments 2. Impact on water at 500 fps Impact temperature 1600° F	Sea water corrosion 0.003 in./yr	Core on bottom of Pacific 200 miles south of equator in uninhabited region east of Society Islands	Yes	Containment of cerium on floor of Pacific for 50 years. Release of 10 ⁻¹³ curies thereafter
5. Failure at Vernier Burnout	289	1. Guidance 2. Propulsion 3. Failure to separate from Atlas	850,000	13.6	17,700	420	1,584,000	1200	2400	Midair propellant fire (Hustler only) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating Skin stagnation heating rate of 100 Btu/ft ² sec for 100 seconds Peak heating rate of 238 Btu/ft ² sec	Shock loads (Hustler) shock overpressures < 3500 psi Overpressures minimized by rarified atmosphere Impact loads 1. Accelerated fragments 2. Impact on water at 500 fps Impact temperature 1600° F	Sea water corrosion 0.003 in./yr	Core on bottom of Pacific 300 miles south of equator east of Society Islands	Yes	Containment of cerium on floor of Pacific for 50 years. Release of 10 ⁻¹³ curies thereafter
6. Failure at Final Stage Ignition	585	1. Failure of final stage to ignite 2. Guidance	1,570,000	2.5	17,780*	--	1,584,000	1200	2500	Midair propellant fire (Hustler only) 6000° F for microseconds 4500° F for 2 seconds Aerodynamic heating Stagnation heating rates intermediate between B-5 above and C-5e below)	Shock loads (Hustler) shock overpressures < 3500 psi Overpressures minimized by rarified atmosphere Impact loads 1. Accelerated fragments 2. Impact on water at 560 fps Impact temperature 2000° F	Sea water corrosion 0.003 in./yr	Core or core remnants on bottom of Pacific 400 miles south of equator	Marginal	1. Containment of cerium on floor of Pacific for 25 years. Release of 5 x 10 ⁻⁴ curies thereafter 2. If core fails on impact, gradual release on floor of Pacific

* Corresponds to A - ΔV of 7242 fps

TABLE 10

Summary of Final Stage Failures

Vehicle Condition Considered	Failure		V ₀ - V _{FS} ΔV (fps)	Misalignment		End Points		% of Core Consumed at Impact	Core Burnup Altitude (ft)	Impact Temperature (°F)	Core Integrity Maintained	Immediate Fate of Radiocerium					Principal Radiological Hazard Presented
	Time (sec)	Cause		Vertical 8° (deg)	Horizontal 8° (deg)	Latitude (deg)	Longitude (deg)					% in Hydrosphere	% in Atmosphere	% on Lithosphere	% in Orbit	Location of Radiocerium	
C. Final Stage Failures (Summary)	585 to 691	1. Propulsion cutoff 2. Guidance Vertical or horizontal thrust misalignment	400 to 6000	0 to 90													
1. Propulsion Failure	585 to 691	Propulsion															
5a*			400	0	0	47.9S	40.0E	100 burnup	234,700	--	No	0	100	0	0	Over Indian Ocean	Direct fallout
5b			600	0	0	80.0S	127.9W	100 burnup	179,837	--	No	0	100	0	0	Over Antarctica	Direct fallout
5c			2000	0	0	36.0S	125.3W	100 burnup	121,250	--	No	0	100	0	0	Over Pacific Ocean	Direct fallout
5d			4000	0	0	20.0S	125.1W	70	--	2379	No	30	70	0	0	Over and in Pacific Ocean	Direct fallout
5e			6000	0	0	Not recorded		28	--	2169	No	72	28	0	0	Over and in Pacific Ocean	Direct fallout
2. Vertical Thrust Misalignment	585 to 691	Propulsion or guidance															
1a			600	5	0	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
1b			600	45	0	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
1c			600	90	0	76.9S	50.5E	100 burnup	172,000	--	No	0	100	0	0	Over Antarctica	Direct fallout
2a			2000	5	0	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
2b			2000	45	0	53.0S	48.7E	100 burnup	163,300	--	No	0	100	0	0	Over Indian Ocean	Direct fallout
2c			2000	90	0	63.9S	128.8W	0 core intact	--	2195	Marginal	100	0	0	0	In Pacific Ocean	Marine dissemination
3a			4000	5	0	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
3b			4000	45	0	77.0S	49.8E	100 burnup	118,200	--	No	0	100	0	0	Over Antarctica	Direct fallout
3c			4000	90	0	46.6S	127.4W	0 core intact	--	1688	Yes	100	0	0	0	In Pacific Ocean	None
4a			6000	5	0	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
4b			6000	45	0	88.7S	49.3E	90	--	2460	No	0	90	10	0	Over and at South Pole	Direct fallout
4c			6000	90	0	35.4S	127.5W	0 core intact	--	2098	Marginal	100	0	0	0	In Pacific Ocean	Marine dissemination
3. Horizontal Thrust Misalignment	585 to 691	Propulsion or guidance															
6a			600	0	5, -5	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
6b			600	0	45, -45	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
6c			600	0	90, -90	64.5S	150.3W -104.1W	100 burnup	185,000	--	No	0	100	0	0	Over Pacific Ocean	Direct fallout
7a			2000	0	5, -5	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
7b			2000	0	45, -45	70.4S	140.3W -104.7W	100 burnup	178,400	--	No	0	100	0	0	Over Arctic Circle	Direct fallout
7c			2000	0	90, -90	36.2S	135.2W -114.1W	100 burnup	105,000	--	No	0	100	0	0	Over Pacific Ocean	Direct fallout
8a			4000	0	5, -5	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
8b			4000	0	45, -45	56.5S	141.1W -110.0W	100 burnup	169,200	--	No	0	100	0	0	Over Pacific Ocean	Direct fallout
8c			4000	0	90, -90	20.5S	135.6W -113.4W	80	--	2267	No	20	80	0	0	Over and in Pacific Ocean	Direct fallout
9a			6000	0	5, -5	Into orbit		--	--	--	Yes	0	0	0	100	In orbit	Ultimate random fallout
9b			6000	0	45, -45	44.3S	143.3W -106.2W	100 burnup	134,960	--	No	0	0	0	0	Over Pacific Ocean	Direct fallout
9c			6000	0	90, -90	11.3S	135.7W -112.5W	44	--	2253	No	56	44	0	0	Over and in Pacific Ocean near equator	Direct fallout

* Denotes IBM-709 trajectories by R. Oehrli et al and velocity increments correspond to the following final stage burning times: ΔV 400 = 103.4 sec, ΔV 600 = 100.8 sec, ΔV 2000 = 85.6 sec, ΔV 4000 = 58.8 sec, ΔV 6000 = 21.5 sec

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year* resulting in the exposure of radiocerium in approximately 50 years. At this time, the initial source of 1.7 megacuries has decayed to 10^{-13} curies and presents no serious consequence.

Case B6 is somewhat marginal due to the aerodynamic heating and impact effects. If one assumes that the cores, with their wall thickness decreased by a factor of 2, do not fail on impact and are subject to corrosion at the bottom of the Pacific, then the fuel will be exposed after 25 years with 5×10^{-4} curies of radiocerium available at this time after decay.

The ascent failures considered result in core integrity in five cases for a period of 50 years or more. Two of the five cases facilitate ultimate safe disposal of cores after their retrieval on the base. The remaining three cases result in intact core burial on the floor of the Pacific Ocean at successive down range distances of 640, 2290, and 2400 miles. The sixth case is somewhat marginal but cannot result in fallout or surface atmospheric contamination.

C. FINAL STAGE FAILURES

Table 10 summarizes the various final stage failures and resolves each failure in terms of the principal long term radiological hazard presented. Of the three principal radiological hazards shown, (1) direct fallout, (2) ultimate random fallout, and (3) marine dissemination, the direct and ultimate random fallout pose the most significant problem.

1. Direct Fallout

Direct fallout is defined as the release of radiocerium into the stratosphere above the Southern Hemisphere as a result of 100% or partial burnup of fuel cores upon failure of the final stage to achieve orbital velocity. It will be shown that most of the radiocerium injected will fall out in the Southern Hemisphere because all of it is injected into the atmosphere or stratosphere above that region, as shown in Fig. 39.

2. Ultimate Random Fallout

Ultimate fallout, as a result of orbital decay and re-entry of the satellite vehicle at some finite time after injection of the vehicle into a defective orbit, is of a random nature. Since the point of release can occur at any geographical location, i.e., over the Northern or Southern Hemispheres, above the Equator, or over the Poles, the re-

* Based upon data provided by the Harbor Island Test Station of the International Nickel Corporation.

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sultant worldwide fallout pattern will vary for each case. The re-entry history of fuel cores following orbital decay will approximate that of Case 5a*. This is characteristic of a high altitude release (234,000 feet).

3. Fallout Pattern**

Worldwide fallout from bomb testing serves as a good example for the fallout pattern derived from an injection of radioisotopes into the stratosphere. From the total yield (90 to 92 megatons) of bomb tests to date, 140 pounds or 9.2 megacuries of Strontium-90 have been produced. Of this amount approximately 5 megacuries were injected into the stratosphere, most of it above the Northern Hemisphere. Figure 40 shows the present and predicted soil levels of Strontium-90***. Figure 41 demonstrates that fallout tends to concentrate selectively in certain latitudinal bands in the Northern and Southern Hemispheres, and also that material injected into the stratosphere above the higher latitudes of a given hemisphere will deposit almost entirely in that hemisphere. Weapons debris injected into the stratosphere will fall out in an average time of 1 to 3 years, depending upon the latitude of injection****. The lowest half-residence time suggested lately is about 4 months, which is applicable only to debris injected into the stratosphere above the Arctic. This conservative estimate of 4 months has been used as a basis for the calculations presented in this report.

Several sources indicate that during the Soviet tests in the fall of 1958, approximately 30 to 50 megacuries of Cerium-144 (about 10,000 grams) were injected into the stratosphere above the Northern Hemisphere. This compares with a maximum injection of 1.76 megacuries of Cerium-144 that would be injected into the stratosphere by a single final stage failure or by a single orbital decay and re-entry from Orbit Number 1.

Direct fallout from aborted missions.- Following final stage high altitude aborts, radiocerium is injected directly into the stratosphere above the Southern Hemisphere/. Case 2b is demonstrative of this. In this case, radiocerium is released as fine particles above an altitude of 163,300 feet at 53.0°S and 48.7°E above the Indian Ocean. To estimate the fallout from this release, the following would be assumed:

* Case 5a also approximates the re-entry history of the satellite from a successful orbit--a condition not described elsewhere in this report.

** T. J. Dobry, Martin Nuclear Division

*** Joint Committee on Atomic Energy, Fallout from Nuclear Weapons Tests, August 1959, page 41.

**** Merrill Eisenbud

/ Where partial burnup occurs and results in a release of some fuel as very large particles, fallout will be prompt over uninhabited regions of the Southern Hemisphere (i.e., over South Pacific and Indian Oceans, and Antarctica where population density is essentially zero).

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Fig. 40. Present and Predicted Soil Levels of Sr-90

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Amount Ce-144 released	1.7×10^6 curies
Half residence time	120 days
Half-life Ce-144	285 days
Deposition zone	30°S to 60°S
Ce-144 deposited in zone*	75%
Area of zone	3.6×10^7 square miles

The half residence time of 120 days employed in the calculations is representative of polar injections; actually injections occurring above the Southern Temperate zone would have a half residence time of about 1 year or longer.

The ground concentration can be calculated by the following equation:

Total deposition (χ_o) in millicuries/mile²

$$= \frac{N_o(Z)}{A} \left[e^{-\lambda_r t} \right] \left[1 - e^{-\lambda_f t} \right]$$

where

N_o = number of curies released initially 1.76×10^9 millicuries

Z = % deposited in zone 30 to 60°S = 75%

A = area of zone = 3.6×10^7 miles²

λ_r = radioactive decay constant = 2.43×10^{-3} day⁻¹

λ_f = residence decay constant = 5.77×10^{-3} day⁻¹

t = time after release = days

$$\chi_o = 36.67 \left[e^{-2.43 \times 10^{-3} t} - e^{-8.2 \times 10^{-3} t} \right]$$

and

$$\frac{d\chi_o}{dt} = 36.67 \left[-2.43 \times 10^{-3} e^{-2.43 \times 10^{-3} t} + 8.2 \times 10^{-3} e^{-8.2 \times 10^{-3} t} \right]$$

* It is assumed that the remaining 25% is deposited elsewhere on the earth's surface.

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$$\chi_o \text{ is maximum when } \frac{d \chi_o}{dt} = 0$$

Therefore

$$0 = 36.67 \left[-2.43 \times 10^{-3} e^{-2.43 \times 10^{-3} t} + 8.2 \times 10^{-3} e^{-8.2 \times 10^{-3} t} \right] e^{-5.77 \times 10^{-3} t} = 0.296$$

Then $t_{\max} = 210.7$ days

and

$$\chi_{o \max} = 15.47 \text{ millicuries/square mile}$$

Figure 41 shows the Cerium-144 deposition in the Southern Hemisphere as a function of time after injection of 1.76 megacuries. Figure 39 shows the zone of 75% deposition.

Further, one might estimate the total maximum dose that an individual could receive in the fallout zone (30 to 60°S) in the first year. The United Nations Radiation Committee Report gives the following equation:

$$I = c \bar{E} \gamma F_d^T(t) *$$

I = Exposure rate mrad/yr

$$c = 0.1 \frac{(\text{mrad})(\text{km}^2)}{(\text{year})(\text{mcurie})(\text{Mev})}$$

E = Average γ energy = 0.2 Mev (for Ce-144-Pr-144)

$$F_d^T(t) = \text{Total activity of the deposit mc/km}^2$$

The above equation yields a dose during the first year of 0.098 millirads for the above ground concentration. This will produce a 30-year does of 0.23 millirad.

The following table compares this dose with the doses received from various natural and man-made sources of radiation.

* Report of the United Nations Scientific Committee on the Effects of Atomic Radiation, 13th Session, Supplement No. 17 (A/3838), New York, 1958, page 101.

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Estimated Dose from Different Radioactive Sources*

	30 Year Genetically Significant Dose (millirad)
Natural sources	3000
Fallout from nuclear weapons	50**
Medical sources	500 to 5000
From 1.76 megacurie Ce-144 injected postulated	0.23

Having estimated the 30-year genetically significant dose, there remains the potential internal hazard from the entry of Ce-144 into food chains and man. This can best be evaluated by using the known amounts of Ce-144 produced in weapons tests as a tracer and actually measuring the amounts present in foods and human tissues. A pilot sampling program has been designed, but is not yet under way.

Ultimate random fallout from orbital decay.- For cases where orbital decay occurs on the first orbit, it is conceivable that about 1.7 megacuries of Cerium-144 would be injected into the stratosphere above the upper latitudes of the Northern Hemisphere. The net effect of such an injection would be to reverse the case previously discussed for the Southern Hemisphere. Therefore, an estimated peak ground accumulation of Cerium-144 of 15.47 millicuries per square mile would be expected between 30° to 60° N. This would yield an exposure of 98 microrad based upon a 1 year exposure time.

For cases where orbital decay occurs at 285 days (orbit 4560), 0.88 megacuries would be injected at random into either hemisphere. This would yield a peak ground concentration of about 7.73 millicuries per square mile or about 49 microrad based upon a 1 year exposure 30° to 60° N or S.

* External radiation doses, p 15, Report of the United Nations Scientific Committee on the Effects of Atomic Radiation, 1958.

** Table IV page 40, Fallout from Nuclear Weapons Tests, JCAE, May 1959.

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Fig. 42. Comparison of Cerium-144 Fallout in Northern Hemisphere (30° - 60° N) for three Injections

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Comparison to Cerium-144 fallout from bomb testing.- Figure 42 compares Cerium-144 fallout in the Northern Hemisphere between 30° to 60°N for a 30 megacurie release, a 1.76 megacurie release, and a 0.88 megacurie release. The same assumptions and mode of calculation were used in each case.

Actual measurements of Cerium-144 fallout deposition have been made by Argonne National Laboratory from soil samples by gamma ray spectroscopy. The samples were collected at Argonne National Laboratory which is located at about 42° North latitude. The following table lists the Argonne data.

Ce-144 Activity in Soil (ANL)*

<u>Month</u>	<u>Activity (mc/mi²)</u>	<u>Estimated Dose Rate (μrad/hr)</u>
September 1958	270	0.05
March 1959	1127	0.21
April 1959	1450	0.27
May 1959	1533	0.28
June 1959	1704	0.31
July 1959	1812	0.33

The above concentrations may be compared with the estimated maximum concentration (15 mc/mi²) within the Northern or Southern Temperate Zones for a 1.76 megacurie release following aborted missions or prompt orbital decay of a satellite vehicle employing the Task 2 generator.

The Joint Committee report of 1958 indicates that a total of 40 megatons of fission occurred during the 1957 to 1958 bomb tests. Of this debris 26 megatons were injected into the atmosphere. This would produce 2.6 megacuries of Strontium 90 and, based upon a 1 to 1 ratio of atoms produced, 93 megacuries of Cerium 144 would have been injected during this period. This is higher by a factor of about 3 than the 30 megacurie value assumed in this report for comparative purposes.

* Quarterly Statement of Fallout by the U.S. Atomic Energy Commission, October 1959, Table I, page 5.

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IX. PAST AND FUTURE TEST PROGRAMS

The analytical work of the Task 2 hazards investigation is supported by experiments in certain areas where theoretical methods are not well established. The areas requiring empirical investigation are shock wave phenomena, propellant energy partition, impact phenomena, complex chemical reactions, and aerodynamic heating effects. The past and future experimental programs are summarized.

A. PAST PROGRAMS

The following test programs have been conducted in the past:

- (1) High velocity impact tests
- (2) Low velocity impact tests
- (3) Liquid oxygen--molybdenum tests
- (4) Missile failure tests.

B. FUTURE PROGRAMS

The following programs are scheduled or planned for fiscal year 1960.

1. High Velocity Impact Tests

High velocity impact tests are scheduled for December 1959. The purpose of these tests is to determine the ability of test fuel cores made of various high nickel alloys to withstand impact at operating temperatures upon various rigid and nonrigid target media. Twelve rocket sled shots are planned with target media to include water, unconsolidated soil, concrete, and granite. Cores will be impacted at 500 feet per second. A fluorescent cerium compound is mixed with the cold cerium fuel so that a fuel leak can be detected.

2. Propellant Energy Release Test

The partition of energy from exploding propellants is a rather complex phenomenon near the source of the release. For this reason tests of energy releases from Atlas and Hustler propellants will be conducted. A determination will be made of the partition of energy into thermal energy (fire) and mechanical energy (shock waves) at po-

Plasma arc testing of generator components will also be conducted in 1960. The purpose of these tests is to determine the time duration that components will withstand given aerodynamic heating inputs. In this way the mode of disintegration of the generator on re-entry can be approximated.

6. Salt Water Corrosion Program

A program is being formulated in which the corrosion, pitting, and solubility of cerium compounds exposed to quiescent or moving sea water will be investigated. The purpose of this program is to predict the evolution rate of cerium in sea water when fuel pellets are exposed following certain aborted missions.

APPENDIX

SUMMARY OF PAST TEST PROGRAMS

A. HIGH VELOCITY IMPACT

Nine rocket sled tests were fired at Aberdeen Proving Ground, Maryland, between January 8 and 13, 1959. The tests consist of impacting eight SNAP I boiler assemblies upon various targets at impact angles of 45 and 90 degrees and at velocities of 478 to 1126 feet per second. One bare fuel core was fired into granite and impacted at 90 degrees at a velocity of 492 feet per second. Specimens were preheated so the temperature at impact was between 1855 and 1875° F.

From a previous re-entry analysis, it was demonstrated that the maximum impact velocity of a SNAP I boiler was 419 feet per second. Assuming impact on the earth's surface, there are three general types of rocks to consider. These are crystalline, consolidated and unconsolidated. Each of these types was simulated as targets. The impact angle favored 45 degrees according to calculations of statistical probability for impact orientation.

Tables 11 and 12 summarize the results of the tests and Figs. 43 through 51 show the test specimens. Specimens 1 through 8 maintained their integrity under the high velocity impact conditions and the simulated fuel within the fuel blocks was not released. Specimen 9 was fired at granite at a velocity of 1118 feet per second (2.6 times the calculated impact velocity from re-entry) and impacted at an angle of 90 degrees.

The conclusions indicate that the SNAP I boilers would:

- (1) Maintain their structural integrity under re-entry impact conditions.
- (2) Bury themselves four to seven feet beneath the surface when impact occurs on unconsolidated rock.
- (3) Penetrate from 1.5 to 2.5 feet into consolidated rock.
- (4) Not penetrate crystalline rock.
- (5) Not release their radioisotope fuel on impact upon any of these targets at terminal velocity.

TABLE 11
Experimental Data

Shot Number	Specimen Number and Type	Vehicle Weight (lb)	Firing Distance On Track (ft)	Pre-Heat Temperature		Time from Furnace		Impact (Calculated) (°F)	Temperature Recorded (Actual)		Impact Velocity (ft/sec)	Position of Screw Cap at Impact	Target Medium	Angle of Impact (deg)	Area (in. ²)	Impact Pressure (Calculated) (lb/in. ²)*		Depth of Penetration (in.)
				Surface (°F)	Inside (°F)	To Impact (sec)	To Recovery (sec)		Surface (°F)	Inside (°F)								
1	1	1	114.84	395	2020	1900	20	840	1870	700	1160	478	Away from target	Unconsolidated rock	45	44.3	2.25 x 10 ⁴	55
2	2	1	114.84	395	2060	1901	19	465	1870	736	--	500	At target	Unconsolidated rock	90	20.5	2.46 x 10 ⁴	87
3	4	1	114.84	395	2060	1910	20	235	1870	1338 (core)	--	503	At target	Granite	90	20.5	2.49 x 10 ⁴	None
4	8	1	114.84	395	2060	1910	32	Temperature lost	1855	--	--	472	Away from target	Consolidated rock	90	20.5	2.19 x 10 ⁴	21
5	5	1	114.84	395	2075	1905	16	225	1875	820	1338	489	At target	Consolidated rock	45	44.3	2.35 x 10 ⁴	32
6	7	1	114.84	395	2060	1901	30	570	1855	585 (core)	--	471	Away from target	Granite	45	44.3	2.18 x 10 ⁴	None
7	6	2	53.10	395	1901	--	19	220	1875	1261 (core)		492	--	Granite	90	9.7	2.47 x 10 ⁴	None
8	9	1	134.09	500	2040	1910	15	--	1875	--	--	1,118	Away from target	Granite	90	20.5	1.23 x 10 ⁵	Target disintegrated
9	3	1	134.09	500	2100	1929	15	1080	1875	450 (core)	--	1,126	At target	Unconsolidated rock	90	20.5	1.25 x 10 ⁵	140

* $P = \rho V^2$
 ρ = density
 V = impact velocity

TABLE 12
Damage Summary Specimens and Targets

Shot Number	Specimen Number	Target Medium	Description of Damage to Target	Comments	Description of Damage
1	1	Unconsolidated rock	Crater covered over after specimen, impacting at 45°, buried itself 55 in.	Soil was frozen when test was performed.	Can sustained no cracks. Side ruptured in area of impact.
2	2	Unconsolidated rock	90 deg impact, crater collapsed after specimen penetrated 87 in.	Side of the box containing soil bent out slightly from shock wave.	Can sustained no cracks. Side ruptured around periphery.
3	4	Granite	90 deg impact caused granite block to split in half.	Granite hit parallel to grain.	Can split open in two places, core compressed, no cracks developed.
4	8	Consolidated rock	90 deg impact caused block to break. Note how pressure wave pushed out plywood sides.	If target was infinite in size, specimen would have cratered it and debris would bury the specimen.	Can sustained no cracks. Side ruptured around periphery.
5	5	Consolidated rock	45 deg impact caused block to break.	Specimen would have buried itself if target was larger.	Screw cap pushed out, walls of can ruptured. Core undamaged.
6	7	Granite	45 deg impact moved granite block about 40 ft.	Specimen ricocheted about 150 yd. Distance would have been greater if target was infinite in size.	Can split open at impact point. Core compressed at surface of impact. Was found about 150 yd from target.
7	6	Granite	90 deg imp ct had no effect on block.	Specimen ricocheted about 8 ft, 180° to face of target.	Bare core compressed, one molybdenum plug stripped out. Coating peeled off.
8	9	Granite	Granite block demolished. Impact was at 90 deg.	Greater part of specimen was found at target.	Can and core demolished.
9	3	Unconsolidated rock	Crater covered over after specimen buried itself 140 in. Impact angle was 90 deg.	Sides of box pushed out due to shock wave.	Can was ripped off core, core undamaged.

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Fig. 43. Specimen No. 1 after Impact at 45 Degrees on Unconsolidated Rock

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Fig. 44. Specimen No. 2 after Impact at 90 Degrees on Unconsolidated Rock

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Fig. 45. Specimen No. 3 after Impact with Unconsolidated Rock at 90 Degrees

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Fig. 46. Specimen No. 4. Showing Exploded View of Can and Core after Impact on Granite at 90 Degrees

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Fig. 47. Specimen No. 5 after Impact with Consolidated Rock at 45 Degrees

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Fig. 48. Bare Core, Specimen No. 6 after Impact with Granite at;90 Degrees

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Fig. 49. Side View of Specimen No. 7 after Impact with Granite at 45 Degrees

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Fig. 50. Specimen No. 8 after Impact with Consolidated Rock at 90 Degrees

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Fig. 51. Specimen No. 9 after Impact with Granite at 90 Degrees
Velocity - 1,118 ft/sec

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B. LOW VELOCITY IMPACT

The low velocity impact tests were carried out at Aberdeen Proving Ground. In the tests, SNAP I cores and boilers were dropped from a height of 70 feet at approximately 1500° F, simulating the conditions of the APU within a typical missile. The specimens were impacted, with their major axis at various angles to the plane of impact, upon reinforced concrete and, in some cases, upon armor plate. One bare core was dropped three times and two others which were encased in boiler housings were dropped three times each. The damage to the APU cores was insignificant; no macroscopic fissures resulted and plastic deformation was minimal. Boiler housings which were dropped only once suffered only minor plastic deformation in all cases. The results of the tests demonstrate that the Ce-144 fuel would not be released from impact of the SNAP I cores during low velocity impact from launch pad abort.

The test apparatus consisted of a 10-kw portable electric furnace, a 70-foot drop tower and its appendages, reinforced concrete target blocks, a 10,000 frame per second Fastax camera and two portable electric generators. The drop tower equipment consisted of a square drop frame, guide wires, an automatic release mechanism and positioning chains (to vary the impact angle). Figure 52 shows the drop apparatus.

The following test procedures were followed. The specimens were placed in the electric furnace and heated to 1600 to 1700° F. When the specimen reached the proper temperature, it was removed from the furnace, as shown in Fig. 53, and suspended beneath the drop frame at a given orientation (45°, 10° and 80°). The specimen was then pulled to the top of the tower and released automatically. The specimens impacted upon a concrete block at a velocity of 65 to 69 feet per second. The cooling time (from removal from the furnace to impact) varied between 60 to 91 seconds. The impact sequence was recorded on Fastax film at 10,000 frames per second and static photographs were taken of the specimens and concrete after each drop. Figure 54 shows a typical drop test.

Damage in all cases was light even for two specimens which inadvertently impacted upon armor plate. It is significant that all specimens were impacted at 1500° F and that one bare core was dropped three consecutive times without sustaining significant damage (drops 3, 6, 9). The other two cores were also dropped three times each, but were placed within a new boiler assembly each time (drops 1, 4, 7 and drops 2, 5, 8). Each time the damaged boiler was removed and the cores examined for damage. Damage to boiler cans was generally light and no punctures or open fissures resulted.

Fig. 52. Low Velocity Impact Test - Specimen in Drop Fixture at 45°

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Fig. 53. Low Velocity Impact Test - Specimen No. 1 Being Removed from Furnace

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Fig. 54. Low Velocity Impact Test - Impact Area of Specimen No. 6 on Concrete

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Table 13 shows the results of the tests. Figure 55 shows the three cores following the tests. Figure 56 shows the cores, can and bare core following drops 4, 5 and 6. Figure 57 shows cores and cans following drops 7, 8 and 9.

C. LIQUID OXYGEN-MOLYBDENUM

An investigation of the molybdenum-liquid oxygen reaction was conducted by the Broadview Research Corporation. One series of tests was performed with bare molybdenum wires 0.050 inch and 0.100 inch in diameter, and coated strips of molybdenum 2 inches x 5 inches by 40 mils heated electrically in an inert atmosphere and then plunged into liquid oxygen. Another series of tests consisted of dropping into liquid oxygen a half-scale core previously heated in an electric furnace.

Figure 58 schematically illustrates the apparatus used in the first series of tests. The molybdenum wire (or strip) clamped in the copper terminals of the sample holder was heated by current supplied by storage batteries. The molybdenum wire was protected from oxidation during heating by an enclosure into which argon gas was continuously fed. The bottom of the enclosure consisted of a thin sheet of aluminum foil. Then the test samples were driven through the foil on the bottom of the enclosure into the liquid oxygen by action of a spring. At temperatures below 1630° F there was no apparent reaction except for occasional occurrence of a thin layer of oxide on the uncoated wires. The strips protected by a coating experienced no reaction except when heated above the melting point of the coating. When the wires were heated above 1650° F and the strips above the melting point of the coating the test samples ignited and were completely consumed. However, in no case was there evidence of the development of detonation.

The half-scale test samples were solid molybdenum cylinders, some treated with a protective coating. The test samples were heated and transported in the steel container shown in Fig. 59. The container with a test sample assembly was suspended at the center of a nichrome-wound furnace. Preheated helium gas was admitted at the bottom of the container and escaped through the crack in the steel lid. A slow stream of helium was maintained through the container during the time it was heated and transferred. At a predetermined temperature, the container was withdrawn from the furnace and transported via a trolley system to a position over a pyrex jar filled with liquid oxygen. Upon dumping the container, its hinged lid fell open, dropping the test sample into a cylindrical cage suspended in the pyrex jar of liquid oxygen. All of the uncoated test samples heated to temperatures as high as 2130° F quenched in liquid oxygen experienced a small percentage weight loss due to oxidation of the metal surface. No cracking, spalling or evidence of melting of the metal surface was observed as a result of any of the tests. The coated half-scale test samples experienced no weight loss when heated to the melting temperature of the coating, approximately 2000° F.

TABLE 13
Results of Low Velocity Impact Tests

Drop No.	Impact Angle	Impact T (°F)	Impact V (fps)	Impact Medium	Impact Damage			Remarks
					To Core	To Boiler	To Target	
1	45°	1510	69	Armor Plate	Slight beveling of one corner	Deformation on end and ⊥ to axis	Stamped impression on armor plate	Missed concrete
2	45°	1537	-- Lost	Reinforced Concrete	No macroscopic damage	Very minor defor- mation (0.1 in.) on end	Concrete disintegrated	
3	45°	1550	66	Reinforced Concrete	Very slight abrasion and bevel at corner		Concrete disintegrated	No damage sus- tained by specimens
4	10°	1490	66	Reinforced Concrete	No macroscopic damage	Deformation to axis	Concrete disintegrated	Core 1, 4 tested
5	10°	1600	69	Reinforced Concrete	No macroscopic damage	Deformation to axis	Concrete disintegrated	Core 2, 5 tested
6	10°	1550	68	Reinforced Concrete	Very slight abrasion, defor- mation 0.01 in.		Concrete disintegrated	Core 3, 6 tested
7	80°	1600	65	Reinforced Concrete	No macroscopic damage	Very minor abra- sion, deformation	Chipped edge	Core 1, 4, 7 tested
8	80°	1610	67	Armor Plate	No macroscopic damage	Gash type plastic deformation	Armor plate dented	Core 2, 5, 8 coils flattened
9	10°	1600	65	Reinforced Concrete	Very slight abrasion and deformation		Concrete disintegrated	Impact area of 6 duplicated Core 3, 6, 9

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Fig. 57. Cores and Cans After Drops 7, 8 and 9

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FIG. 58. Apparatus for Liquid Oxygen-Molybdenum Tests

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Fig. 59. Steel Container Used in Heating and Transfer of Test Assemblies

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The outstanding feature of the investigation was the failure of the uncoated half-scale model test assemblies to support combustion when immersed in liquid oxygen. Even when heated enough to cause the samples to be consumed, there was no evidence of any explosive reaction. Full-scale test assemblies having coatings intact will be safe from loss of integrity below the melting point of the coating.

A detailed summary of the investigation can be obtained in report RD-58.1 - 1.035-8, "An Investigation of the Reaction Between Hot Molybdenum and Liquid Oxygen", submitted by the Broadview Research Corporation.

D. MISSILE ABORT

An investigation was conducted to determine the integrity of the fuel block under conditions of a typical missile abort by the Broadview Research Corporation. Several test specimens of different model scales were exposed to conditions resulting from different size missiles.

The experiments undertaken consisted of heating the test specimens to 1500° F and dropping them into fuel propellants of a simulated failing missile. Upon detonation, the test specimens were subject to conditions similar to those which would occur at a missile failure at the launch site. In addition to preliminary tests, 19 tests were conducted that yielded information. Figure 60 shows the scaling information used in the tests and a schematic of the missile construction.

The site at which the tests were conducted contained a fence compound area with a small building for the recording instruments and storage of the simulated missiles. The site also contained a pad area located 500 feet from the compound area on which the actual test was performed. The pad area contained the electric furnace in a concrete bunker, test specimen transfer mechanisms, concrete testing and measuring pads, missile suspension equipment and supporting electrical equipment. Figure 61 shows the layout of the pad area and the location of the equipment used in the test.

Table 14 tabulates the important exposure conditions to which each test specimen was subject to. Visual examination of the test assemblies reveal no obvious indications of damage other than a few small dents and abrasions which were attributed to impact and the concrete pad. As can be seen in column 9 of Table 14, the detectable weight change was very small in all cases.

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Fig. 60. Scaling Information and Diagram of Model Missile Construction

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Fig. 61. Pad Area

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Because of the small number of tests run in this investigation, it is difficult to draw any significant conclusions on a purely statistical basis. However, the results strongly suggest that failure of the fuel core due to missile abort is highly improbable. A detailed summary of the investigation has been submitted in report RD-58-2.1035.9, "An Investigation of the Integrity of Certain Test Assemblies Under Missile Failure Forces, Explosive Shock, Temperature, and Oxidation Test Program", by the Broadview Research Corporation.

TABLE 14
Test Specimen Exposure Conditions and Resulting Weight Changes

Shot No.	Specimen No. and Scale	(1) Temperature (°F)	Delay Time Failure to Detonation		(2) Exposure Time of LOX (sec)	(3) Initial Condition of Coating	Missile Yield (% TNT)	(4) Weight Change (gm)
			Planned (sec)	Measured (sec)				
1/18 Scale Missile								
6	2 - 1/2	1650 ±30°	0.9	1.0	0.0	0+	25	0.0
7	4 - 1/2	1650 ±30°	0.9	1.0	0.0	0	24	- 0.6
8	3 - 1/2	1500 ±30°	0.9	1.0	0.0(a)	0	19	- 0.6
13	15 - 1/3	1570 ±20°	0.9	1.0	0.0	0	39	- 0.6
14	16 - 1/3	1630 ±20°	0.9	1.0	0.0	0	--	- 0.6
15	21 - 1/6	1600 ±20°	0.9	1.0	0.0	0	39	lost
	24 - 1/6							- 0.2
	25 - 1/6							- 0.2
16	22 - 1/6	1380 ±20°	0.9	1.0	0.0	0	34	- 0.4
	23 - 1/6							- 0.2
	26 - 1/6							- 0.2
12	1 - 1/2	1450 ±30°	2.5	1.7	0.7	0+	26	- 1.0
10	5 - 1/2	1510 ±30°	2.5	1.0(b)	0.0	0	37	- 0.6
17	4(c) - 1/2	1550 ±20°	2.5	1.1(b)	0.0	0	28	+ 0.5
9	8 - 1/2	1630 ±30°	2.5	1.0(b)	0.0	I	35	- 1.6
11	7 - 1/2	1510 ±30°	2.5	1.3	--	I	12	- 0.9
18	9 - 1/2	1550 ±20°	2.5	1.0(b)	0.0	II	45	-14.3
19	10 - 1/2	1570 ±50°	2.5	--	--	I	36	- 0.0
1/10 Scale Missile								
21	11 - 1/2	1460 ±50°	1.3	1.4	--	I	--	- 1.0
	12 - 1/2					0		- 0.2
23	6 - 1/2	1600 ±50°	3.5	1.7(b)	0.0	0	77	- 0.4
	13 - 1/2					I(b)		- 1.0(d)
24	17 - 1/2	1500 ±20°	3.5	1.5(b)		0	71	- 0.6
	20 - 1/3							- 0.5
258	14 - 1/2	1580 ±20°	3.5	1.8(b)	0.2	0	59	- 0.3
	18 - 1/3							- 0.5

NOTES:

- (1) All temperatures monitored by a thermocouple were measured within 10 seconds of explosion.
 - (2) Time is determined as follows: From the high speed film, the time between failure and detonation and between failure and release of the test assembly from the heat sink are measured. Latter time is added to calculated time of fall and subtracted from former time to yield exposure time.
 - (3) Two test assemblies marked (+) were coated with Chromalloy; remainder had Martin coating. Test assemblies marked (0) were undamaged, those marked (I) slightly damaged and those marked (II) were heavily damaged.
 - (4) Probable error in each observation; 0.3 gram for 1/2 scale test assembly, 0.2 gram for 1/3 test assembly and 0.03 gram for 1/6 scale test assemblies.
- + See (3)
- (a) High speed film show test assemblies did not reach mixture before LOX detonator initiated it.
- (b) Indicates test assemblies initiated the propellant mixture.
- (c) Test Assembly No. 4 was used in two tests.
- (d) Test Assembly No. 13 was altered in field.

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