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CRONUS

Controlled Reentry Orbiting NUClear System

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MARTIN

RTG/RB

CIR

AGE

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Study for 250-Watt
Controlled Reentry Orbiting NUClear System

VOLUME II

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AEC Research and
Development Report

CRONUS

Controlled Reentry Orbiting Nuclear System

SPECIAL REVIEW FINAL DETERMINATION	Reviewer	Class.	Date
	KAW	U	1-12-82
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Study for 250-Watt Controlled Reentry Orbiting Nuclear System

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November 2, 1966

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FOREWORD

The United States Atomic Energy Commission awarded the Martin Company Contract AT(29-2)-2050 for a six-month study, effective May 2, 1966, to develop and evaluate a feasible concept(s) for a 250-watt(e) radioisotopic thermoelectric generator providing controlled intact reentry of the fuel from earth orbit. A concept(s) for handling and fueling the RTG at a Titan III launch site was also to be developed and evaluated. Based upon the selected system concept, a development program plan was to be prepared, including estimated costs, for the design, development, flight test and delivery of operationally flight qualified hardware systems.

The flight and ground systems have been integrated by Martin into a single system concept designated CRONUS, an abbreviation for Controlled Reentry Orbiting Nuclear System. Major CRONUS systems include the 250-watt(e) generator/reentry body (RTG/RB), the controlled intact reentry (CIR), and required ground support (AGE) systems which include fuel loading (GHE).

The following is a complete list of the documents prepared and submitted in accordance with Contract AT(29-2)-2050.

Volume I--Summary

Volume II--Technical

Volume II--Appendices

Volume III--Development Program Plan

Volume IV--Specifications and Data Management

Volume V--Budgetary Estimate

Volume VI--Quality Assurance Plan

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I. INTRODUCTION

Volume II represents a complete and accurate accounting of the technical efforts pursued under Contract AT(29-2)-2050. Organization of this volume is as follows:

Chapter II	Mission Description	Discussion of the mission for which CRONUS has been designed. Includes launch complex and vehicle descriptions and ground and flight operations.
Chapter III	Selected Design	Presentation of the selected concept for controlled intact reentry (CIR), reentry body and generator (RTG/RB) and aerospace ground equipment (AGE). Includes performance and operational characteristics.
Chapter IV	CRONUS Integration	Integration of CRONUS into the launch complex, launch vehicle and spacecraft are discussed.
Chapter V	Nuclear Safety	Evaluation of the safety aspects of CRONUS including radiation shielding analyses.
Chapter VI	Reliability	Assessment of the reliability of the RTG from a power standpoint, the CIR/RB and GHE from a safety-imposed reliability goal, plus the GHE from a smooth countdown consideration.
Chapter VII	Design Evaluation	Explanation of how the selected concept evolved and the alternative choices.
Chapter VIII	Design Credibility	Assessment of the selected design with regard to selected components and materials, analyses supporting the design and the recognized problems associated with the design.

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Chapter IX	Improvement Potential and Design Flexibility	Discussion of weight reduction and sensitivity and flexibility of the reference concept with regard to the alternate fuel (Pu-238), alternate land complex (WTR) and the adaptation of the GHE to a side loading technique.
Volume II	Appendices	Contains techniques and results of analytical efforts in support of CRONUS as well as certain other appendices supporting the selected design (e.g., thermoelectric properties, idealized GHE concepts, etc.).

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II. MISSION DESCRIPTION

The 250-watt airborne system will be mounted on a payload and launched by Titan III into a low earth orbit.

To achieve this, it is necessary to send the airborne system to the **Spacecraft Integrating Contractor who integrates it with the spacecraft**, delivers it to the launch complex, mounts the payload on the Titan III and installs the propulsion devices in a safe-arm area (SMAB). The launch vehicle is then prepared for flight. Near the end of the count-down (T-20 minutes), a scheduled hold allows the RTG to be fueled. Following fueling, the countdown is continued and Titan III is launched.

The spacecraft is placed into two orbits. First, it is injected by the Titan III transtage into a transfer orbit. Near apogee of this orbit, the transtage is reignited. A final 600-nautical mile circular orbit is achieved. The characteristics of these orbits are as shown.

	<u>Transfer</u>	<u>Final</u>
Apogee (naut mi)	600	600
Perigee (naut mi)	100	600
Eccentricity	0.067	0.0
Period (min)	98	108
Inclination (deg)	30 ± 1 ; 90 ± 1	30 ± 1 ; 90 ± 1
Location of perigee	First shutdown of transtage	--
Orbital lifetime ⁽¹⁾	82 days	>5000 years

(1) Based upon $W/C_D A = 22 \text{ lb/ft}^2$

The low inclination orbits can be achieved from the Eastern Test Range. The high inclination orbits would be achieved by a launch in the southern direction from the Western Test Range.

A. LAUNCH SITE

A typical Titan III launch site is shown in Fig. II-1 (Pad 40 or 41 at Cape Kennedy, Florida).

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Fig. 11-1. Artist's Concept, Positioning Titan IIIC for Final Checkout and Launching

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This complex is equipped to handle the larger Titan version which has two solid rockets strapped onto the sides of the basic liquid propellant Titan core. Titan IIC is erected in a vertical integration building on a railroad-type transporter which is pulled to the solid motor assembly building (SMAB) and then to the launch pad by two locomotives (see Figs. II-2, II-3 and II-4). An electrical umbilical mast is fixed to the transporter. The transporter, which also serves as a thrust mount for the Titan, is positioned over a flame bucket and beside a fixed umbilical tower. The umbilical tower measures approximately 25 by 44 by 170 feet high (see Fig. II-1). A mobile service tower (MST) riding on rails can be moved around the umbilical tower and launch vehicle on its transporter (see Fig. II-5). The umbilical tower is protected on the side toward the launch vehicle by flame shields with the exception of the area of the retractable platforms (see Fig. II-1). Platforms on the MST provide access to the three sides of the launch vehicle not made available by the platforms from the UT. A plan view showing the clearances between the various structures and their direction of travel is shown in Fig. II-6.

B. SPACECRAFT ASSEMBLY AND INSTALLATION PROCEDURES

The RTG/RB-CIR assembly will be delivered to the Spacecraft Integrating Contractor following fabrication and checkout. There the RTG/RB-CIR will be mated to the reference satellite and perform integration tests to demonstrate systems operational compatibility. Upon completion of integration testing, the Spacecraft Integrating Contractor will package the spacecraft as an assembled unit in an appropriate shipping container for transfer to ETR.

At ETR the spacecraft will be delivered to the vertical integration building (VIB) shown in Fig. II-2 where it will be unpacked and hoisted to interface with the Titan IIC core. Installation will be accomplished through the use of appropriate slings, jigs and tools provided by the Integrating Contractor. Upon completion of the mechanical and structural interface connections, the Titan fairing will be installed and the spacecraft access actuating and latching mechanisms checked out and adjusted if required. The appropriate electrical umbilicals will be mated with the payload in order that ground checkout can be performed.

With all electrical umbilicals connected to the spacecraft, the launch vehicle-spacecraft assembly will be carried on the transporter thruster mount to the solid motor assembly building (Fig. II-4). In the SMAB, the solid stage zero rocket motors will be attached to the Titan IIC core.

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Fig. II-2. Vertical Integration Building for Erecting Titan IIIC

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Fig. 11-3. Titan 111C Transporter

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Fig. II-4. View of Solid Motor Assembly Building

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Fig. 11-5. Mobile Service Tower

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Fig. 11-6. Titan III Complex at ETR

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The Titan nose fairing will be removed and reinstalled after the spacecraft retrorocket and spin motors are installed on the RTG/RB-CIR and optically aligned.

The transporter/thruster mount then carries the launch vehicle-payload assembly to the launch stand for final checkout, countdown, fuel block loading and launch. All the maintenance and operational functions which were performed in the VIB will be performed again on the launch stand and, in addition, a fuel handling machine-fuel handling machine control console functional checkout and fuel loading demonstration will be performed.

The fueling is described in detail in Chapter III.

C. TITAN III FLIGHT PLAN

The Titan III or standard space launch system (SSLS) has a wide variety of propulsion stage combinations which optimize propulsion requirements for given payload weight ranges. These payload weights range from a few thousand pounds in high altitude orbits to 25,000 pounds in low altitude orbits. In addition, the capability to place several payloads in different orbits with one vehicle has been demonstrated. Because of this versatility and because of the lack of a projected, specific payload definition at this early stage of the 250-watt generator program, the flight plan for the vehicle configuration which provides the maximum payload weight capability (Titan IIIC) is described.

The Titan IIIC utilizes two solid propellant rocket boosters (commonly identified as stage zero) strapped in parallel to the first stage of the CORE vehicle which consists of a two-engine storable liquid propellant first stage, a single engine storable liquid propellant second stage and a two-engine storable liquid propellant third stage (also called transtage). This latter stage serves as an orbit injection and orbit transfer stage.

An outboard profile of Titan IIIC is shown in Fig. II-7. Not shown in this figure is the standard payload fairing which is also a part of Titan IIIC. This fairing will have to be modified for use with CRONUS (Chapter III). Titan IIIC is in control of the flight until payload separation. The Transtage (Fig. II-8) has the capability of maintaining attitude and flight control during coasting periods as well as during the main thrusting periods. Although it has the capability of restarting three times, only two burns are required for the CRONUS flight profile.

The flight sequence occurs as follows:

- (1) Ignition of the two solid motors on the launch pad

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Fig. II-7. Outboard Profile of Titan IIIC

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Fig. II-8. Transtage Spacecraft Configuration

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- (2) Ignition of the CORE first stage engines immediately prior to burnout and separation of the two solid motors.
- (3) Ejection of the payload fairing:
 $t \approx 160$ seconds
 $h \approx 230,000$ feet
 $q \approx 0$
- (4) Ignition of the CORE second stage engine immediately prior to shutdown and separation of the first stage engines.
- (5) Ignition of the CORE third stage engines (transtage) after burnout (or shutdown) of the second stage engine.
- (6) Shutdown of the third stage engines after injection into an elliptical (100 to 600 naut mi) orbit has been achieved. The systems coasts in a controlled fashion.
- (7) Reignition of the third stage engine after a coast period to apogee of the elliptical orbit (600 naut mi).
- (8) Shutdown of the third stage engine after injection into a 600-nautical mile circular orbit has been achieved.
- (9) Separation of the third stage from the payload.

The primary trajectory characteristics are shown in Fig. II-9. The circular orbit-payload capability of the Titan IIIC is shown in Fig. II-10. As a means of comparison, the circular orbit payload capability of the Titan IIIA is included to illustrate the capability of a smaller version of the Titan III series; i. e. , a configuration that does not utilize stage zero (two solid motors). In this latter case, the CORE Stage I is ignited on the launch pad and the follow-on sequence of the flight plan remains unchanged.

D. TRANSTAGE SEPARATION

The transtage stays attached to the satellite (and RTG/RB-CIR) until the final orbit has been achieved and verified from the ground. Prior to this, the transtage controls the spacecraft, attitude, the MDS and range safety functions.

After the final orbit has been verified, the transtage may orient the spacecraft into a suitable attitude and S/C separation would then occur.

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Figure II-9. Titan IIIC Launch Trajectory

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Figure II-10. Minimum ETR Performance, Titan III

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If the S/C is a Nimbus satellite, the spacecraft attitude control system will take over and refine and update the S/C attitude. The CRONUS system will deliver regulated power (up to 250 watts, as required) to the spacecraft for a period up to five years. Presumably, the CRONUS mission requirements will then have been completed.

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III. REFERENCE CONFIGURATION

This chapter presents the reference configuration. The concept is not weight optimized. The reader is referred to Chapter IX for a discussion of the weight optimized configuration as well as several advanced growth potential concepts. The chapter provides an overall description of the selected concept in Section A. More detailed descriptions of each system are given in the following sections.

A. CRONUS

CRONUS (Controlled Rentry Orbiting Nuclear System) is a combination of three systems and provides the following:

- (1) A minimum of 250 watts (electrical) at 28 volts $\pm$ 10% for utilization in an earth orbital mission with a design life of five years.
- (2) A capability of assuring controlled intact reentry of the nuclear heat source into a preselected area if required because the desired orbit was not achieved.
- (3) The means for servicing the airborne system at the launch complex.

The three systems making up CRONUS are:

- (1) The Generator System (RTG/RB). The power source (RTG) is customized to form part of a suitably shaped reentry body (RB). Special components required for surviving reentry (such as an ablative heat shield) form the remaining part of the RB. Consequently, the power and reentry portions are designated the RTG/RB system.
- (2) The Controlled Intact Reentry System (CIR). The means of propulsion, spacecraft attachment and separation devices, various electronic equipment (such as command receivers, telemetry, etc.), and similar components required for the proper operation and deorbiting of the RTG/RB system are contained in a separate package, designated the CIR system.
- (3) The Aerospace Ground Equipment (AGE). The equipment required to transport and handle the nuclear heat source is designated the ground handling equipment (GHE). Equipment needed for maintenance and verification of the ready status of the RTG/RB and CIR

LEGEND

1. Fairing Hinged Nose Cap	41. Link Release, Pyrotechnic
2. Fairing	42. Pyroswitch
3. RTG/RB	43. Separation spring, RTG/RB to CIR
4. CIR	44. Carriage Stop
5. Spacecraft	45. Heat Shield Back-Up Bulkhead
6. Spacecraft Adapter	46. Heat Shield Protection Pad
7. T-III Transtage	47. Separation Rocket Ejection Lugs
8. RTG Heat Source	48. Separation Rocket Ejection Lugs Stops
9. Hinged RTG/RB	49. Carriage Position Lock, Pyrotechnic
10. Fairing Heat Sink	50. RTG/RB Aft Cap Hinge
11. Fairing Nose Cap Pneumatic Actuator	51. C-Band Beacon Transmitter
12. Nose Cap to Aft Cap Link	52. C-Band Beacon Battery
13. Separation Tracks	53. Thermal Block Gasket
14. Separation Track Struts	67. Radiator/Heat Shield Splice Frame
15. Separation Carriage	68. Radiator
16. Spin Bearing	
17. Separation Rocket & Mount	
18. Deorbit Rocket	
19. Spin-Up Rocket	
20. Despin Rocket	
21. Command & Telemetry	
22. Separation Band RTG/RB to CIR	
23. Latch Fingers	
24. Electrical Disconnect RTG/RB to CIR	
25. Electrical Disconnect (Squib Fired) RTG/RB-CIR to Spacecraft	
26. Subsystems Equipment Mounting Shelf	
27. Indexing Fitting for Fuel Handling Machine (On Fairing)	
28. C-Band Beacon Antenna	
29. Latch For Hinged Aft Cap	
30. Heat Shield, Ablative	
31. Mounting Bulkhead for RTG/RB	
32. Power Conditioner Resistor Box	
33. Power Conditioner Transistor Box	
34. Pyrotechnic System Solid State Switch	
35. Battery	
36. Command & Telemetry Module	
37. Pyrotechnic Safe Arm Units	
38. Low Voltage Relay Switch	
39. Equipment Access Panel	
40. Separation Band Pyrotechnic Release	

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III-2

Fig. III-1. CRONUS General Arrangement

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systems is identified as maintenance ground equipment (MGE). That equipment which functions during the countdown phase of the launch and is an integral part of CRONUS during countdown, exclusive of the GHE, is designated operating ground equipment (OGE). The combination of GHE, MGE, and OGE is designated aerospace ground equipment (AGE).

The first two (RTG/RB and CIR) of these systems are often discussed in combination and hence, for convenience, are designated the airborne system.

1. Layout and Summary Description

A layout of the airborne system is presented in Fig. III-1. The system is shown as it appears on the Nimbus satellite (without solar panels or command antenna and ground plane). The standard Titan III fairing is shown, in outline, as modified for CRONUS.

The major components of the airborne system, as shown in Fig. III-1, are as follows:

The RTG/RB system is above the spacecraft. It is a blunted cone. The narrow end of the cone is pointed up in the launch vehicle. The RTG/RB, with its reentry heat shield, is fastened to the CIR system and the heat source and thermoelectric elements are contained in the reentry body. The walls of the cone are the radiators for the power source.

The CIR is between the RTG/RB and the spacecraft. It resembles a truncated cone. The large end encloses the heat shield and supports the entire RTG/RB. The narrow end is attached to the spacecraft/CRONUS separation structure.

Inside the CIR cone are the various auxiliary systems (telemetry, command receivers, decoders, sequencers, voltage regulators, etc.). External to the cone are the propulsion systems (deorbit rockets, separation rocket, and spinup and despin rockets). The cone skin provides protection for the internal equipment, and also allows for thermal environment control.

The CIR is attached to the spacecraft by the separation system that consists of a separation carriage with a spin bearing and separation tracks. The separation rocket motor is mounted on the carriage. The carriage and spin bearing are locked to the tracks by a pyrotechnic device. The tracks are attached to the spacecraft.

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LEGEND:

- 3 RTG/RB
- 4 CIR
- 12 Nose cap to aft cap link
- 13 Separation tracks
- 15 Separation carriage
- 17 Separation rocket and mount
- 41 Link release, pyrotechnic
- 50 RTG/RB aft cap hinge

Fig. III-2. Airborne System of CRONUS

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Fig. III-3. Fuel Block Loading Concept, Tower Boom/Top Loading

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- (6) The fueling machine is retracted and the fairing and RTG/RB caps closed.
- (7) The countdown is resumed and at approximately T-10, the fueling boom is retracted; the RTG/RB-CIR systems are maintained in monitor status until launch.

B. GENERATOR SYSTEM (RTG/RB)

The generator system includes all subsystems required to produce useful electrical power and to protect the radioisotope payload during reentry. These subsystems are identified in the following:

<u>Subsystem</u>	<u>Function</u>
Heat source subsystem	Supplying thermal energy to the thermoelectric subsystem
Thermoelectric subsystem	Converting thermal energy to electrical power
Radiator subsystem	Rejecting waste heat to the space environment
Heat shield subsystem	Protecting RTG/RB components from aerodynamic heating
Aft cap subsystem	Providing aerodynamic stability and orientation
RF beacon subsystem	Emitting an RF signal for tracking purposes
Voltage regulator	Limiting voltage excursions and disposing of excess electrical power.

The subsystems associated with electric power production (RTG) are the heat source, thermoelectric, radiator and voltage regulator. Those associated only with the reentry function are the heat shield, aft cap and the RF beacon. During reentry, the radiator performs a secondary function in protecting the internal RTG components from aerodynamic heating. It also maintains a smooth aerodynamic shape, and provides mechanical support for all of the subsystems except the voltage regulator. The voltage regulator is

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physically located within the CIR, which is separated from the RTG/RB prior to reentry.

An exploded view of the RTG/RB is shown in Fig. III-4. The RTG is conical in shape, with the cone apex omitted above a plane sloped 45 degrees to the cone axis and with a shallow, spherical segment base with a radiused edge where it joins the cone surface.

This right circular cone body has a half angle of 18 degrees and a base diameter of 46 inches (at the theoretical intersection of the cone and spherical segment base). At the small end, the plane of the 45-degree sloped surface intercepts the cone axis 20-1/4 inches down from the cone apex. The spherical segment radius of the base is 73-1/2 inches and the edge radius, where the base joins the cone, is 1-1/2 inches. The overall height is approximately 59-1/4 inches.

The portion of this body at the small end, above the plane perpendicular to the cone axis and 20.85 inches below the body's upper extremity, is a separate assembly. This aft cap assembly, composed of a canted bulkhead, skin and ring frames, houses the tracking beacon, the tracking beacon antenna and the necessary battery. This assembly is hinged to the main portion of the body below, with two external hinge fittings four inches apart. The aft cap is opened for fueling and closed during flight. Equally spaced around and mounted on the lower ring are eight spring latches that engage holes in the heat source guides at the top of the main body when the aft cap is closed. Each of these spring latches is incorporated in a solenoid that, when activated, acts to override the spring and open the latch.

At the upper extremity of the aft cap and above the hinges is a single lug to which the launch fairing actuating link is latched prior to liftoff.

It is recognized that these three protuberances may be adversely affected by aerodynamic heating during reentry. If, after testing, this should prove to be the case, they will be modified to alleviate the condition.

The conical portion of the RTG/RB extending immediately below the aft cap is the RTG radiator. It also is a structural part of this body and has the RTG thermoelectric units mounted to its inner surface. It is 31 inches long from end to end and it has a ring frame at the large end to which the heat shield is attached.

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LEGEND

- 9. Aft Cap
- 28. C-Band Beacon Antenna
- 30. Heat Shield, Ablative
- 45. Heat Shield Backup Bulkhead
- 50. RTG/RB Aft Cap Hinge
- 51. C-Band Beacon Transmitter
- 53. Thermal Block Gasket
- 60. Thermoelectric Subsystem
- 61. Hot Shoe
- 62. Flexi-Mod Cold End Hardware
- 63. Thermal Insulation (Advanced Min-K)
- 64. Thermal Insulation (Micro-Quartz)
- 65. Silica Cloth Wrap
- 66. Radiator Subsystem
- 67. Ablator Mounting Ring
- 68. Radiator
- 69. Receptable, Quick-Disconnect
- 70. Wiring Harness
- 71. Heat Source Installation Guide
- 72. Heat Source Support
- 73. Locking Disc Support
- 74. Locking Disc
- 75. Latch Ring
- 76. Limit Switch
- 77. Locking Disc Spring
- 78. Support Cap (Upper)
- 79. Support Cap (Lower)
- 80. Support Tube
- 81. Ball Detent
- 82. Fuel Capsule
- 83. Latch Mechanism (Heat Source)
- 84. Latch Hook
- 85. Capsule Retaining Band

Fig. III-4. RTG/RB System

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The heat shield structure comprises the remainder of the RTG/RB. It consists of a dual channel ring frame surrounding and attached to a honeycomb panel at its lower flanges. The honeycomb panel is contoured to a spherical radius shape and is the structural backup for the ablative heat shield. This structure is completely covered by the ablator material which is of a uniform thickness over the face and tapers in thickness as it extends up the web of the ring frame toward the radiator attachment flange.

The heat shield structure is bolted to the radiator through a thermal block gasket in the faying surface.

The weight of the RTG/RB is summarized in Chapter IX with the weights broken down by subsystems. However, the weight of the voltage regulator is not included since the voltage regulator is physically located in the CIR. The center of gravity has been determined to be on the centerline and at Station 73.9 or 18.0 inches aft of the forward face of the heat shield.

The moments of inertia are:

Roll 14.3 slug-ft²

Pitch 16.0 slug-ft²

Yaw 16.0 slug-ft²

1. RTG Concept

The RTG includes four subsystems--the heat source, the thermoelectric, the radiator and the voltage regulator. A summary of RTG characteristics is given in Table III-4 and a weight summary is provided in Chapter IX.

a. Heat source subsystem (numbers in parentheses refer to Fig. III-4)

The heat source subsystem consists of 36 fuel capsules (82) held in a circular array between the upper and lower support caps, as shown in Fig. III-5. The support caps (78 and 79) are lined with thermal insulation to minimize parasitic heat loss. The ends of the fuel capsules are retained by depressions in the capsule retaining bands, which bear against the fibrous insulation (63) and transfer compression loads from the insulation to the fuel capsules.

The upper support cap (78) is joined to a central support tube (80) which in turn is splined to the other (lower) cap (79) to prevent the caps from rotating and the fuel capsules from leaning and collapsing. A threaded ring is used to assemble and hold the support caps together,

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TABLE III-4

RTG Characteristics Summary

	<u>Normal</u>	<u>BOL</u>	<u>Open Circuit</u>	<u>EOL</u> <u>Normal</u>
Output Power (watts) Net	360*			250
Output Voltage (volts) Net	28			28
Fuel Inventory (watts)	6420			5660
Efficiencies (%)				
Thermoelectric				5.43
Thermal				89
Joule				92
Regulator				99.5
Overall				4.42
Temperatures (°F)				
Radiator	527		540	505
Cold Junction	578		590	550
Hot Junction	1590		1795	1450
Hot Shoe	1640		1845	1495
Capsule, Minimum	1720		1910	1580
Capsule, Maximum	1787		1977	1640
Materials				
Fuel		SrTiO ₃		
Separator		Platinum		
Capsule		Haynes-25		
Hot Shoes		SiMo alloy		

* Power available at the RTG output terminals based on a fixed external load.

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TABLE III-4 (continued)

Elements	SiGe
Radiator	Beryllium
Heat Source Insulation	Advanced Min-K, Micro-Quartz
Module Insulation	Advanced Min-K, Micro-Quartz
Heat Source	
Capsules (No.)	36
Heat Source OD (in.)	21.5
Heat Source OL (in.)	19.2
Capsule OD (in.)	1.55
Capsule OL (in.)	13.12
Capsule Wall Thickness (in.)	
Maximum	0.14
Minimum	0.10
T/E Characteristics	
Circuitry	Series-parallel
Element Length (in.)	0.75
Couples (No.)	396
N Element Area per Element (in. ²)	0.091
P Element Area per Element (in. ²)	0.056
Contact Resistivity (pr ohm cm ²)	1800
Area (ft ²)	
Radiator	24
Module (facing heat source)	7.0
Heat Source (lateral envelope)	6.1

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Fig. III-5. CRONUS Heat Source

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compressing the capsules and insulation only enough for handling purposes. The ball detent (81) on the inside surface of the support tube provides an attachment point for handling the heat source. A latch mechanism (83) is provided at the lower end of the support tube to engage with the heat source cask or RTG.

The overall size of the heat source is 21.5 inches in diameter by 19.2 inches in height. It is installed remotely through a 22.5-inch diameter opening in the top of the radiator subsystem with one-inch diametrical clearance.

(1) Fuel capsule design

A drawing of the complete fuel capsule is shown in Fig. III-6. Each capsule contains four or more strontium titanate hot pressed pellets which are individually wrapped in 0.005 inch thick platinum foil separators. Such separators are presently used by ORNL personnel as a handling aid and to prevent contamination of the weld zone in the primary (outer) capsule. With this design, it is expected that the Bureau of Licensing contamination requirement of $0.05 \mu \text{ curies}/100 \text{ cm}^2$ will be met without extensive scrubbing of the outer capsule surface.

The Haynes-25 primary capsule is made in two halves which are welded together in a central half-lap weld joint. Both fusion and electron beam welds are used for maximum impact resistance. The single fusion weld is performed in an inert gas atmosphere prior to the electron beam welds which are performed in a vacuum environment. In this way, the capsule will be filled with an inert gas to promote heat transfer between the fuel, the platinum separator and the outer capsule.

A Haynes-25 honeycomb spacer is provided within the weld zone to prevent the fuel pellets from being damaged by the electron beam. Use of the spacer will allow maximum penetration of the electron beam welds to be achieved. The inside parting line between the two halves will be joined completely by the last electron beam weld.

Each half of the capsule will be machined from a single piece of Haynes-25 round bar. Particular attention has been paid in designing the ends of the capsule so they are free of geometrical discontinuities. For instance, the end and side walls have nearly the same thickness so that deformations occurring during impact will be shared by the adjacent sections. A generous internal radius has been provided to reduce stress concentration, and low density sintered Haynes-25 spacers will be provided (as required) to fill any space not occupied by the fuel pellets. Calculations have shown the impact resistance of this end configuration to be superior to existing designs using welded end caps.

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Fig. III-6. Fuel Capsule for Hot Pressed Sr-90 TiO_3 Pellets

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A hollow skirt has been provided at each end of the capsule to perform two separate functions. First, it provides additional impact resistance by absorbing kinetic energy in crushing. Second, it acts as a conductive fin and so extends the heat transfer area of the heat source.

The fuel capsule wall thickness has been selected to be 0.141 inch, tapering to 0.098 inch between the ends and the weld zone.

(2) Thermal insulation

The upper and lower support caps are each lined with a three-inch thick ring of advanced Min-K fibrous insulation having the capability of operating at temperatures up to 1900° F. This material is currently being developed by Johns-Manville Engineering and Research Center, Manville, N. J., under USAEC Contract No. AT(930-1)3633. It is expected it will be available within one year.

The central support tube is lined with a cylinder of Micro-Quartz blanket-type insulation, which is also produced by Johns-Manville.

Micro-Quartz was selected because of its low density (6 lb/ft³) and its correspondingly low ~~k~~ product. **A thickness of 3.4 inches was selected** for this component, which has a thermal conductance equivalent to three inches of Min-K.

The exposed surfaces of the advanced Min-K rings are machined to a conical shape to provide an interference fit with the upper and lower insulation rings in the radiator subsystem. This will **provide good contact** between mating insulation components after the heat source is installed in the RTG.

(3) Latch mechanism

The latch mechanism provides a direct mechanical attachment between the upper support cap and the mounting structure below the heat source. Six latch hooks are attached to the central support tube with each hook individually spring-powered to swing outward and engage with the latch ring in the heat source support structure.

The latch hooks are disengaged by the fueling probe (part of the fueling handling machine) during a transfer or handling operation, and are permitted to engage with the support structure prior to removal of the fueling probe. When all latch hooks are fully engaged, a locking disc in the support structure locks the hooks into engagement and provides a "latched" indication at **Launch Control Center**. **If one or more of** the hooks are not fully engaged, a no-latch indication will be given at **Launch Control Center until the situation is corrected**

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Fig. III-7. Heat Source Latch Mechanism

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The latch hooks have an epicyclic contact surface which provides a cam action as they swing outward into their engaged positions. This action allows the latch hooks to accommodate a wide variation in mechanical tolerances and thermal expansion and provides a self-tightening action during vibration. Any vibratory motion of the upper support cap will permit the hooks to swing outward to take up the resulting clearances as shown in Fig. III-7. In this way, the fuel capsules and thermal insulation will be compressed to the extent that peak (vector) dynamic forces will be overcome.

b. Thermoelectric subsystem

The energy conversion section of the RTG/RB is comprised of Flexi-Mod thermoelectric modules and associated thermal insulation. Each module can be further subdivided into thermoelements, hot shoes, cold stacks and electrical connector straps.

(1) Operation

The decay heat of the radioisotope fuel radiated from the capsules is absorbed by the large-area silicon-molybdenum (SiMo) alloy hot shoes and transported by conduction to the N- and P-doped silicon-germanium (SiGe) thermoelements. After direct conversion of a small amount of this heat to electricity, the remainder, along with the parasitic heat conducted through insulation, is rejected to space through the beryllium radiator.

Fuel excess due to decay during the five-year operative period amounts to 13% with Sr-90 and results in diminishing component temperatures. Consequently, the design point must be selected from end-of-life conditions (i. e., lowest operating temperatures for the RTG. Operating parameters at other times in life will always produce at least the required minimum power but at beginning of life, in particular, must be determined for critical temperature assessment.

(2) Module

The thermoelectric subsystem consists of 33 Flexi-Mod modules connected in series with each other, each containing 12 SiGe couples wired in a cross-connected series-parallel circuit. The Flexi-Mod concept is illustrated in Fig. III-8 with a more detailed couple view in Fig. III-9.

This conversion package was developed by Radio Corporation of America and derives its name, Flexi-Mod, from its ability to conform to a gradual contour upon flexure of its cold strap connectors used as the assembly structure. This freedom to flex is the chief construction

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Fig. III-8. Selected SiGe Module Configuration

Fig. III-9. Thermoelectric Couple Using Flexi-Mod Fabrication Techniques

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feature of the module with the result that it is easily integrated with the conical beryllium radiator. Each element in the module is bolted to the radiator by means of a terminating stud. This method of fabrication eliminates the need for an intermediate adapter plate.

The basic unit of an individual module is a thermocouple assembly made of silicon-germanium thermoelectric alloy. The two active elements or legs of the thermocouple are of suitably doped material and are joined at the hot junction end by a heat acceptor plate (hot shoe). The thermocouple assemblies are supported in cantilever fashion from the cold end and require no other mechanical structure.

These elements are 0.75 inch in length and are formed, in cross section, as a half circle for close packing to minimize electrical circuit losses in the hot shoe. The alloys maintain good thermoelectric properties at high temperatures, are not volatile at these temperatures, and are not susceptible to poisoning in oxidizing atmospheres. Thus, they can operate in either air or vacuum environments.

The hot shoes are made of a silicon-molybdenum alloy with higher thermal and electrical conductivities than those of the SiGe alloys of which the element legs are made. The higher thermal conductivity enables the hot shoe surface to accept radiant thermal energy at a relatively low flux density and then to concentrate it for processing by the active element legs at a much higher thermal flux. The 1.27 by 1.64 inch shoes are specially shaped to lower couple weight and provide increased resistance to mechanical vibration and shock.

The shape illustrated is similar to a hip roof, tapered on four sides from a central ridge. The hot shoe is 0.160 inch thick at the center tapering to 0.030 inch at the edges. By retaining the full 0.160 inch thickness along the central ridge, maximum cross-sectional area is provided at the point between the two elements in order to minimize electrical resistance losses in the hot shoe and maximize the joule efficiency of the RTG. The 0.030 inch edge thickness is a departure from standard fabrication techniques and will require grinding to shape. It is not anticipated that any other surface treatment will be required for the hot shoe. Anticipated shoe emissivity is 0.8 or greater.

The individual thermocouples are mounted on a supporting structure by means of a stack assembly consisting of a number of elements and performing several functions. The assembly provides a means of mechanical support. To do this, it is generally necessary to go from the very low thermal expansion material of the silicon-germanium alloy and the tungsten cold shoes to a relatively high thermal expansion material such as copper, aluminum, stainless steel or beryllium which

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is used as the radiator. Therefore, stress compensation must be provided in the stack assembly to enable the whole structure to maintain its integrity throughout the range of temperatures from room through processing to operating. This compensation is a designed result of a combination of materials, geometries and processes which produce the desired stress-balanced structure.

The stack assembly must also maintain electrical isolation from the common radiator and yet present a low thermal impedance to the thermal energy being processed through the thermocouple. A thin aluminum oxide disc is placed in the stack as a means of satisfying both of these requirements. The same basic principles of stack construction which were employed in the SNAP 10A and SNAP 17A are proposed here except that materials and processing changes are required due to operating temperature and geometry differences.

The method of fastening the module to the radiator is illustrated in Fig. III-9. As shown, each couple carries its own threaded mount stud for mating to the beryllium radiator where satisfactory metallurgical techniques do not exist. With this concept, it is possible to combine both the reliability of mechanical fastening with the thermal advantage of intimate contact. As seen in the illustration, a collar of a metallurgically compatible material is introduced between the couple mount stud and the radiator. The interface surfaces, when properly treated, will produce an intimate contact under the fastening pressures and operating temperatures of the finished device.

Electrical connections between the required 396 couples are effected by means of nickel-plated copper straps. The couples are arranged in two series-connected strings with each set of adjacent couples joined together in parallel by cross-connections. This technique uses the advantages of both wiring methods--high voltage (28 volts direct) from the series connection and high reliability from the parallel connections. The selected arrangement is illustrated in Fig. III-10.

Failure of a single couple is not catastrophic to the operation of the RTG. If a couple fails in the open circuit mode, the couple current will be shunted through to the adjacent parallel couple by the conductor straps which provide a cross-connection. Since twice the design current will then be flowing through the adjacent couple, deregulation will cause the power output of that couple to be nearly zero. As a good approximation, it may be stated that the failure of a single couple by open circuit will cause a loss of power at the output terminals equal to the output of two couples.

If a couple fails by short circuit, then power from the adjacent parallel couple will be shunted through the short. Again, the power of both couples will be lost but circuit continuity will be maintained.

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A. Series Circuit

B. Parallel Circuit

C. Selected Series--Parallel Circuit Having Two Cross-Connected Parallel Paths

Fig. III-10. Thermoelectric Subsystem Electrical Circuits

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Tie straps will be used to perform electrical connections between the connector straps of adjacent modules. The tie straps are identical to the connector straps in their material, finish, width and thickness to provide a good mechanical and electrical interface. Gold foil will be sandwiched between the connector and tie straps, and mechanical fasteners will be utilized to complete the connection.

(3) Thermal insulation

Thermal insulation in the thermoelectric subsystem consists of strips of advanced high temperature Min-K between the couples, supplemented by loose Micro-Quartz which is packed into any remaining openings between the elements and the Min-K strips.

A technique was developed on SNAP 17A for insulating SiGe modules that includes fitting strips of Dyna-Quartz fibrous insulation (to be replaced by advanced high temperature Min-K in this design) and then packing low density Micro-Quartz (4 to 6 lb/ft³) into the remaining voids to an approximate density of 12 pcf. This operation is performed using dental tools, and care is required to avoid injury to the thermoelectric couples.

If the Micro-Quartz is not packed to the 12 pcf density, subsequent shrinkage will occur and openings will form to permit thermal radiation to stream through the module. If the 12 pcf density is exceeded, the bulk conductivity of the Micro-Quartz will increase.

The selection of an advanced high-temperature Min-K is based upon current developments at Johns-Manville to evaluate existing fibrous thermal insulation products and to develop an improved product which will be suitable for operation to 1900° F. An advanced Min-K should be developed within one year which will have mechanical and thermal properties that are superior to those of Min-K 2000, and which will be suitable for use to 1900° F. In the selected RTG configuration, design and analysis have been based on the known properties of Min-K 2000.

c. Radiator subsystem

The primary function of the radiator subsystem is to reject waste heat, by radiation, from the RTG to space. It also has several secondary functions. The radiator provides an interface for the hinged conical aft cap at the narrow end of the radiator shell to permit access to the interior of the RTG. During the fueling operation, it aids in guiding the heat source into its installed position and provides an indication that the latch mechanism is fully engaged. It includes thermal insulation at the closure between the heat source and the thermoelectric subsystem and contains the instrumentation and power leads for the RTG.

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During ascent and later reentry, the radiator system provides structural support for other RTG components as well as protection for these components during aerodynamic heating.

The radiator subsystem and its components are shown in detail in Fig. III-4. The components are:

- (1) Radiator (68)
- (2) Aft cap hinge plates
- (3) Heat source installation guides (71)
- (4) Heat source support assembly (72)
- (5) Thermal insulation (64)
- (6) Electrical wiring harness (70)
- (7) Electrical connector (69)

(1) Radiator

The most significant item in this subsystem is the radiator. This component is a truncated conical shell made of beryllium, 0.50 inch thick at its central section and tapered to 0.125 inch at each end. There are 396 couple mounting holes spaced around the cone at its thickest section. Other holes are provided for the electrical connector and for mechanical fasteners used to attach structural components to the beryllium shell.

(2) Aft cap hinge plates

The aft cap hinge plates are located at the narrow upper end of the radiator shell to provide a pivot support for the aft cap. These plates are made of Haynes-25 sheet with holes to accept a hinge pin. The aft cap is opened prior to launch for RTG checkout and fueling. When the aft cap is closed, it is secured by eight latches, and support from the hinge plates is no longer required.

(3) Heat source installation guide

The heat source installation guides are also fabricated from Haynes-25 sheet and are fastened internally to the radiator. They serve to align the heat source during the fueling operation while their upper ends also act as latch striker plates for the aft cap latches.

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(4) Heat source support assembly

The heat source support assembly is a biconical structure made of 0.031 inch thick Haynes-25 sheet. It is attached at its outer periphery to the radiator by means of a T ring just below the thermoelements. The assembly is designed to support the 181-pound heat source for peak accelerations of 50 g in either the axial or lateral direction.

Centrally attached to the support cone is a spring-loaded locking disc and its limit switch. During the fueling operation, the fueling probe depresses the locking disc before releasing the heat source latch hooks. This action opens the limit switch and gives a no-latch indication through launch complex telemetry. When the fueling probe is retracted, the latch fingers pivot outward to engage with the latch ring and the locking disc telescopes upward between the latch fingers to lock them in place. This action closes the limit switch and gives an indication that the heat source is latched and locked into place. If one or more latch fingers do not engage completely, they will prevent the locking disc from extending and a no-latch indication will be noted until the situation is corrected.

(5) Thermal insulation

The thermal insulation consists of an upper and lower ring of Micro-Quartz fibrous insulation. This flexible material is fabricated in mats one inch thick or less. The rings illustrated in Fig. III-4 are fabricated by cutting and laying multiple mats over a form, then binding the Micro-Quartz in silica cloth for shape retention and abrasion resistance. The upper ring will be installed by lacing it to the heat source guides while the lower ring will be supported by a wire form. Both rings have conical surfaces which mate with the heat source with an interference fit to prevent heat loss by radiation streaming.

(6) Electrical wiring harness

The electrical wiring harness includes thermocouples and voltage instrumentation leads, heat source latch verification leads, aft cap power and verification leads and RTG output power leads. It terminates at the electrical connector.

(7) Electrical connector

The receptacle will be one half of a quick-disconnect connector. It will have a shorting feature at the output power leads such that when disconnection occurs (at the time of separation from the CIR), the thermoelectric subsystem will be short circuited so as to cool the hot junctions by means of Peltier cooling. Shorting will lower the heat

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Fig. III-11. Progressive Shunt Regulator

Fig. III-12. Regulator Section Block Diagram

Fig. III-13. Regulator Section Schematic Diagram

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source temperature approximately 200° F below comparable open circuit temperatures prior to reentry. Although this is not necessary from the standpoint of fuel capsule integrity, it is desirable from the standpoint of the added safety factor. The thermal analysis for descent (see Appendix C) has conservatively assumed that the thermoelectric subsystem was open circuited prior to reentry.

d. Voltage regulator

A shunt voltage regulator has been included in the generator system to limit voltage excursions during switching or shutdown of spacecraft equipment. The regulator consists of six shunt-connected regulator sections as shown in the subsystem block diagram (Fig. III-11). Each section contains a voltage sensing and comparing circuit, an amplifier and a shunting element as illustrated in Fig. III-12. Figure III-13 shows the components necessary to perform the required functions.

This circuit acts as a feedback control system. A differential amplifier (Q_1 and Q_2) is used as a comparator which samples a portion of the generator voltage (through Resistors R_1 and R_2) and compares it to the reference provided by a zener diode, Z_1 . If the RTG voltage attempts to go above the design voltage, a difference voltage, V_D , is developed by the differential amplifier, which is converted to a proportional current, I_D , by the transistor, Q_3 . (Variable Resistor R_1 is set such that I_D is zero when the RTG voltage is 28 volts.) Current I_D is amplified by Transistors Q_4 and Q_5 and supplied to the base of the shunting transistor, Q_S . The base current is sufficient to enable Q_S to shunt the necessary current to maintain the 28-volt output. Resistor R_S limits the shunting current to 2.1 amperes and shares the power dissipation requirements of Q_S .

At BOL, the RTG will produce 360 watts. If all the spacecraft equipment is shut down, all 360 watts must be dissipated in the six regulator sections. This corresponds to 60 watts or 2.1 amperes per section. The value of shunt resistance R_S to pass 2.1 amperes is 13 ohms, and the maximum power dissipated by Q_S of any section would be 15 watts which would occur at the half-power condition for that section.

The sections are activated progressively such that only one of the six shunting transistors, Q_S , is in the maximum dissipation region at

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Fig. III-14. Progressive Shunt Regulator Package

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any given time. This is accomplished by adjusting R_1 in each section such that they would be activated at progressively higher voltage levels. For example, Section 1 would be set to activate when the RTG voltage increases to 28.01 volts. Due to Resistor R_S , the maximum current which would be shunted by Section 1 is 2.1 amperes. If this amount of current was not sufficient to maintain the voltage, it would increase further. If the voltage increased to 28.05 volts, Section 2 would be activated. It should be noted that Section 2 would not become activated until Section 1 was shunting its maximum current and Q_{S1} was in saturation. The power dissipated by Q_S when in saturation is approximately one watt.

As the voltage increases further (due to a further decrease in system power demand), the remaining five sections are progressively activated. The maximum total power to be dissipated by the shunting transistors would occur when Q_{S6} was in the maximum dissipation region. There would be a contribution of one watt each from Q_{S1} through Q_{S5} which would be operating in the saturation region, and 15 watts from Q_{S6} which would be operating in the maximum dissipation region. This would give a maximum total power dissipation of 20 watts.

The six Q_S transistors are placed on a single aluminum plate which provides a common heat sink. The entire regulator package, excluding the six dissipating resistors, occupies approximately 40 in.³ (3 by 3 by 4.5 inches) and weighs approximately one pound. The dissipating resistors are mounted remotely from the regulator package since they operate at a high temperature. The resistors require an additional 20 in.³ and weigh approximately one-half pound. A conceptual package layout is shown in Fig. III-14.

2. Reentry Body (RB) Concept

The RB includes three subsystems--the heat shield, the aft cap and the beacon. In addition, the beryllium radiator of the RTG provides protection for RTG components during reentry. The subsystems are illustrated in Fig. III-4.

a. Heat shield subsystem

The ablative heat shield (30) consists of an elastomeric silicone compound (Martin ESA 3560H) which is supported in a low density

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flexible plastic honeycomb matrix. The honeycomb matrix is bonded to the heat shield backup structure by HT-424 adhesive.

The heat shield backup structure (45) consists of a bonded phenolic glass laminate honeycomb panel assembled to conform to a spherical radius. The two prime functions of this component are to provide a continuous structural support for the ablative heat shield and to react the airloads imposed upon the vehicle forebody during reentry. The honeycomb panel is attached to a titanium skirt structure which forms a structural transition between the panel and the beryllium radiator. This skirt structure is sufficiently flexible to accommodate the large differential thermal expansion between the honeycomb panel and the beryllium radiator and at the same time retain the structural continuity and integrity between these two components.

An insulation ring (53) exists between the radiator/heat shield splice frame and the titanium skirt structure in order to control the heat flow from the beryllium radiator to the heat shield backup structure. The ring is made of Min-K 2000.

b. Aft cap subsystems

A conical frustum is aft of the radiator. It is made of Haynes-25 using a skin-frame construction. Its functions are to close off the RTG (to protect it from aerodynamic heating), to serve as an aerodynamic destabilization system in case of inverted entry, and to provide a platform for the beacon antenna.

A hinge between the radiator and the cone frustum permits opening the reentry body for ground handling, fueling and checkout. This device can be operated remotely (both opening and closing).

An attachment point is provided on the cone frustum for a physical link between the fairing and the cone. As the fairing is opened or closed, the cone frustum is also opened or closed. At launch, automatically activated pin pullers disengage the physical link.

Eight remotely activated latches are provided on the cone frustum to lock it to the radiator after RTG fueling. These latches are fail-safe and can be released remotely. Their secure closing and opening are verifiable by remote techniques.

c. Beacon subsystem

The reentry body will contain a C-band beacon and its associated power supply. This unit is mounted in the hinged aft cap and will be utilized for reentry tracking purposes where abort or

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deorbit may be effected to an impact point where tracking coverage is available from ground or range ship sources. The expected temperature environment within the RB makes an independent telemetry system impractical except as a substitute for the RTG during development tests. The C-band beacon unit may require development of high temperature circuitry and power supply. The beacon is on stand-by during ascent and activated only at separation.

d. Reentry characteristics

The RTG/RB is designed to survive reentry from any condition that could be encountered during ascent, transfer orbit or injection into the final orbit. Included in these various reentry conditions is the possibility that the CIR will deliver a velocity input up to 1500 fps in any direction.

There are two critical design conditions for reentry. These are:

- (1) Steep entry angle, orbital velocity
- (2) Shallow entry angle, orbital velocity.

The maximum heating rate and maximum aerodynamic deceleration are controlled by the first condition. The maximum aerodynamic heat absorbed by the reentry system is controlled by the second condition. This condition can occur only when some part of the Titan III MDS or the CIR system malfunctions.

The design conditions that arise for the reentry body are therefore as follows:

Maximum acceleration, $a = 25.5 \text{ g}$

Maximum heat rate, $q = 95 \text{ Btu/ft}^2\text{-sec}$

Maximum total heat, $q = 10,680 \text{ Btu/ft}^2$ based on $W/C_D AR_n = 3.6 \text{ lb/ft}^2$

Detailed reentry trajectories are presented in Appendix HH. The impact points of the RTG/RB are shown in Figs. III-15 and III-16 for the 30-degree orbit and the 90-degree orbit, assuming a Titan IIIC launch vehicle. The dispersion about these impact points is discussed in Appendix P.

e. Aerodynamic properties

The reentry body is aerodynamically stable at hypersonic velocities in only one attitude heat shield forward. It is neutrally stable in roll

*Reference values at stagnation point.

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Fig. III-15. Potential Impact Points, 30-Degree Orbit

Fig. III-16. Potential Impact Points, 90-Degree Orbit

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Fig. III-17. Hypersonic Lift Coefficient as Function of Angle of Attack for the CRONUS RB

Fig. III-18. Hypersonic Drag Coefficient as Function of Angle of Attack for the CRONUS RB

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although the aerodynamic coefficients are distinct functions of the roll angle at high angles of attack. This is due to the shape of the aft cap. To hasten the turning of the vehicle in case of a spin stabilized, narrow end forward reentry, the narrow end of the cone is truncated with a 45-degree slope.

The center of mass of the reentry system is on the roll axis. During normal entry, therefore, the system will act as a ballistic (zero lift) reentry body. In case of a malfunction and the MDS system does not cause separation, the system may enter the atmosphere with a high spin and/or pitching rates, and with abnormal attitude with respect to the atmosphere. The aerodynamic properties of the vehicle have therefore been calculated for all angles of attack, side slip and roll (Appendix Y). For ease in visualizing the behavior of the body, the actual aerodynamic coefficients have also been averaged over roll.

f. Aerodynamic coefficients

The hypersonic aerodynamic coefficients for the RTG/RB were determined by using a digital code which considered modified Newtonian impact theory. A maximum pressure coefficient of 1.833 ($C_{P_{\max}} = 1.833$) is used in this code. It calculated the following coefficients as a function of total angle of attack, η , and roll angle, ϕ , in the body axis system:

- (1) Axial force coefficient (C_X)
- (2) Normal force coefficient (C_Z)
- (3) Side force coefficient (C_Y)
- (4) Rolling moment coefficient (C_l)
- (5) Pitching moment coefficient (C_m)
- (6) Yawing moment coefficient (C_n).

These coefficients are discussed and presented in Appendix Y. Average coefficients for lift (C_L), drag (C_D), and pitching moment versus angle of attack (α) are shown in Figs. III-17 through III-19. In Fig. III-20, the hypersonic lift-to-drag ratio versus angle of attack is presented for the RTG/RB.

Figure III-21 shows the drag coefficient at zero angle of attack (blunt nose first) as a function of Mach number (M). The values for C_D between $M = 0.5$ and 9.6 were obtained from Ref. III-1.

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Fig. III-19. Hypersonic Pitching Moment as Function of Angle of Attack for the CRONUS RB

Fig. III-20. Hypersonic Lift to Drag Ratio as Function of Angle of Attack for the CRONUS RB

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Fig. III-21. Drag Coefficient at Zero Angle of Attack as Function of Mach Number for CRONUS RB

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$S' = 1.95 \text{ ft}$ = surface distance from stagnation point to shoulder on blunt nose
 S = surface distance from stagnation point
 $C_P = (p - p_\infty) / q_\infty$ = pressure coefficient
 p = local pressure
 p_∞ / q_∞ = free stream pressure and dynamic pressure, respectively
 $\dot{q}$ = local heating rate
 $\dot{q}_{S_{\alpha=0^\circ}}$ = stagnation heating rate on blunt nose at $\alpha = 0^\circ$
 α = angle of attack

Fig. III-22. Pressure and Heating Distribution over CRONUS RB at Zero Angle of Attack

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The hypersonic ballistic coefficient ($W/C_D A$) at zero angle of attack for the RTG/RB is 22 lb/ft². The value of ballistic coefficient divided by nose radius is 3.6 lb/ft³. At subsonic speed the ballistic coefficient is 34 lb/ft².

The pressure coefficient, C_P , and the heating ratio, $\dot{q}/\dot{q}_S$ $\alpha = 0^\circ$, are presented in Fig. III-22 for zero angle of attack (blunt nose first) as a function of surface distance. These distributions are over the blunt nose and the conical afterbody. The q is defined as the local heating rate and $\dot{q}_S$ $\alpha = 0^\circ$ is the stagnation value at zero angle of attack. Other distributions for pressure and heating are discussed and presented in Appendix Y for angles of attack greater than zero degrees.

C. CONTROLLED INTACT REENTRY SYSTEM (CIR)

In an emergency, the CIR must perform the following functions:

- (1) Receive separation commands
- (2) Separate from spacecraft
- (3) Deorbit upon ground command
- (4) Separate from RTG/RB.

During functions (2) and (3), the airborne system is spin stabilized. Therefore, a fifth function must be performed--namely, keeping the residual pitch and roll rates to a minimum.

The CIR must also perform some functions during the normal operation of the airborne system. First of all, it is the physical link between the spacecraft and the RTG/RB. It is also the housing for the voltage regulator subsystem of the RTG/RB. All of the RTG/RB electrical signals are made to the CIR. From there, the information is telemetered or sent to AGE via an umbilical. The power is conditioned and then sent to the spacecraft. Thus, the electrical connections between RTG/RB and the spacecraft and launch vehicle are through the CIR.

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LEGEND:

- 13. Separation Tracks
- 14. Separation Track Struts
- 15. Separation Carriage
- 16. Spin Bearing
- 17. Separation Rocket and Mount
- 18. Deorbit Rocket
- 19. Spinup Rocket
- 20. Despin Rocket
- 21. Command and Telemetry Antenna
- 22. Separation Band RTG/RB to CIR
- 23. Latch Fingers
- 24. Electrical Disconnect RTG/RB to CIR
- 25. Electrical Disconnect (Squib Fired) RTG/RB-CIR Spacecraft
- 26. Subsystems Equipment Mounting Shelf
- 31. Mounting Bulk for RTG/RB
- 32. Power Conditioner Resistor Box
- 33. Power Conditioner Transistor Box
- 34. Pyrotechnic System Solid State Switch
- 35. Battery
- 36. Command and Telemetry Module
- 37. Pyrotechnic Safe Arm Units
- 39. Equipment Access Panel
- 40. Separation Band Pyrotechnic Release
- 42. Pyroswitch
- 43. Separation Spring, RTG/RB to CIR

Fig. III-23. CIR System

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The CIR consists of four subsystems:

- (1) Structure
- (2) Propulsion
- (3) Ordnance
- (4) Electronics.

The purpose and general description of the components comprising the CIR system was given briefly in Section A. The CIR subsystems and design/performance considerations are discussed in greater detail in the sections that follow. The airborne system drawing (Fig. III-1) shows the CIR as well as its relationship to the other systems. A trimetric view of the CIR is presented in Fig. III-23.

The weight of the CIR is summarized in Chapter IX. Included in this weight is that of the voltage regulator subsystem of the RTG/RB, which is physically located in the CIR. The center of gravity has been determined to be on the centerline and at Station 48.8. The moments of inertia of CIR are:

Roll	26.7 slug-ft ²
Pitch	16.9 slug-ft ²
Yaw	16.2 slug-ft ²

Before proceeding with the discussion of subsystems, however, it is appropriate to present the flight operational sequence.

1. Flight Operational Sequence

CRONUS, excluding the AGE, is basically an RTG power supply, plus an orbital parachute. The basic self-explanatory functions it has to be able to perform are shown in Fig. III-24.

Some of the details in this plan must be described carefully. The exact strategy under which separation from the spacecraft occurs, and the sequence of events which initiate the ground-based command to fire the deorbit rockets, must be described.

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The procedure for mission abort, hence separation of the reentry body system from the spacecraft and controlled intact reentry of the RB, is described for each phase of the mission for which potential abort condition exists.

a. Period from liftoff to payload fairing release

At launch, the physical link between the fairing and the RTG/RB aft cap is released by activating the fairing attachment pin puller.

The ejection system will be locked out to prevent separation of the CIR-RTG/RB from the launch vehicle prior to ejection of the aerodynamic fairing at 160 seconds into the Titan III flight (approximately 300,000-ft altitude). In the event of an abort, the CIR-RTG/RB will go down with the vehicle. At fairing release, the CIR will be armed.

b. Period from fairing release altitude through transfer orbit injection

If an abort command is received either from the LV or from a ground originated command, a standard separation procedure will be followed; i.e., a retainer pin holding the CIR-RTG/RB systems to the spacecraft will be released, and the separation rocket and two spin rockets will be ignited. The CIR-RTG/RB systems will clear the spacecraft in approximately 0.25 second. The abort signal which leads to separation will be derived either automatically through the malfunction detection system (MDS) or through ground command. An automatic malfunction command generated by the MDS located within the spacecraft launch vehicle will, at any time prior to separation of the transtage from the spacecraft, immediately cause the CIR-RTG/RB to be separated from the spacecraft. The deorbit function, however, will always be commanded from the ground.

Ground command will issue, as soon as possible and simultaneously if possible, the deorbit signal regardless of whether the spacecraft separation signal is automatic (MDS) or by ground command. Fixed delay periods will be incorporated in the deorbit ordnance circuit to assure an adequate separation distance before the deorbit impulse is delivered or despin and structure separation can occur.

c. Period from transfer orbit to circular orbit injection

If the separation command is received, the standard separation procedure will be followed. However, deorbit command will be given as soon as

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Fig. III-24. Flight Operational Sequence

possible after it can be established that the RTG/RB will not impact in a restricted area on the earth's surface.

d. Period during circular orbit injection

If the MDS indicates a malfunction, the normal separation sequence is followed. However, some malfunction may be disguised as a normal flight (i.e., guidance system error). In these cases, the MDS system does not automatically separate the CIR-RTG/RB from the spacecraft. The transtage will remain attached to the spacecraft and will maintain the normal (parallel) flight path orientation of the vehicle. A ground command for separation and deorbit will be issued at the most appropriate time for controlled reentry if the actual orbit does not meet certain minimum specifications. Since Nimbus cannot operate with the transtage attached, this is equivalent to terminating the mission if a proper lifetime is not achieved.

e. Circular orbit (600 naut mi)

The standard separation procedure will be followed for any malfunction of the transtage prior to separation of the transtage from the spacecraft. The separated CIR-RTG/RB will not be deorbited if the spacecraft is in the process of being placed in its proper earth orientation by the transtage at the time the transtage malfunction occurs.

It is possible that spacecraft orientation by the transtage will not be attempted immediately after a successful circular orbit has been achieved. This situation could arise for several reasons. Included are:

- (1) Achievement of a successful circular orbit has not been adequately confirmed, in which case all or a portion of one orbit may be required.
- (2) Ground command may desire to await a more advantageous position with respect to observation of the orientation maneuver.

Some transtage malfunctions in this period may require separation of the CIR-RTG/RB from the spacecraft. However, the desirability of deorbiting will be left to the discretion of ground command.

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Separation of the transtage from the spacecraft will automatically disengage the MDS system to the CIR. No attempt will be made to either separate or deorbit the CIR-RTG/RB after this time.

f. Sequence

Flight Sequence

<u>Time</u>		<u>Event</u>
	Prior to launch	Turn on telemetry Verify cone frustum closed Verify RTG latches closed Verify RTG temperatures proper Verify RTG power outputs
0	Launch	Turn beacon on standby
160 sec	Fairing release*	Arm CIR
259 sec	BO Stage 1	
474 sec	BO Stage 2	
25,171 fps (760 sec)	Velocity sufficient to impact Africa	
23,644 fps	Circular velocity	
841 sec	BO ₁ , transtage	
3783 sec	Start, transtage	
3831 sec	BO ₂ , transtage	
	Verify final orbit	Disarm CIR, turn off beacon and telemetry
	Turn transtage	
	Separate transtage	

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Flight Sequence (Cont'd)

*The following events take place, provided no abort command is received by the CIR. If such a command is received, the abort sequence is followed.

Assume that an abort signal has been given ($t_a = 0$) either automatically through MDS or by the decision of the ground command officer. The following sequence of events will then take place:

- (1) The initiators for the pull pin separation system and the igniters for the separation rocket and two spin rockets will receive the signal simultaneously such that:
 - (a) The structure holding the CIR-RTG/RB systems to the spacecraft will be released first.
 - (b) The spin rockets will build up thrust slightly ahead of the separation rocket and begin to spin the CIR-RTG/RB.
 - (c) The separation rocket will begin to propel the CIR-RTG/RB down the track.
- (2) The CIR-RTG/RB will clear the three-rail track system in about 0.25 second at which time it will be spinning at approximately 27 rpm.
- (3) The CIR-RTG/RB will continue to spin for another 0.37 second until a rotation of 62 rpm is achieved. The separation rocket is retained on the rail system.
- (4) Only a ground command signal can be sent to the CIR at the same time as the separation signal or anytime thereafter (a delay of hours is perfectly satisfactory). However, in any event, ignition of the deorbit rocket cannot take place until two seconds after receipt of the ground command signal.
- (5) The deorbit signal also activates the despin rocket circuitry and the CIR-RTG/RB separation circuitry such that the two despin rocket igniters are activated 15 seconds after ignition of the deorbit rocket.
- (6) The CIR-RTG/RB is despun in 0.63 second, five seconds after the deorbit rockets have burned out.

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- (7) The RTG/RB is separated from the CIR two seconds after despin.
- (8) The RTG/RB enters the atmosphere with a low residual spin rate.
If the CIR has not been separated from the RTG/RB, the aerodynamic initial heating of the separation band will separate the two bodies.

An alternative to Item 5 is:

It may be desirable to bypass the application of the deorbit V during a suborbital abort. For this reason, a (third) separation signal can be received by the despin rocket igniters and the CIR-RTG/RB separation clamps to assure that a clean nonspinning RTG/RB enters the atmosphere.

2. Structure Subsystem

The functions of the various pieces of the structure subsystem are described and are summarized in Table III-7. They are illustrated in Figs. III-1 and III-23.

TABLE III-7

CIR Structure Functions

	Support RTG/ RB	Separation CIR-RTG/RB	Attach to S/C	Separate from S/C	Support Propul- sion	House and Support Electronics and Ordnance
Frame Structure	X		X		X	X
Support Ring	X					
Attachment Finger	X	X				
Finger Re- tention Band	X	X				

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TABLE III-7 (Cont'd)

	<u>Support</u> <u>RTG/</u> <u>RB</u>	<u>Separa-</u> <u>tion CIR-</u> <u>RTG/RB</u>	<u>Attach</u> <u>to S/C</u>	<u>Separate</u> <u>from S/C</u>	<u>Support</u> <u>Propul-</u> <u>sion</u>	<u>House and</u> <u>Support</u> <u>Electronics</u> <u>and Ordnance</u>
Separation Spring		X				
Shelves						X
Spin Bearing				X		
Support Carriage			X	X		
Tracks			X	X		
Track Rollers			X	X		
Separation Motor Mount				X	X	
Separation Release			X			

a. Frame structure

An inverted conical skin stringer frame structure of aluminum alloy (4) is used to house the subsystem equipment and to support the RTG/RB on the spacecraft. The base end of the cone is topped by a RB support ring that nests about the outside of the RB heat shield face and by the RB attachment device. The small end of the cone terminates at the spin bearing inner face which is part of the S/C separation system. Inside the cone is a shelf for support of electrical/electronic subsystem equipment and certain ordnance subsystem components.

b. Support ring

The purpose of this structure is to support and protect the ablative heat

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shield on the RB. The support ring is made in two parts. The outer cylindrical part is composed of three turned and spun aluminum alloy pieces, a zee section, a channel section and a spun flared section. The inner part is a bonded aluminum honeycomb ring which is contoured to match the heat shield surface. It is covered with a one-tenth inch thick silicon rubber material which is curved at the outer edge in order to nest around and protect the ablative heat shield.

c. Attachment fingers

Eight spring-loaded attachment fingers (23) are provided to hold the RB down in the support ring. Provisions are made to prevent vacuum welding of the finger face to the beryllium radiators by coating the finger faces with teflon and to prevent binding of the finger pivot by coating the pivot end of the finger with molybdenum disulfide.

Seven of the eight fingers are identical. One of the fingers is modified to contain an electrical attachment plug to connect into the RB. This quick-disconnect plug is engaged as long as the fingers are in the closed position, and is positively disconnected as the fingers are opened. Each finger is pushed to the open position by a compressed spring.

d. Finger retention band

A steel band (22) surrounds the support ring and fingers. This holds the eight attachment fingers in the closed position. The only way these fingers can open is if this band is released. As long as the band is in place, the RB is held securely in the support ring. As soon as the band is opened anywhere along its periphery, all fingers are released simultaneously.

The band comes in two halves, each of which is joined to a pyrotechnic band release mechanism. When either of these release mechanisms is activated, the band is opened.

Two pyrotechnic band release devices are provided. Two Hi-Shear Corporation clamp separators, SC1001-4, with dual PE470 initiators and a HE350 initiation command module are considered for this function (Fig. III-1, Detail A, 40).

e. Separation springs

Four matched compression springs (43), equally spaced around the outer

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region of the heat shield, are used to separate the RTG/RB from the CIR.

These springs each provide 80 pounds of force when deflected four inches, as they are when the two bodies are joined. This results in a differential velocity between the bodies of approximately 5.5 fps at the end of the stroke.

The release of the attachment fingers, previously described, allows these springs to activate.

f. Shelves and access door

Shelves (26) are provided in the frame structure to physically support the ordnance and electronic devices. An access door (39) is provided in the frame structure to allow servicing of the internal equipment.

g. Spin bearing

The spin bearing (16) is an X roller bearing (invented by William Messinger and under U.S. Patents owned by Messinger Bearings, Inc., Philadelphia, Pennsylvania). This bearing type is unique in that it does multiple duty; it acts as a radial, thrust and moment bearing providing roller action between the races under all three loading conditions.

The bearing chosen for this application has a 20-inch OD with a 3/4 by 3/4 inch cross section. The rollers are 3/8 inch in OD by 3/8 inch long. The outer race is split to permit assembly. A cross section of this bearing appears in the sketch.

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The cylindrical axes of the roller are alternated perpendicular to one another around the bearing.

The races and rollers are made of bearing steel. For this vacuum environment, the roller cage is contemplated to be fabricated from teflon and the rollers coated with a 0.0002 to 0.0003 inch phenolic epoxy-bonded layer of molybdenum disulfide. (From NASA SP-38, "Advance Bearing Technology," 1964.) This reference gives the sliding coefficient of friction and rider wear for phenolic epoxy-bonded molybdenum disulfide of these thicknesses as $C_F = 0.025$ and rider wear approximately 1×10^{-10} in.³/hr under the following conditions: ambient pressure = 8×10^{-7} to 2×10^{-6} mm of Hg; sliding velocity = 390 ft/min; load = 1000 grams; duration of run = 1 hour, material = 440 C on 440 C. This reference also gives the evaporation rate for this coating as approximately 10^{-8} gm/cm²-sec at 70° F and 1 to 2×10^{-6} mm Hg ambient pressure. For teflon, the same reference gives the evaporation rate at 70° F and 10^{-7} mm Hg as 10^{-9} gm/cm²-sec. The 0.0002 inch MoS₂ coating will not last longer than three days which is adequate for the CIR system.

The time from separation onset, which includes spinup, to the RTG/RB-CIR leaving the guide rails is only 1/4 second or approximately 1/4 revolution. This is all the dynamic bearing life that is required. Since the C_F and wear values of MoS₂ defined in the reference are low for sliding friction and this is primarily rolling friction, no friction or wear problems are anticipated.

h. Separation carriage

This structure is essentially an equilateral triangle with curved sides, to receive the spin bearing outer race (15). It is an extruded channel constructed of aluminum alloy with the flanges extending away from the center plus other angles and plates for mounting the bearing and the separation rocket motor.

Each corner of this structure is provided with a fitting that supports three track rollers. Two rollers in each group ride the flanges and the third rides the web of the separation track as shown in the sketch.

The axes of the rollers riding the center track are oriented 90 degrees to those riding the outer tracks.

The separation rocket motor mount is attached to the corner of this structure which is adjacent to the center track.

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i. Separation tracks

Three extruded channel guide rails (13) are mounted on top of the spacecraft. Each of the two outer rails are supported at their outboard ends by a pair of tubular struts and are flanged toward each other. The center rail is entirely within the bounds of the spacecraft top and is flanged up.

These tracks receive the rollers, three groups of three, on the separation carriage. Their outer ends are cut to a shape that assures that all nine rollers disengage at the same instant. The end of the center track, opposite to the direction of separation, is blocked off and acts as a stop for centering the RTG/RB-CIR on the spacecraft. This same block has a plate on top that extends across the end of the channel flanges. The central roller, the one in the vertical plane, of the three riding this track is trapped under this plate when the carriage supporting the RTG/RB-CIR is positioned against the stop.

j. Track rollers

A needle bearing, similar to NAS 505-4 but modified with coatings to replace the lubricant for hard vacuum use, is contemplated for these rollers.

For recall within the first few days, the rollers (needles) will be coated with 0.0003 inch of MoS_2 phenolic epoxy bonded. The outer surface of the outer race of these rollers will also be coated with this film.

k. Separation release

The carriage and spin bearing inner race, which supports the RTG/RB-CIR system on top of the spacecraft, are both locked to the spacecraft by a pyrotechnic pin puller mounted on the spin bearing inner race.

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A Hi-Shear SP 1200 dual cartridge pin puller with a HE 350 initiation command module and a PE 470 initiator was selected for this duty (49). The PE 470 initiator may be coupled with a booster charge to assure sufficiently high pressure to pull the pin under any loading condition.

The shear pin part of the pin puller engages a block mounted on the top of the spacecraft so that the pin is loaded in single shear. This block has a V groove that rests against one side of the shear pin and is adjustable in position, parallel to the separation tracks, to provide a positive preload of the separation carriage against the carriage stop at the closed end of the center rail.

1. Separation motor mount

The separation rocket motor (17) is mounted in a jettisonable bracket attached at the corner of the separation carriage adjacent to the center guide track.

The rocket motor and mount must be jettisoned, since it starts to spin with the rest of CRONUS when the separation carriage leaves the guide tracks. The dynamic unbalance of this mass if it were left attached might not be tolerable and sufficient balance weights are undesirable.

The rocket motor is permanently bolted to the mount and the mount is attached to the separation carriage by four bolts in slotted holes. The axes of the slots are parallel to the separation motion direction and the open ends of these slots are in this motion direction. The acceleration produced by the rocket drives the closed ends of these slots against the bolts, thus transferring the rocket thrust to the separation carriage.

Two lugs, one attached on each side of the mount, protrude below the bottom of the carriage. These lugs intercept two other lugs mounted on the spacecraft. The lugs on the spacecraft are located so that they encounter their mates on the mount just as the carriage rollers leave the guide tracks. Blocking further motion of the rocket motor and mount causes the mounting bolts in the carriage to move out of their slots, leaving the rocket and mount behind.

Teflon washers will be used on these bolts both above and below the mount flange with the slotted holes. This is to minimize the sliding friction and any possibility of vacuum welding of the mount to the carriage.

When the separation carriage is moved against its stop at the closed end of the center guide track, a lug on this stop rests against the rocket motor mount, thus preventing it from coming loose before the desired moment.

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Fig. III-25. Schematic Diagram of Ordnance Subsystem

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3. Ordnance Subsystem

The abort sequence has been described in Section III-C. The ordnance subsystem consists of the devices and circuitry required to activate the abort functions according to the predetermined abort sequence logic. A schematic diagram is shown in Fig. III-25.

a. Ordnance components

The primary components of the ordnance subsystem, other than the rocket igniters which are included with the propulsion subsystem, are:

- (1) Solid state switching device (34). (Ref. Fig. III-23) Twelve of these switches are provided. Martin provides a large number of these devices for the Titan III.
- (2) Six time delay devices (42) of the type manufactured by Atlas Chemical Industries are required. Their OM Type 450 delay devices can meet both the 2- and the 15-second requirements for weights of 24 and 48 grams, respectively. A positive visual and electrical indication of status is included. These switches consist of a gas generating module which drives a piston-like column of parallel contacts down a sleeve containing a matching series of parallel contacts. The number of contacts depends on the number of circuits to be connected. The length of the sleeve determines the amount of delay time obtained.
- (3) Twelve safe-arm devices including initiators (37) of the type manufactured by Hi-Shear Corporation are used. Their initiator command module (HE 350 type) combines standard safe-arm and checkout capability with positive protection against accidental activation through stray RF or EM energy. The completely remote checkout and safety features will eliminate the necessity for ordnance personnel to be at the launch pad late in the countdown and thus possibly conflict with installation of the fueled RTG.
- (4) Two clamp separators (40) of the type manufactured by Hi-Shear Corporation are required. The SC 1001-4 type gas piston-operated clamp operates on the shear pin principle. The combustion products and piston can be retained internally, and redundant power cartridges can be included.

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b. Discussion

Power is provided by either the RTG or a battery. The RTG is the prime power source with the battery being switched into the circuit automatically (low voltage switch relay) only in the event the RTG power system fails or falls below an established power output level. The RTG and battery circuit are mutually and independently redundant.

Redundant solid state switching devices are used in each circuit for separation of the power sources from the ordnance system. After fairing separation, similar redundant devices isolate the power source from the safe-arm and pyrotechnic components. Safe-arm devices are utilized in the reference configuration to provide maximum safety with respect to inadvertent or RF energy ignition of solid rocket motors and separation devices prior to liftoff. One set of redundant switches will be activated by the MDS separation signal source and the second set will be activated by the separate ground command deorbit signal source.

A two-second pyrotechnic delay device is included in the deorbit circuit to ensure that accidental overlapping signals from the two signal sources cannot cause the deorbit system to function before the CIR-RTG/RB has been separated from the spacecraft. Furthermore, to ensure that an inadvertent deorbit command, prior to a separation command, will not cause the retrorockets to fire before separation, the deorbit circuit requires that the separation system switching devices be closed before power can be made available.

Redundant circuits are used for CIR-RTG/RB spacecraft separation (single point), separation rocket ignition and spin rocket ignition. The safe-arm devices permit checkout to the bridgewire and provide inherent means for prevention of inadvertent activation from stray RF or EM energy. Initiators also include this latter safety feature. All rocket igniters are double squibbed and contain two bridgewires.

Redundant circuits are used for deorbit rocket ignition, despin rocket ignition and separation of the CIR from the RTG/RB. Safe-arm devices for the rocket ignition systems follow the design concept described previously. Redundant pyrotechnic delay switches (15-sec) ensure proper and reliable time functioning of the safe-arm devices for the despin rocket ignition as well as activation of similarly redundant pyrotechnic delay switches (2-sec) in the structure separation system. Blocking diodes are utilized in each of the redundant pyrotechnic delay circuits.

The two-piece tension band used to retain the eight spring-loaded clamps that hold the CIR to the RTG/RB is held in place by two pyrotechnic activated clamp separators one clamp at each of the two opposing

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band separation points. Only one of the clamp separators need function to assure a clean separation of the CIR from the RTG/RB. Each clamp separator utilizes two power cartridges.

Ground checkout of the entire ordnance system is achieved through a service disconnect. The checkout system involves all ordnance system components up to the solid state switching devices that initially isolate the airborne power supplies. The ordnance circuit for each power supply is checked out separately. Checkout consists primarily of electrical continuity checks; however, both electrical and visual checks (lights) of the pyrotechnic delay switches are provided.

A schematic of the signal-time sequence for separation and deorbit is shown.

4. Propulsion Subsystem

The CIR propulsion subsystem consists of three motor packages or a total of nine rocket motors. These assemblies are shown in Figs. III-1 and III-23 and are described in the following. Each motor will have its own ignition system etc., that is necessary for its operation.

a. Separation rocket

A single cylindrical solid propellant rocket motor (17) having a thrust of 2000 pounds and a burning time of 0.23 second will separate the CIR-RTG/RB systems from the spacecraft at a velocity of 20 fps. The high chamber pressure rocket consists of 10 singly perforated extruded rods of composite propellant bonded to a spider structure attached to the head end of the chamber. The propellant burns on all exposed surfaces in order to yield the high thrust for a short time.

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The motor, which is 4.5 inches in diameter and approximately 7.5 inches long, is a scaled-down version of the Atlantic Research Corporation's MARC 17D1 rocket motor. Over 400 have been manufactured.

The chamber is 4130 steel; mounting fixtures are provided in the head end dome which also contains the single igniter mount. A three-nozzle aft closure is used to minimize overall length.

b. Spin rockets

Two Atlantic Research Corporation (MARC 5A1) solid propellant spin motors (20) each with a thrust of 120 pounds and a burning time of 0.5 second will spin the CIR-RTG/RB systems to 62 rpm. The composite propellant which burns both on the end of the grain and radially is extruded. Each rocket is 2 inches in diameter and 8.3 inches long, and contains a flat plate mounting bracket welded radially to the steel case. The single nozzle has an expansion ratio of 32. The single head end-mounted igniter has an ignition interval of less than 40 ms. Over 100 production motors have been delivered for such programs as Explorer and Discoverer.

The two despin rockets and mounts will be identical to the spin motors.

c. Deorbit rockets

Four Thiokol Chemical Corporation SARV Retro MK1 solid propellant rockets (18) each having a thrust of 755 pounds and a burning time of 9 seconds will provide a deorbit velocity increment of 1500 fps. The 10-inch diameter, 12-inch long rocket utilizes a case bonded composite propellant in an internal burning eight-point star configuration. The 4130 steel case provides attachment fittings at the aft end chamber wall extension and can be modified to provide an additional attachment on the fore end of the cylindrical chamber wall. The single igniter is inserted through the nozzle.

The rocket has been developed and has served as the prototype of a somewhat larger unit (Mark IV) which is currently being manufactured in large quantities for space applications.

The rockets are mounted so that the nozzles are canted out at 15 degrees to the RTG/RB centerline. The nozzle cone exists also are skewed such that the thrust vectors are effectively canted slightly more. The combined misalignments of manufacture, assembly of the rocket and installation in the CIR will be less than 0.10 degree (3σ). Attachments will be adjustable, and alignment will be accomplished using optical equipment.

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a. SGLS Ground Station--Inputs and Outputs

b. SGLS Vehicle Flight Unit--Inputs and Outputs

Fig. III-26. Space Ground Link Subsystem

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Fig. III-27. Simplified System Block Diagram

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Fig. III-28. SGLS Vehicle Equipment Configuration

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Fig. III-29. SGLS Vehicle Flight Unit

TABLE III-8
Vehicle Equipment

<u>Item</u>	<u>Weight (lb)</u>	<u>Power (watts)</u>	<u>Volume (in. ³)</u>
Multiplexer	2.0	--	122
Receiver	3.4	1.44	109
Medium Power Transmitter (2 watts)	2.6	24.78	59
Power Converter (communication)	1.45	1.79	41
Power Converter (transmitter)	2.1	3.7	59
Power Converter (DTU)	1.3	0.2	72.5
Baseband Assembly Unit	1.3	0.2	72.5
256 Inputs	8.0	11.0	219
Signal Conditioner	1.5	0.5	63
Command Decoder	2.25	2.5	44
Programmer	6.0	9.0	130
	32.0	68.1	992

Dimensions: 7.25 by 5.0 by 27.36 inches

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which adds fore and aft and gives omnidirectional coverage in the transverse plane of the vehicle. The output of the medium power transmitter is distributed among the three turnstiles by a diplexer and will provide effective communications for ascent and orbit.

c. Power supply

The power supply required for the deorbit systems will include the power necessary for SGLS for nine hours and that required for ordnance activation (the latter requirement is minimal). The SGLS will draw about 2.43 amperes at 28 volts (68.1 watts). This power may be drawn from the fully operational RTG at beginning of life. However, a backup battery will be necessary and silver-zinc, manually activated cells have been selected. It will be necessary to select a unit with somewhat larger capacity than that required for use to account for possible storage effects and operating temperature degradation. A representative standard system which will meet requirements is the Eagle Picher MAP-4029. Battery usage will include activation during checkout, followed by parallel operation with the RTG during ascent. The battery may be packaged in several small units to aid in balance for spin stabilization.

d. SGLS usage

The SGLS will be activated at launch and permitted to radiate for telemetry and command purposes through the RF transparent fairing. The command system will serve to verify the safe condition and continuity of each of the 12 ordnance devices associated with the CIR system. Telemetry will be utilized to establish programmer status, verify command execution, monitor RTG and battery voltages, RTG/RB & CIR temperatures and the vibration environments during launch and ascent. Eastern Test Range coverage of these functions will be provided by S-band radar installations to be converted under other USAF programs to SGLS ground stations. Stations required include:

Cape Kennedy,
Antigua,
ETR Ship,
Ascension,
Pretoria,
Carnarvon
Guam,
Kwajalein
Hawaii,
Vandenberg and
Houston.

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6. Separation Analysis

a. CIR-RTG/RB-S/C separation

A rail separation system is used for the separation of the CIR-RTG/RB from the spacecraft. The operational sequence of this separation system is as follows.

When the separation command is received by the CIR, the separation sequence is started, provided the launch vehicle fairing has been jettisoned. (Failure of proper fairing separation calls for a flight abort, without CIR operation.) The source of the separation command can be either the MDS system or a ground station.

The initiation of the separation system triggers the following events:

- (1) Blow the Hi-Shear SP1200 dual cartridge pin puller. This releases the spin bearing so it can turn and the carriage so it can slide along the separation rails.
- (2) Ignite the two spin motors.
- (3) Ignite the separation motor.

The CIR-RTG/RB is then accelerated along the tracks with the separation carriage carrying the main systems. At the same time, it is being spun about its longitudinal axis on the spin bearing. This bearing is located on the separation carriage. A time history of this motion is shown in Fig. III-30.

The separation motor burns out prior to the carriage reaching the end of the track. As the end is reached, the separation motor is automatically jettisoned. The spin motors, however, burn longer than the separation motor. The angular velocity is still being increased after the airborne system has been separated from the S/C in order to reduce any pitching moment induced during separation.

While the carriage is in contact with the track, any longitudinal acceleration of the launch vehicle is also transmitted to the airborne systems. Consequently, any clearance problems between the L/V and the airborne systems can arise only after the carriage has left the tracks. The relative motion between the launch vehicle and the airborne system is shown in Fig. III-31, assuming a 3 g longitudinal acceleration. As this figure clearly indicates, there is no clearance problem between the L/V and the airborne system.

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Fig. III-30. CRONUS/LV Separation Time History

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Fig. III-31. Relative Motion Between Launch Vehicle and Airborne System

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There are some forces of constraint in the tracks which are the main forces which the launch vehicle feels during separation. These forces, which have been computed, will affect the motion of the launch vehicle. However, the net lateral, longitudinal and transverse forces of constraint are small as compared to the separation thrust (~2000 lb). Consequently, as long as the mass below the airborne system is at least as large as the airborne system, the perturbation in motion on the L/V due to the forces of constraint is negligible.

Considering L/V longitudinal acceleration, the two outer tracks can withstand both positive and negative longitudinal forces. The middle track, however, can only sustain compression longitudinal forces. The middle track is subjected to this type of force for all reasonable L/V attitude change rates acting in the plane of separation motion. Only when there is a 10 g negative acceleration on the L/V (due to an attitude change rate of at least 200 deg/sec) might the rollers leave the middle track prior to clearing the end. Since this is considerably in excess of the environment for which the S/C and the booster has been designed (not to mention the MDS sensors), no physically realizable L/V pitch rate will inhibit separation.

A L/V roll rate of the order of magnitude of 300 deg/sec may affect the separation system. This rate is also considerably in excess of what the L/V could survive. Consequently, no reasonable L/V roll rate will inhibit separation. Therefore, the separation system will cleanly separate the airborne system from the L/V.

b. Deorbit

The deorbit rockets are ignited at the discretion of the ground commander. This same signal also activates the delay circuits which sequence the despin and the CIR-RTG/RB separation function to occur after the deorbit impulse has been delivered. The four deorbit rockets decelerate the spinning CIR-RTG/RB from spacecraft velocity. Nominally, this velocity is decreased by 1500 fps in approximately 10 seconds. This velocity impulse is sufficient to deorbit the RTG/RB from all expected orbits and flight conditions, with sufficient margin to guard against coning, or other malfunctions, and also to decrease the range sensitivity to acceptable levels.

c. Despin

Six seconds after nominal thrust decay of the deorbit rockets, the two Atlantic Research MARC 5A1 despin rockets are ignited. These rockets despin the CIR-RTG/RB to approximately 0 rpm. Some low level residual spin will result from installation tolerances and performance tolerances in the spin/despin and deorbit rockets; however, it has been ascertained that the anticipated magnitude of residual spin will not affect the critical aerodynamic stabilization period during reentry.

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d. CIR-RTG/RB separation

The CIR and the RTG/RB should be physically separated prior to reentry. The chosen separation scheme is to open up the separation band by activating either or both of the two band cutters. The release of the separation band permits the eight spring-powered fingers to open. One of these fingers also releases the CIR-RTG/RB electrical connector. The two bodies can now be separated.

The separation force is provided by the four separation springs. These springs (20 lb/in.) are compressed four inches when the two bodies are joined. They are located at 90-degree intervals around the outer region of the heat shield.

These four springs induce a 5.5 fps relative velocity between the CIR and the generator. Any relative tumbling of the body will only increase the relative separation velocity without introducing clearance problems. The springs are sufficiently light so as to avoid any non-negligible tipoff moments during separation, even if there is a severe unbalance in some of the springs.

Since the time between separation and entry into the atmosphere will be on the order of several minutes (8 to 20), the separation distance between the CIR and the RB, when aerodynamic forces start to act on both, is at least 1500 feet. Consequently, the probability of a collision between the two reentering pieces is considerably less than 10^{-6} .

D. AEROSPACE GROUND EQUIPMENT (AGE)

The AGE system selected for CRONUS consists of ground handling equipment (GHE), maintenance ground equipment (MGE) and operating ground equipment (OGE). The GHE includes that equipment required to transport and handle the nuclear heat source; the MGE is that needed to accomplish the maintenance and verify the ready status of the RTG/RB-CIR systems; the OGE includes those equipment items which function during the countdown phase of the launch and are an integral part of the launch vehicle countdown exclusive of GHE.

1. Ground Handling Equipment

The selected GHE concept is shown in Fig. III-3. To review briefly, the concept is a top loading scheme servicing the payload from a Titan III umbilical tower by an UT mounted, palletized boom and a fueling machine (FM). The FM traverses the boom in transferring the nuclear heat source from the UT-positioned cask to the RTG/RB. Operation of the GHE in the loading operation is as follows:

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Fig. III-32. Fuel Block Loading Concept Tower Boom/Top Loading

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a. Structural support pallet

The fuel block loading equipment has been designed using an ease of removal philosophy to facilitate movement to other launch sites and test areas. All mechanisms and structures required to transfer the fuel block from the cask to the launch vehicle are contained within or are assembled to a single structural pallet.

The platform of the pallet base structure is used during cask installation to extend under the cask hoisted by the MST crane. The cask is lowered onto the platform and is rolled into a position on the pallet which is within the envelope of the umbilical tower.

All regulators and control valves supplying GN_2 to the drive motors, actuators, air bearings, purge systems, etc., of the fuel handling machine are installed as a system on the pallet.

An interfacing GN_2 service line is connected through a disconnect in the launch facility to a manifold on the pallet. A similar electrical disconnect on the pallet provides facility interface signal voltage to the various system controls, low voltage instrumentation lines, and electrical power consuming components and subsystems in the GHE.

b. Boom assembly

The fuel boom used to support the carriage carrying the fuel block is hinged to the static structure mounted on the pallet. The boom is of box construction using 1/4-inch structural plate. The approximate dimensions are 10 in. by 16 in. by 19 ft long. The weight of the boom is 900 pounds.

A motorized cable reel is mounted on the static structure to raise the boom to a vertical position for launch.

A negator spring-powered electrical and pneumatic line rewind reel is provided to take up the slack in the power and control lines to the fueling machine as the machine traverses the fueling boom. This rewind capability enhances the equipment operation and reliability by eliminating the possibility of twists and sharp bends in the electrical and pneumatic line and possible fouling with the traversing fueling machine.

Although the fuel block loading system is fully instrumented to indicate the position of the fuel block and all critical mechanical functions, a closed circuit television system is provided to observe the loading operation. A minimum of two TV cameras are required; one on the tower structure to monitor the cask operation and the removal or

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insertion of the fuel block, and the other on the fueling boom to monitor the remote connection of the FM with the L/V and RTG installation or removal of the fuel block.

Television monitoring of this system is not mandatory from an operational viewpoint but is included to give the operator a sense of control. In the operation of the system, TV cameras show the external functions of the fuel handling equipment so the operator can compare it with other indicators at the control console.

c. Fueling machine

The fueling machine is illustrated in Fig. III-33. It consists of the fueling machine support structure roller mounted and driven along the support boom, fueling machine traversing motor and drive, air bearing compliance mechanism, fueling probe, position locators, fuel block shroud and associated pneumatic and electrical leads.

The fueling machine traversing motor is used to drive the fueling machine to any required position on the fuel boom. The motor drive pinion gear engages a rack installed on the underside of the fuel boom.

The fueling machine support structure incorporates a series of rollers which travel in tracks mounted on each side of the fuel boom. Stops at both ends of the tracks prevent overtravel of the carriage.

Position locators, which are attached to the fueling probe assembly, align the fuel probe with the RTG or fuel cask during insertion and retraction of the fuel block. Stops within the fueling machine limit rotation of the locators about the probe vertical axis and assure their engagement with the position indices mounted on the payload fairing and fuel cask, as the fueling machine is moved into position over the cask or RTG.

When the fuel block is removed from the RTG or cask, it is drawn within a shroud. A GN_2 supply line attached to the shroud provides a continuous inert atmosphere around the fuel block during transfer operations. This is done to prevent propellant fumes or other vapors from coming into contact with the hot fuel block surface which would be a possible source of ignition.

Positioning springs in the air bearing compliance mechanism maintain the fuel probe in a centered position at any time when there are no external forces acting on the probe or fuel block.

The FM is supported by a compliance system which compensates for the relative motion between the RTG, when installed in the launch

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Fig. III-33. Air Bearing Compliance Mechanism

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vehicle and the fuel block loading mechanism supported from the umbilical tower. The compliance system permits five degrees of freedom of motion for the fuel block as indicated in Fig. III-33. Insertion and withdrawal of the fuel block from the cask or RTG is achieved by extending and retracting the fueling probe. Hydrostatic gas bearings have been selected to provide a minimum friction system for the compliance mechanism.

Alignment of the fuel probe/compliance device is accomplished by the engagement and centering of the position locators in the position indices. The position locators incorporate probes which are extended into recesses in the indices to cause the compliance device to follow the payload when relative movement between the L/V and fueling boom occurs.

Calculations show that a system air supply pressure of 91 psi at 120° F temperature and a flow rate of 0.0157 lb/sec will meet the system requirements. Nitrogen may also be used to operate the system with negligible change in operating pressure and flow rate. The physical dimensions of the bearing system are compatible with the remainder of the fuel loading concept.

Appendix FF provides a more complete description of the air bearing including supporting analysis.

A trimetric drawing of the mechanisms for handling the fuel block between the fuel cask and RTG is shown in Fig. III-34. The system is designed for vertical loading, and incorporates a pneumatically actuated ball-lock fueling probe to engage with, and lock onto, the fuel block for pickup and transfer operations. The probe also releases the gripper mechanism which locks the fuel block into either the RTG or the fuel cask. The unit has a fail-safe design so that if system pneumatic pressure is lost during fuel transfer, the fuel block cannot be released until pressure is restored, or, it will not allow the fuel block to be picked up until pressure is restored. This is accomplished by the plunger retraction spring which holds the plunger in the locked position at any time sufficient GN₂ pressure is not applied to cause the plunger to move to the release position.

Operation and control of the mechanism is accomplished by a remote operator, stationed in the launch control center (LCC) except for those functions which are automatic within the system, such as:

- (1) Automatic travel of the plunger to locked position whenever system pressure is released.
- (2) Automatic latching of the fuel block in the RTG or cask whenever the probe is released and withdrawn from the fuel block.

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Fig. III-34. Fuel Block Loading Concept Tower Boom/Top Loading

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- (3) Automatic locking of the fuel block gripper fingers in latched position as the fuel handling probe is withdrawn.
- (4) Automatic holddown of the fuel block, by the probe centering collar, in the cask or RTG until the probe is withdrawn and the block is locked in place.

Figure III-34 shows the fuel block and handling probe in their relative positions for engagement and disengagement. The fueling probe is in the pressurized condition with the probe plunger extended and the plunger recess groove aligned with the locking balls. This allows the balls to move to a position in which they do not protrude beyond the outside diameter of the probe so that the probe may be inserted or retracted from the fuel block center opening without interference.

A fuel block pickup operation is accomplished in the following procedures:

- (1) With the probe in the pressurized condition and the locking balls retracted, an inserter actuator (not shown) will be extended to move the probe into the fuel block.
- (2) The probe centering collar engages the conical surface of the fuel block and causes the probe to become centered. Translation of the probe to allow centering will be accomplished by action of the air bearing compliance device described in a preceding section.
- (3) The probe continues to move down so that the gripper finger locking disc actuator (see Detail A) causes the disc to be depressed, freeing the gripper fingers for rotation to disengage from the holddown ring (see Detail C).
- (4) Continued movement of the probe causes the conical surface to engage the gripper upper arms and actuates the fingers to rotate to the unlatched position.
- (5) Final travel of the probe into the fuel block aligns the locking balls with the annular detent ring in block assembly.
- (6) Pneumatic pressure is released from the probe plunger and the plunger retraction spring causes the plunger to retract. This causes the land surface to align with the balls and creates a positive lock with the balls engaged with the fuel block detent ring.

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- (7) The probe inserter actuator is retracted, lifting the fuel block clear of the surrounding structure of the RTG or cask so that it may be transferred as required.

The fuel block installation and handling is symmetrical; that is, identical procedurally and hardwarewise in the cask and RTG.

To install the fuel block into either the RTG or cask, the procedure is as follows:

- (1) The fuel block is locked onto the fueling probe, in the manner indicated prior, with the fuel probe in the unpressurized condition.
- (2) The fueling machine is positioned to engage the unit into which the fuel block is to be placed and is maintaining alignment through the compliance device as required.
- (3) The inserter actuator of the fuel block handling probe is extended so that the lower conical surface of the fuel block engages with and centers on the conical surface shown in Detail C.
- (4) Extension of the probe plunger causes the gripper finger disc actuator to depress the disc and allow the fuel block to seat in position without interference.
- (5) Pneumatic pressure is applied to the probe to cause the plunger to extend and align the recess with the balls, allowing them to move into the recessed position and clear the outside diameter of the probe.
- (6) The inserter actuator is actuated to raise the probe. As the probe is retracted, the fuel block is held in place in the RTG or cask by action of the centering collar spring which causes a force to be applied to the fuel block until the probe assembly clears the gripper fingers, allows them to rotate to the latched position, and the gripper locking disc to move into the lock position. A sensor in the RTG gives remote indication that the fuel block is locked in place.
- (7) Further travel of the probe allows the collar retaining guide pins to bottom out in their grooves (see Detail A) and lift the collar clear as the probe is raised.
- (8) The probe will be retracted to an elevation sufficient to clear all obstacles and allow it to be moved to the required stowage location.

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A switch is installed on the fueling probe to give remote indication of when the probe is in the locked condition, either with or without the fuel block attached. The design incorporates a safety feature which precludes the application of pneumatic pressure and inadvertently releases the fuel block.

The design of the fueling probe, fueling probe locking mechanism and RTG/cask holddown ring requires no rotational alignment about the vertical axis.

d. Fuel block cask

The cask shown in Fig. III-35 is provided in compliance with the applicable ICC licensing regulations and using AFSCM 80-5, Handbook of Instructions for Ground Equipment Designers; MIL-S-8512B, General Specifications for the Design of Special Support Equipment; and other specifications as a design guide. The cask is an integral part of the selected fueling concepts and its configuration is influenced by concept considerations. The cask is skid mounted and includes provisions for interfacing with specific UT platform locations. When installed on the UT, the cask is interfaced electrically with facility lines providing control and instrumentation monitoring remotely at the LCC. The cask on the tower is also connected to a purge gas (nitrogen supply).

The general configuration of the cask is that of an upright cylinder which, for transportation purposes, fastens to a skid and is enclosed by a woven wire cage for thermal protection. Access to the cask, after removal of the cage, is through the top which is provided with a hinged lid. The lid is opened and closed by an actuator connected to the pallet structure. The lid is secured to the cylindrical body of the cask after closing by bolts. Radiation leakage through the lid-to-body joint is prevented by a stepped concentric labyrinth having sloped vertical walls to facilitate centering and accommodating the lid rotation.

The cask cavity is sized to give one-quarter inch radial clearance between the walls and the fuel block. A hollow frustum of a cone-shaped bracket is fastened to the bottom of the cask (see Fig. III-34) so that the fuel block can lock to the cask.

The skid is of structural steel shapes welded together and provides lifting and tiedown rings and fork-lifting means. The top of the skid is structural steel plate.

Crane hook trunnions on the side of the cask provide the primary means of cask handling and are located above the center of gravity for stability purposes. The ends of the trunnions are threaded to accept tiedown fittings to be used during transport. An eye in the center of

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Fig. III-35. Transportation Cask

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the lid is used to lift the lid during assembly of the cask or the opening or closing of the lid under unusual or emergency conditions.

The woven wire cage prevents handling personnel from being burned by the surface of the cask. Cutouts in the wire mesh are provided for access to the trunnion fittings for transportation tiedowns. The cage is attached to the skid by quick release fasteners for easy removal.

At the launch site the cask is removed from the skid, hoisted to the fuel transfer level on the umbilical tower and fastened in place on a sliding platform. An electromechanical screw jack is then fastened between the arm on the lid and the pallet of the fueling equipment so that at the proper time the lid may be opened or closed remotely by actuating the mechanism. To remove the fuel block, the lid is opened and the position locators of the fuel handling machine are mated with the position indices high on the side of the cask. The fueling probe then enters the cavity, unlocks the fuel block from the fuel block centering cone and extracts it from the cask.

To preclude the entrance of combustible vapors at the launch site, a nitrogen purge gas connection is provided on the side of the cask and nitrogen gas is flowed continuously **after the opening of the cask lid.**

Uranium alloy (U-10% Mo) was selected for the shield. The purpose of the molybdenum is to dimensionally stabilize the uranium such that the cask does not grow due to heating of the cask by the fuel block. Corrosion, however, is not inhibited by this alloying and the uranium shield must be isolated from the atmosphere by a stainless steel shell.

The shield is to be cast in two separate parts, i. e. , the container and the lid. These parts are then finish machined and fitted with closely fitting stainless steel shells so they are completely enclosed. The seams of the shells will be welded so that the uranium is completely sealed off from the atmosphere. This encasement is used to prevent corrosion of the uranium alloy. The shell thickness will be approximately 1/4 inch. In the sizing of the clearances between the uranium and the shell, differences in thermal expansion, material growth due to temperature cycling and tolerances will be taken into account to achieve a minimum gap. All hinges, fasteners, brackets, supports, etc. , will be welded to the outer shell and will employ webs and stiffeners and other local structure as required to ensure sufficient strength.

In a cask designed without cooling fins and with surfaces treated to have high heat radiation emissivity, the outside wall temperature would be 400° to 500° F and the fuel capsule wall temperature would be less than 1000° F. These temperatures present no unusual or difficult problems to the fuel, the cask and its materials and components, cask transportation and cask handling.

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As indicated in other sections of this report, surface temperatures of the cask and fuel block during transfer have been scrutinized relative to potential fuel vapor ignition at the launch site. The fuel block hazard is controlled by the surrounding flow of a GN_2 purge. Several methods of cooling the external surfaces of the cask are considered in the event the surface temperature of the passively cooled cask is deemed objectionable. Additional finned structure, forced convection, a water jacket and a refrigeration cycle are alternates (in order of increasing effectiveness) with resultant penalties in cost, complexity, more difficult handling, etc.

2. MGE and OGE

The maintenance ground equipment and operating ground equipment for the control, instrumentation and checkout of the RTG/RB, CIR and nuclear GHE are installed in the launch control center (LCC) of the vertical integration building (VIB). This equipment is incorporated into standard racks and consoles (commercial equipment, Emcor, Ingersol Products, Division of Borg-Warner or equivalent) providing the visual displays, instrumentation and controls applicable to the launch site and system maintenance functions.

The displays, meters, indicators and panel general arrangement of the consoles and other MGE and OGE will be accomplished applying MIL-STD-803, "Human Engineering Design Criteria for Aerospace Systems and Equipment."

The MGE and OGE are grouped into two major areas: the equipment associated with the nuclear fueling operation; and the equipment associated with the checkout, monitor and instrumentation of the RTG/RB and CIR.

a. Nuclear GHE control and monitor console

The items of instrumentation readout or display required at the LCC for the fuel block loading system are as follows:

- (1) Position locator lock pin actuator switch
- (2) Position locator lock pin position indicator
- (3) Position locator cask alignment indicator switch
- (4) Fuel block lock pin position switches
- (5) Cask lid position (open or closed)

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- (6) Boom position (up and locked, down and locked)
 - (7) FHM position (stowed position and end of boom)
 - (8) Fuel block latch position in RTG
 - (9) RB nose cone latch position.
- (1) Position locator lock pin actuator switch

This switch is located on the lower cylindrical surface of both position locators. The purpose of the switch is to provide an electric signal to actuate the lock pin at the L/V interfacing end of the position locator. Actuation is initiated when the switches are contacted by the position index fittings on the launch vehicle fairing. Lock pins in the bottom of the position locators are acutated down into holes at the bottom of the position index fittings, locking the FM to the launch vehicle and fixing the FM in phase with the L/V.

- (2) Position locator lock pin position indicator (L/V)

Position indication consisting of lights will be provided on the nuclear AGE control and monitor console to indicate the full-up and full-down position of both position locator lock pins. Transfer of the fuel block into the RTG will not proceed until both lock indicator lights are on. Locking of the position locators in the index fittings will be verified by observation of zero relative motion between the position locators and the mating L/V position indices on the provided closed circuit TV.

- (3) Position locator cask alignment indicator switch

Switches will be located on the lower cylindrical surface of the position locators designed to contact the position indices on the cask. The switches will serve two functions: to stop the drive motor on the FM as soon as the position locators have bottomed in the cask position index fittings; and to activate an indicator light on the control console to indicate that the FM is aligned with the cask. Fuel block transfer can now be initiated.

- (4) Fuel block lock pin position switch

Position switches will be provided in the loading actuator to operate indicator lights on the operator's console at the LCC. The lights will indicate the full-up and locked or full-down and unlocked position of the fueling probe plunger which locks the fuel block to the loading actuator.

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(5) Cask lid position switch

Position switches will be provided on the cask lid actuator to operate lights on the fuel loading console to indicate that the cask lid is in the full-open or full-closed position.

(6) Boom position switch

Switches will be provided on the fuel loading boom to operate lights on the fuel loading console to indicate if the boom is in the full-up and locked or full-down and locked position.

(7) FM position switch

Two switches will be provided on the boom to operate lights on the fuel loading console indicating the position of the FM. One switch will be located at the extreme end of the boom toward the launch vehicle. The switch and a stop will be adjustable so that the FM can be positioned over the centerline of the launch vehicle. The light will indicate when this position is reached. The second switch will be located at a point between the cask and the boom hinge point. This switch will serve two functions. It will control the air valve to the air motor, stopping the FM in the stowage position (the cask lid can be opened when the carriage is in this position), and it will also operate a light on the operator's control console to indicate the position of the FM.

(8) Fuel block latch position switch

A switch will be provided in the RTG to operate a light on the fuel loading console which will indicate the position of the latch which locks the fuel block in the RTG.

(9) RB aft cap latch position indicator

A switch will be provided in the RB to operate a light on the operator's console to indicate the aft cap on the RB is closed and locked.

b. RTG/RB-CIR checkout equipment

The MGE and OGE checkout equipment for the RTG/RB-CIR payload will consist of that required to service airborne equipment, to verify the conditions of airborne systems, and to control airborne systems during ground operation.

The system is based on the telemetric transmission of a high percentage of system data from the vehicle, with the telemetry system providing data for evaluating system checkout prior to the launch countdown phase. Critical signals associated with airborne systems are to

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be selected on the basis of performance requirements for the launch phase and will be transferred to ground equipment through an electrical umbilical. Signals associated with the control of airborne systems from the ground will also be transferred through an electrical umbilical.

A block diagram of the RB checkout equipment required and its relation to airborne systems or equipment is included in Fig. III-36.

(1) Space ground link subsystem (SGLS)

The checkout of the spacecraft payload will utilize the SGLS ground equipment to support the airborne telemetry (T/M) system. The support the system provides will include the collection of all data transmitted by T/M, the transfer of data to a format for further transfer to a data analysis system and the checking out of the operation of the T/M system. The SGLS ground equipment is considered to be a customer-supplied facility at the launch site.

(2) Go no-go analyzer

The checkout equipment will include a system to operate from data supplied by the SGLS which will analyze the data and provide an indication of the satisfactory (go) or unsatisfactory (no-go) level of each of the signals transferred from the spacecraft by telemetry. The equipment will include:

- (1) A buffer unit to accept the digitized output from the SGLS and interface it with the analyzer equipment.
- (2) A data analyzer or computer to operate on input data from the SGLS for comparison with programmed limits supplied to the analyzer to determine the acceptability or nonacceptability of data.
- (3) A printout or indication of the go no-go nature of data.

The system will provide an automatic analysis of airborne system measurements and the display of these data. However, it will not be an automatic checkout system providing stimuli to the vehicle. Stimuli will be provided through the checkout console for those items requiring stimulus other than the ambient conditions of the checkout phase.

(3) Launch and checkout control console

The console will control and monitor airborne and ground systems operation during the checkout and launch phases. The console will include controls to operate the ground electrical power, the air conditioning equipment, the selection of ground power for airborne systems,

Fig. 111-36 AIRBORNE SYSTEMS CHECKOUT EQUIPMENT

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spacecraft latches and other electrical controls to spacecraft systems requiring ground activation during checkout and launch phases. Monitors will be provided for those spacecraft systems with outputs such as fairing and RTG latches, ordnance safe indications and others. The status of the systems necessary for the launch sequence will be provided. Circuitry, either to convert inputs to a status that can be mounted or displayed or to summate inputs, will be included in the console.

(4) C-band checkout set

The equipment to check out the C-band communication equipment will include the antenna connector, the cabling from the antenna connector to the test set and a C-band test set. The antenna connector will be a hat designed to mate the spacecraft around the antenna which will provide a controlled environment for RF transmission. The cabling from the hat to the test set will be RF cable with a minimum of disconnection points. The C-band test set will be a rack of commercial-type test equipment associated with C-band including such items as: pulse generator, signal generator, power meter, frequency converter, frequency counter, high gain oscilloscope and attenuators and connections.

(5) Alignment kit

The installation of the propulsion system rocket motors will require their precise optical alignment on the CIR. The alignment kit will provide optical sighting equipment, pads and monitoring equipment for installation of the sighting equipment, and tools for aligning and adjusting the installation of rocket motors.

(6) Ordnance checkout set

The ordnance checkout set will be made up of an ordnance simulator and stray voltage detector. The ordnance simulator will be made up of devices to replace and simulate each ordnance item, a control unit to provide stimuli to the ordnance circuits and indications of circuit continuity, and the necessary adapter cabling to interconnect the control unit, simulators and the spacecraft cabling. The stray voltage detector will be an adaptation of a commercial unit with the necessary adapter cables (if different from those mentioned above) to permit determination of the absence of stray voltages prior to the installation of ordnance.

(7) RTG/RB-CIR ground system connections

The AGE will include items required to interface the spacecraft and the ground equipment. Those items of connecting equipment, not

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multipurpose in nature and servicing a single item of AGE such as air conditioning and pneumatic service units that during test are connected directly to spacecraft system, and ordnance items are not a part of this item but are to be part of the single item they serve. Included are the electrical umbilical and service connections to the spacecraft, the umbilical and service cables, junction boxes for accepting umbilical and service cable wiring and for distribution to the applicable ground equipment (three junction boxes will be required), and the interconnecting cabling between the junction boxes and between the junction boxes and the ground equipment.

In addition, the cabling connecting the launch control console and the items supplying inputs to it or controlled by it (such as the 28-volt dc power supply, the air conditioner controls, etc.) are to be included in this item.

(8) Air conditioner

An air conditioning unit, including controls for remote operation from the launch control console, and the necessary disconnects and ducting to interface it with the spacecraft will be required. The capacity of the unit will be such that the necessary temperature environment of the spacecraft can be maintained at the launch site for the duration of checkout and launch testing. The air conditioning equipment less the necessary controls are considered GFE.

(9) Pneumatic pressure

A GFE source of pressure for the pneumatic latching cylinders will be required. Control, regulation and monitoring of this pressure will be from the **nuclear GHE control console**.

(10) Power supply

A 28-volt dc regulated power supply with controls for local or remote operation from the launch control console will be required. The controls associated with the power supply, but most likely located in the launch console, that are required to apply to and transfer power from the spacecraft will be a part of the **AGE**.

(11) Communications

A communications system of the interphone type is required to link the launch console, the go no-go analyzer and stations at the spacecraft and other remote locations. It is anticipated that ties to launch site intercommunication equipment will be available.

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3. Description of Fueling Operation

a. Preparation for loading

The conditions necessary to a nuclear fueling operation within the coordinated system are as follows:

- (1) L/V checkout and verification is complete.
- (2) Chemical fueling of the L/V is complete.
- (3) The fueling boom and associated equipment have been installed on the umbilical tower.
- (4) The nuclear fuel block in the cask is hoisted and installed on the designated UT platform using the bridge crane on the MST.
- (5) The cask signal, control monitor and purge interface connections have been accomplished and these activities verified by the nuclear GHE operator in the LCC.

b. Simulated loading and system checkout

The nuclear fueling boom is actuated from its retracted position to the horizontal. The fairing and RB caps are signaled open by the nuclear GHE operator in the LCC. Air conditioning of the fairing space is continuous, exhausting air through the fairing cap opening. Air to the air bearing compliance mechanism is provided activating the system. The compliance mechanism is maintained in its midposition by included centering springs. The remote GHE operator commands the FM run-out on the positioned boom and engaged with the mating L/V indices. The FM is held against the indices by a continued application of motive power (at the stall torque of the air motor), the indices lock pins are engaged and the air motor shut down. This simulated fueling operation is observed by the remote operator through the boom-installed television cameras and verified by personnel at the payload level on the MST. Operation of the purge gas to the fuel block shroud and the operation of all lock indicators are verified by the remote operator and on-site observers. The FM is retracted to the full retract position by the remote operator, and the fairing and RB caps are remotely closed by him. Operation of the mechanisms is reviewed by the remote operator and the readiness state of all systems verified.

c. Fuel block loading

All personnel are evacuated from the MST and the MST retracted. In the countdown, the count is approximately T-80. At T-20, all personnel are retired to an area safe from nuclear hazards and a 30-minute

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programmed hold is effected for the actual nuclear fueling. The cask lid is remotely opened by the LCC operator. The fuel machine is advanced to a position over the cask (a programmed location) and the pickup probe is extended into and locked onto the fuel block. Positive attachment of the FM to the fuel block is verified by the operator in the LCC at the nuclear AGE control and monitor console. Purge gas (GN₂) flow to the FM shroud is started as soon as the FM is positioned over the loaded cask.

The operator opens the L/V fairing and RB cap on signal through the L/V umbilical. The status of the system is checked at the remote console and verified by a television monitor. On ascertaining the system is go, the operator withdraws the fuel block from the cask, activates the compliance mechanism, traverses the FM the full length of the boom and interfaces the FM position locators with the L/V indices. The FM and RTG are moving in phase with each other (relative displacement is zero). The condition of the system is verified at this point and the fuel block probe is extended, inserting the fuel block into the RTG. The step-by-step procedure for inserting the fuel block into the RTG is described in a preceding section. The lockup of the block is verified and the pickup probe disconnected from the block and withdrawn. After a system verification, the FM is fully retracted within the envelope of the UT structure. The fairing and RB cap are remotely closed on signal and the RTG placed on monitor through the L/V umbilical to the launch control console in the LCC.

The countdown picks up at T-20, and at approximately T-10 (or as late as possible in the count) the boom retraction system is actuated and the boom fully retracted within the structural limits of the umbilical mast.

d. Fuel block unloading

Removal of the nuclear fuel block from the RTG in the event of a launch abort is the reverse of the loading sequence. The transfer boom is positioned to the horizontal, the FM is run out to a position approximately six feet from the L/V, and the operation is maintained by remote viewing television cameras. The compliance system is activated. On command, the FM is advanced, indexing on the L/V and the fuel block probe is engaged to the nuclear fuel block. The condition of the interface locks is verified and the block is withdrawn from the RTG and installed in the cask. The cask lid is remotely closed, the fairing and RB cap are closed and the nuclear AGE is retired to its static storage position. The MST is rolled to its service position surrounding the UT, UM, L/V and transporter, and service to the L/V is accomplished in the usual manner.

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e. System operation to launch

The fuel block loading operation is performed remotely within the last 50 minutes before launch. The loading sequence is as follows:

- (1) Verify that FM position light shows FM in stowed position.
- (2) Press cask lid open switch and verify that cask lid open light is on.
- (3) Actuate FM drive motor in reverse to position FM over cask.
- (4) Verify that both locator probe indicator lights are lighted.
- (5) Actuate fueling probe, inserting probe into fuel block.
- (6) Actuate the fuel block pickup lock.
- (7) Verify that the pickup lock light is on.
- (8) Switch nitrogen purge on in the shroud.
- (9) Actuate fueling probe to lift fuel block into the shroud.
- (10) Actuate FM drive motor in forward direction to drive FM along boom.
- (11) Verify that position locators are engaged in L/V position index fittings when position locator light is lit.
- (12) Position locator lock pins are automatically actuated as soon as the position locators are bottomed in the position index fittings, depressing the position locator switches. Verify that the position locator lights and the position locator lock pin lights are lit.
- (13) Actuate the fueling probe, lowering the fuel block into the RTG.
- (14) Verify that the RTG fuel block latch light indicates unlatched.
- (15) Actuate fuel block pickup lock and verify that the unlocked indicator light is on.
- (16) Actuate the fueling probe to retract the probe from the fuel block.
- (17) Verify that the RTG fuel block latch indicator light is on.

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- (18) Retract position locator lock pins and verify that lock pin released light indicates.
- (19) Actuate FM drive motor in reverse and stop in stowed position.
- (20) Retract transfer boom to a vertical position by actuating winch. Verify that up and locked lights operate correctly.
- (21) Monitor RTG on console.

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IV. CRONUS INTEGRATION

Chapter III presented the selected design for CRONUS, and this chapter discusses the integration of CRONUS into the Titan launch vehicle and Nimbus spacecraft, and the launch complex.

A. TITAN AND NIMBUS COMPATIBILITY

CRONUS imposes only a few requirements on the launch vehicle and spacecraft. The problem of compatibility, as it applies to Titan IIIA, B, or C and to Nimbus can be discussed in six different areas:

- (1) Trajectory
- (2) Fairing
- (3) Communications
- (4) Electric power conditioning
- (5) Structural
- (6) Thermal.

1. Trajectory Compatibility

All Titan III launch vehicles can place large payloads into orbit at an altitude of 600 nautical miles. Insofar as the operation of the airborne system is concerned, there would be no material difference between any Titan family booster. Insofar as safety is concerned, Titan IIIC has a longer downrange distance at injection and a lower angle of attack than the other two boosters. Consequently, that vehicle is the "worst" launch vehicle with regard to impacting of the RTG/RB on Africa or the Antarctic. Titan IIIA and IIIB are closer to the shores of the United States when the fairing is released. However, it is felt that these differences between the Titan family trajectories are not critical to either the safety of CRONUS, or the reentry environment for which CRONUS is designed.

The mission profile after attaining near orbital velocities is very similar to the Titan IIIC profile. The basic failure rates, therefore, should be the same for the various Titan III launch vehicles. Consequently, the launch vehicle reliability concern for CRONUS is essentially a function of the time required to go from near orbital velocities to final injection. This time is almost the same for all launch vehicles (actually, Titan IIIC has the longest time). Consequently, there is no real difference in the L/V reliability compatibility of the different Titan III vehicles.

STANDARD AERODYNAMIC FAIRING--TITAN IIIA AND C
WITH MODIFICATION FOR 250-WATT CRONUS

NOTE: ALL DIMENSIONS IN INCHES

Fig. IV-1. Standard Aerodynamic Fairing--Titan IIIA and C
with Modification for 250-Watt CRONUS

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As shown in Fig. IV-1, the fairing has a separation plane running along the centerline from the bottom of the fairing to a point on the nose cap and then to the side of the nose cap at an oblique angle. The nose cap will be jettisoned with one part of the fairing when separation occurs. An explosive charge contained within a bellows is used for separation. Modifications to the fairing for fuel block loading access require an additional separation plane horizontally across the fairing at a point approximately 51 inches from the top. The 51-inch section of nose cap will be hinged on one side and opened by a pneumatic actuator. A number of stud-type shear pins would be used between the hinged nose cap and fairing interface. Approximately two remote tension latches would be required to hold the nose cap onto the interface structure. Several methods have been considered for separation of the fairing in space. One method would require a sequential operation ejecting the 51-inch section forward and to the side by using asymmetrical charges. The remaining two halves would be separated in the usual manner.

The most attractive method of separation is to maintain the existing configuration. The addition of a horizontal cut across the explosive vertical separation plane for RTG access requires a separate igniter in the nose cap portion of the vertical explosive separation plane. The oblique angle separation plane, on the nose cap remains as is in the existing fairing configuration. There are no fasteners through the oblique angle separation plane, and separation will occur exactly as it is at present. The nose cap portion above the oblique angle separation and from the nose cap hinge to the explosive separation plane on the centerline will separate with one-half of the fairing. The remaining portion of the nose cap below the oblique angle separation plane which is locked to the lower fairing section will separate with the other half of the fairing.

The fairing on Titan IIIB may represent a different problem. First of all, the Agena stage (which replaces the 10-foot diameter transtage of Titan IIIA) has a 5-foot, 5-inch diameter. Consequently, if this 65-inch diameter fairing is used, there will be a tight fit between the airborne system and the inside of the fairing. Some clearance must be provided but the exact amount depends upon the exact elastic properties of the L/V and the fairing. Most likely, the outer diameter of the airborne system must be reduced a few inches to be compatible with a 65-inch diameter fairing. Although this does not incur problems concerning the concept, a few minor changes must be made to the solid motor attachment and separation rail length.

Some changes must be made to a 65-inch fairing to accommodate the selected ground handling concept. Similar changes must be made for an Agena fairing. Since some modifications would be necessary, minor structural bulges in the fairing in order to fit it around the airborne system are entirely feasible.

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If a standard Titan III fairing is used on Titan IIIB, then the fairing compatibility problems are the same as previously discussed.

3. Communications

The following signals must be transmitted from the launch vehicle to CIR-RTG/RB during flight:

<u>Signal</u>	<u>Function</u>
Liftoff	Activates fairing link pin puller
Fairing release	Arms CIR system
Abort (includes malfunction, incipient destruct, or ground command (backup))	Initiate separation sequence, turn on beacon
Final orbit	Disarm CIR, turn off beacon, telemetry

These signals will be transmitted through the CIR/LV electrical connection. The conditioned power generated by the RTG/RB will be transmitted, through the electrical connection, to the spacecraft. This is a squib fired connector. The fixed side is mounted in the CIR structure just above the lower ring frame and near one side of the separation rocket mount. The jettisoned side is mounted on a bracket extending above and attached to the spacecraft. At the separation signal, the squib fires, driving the connector apart and allowing the CIR-RTG/RB to spin and separate.

a. Malfunction detection system (MDS)

There is a major interface between the CIR-RTG/RB, S/C and launch vehicle in malfunction detection. It is a part of the CRONUS concept that the launch system (i. e., everything below the CIR-RTG/RB) indicates when a flight critical failure has occurred, when an explosion or structural malfunction may be about to occur (i. e., over and under pressures in the tanks), or when a ground-based destruct command is about to be executed.

Particular malfunctions which are of interest to CRONUS are the pitch and yaw rates (rates on the order of 30 deg/sec are acceptable so present MDS limits are more than adequate) and any marked deviation of the vehicle attitude from the planned attitude (an angular deviation of five degrees is tolerable). This latter constraint is needed only during

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the latter phases of the transtage flight, during coast, and during the second transtage burn. Flight conditions prior to possible land impact are essentially independent of vehicle attitude. Strong L/V deviations are monitored by range safety and, also, by a backup ground guidance system.

Both the man-rated Titan IIIC and the similar Titan IIIA launch vehicles have such a malfunction detection system. Some potential modifications which may be required in the presently available MDS have been described. In this regard, both Titans IIIA and IIIC are similar.

Titan IIIB presents a different problem. Available information indicates this vehicle is presently assigned only for specific missions. Consequently, no MDS has been incorporated. The addition of this system (either in the booster, spacecraft or possibly in CRONUS) is necessary for Titan IIIB to be compatible with the safety requirements.

4. Electric Power Conditioning

Energy storage is necessary when the power demanded by a spacecraft load exceeds the capability of the onboard generator. The load profile for Nimbus B for each 107-minute orbit includes the following features:

	<u>(watts)</u>	<u>(sec)</u>
Average load	235	
Minimum load	213	
Maximum load	390	30
Followed by	290	5
Secondary peak load	337	3

Although considered as a separate supplemental subsystem to the CIR-RTG/RB systems, an energy storage and power conditioning (ES/PC) subsystem has been studied. Appendix V gives the findings of this study.

5. Structural Interfaces

The longitudinal and lateral loads during ascent are not critical for CRONUS. Consequently, there would be no difference between the various members of the Titan III family with respect to loads.

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The acoustic environment is most severe for Titan IIIC. Even for this configuration, acoustics do not appear to be any problem. The vibration and shock environment, which is a function of the launch vehicle, is also a function of the explicit spacecraft upon which CRONUS is placed. It is felt that suitable mounting techniques can easily be made for any launch vehicle. In fact, the dynamic effect of the air-borne system on the launch vehicle may be more serious than the L/V dynamic effect on the airborne system since the payload stiffness is often an important factor in the launch vehicle autopilot loop.

The reference spacecraft designated for this study, Nimbus, has been used in a very liberal sense. That is, since it is obvious that major structural modifications would be required to enable it to support the additional weight of the CIR-RTG/RB, other modifications would also be possible. With this in mind, the command antenna with its ground plane was omitted to provide a mounting interface between the spacecraft and the CIR-RTG/RB. No effort has been expended on defining a new Nimbus command antenna or in relocating and resizing the Nimbus attitude control system thrusters. The solar paddles of Nimbus were also removed.

The structural interface is the Nimbus control section structure. Three separation tracks are attached to the top of this structure and their four supporting struts are attached to it at other locations.

There is also a structural interface between the launch vehicle and the CIR-RTG/RB, that is, the link that connects the hinged nose cap of the launch fairing to the hinged aft cap of the RTG/RB. This strut is released from the RTG/RB aft cap, along with its release actuator at the liftoff signal. An attachment lug is provided on the aft cap for this strut.

6. Thermal Interface

Nimbus does not have a direct view of the RTG/RB radiator. The temperature of the CIR structure is expected to be considerably less than 200° F due to the effective insulation properties of the pads on the attachment fingers, the insulation ring in the RTG/RB, and the relatively low heat production in the power conditioner and other CIR components.

On the ground, air cooling is provided. During the initial portion of the ascent, however, there might be an internal temperature problem for Nimbus if there were no beryllium heat sink on the fairing. With this device, however, the reradiation from the fairing should be negligible.

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B. LAUNCH COMPLEX (ETR) INTEGRATION

1. Modifications

The modifications required at the Eastern Test Range (ETR) Titan III complex to facilitate the installation of the AGE are as follows:

- (1) Removal of the umbilical tower movable platform at the payload level to provide access for the installation of the fuel block loading equipment.
- (2) Addition of holddown provisions for the pallet to attach to the fixed structure of the umbilical tower.
- (3) Reenforcement of the umbilical tower structure to support the nuclear fueling equipment installation.
- (4) Removal of the umbilical tower movable platform and a section of the fixed platform at the level above the boom installation to clear the boom retraction mechanism.
- (5) Removal of the movable platform extensions above the boom installation to clear the boom when in the retracted position.
- (6) Installation of pneumatic, electrical, instrumentation and communication lines to the fuel handling installation for remote actuation, control and monitoring of the system.

The vertical integration building (VIB) will require the installation of console(s) for remote control and monitoring of the nuclear fuel loading system and for checking out and monitoring the RTG/RB-CIR.

2. Services

Services as follows are required at the payload level. These services now exist at the launch site; the modifications consist of routing these services to the point of use:

- (1) Five-kilowatt, 220-volt ac, 3-phase electrical power for operation of the boom retraction device.
- (2) Nitrogen gas, 3000 psi at 2 lb/sec for operation of locks, air bearing, air motors, actuators, purging, etc., as applicable in the AGE concept.
- (3) Electrical power (28 volts, dc) for instrumentation readout and position indication.

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- (4) Air conditioning as determined for environmental control of the system in the L/V fairing before and after fuel block installation. At present, 70 lb/min of 60° to 65° F and 42 to 50% relative humidity conditioned air is available at the site.

3. Service Routing

Additions to the existing system are required for routing of control and monitor signals between the launch site and the LCC in the VIB. Cable and other service and signal routes between sites at the complex, i. e., between the launch site and the VIB, are through underground conduit and it is expected that for purposes of the nuclear AGE, C/O and monitor functions, routing of the necessary additional cable through existing conduits would be accomplished. Within the VIB, cabling and routing are accomplished by equipment interconnecting cables routed through cable trays under the floor. Sufficient excess volume in the cable trays is indicated.

4. Explosion Proofing

Although the enclosure "Payload Environment" to the ground rules letter addendum to the Contract indicates that N_2O_4 and UDMH propellant vapors will not be experienced within the payload fairing and, therefore, by implication, not experienced at the launch site, Martin Company experience indicates that under certain environmental ambient temperature conditions propellant vapors may be deposited on the launch site. The Martin Company has taken cognizance of this possible condition in the configuration of the nuclear handling equipment and has included within the design a mode of explosion proofing the exposed fuel block during the transfer from the cask to the RTG.

The propellant fuel used in the Titan III presents an explosion hazard when a small percentage of the vapors are mixed with air. For this reason a 100-foot exclusion radius has been established around the launch vehicle for all uncontained spark-producing or hot wire devices (Ref. IV-1). The fuel used in the Titan III is a mixture of unsymmetrical dimethylhydrazine (UDMH) and hydrazine. Each has different flammability properties. UDMH is flammable in air over a wide range of concentrations (2.4 to 95% by volume (Ref. IV-2)). The auto-ignition temperature of UDMH in air is 482° F (Ref. IV-2). UDMH forms an explosive mixture with air at temperatures above 70° F and requires only a spark or hot wire to produce combustion. Hydrazine vapor and air burn readily at temperatures above 104° F when a spark or hot wire is introduced (Ref. IV-2).

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It is anticipated that the fuel block cask with the fuel block will be positioned on the umbilical tower before the launch vehicle is fueled. The temperature of the fuel block in air is in excess of 700° F and will be a source of ignition if fuel vapors are allowed to come into contact with the hot fuel block during the nuclear fueling operation. A nitrogen purge will be provided in the cask during the fueling operation when the cask lid is opened.

The fuel block is purged with GN₂ during the transfer from the cask to the RTG or from the RTG to the cask.

The fueling machine is equipped with an inverted can or shroud mounted on the compliance device and surrounding the fuel probe. (Figure III-3) Nitrogen will be flooded into the shroud during the transfer operation, thus enveloping the fuel block with an inert gas. During the transfer from the cask to the fueling machine, the shroud will be positioned directly over the cask and nitrogen will be fed into the shroud and the cask. Nitrogen will continue to be fed into the shroud while the fuel block is transported along the boom to the RTG. The shroud will be positioned directly over the RTG and nitrogen will be fed into the RTG as well as into the shroud. The fuel block will be lowered into the RTG while the area is being flooded with nitrogen. The air conditioning system will purge the propellant vapors and maintain a positive pressure inside the payload fairing, eliminating propellant vapors after the nose caps have been closed.

5. Countdown Considerations

The Titan IIIC countdown requires personnel on the launch stand until T-20 minutes. A hold in the countdown of 30 minutes will be required at the T-20 minute point for fuel block loading.

Items in the Titan IIIC countdown which are performed after T-60 (T-60 is chosen for consideration here because it is believed the fuel block loading within the last hour prior to launch is desirable) minutes and require personnel on the launch pad are as follows (Ref. IV-3):

- (1) T-60--Secure L/V door L-7 and L-9
- (2) T-60--Remove Stage II oxidizer vent line
- (3) T-55--Start VECOS (vehicle checkout system) run
- (4) T-50--Complete camera installation and adjustment
- (5) T-50--Flyaway cable in flight position

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- (6) T-35--Verify all umbilical lanyards connected
- (7) T-35--Complete the VECOS run
- (8) T-35--Verify MST locked in park position
- (9) T-30--Clear all but arming personnel from launch pad
- (10) T-30--Install solid rocket motor arming plug
- (11) T-20--All personnel cleared from launch pad.

Some of these operations may be performed prior to T-60 minutes; however, the majority of the operations cannot be rescheduled. Discussion with site personnel led to the conclusion that the countdown remains intact and that a hold be scheduled at T-20 minutes for fuel block loading. After fuel block loading, the countdown picks up at T-20.

6. Relative Motion of Launch Vehicle and Umbilical Tower

In any nuclear fuel handling scheme involving launch site structures, i.e., umbilical towers, mobile service towers, or any other structure or moving tower and the L/V, compensation for the relative motion of these structures is required because of the influence of this motion on the fuel transfer operation.

Relative motion of the UT and L/V in the selected concept is accounted for by the kinematic freedoms designed into the compliance mechanism of the fueling handling machine.

A relative motion of ± 3 inches in the station plane of the launch vehicle has been provided for in the configuration of the selected concept. This relative displacement is an approximation of that realized between the umbilical tower and launch vehicle in the design criteria wind conditions of 46 mph at 30-foot elevation above the ground. Analytical assumptions and test data refinement can reduce the estimate of the total excursion, however, not sufficiently to eliminate the relative motion of the site equipment and vehicle from consideration.

Of secondary interest, and associated with the considerations of relative motion, is the frequency of motion between the structures in question. Test data and criteria are available defining the natural frequency of the launch vehicle in a 46-mph design wind condition. Conservative assumptions have been made on the oscillatory motion of the umbilical tower under the same wind condition for purposes of this study.

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The relative displacement and frequency between the umbilical tower are approximated as ± 3 inches in the station plane at approximately 0.5 cps (Appendix OO). In subsequent phases of the program, it is planned to determine the relative motion criteria as applicable to the limiting values of this motion so that the resulting design is not penalized.

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V. SAFETY ANALYSIS

A. OBJECTIVES

1. Safety Criteria

CRONUS is designed to provide intact reentry of the heat source. Therefore, direct radiation will be the most probable nuclear hazard. In the event of accidents, less probable hazards such as inhalation and ingestion of radioactive debris can be created. National and international agencies have established recommended and/or permissible levels of radiation which they feel will not jeopardize the health of a human being. It is the intent of the nuclear industry to design systems which do not normally allow humans to receive more than the established radiation limits. However, the fact that accidents can occur which may cause radiation injury to people creates a difficulty in assessing the overall safety of any nuclear system.

The criteria for assessing the nuclear safety of CRONUS have been defined in the contract statement of work as follows:

- (1) "The probability that radiation injury of people within a small geographical area will result from failure of the controlled reentry system shall be no greater than 1 in 10^6 ."
- (2) "The probability of release of the radioactive fuel to the atmosphere so that it is widely dispersed shall be less than 1 in 10^7 ."
- (3) "The probability of release of radioactive fuel due to ground handling equipment failures shall be less than 10^{-6} ."

Basically, these criteria state that any intact reentry will not cause radiation injury to people with a probability greater than 10^{-6} , that reentry burnup will not occur with a probability greater than 10^{-7} , and that the probability of fuel release during all terrestrial handling will not exceed 10^{-6} .

The criteria stated can be broken into two categories: mechanical failure and biological. The ability to meet mechanical failure criteria can be determined and evaluated by reliability analyses of systems, components and materials failure probabilities. These mechanical failure criteria are independent of the biological, or dose dependent, consequences to the general population resulting from an accident or system failure. The dose dependent criterion requires a probabilistic analysis of a given number of people receiving a given dose. The nuclear safety analysis for CRONUS has been based on consideration of both the mechanical and biological categories.

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2. Analytical Approach

The sequence of considerations, definitions and analyses leading up to a final probabilistic evaluation is as follows:

- (1) Establishment of a circumstance (transportation of fuel, launch vehicle loading, flight, orbit or reentry conditions).
- (2) Postulation of conceivable accidents from Item 1 (collision, explosion, fire, thrust cutoff, trajectory misalignment, etc.).
- (3) Determination of accident occurrence probability for Item 2.
- (4) Determination of fuel disposition from Item 3 (capsule melt-down, capsule rupture, on land, in water, ground burial, etc.).
- (5) Determination of hazard from Item 4 (external radiation, inhalation, ingestion).
- (6) Determination of dose resulting from Item 5 (rem/individual).
- (7) Determination of the probable number of receptors receiving dose from Item 6.
- (8) Multiplication of items to yield final PDN (probability, dose, number of people) values for each accident and/or hazard.

Some of the considerations listed lend themselves quite well to a numerical analysis, such as the determination of a dose rate or the probability of a vehicle failure during a launch trajectory. Other considerations are not so discrete. For instance, although one may know (from reliability analysis or actual flight data) the probability of failure for a third stage ignition, the types of ignition failures which lead to either reentry burnup or intact reentry are much more difficult to define quantitatively. A simple premature burnout or ignition failure will not jeopardize the orientation or structural integrity of the RTG/RB, but thrust termination by an in-flight explosion could yield an orientation of high tumble rate or structural damage which could result in partial or total burnup.

A PDN analysis, used to evaluate the biological hazard, considers an entire concept or design in terms of the probability of a number of people receiving any given dose from any conceivable operation or accident postulated for the system. In this type of evaluation, the limits

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recommended by various agencies such as the International Committee for Radiation Protection (ICRP) are used as a reference to evaluate the risk of putting a nuclear system into operation.

3. Mission Requirements

The mission requirement for CRONUS calls for direct injection into an elliptical, coasting orbit with a final burn to circularize at the design altitude. It is desirable to have system recall continuously throughout ascent to final orbit and up to the end of the five-year mission lifetime. However, throughout this time period, the safety criteria previously defined would have to be met.

As shown in Table V-1, the 10^{-6} criterion for radiation injury from strontium can be met if the controlled intact reentry system (CIR) is about 83% reliable. This reliability is considered attainable within the present state of the art for the limited time in space required to ascertain correct orbit attainment. It is not developed for a five-year recall capability. To establish a strontium-90 reentry system with a five-year reliability of no less than 83% is not felt to be feasible within the constraints of a two or three year program. As a result of these considerations, it is recommended that the strontium system be immediately recalled if the achieved orbit is unsafe.

The obvious question which arises is, "What is considered an unsafe orbit?"

If a nominal 600-mile circular orbit is achieved, the fuel will naturally decay to a safe level during the orbit lifetime and the CIR system will not be required. If a nominal 600-mile orbit is not achieved and the predicted orbital lifetime is less than a year, the fuel will not have decayed to a safe level and the CIR system will be required to avoid a potential burnup or random land deposition. It will accomplish this by directing reentry impact to occur in deep ocean areas. At some transitional orbit lifetime, the philosophy of recall yields to a no action decision which results in a natural fuel decay while in orbit. The transitional orbit is not definable at this time but can be identified by subsequent studies and redefinition of governing safety criteria.

If the burnup criterion (10^{-7}) is looked at with the intent to identify the transitional orbit, it is found that any decision to leave the system in orbit (be it 10, 50 or 500 years) will probably result in a burnup reentry. Burnup could result from heat shield degradation through outgassing, radiation damage, micrometeor impact or other unknown phenomena which can occur over the years. If the criterion of radiation injury (10^{-6}) is used in hopes of identifying the transitional orbit it may be that if recall is not activated early after launch (within days)

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TABLE V-1
Hazard Probability Summary

	Event Probability	Event Probability	Product	Event Probability	Product	Event Probability	Product	Conclusion
Transportation Criterion	Highway accident (2.3×10^{-5})	Impact (1.0)	2.3×10^{-5}	Capsule rupture (0.01)	2.3×10^{-7}			Meets 10^{-6} criterion
		Fire (0.75)	1.7×10^{-5}	Meltdown (0.01)	1.7×10^{-7}			
		Ejection from vehicle (0.25)	5.7×10^{-6}	Buried with meltdown (0.05)	2.8×10^{-7}			
	Railroad accident (1.5×10^{-6})	Impact (1.0)	1.5×10^{-6}	Capsule rupture (0.01)	1.5×10^{-8}			Meets 10^{-6} criterion
		Fire (0.75)	1.12×10^{-6}	Meltdown (0.01)	1.2×10^{-8}			
		Ejection from vehicle (0.25)	3.7×10^{-7}	Buried with meltdown (0.05)	1.9×10^{-8}			
	Air accident (1.59×10^{-4})	Impact (1.0)	1.6×10^{-4}	Capsule rupture (0.01)	1.6×10^{-6}			Does not meet 10^{-6} criterion
		Fire (0.75)	1.2×10^{-4}	Meltdown (0.01)	1.2×10^{-6}			
		Ejection from vehicle (0.50)	8×10^{-5}	Buried with meltdown (0.10)	8×10^{-6}			
Injurious Dose Criterion	Low orbit (4.6×10^{-3})	Reentry control failure (0.17)	7.8×10^{-4}	Populated land impact (0.26)	2.03×10^{-4}	Injurious dose Sr-90 (4.7×10^{-3})	9.6×10^{-7}	Meets 10^{-6} criterion
						Injurious dose Pu-238 (7.3×10^{-8})	1.5×10^{-11}	
Burnup Criterion	Burnup altitude (8.2×10^{-3})	Reentry control failure (0.17)	1.4×10^{-3}	Restriction for conditions which yield burnup of fuel (7.1×10^{-5})			1×10^{-7}	To be de- termined in Phase I studies

Failure probability - (1 - reliability)

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the reliability of the CIR system may degrade to a value which jeopardizes the injury criterion. It appears that the present criteria are insufficient to completely resolve the mission safety requirements.

Although the CRONUS statement of work has defined acceptable risk levels, later contracts and nuclear safety reports in the industry (Refs. V-1 and V-2) have further defined acceptable risks exclusively in terms of the ultimate biological consequence to the general public. For example, one may define an acceptable risk for some or all accidents as being no greater than 1×10^{-4} for a dose of 100 rem, i.e., the probability of one person receiving 100 rem shall be no greater than 1×10^{-4} , the probability of 10 persons receiving 100 rem shall be no greater than 1×10^{-5} , etc. One may also wish to define criteria in terms of the probability of receiving a given dose rate instead of a total integrated dose. For these criteria a transitional orbit can be identified since the ultimate hazard is defined in terms of dose or dose rate.

In view of the later developments in the area of risk definition, it seems appropriate to recommend that additional studies be performed to establish more comprehensive criteria for CRONUS. These studies should also establish standard calculational techniques for determining the risk probability. An example of the PDN (probability, dose, number of people) technique for determining the probability of receiving a given dose from an intact system is presented in Section E of this chapter and is representative of the type of calculations which can be derived to describe any given hazard.

The plutonium system is much more flexible. Since the dose rates for the plutonium system are very low compared to the strontium system, the probability of receiving an injurious dose (100 rem) from the plutonium heat source is a factor of 10^5 less than that for the strontium system. The conclusion to be drawn from the comparison (which is shown numerically in Table V-1) is that the plutonium system can always meet the 10^{-6} criterion regardless of the reliability of the reentry system. The decision to attempt to bring the plutonium system back becomes a political or economic consideration and is beyond the scope of nuclear safety analysis, with one extremely important exception. If, when recall is activated, the system malfunctions in a manner such that burnup occurs, a fuel release criterion might be jeopardized. This can be evaluated when new criteria have been established.

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4. Conclusions

The nuclear safety of the CRONUS design has been analyzed based on the three criteria presented. The conclusions of the analysis are:

- (1) The probability of inflicting radiation injury to anyone is less than 10^{-6} for both strontium and plutonium systems.
- (2) The probability of burnup cannot be reliably and numerically demonstrated and the 10^{-7} criterion is considered insufficient to define a transition orbit lifetime separating the recall from no action philosophy. New criteria are required.
- (3) The probability of releasing fuel during all ground handling procedures is less than 10^{-6} , except for transportation by air.

The summation of occurrence events and PDN calculations is shown in Table V-1. The basis for these numbers is explained in subsequent sections of this chapter. One significant item has been omitted from the fuel release tabulation of transportation accidents--water deposition. If the fuel is deposited in water and is not recovered, the encapsulation will corrode and release the fuel in a period ranging from 25 to 100 years. Estimates of shallow water recovery were not attempted during this study, but it was assumed that timely recovery would be accomplished.

Criticality calculations have shown the plutonium system to be adequately subcritical under all normal transportation and operational conditions. The effects of geometry change due to impact must be evaluated in future phase studies.

B. FLIGHT CHARACTERISTICS

1. Launch Vehicle and Test Range

A Titan IIIC configuration and a launch out of the Eastern Test Range were chosen to illustrate, in detail, the launch failure probabilities associated with a given mission, and to establish the probable impact areas associated with a failure to achieve the desired orbit of 600 to 700 nautical miles. Ascent trajectory and reliability data for the Titan IIIC configuration were provided by the Martin Company at Denver.

2. Instantaneous Impact Points

Figure V-1 illustrates the ground trace for the first orbit of both an ETR and WTR launch. Although the flight for a launch out of WTR passes over about three times as much land during the first orbit as does a flight out of ETR, the steep ascent profile and early ejection into coasting orbit associated with both launch sites reduce the effect of this land difference.

For example, at 760 seconds out of ETR a loss of thrust of the transtage (Titan IIIC) would result in an impact just off the coast of Africa. Within seconds, the instantaneous impact point (IIP) will have shifted from the coast of Africa to a random deposition anywhere on earth between the 30-degree parallels. This is the result of the rapid injection described by the ascent trajectory. The IIP will travel across the continent of Africa (a distance of about 900 statute miles) in only 1.5 seconds.

Since the IIP changes so rapidly during the ascent trajectories from either launch site, a "short orbit" condition was defined as that resulting from any failure which would occur from the time of the initial IIP continental contact to and including successful shutdown after the circularizing burn which occurs at the nominal design orbit altitude. Initial IIP continental contact out of ETR occurs at $T = 760$ seconds. Detailed calculations to determine the probability of impacting individual land areas were not conducted during this initial study phase since they do not significantly affect the overall safety feasibility of the design concept. (In this study, land impact was determined based on the land-to-water ratio underlying the orbit flight path.) However, a discrete impact analysis must be conducted prior to range safety approval to launch the system in order to provide direction and data for the CIR abort procedures.

Reentry impact from the defined short orbit can occur at any point on the earth's surface for a WTR launch and for the equatorial orbit between the 30-degree parallels. The land-to-ocean ratio and population densities within the 30-degree parallels are approximately equivalent to those of the entire world.

3. Flight Failure Probabilities

Failure probabilities for the ETR launch configuration are shown in Fig. V-2 as an accumulated failure probability as a function of time after initial ignition, based on a design goal analysis. A successful achievement of orbit for the Titan IIIC is predicted for 92.5% of the launches.

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Fig. V-1. First Orbit Ground Trace for ETR and WTR

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Fig. V-2. Accumulated Failure Probability as a Function of Time from Initial Ignition for Titan IIIC Out of ETR

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Fig. V-3. Vehicle Velocity, Altitude and Downrange Distance as a Function of Time from Liftoff for Titan IIIC Out of ETR

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The probability of achieving a short orbit was assessed to be 0.0046 for the Titan IIIC out of ETR. The IIP of the launch trajectory reaches Africa at 760 seconds into the flight or at an accumulated failure probability value of 0.0705. Since the total failure probability is 0.0751, the difference, 0.0046, represents those instances where a land impact is possible. Table V-2 summarizes the failure probabilities of Fig. V-2 of fuel deposition.

TABLE V-2
Occurrence Probability

On-pad abort (0 to 5 sec)	0.0050
Shallow water deposition (5 to 150 sec)	0.0330
Deep ocean deposition (150 to 760 sec)	0.0325
Random deposition from short orbit (760 to 3831 sec)	0.0046
Successful orbit (after 3831 sec)	<u>0.9249</u>
	1.0000

Figure V-3 represents the launch trajectory profile out of ETR and includes the altitude, velocity and downrange distance of the vehicle as a function of time after liftoff. These data and those for impact are summarized in Table V-3.

The fuel deposition modes identified in Table V-2 are illustrated in relationship to the flight profile as shown in Fig. V-4. Shallow water deposition (in less than 6000 ft of ocean) can occur if the IIP is within 250 miles of the coast of Cape Kennedy or about 200 miles off the coast of Africa. Other shallow water depositions can result from the short orbit condition; however, the probability is much less than for the coasts of Florida and Africa. Aborts which yield shallow water depositions off the coast of Florida cannot utilize the reentry system to divert the ballistic return to earth since the shroud is not jettisoned until the IIP has moved off the coast about 1000 miles. Controlled reentry can be used to avert the shallow water deposition off the coast of Africa and any shallow water deposition from the short orbit. Ocean depth contours for ETR and WTR are shown in Figs. V-5 and V-6.

When an abort occurs which yields a deep ocean deposition, the controlled reentry system is not required. Furthermore, early activation of the system does not significantly change the ultimate splash point as illustrated in Fig. V-4 for a thrust failure at Stage II burnout/Stage III start. The difference in impact point is only 175 statute miles.

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Summary of Flight Impact History for Titan IIIC Out of ETR

<u>Event</u>	<u>Time (sec)</u>	<u>Velocity (fps)</u>	<u>Altitude (ft)</u>	<u>Range (stat mi)</u>	<u>Ballistic Impact Range (stat mi)</u>	<u>Controlled Impact Range (stat mi)</u>
Ignition	0	0	0	0	0	0
Stage zero burnout Stage I start	119	5,998	131,815	34	<150	Not applicable
Stage I burnout Stage II start	259	14,250	312,431	243	930	755
Stage II burnout Stage III start	474	21,620	515,580	894	2,320	2065
Stage III shutoff Coasting period	841	26,588	612,786	2,415	Elipitical orbit random impact	As controlled
Stage III restart	3783	22,713	4,257,522	10,906	Elipitical orbit random impact	As controlled
Stage III burnout	3831	23,644	4,257,250	14,157	Long-lived orbit	As controlled

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Fig. V-4. Launch Trajectory and Abort Profiles for Titan IIIC OUT of ETR

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Fig. V-5. ETR Ocean Depth Contours

Fig. V-6. WTR Ocean Depth Contours

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Contrast this to an abort at the critical 760-second interval which would normally result in an impact on the coast of Africa. Successful operation of the reentry system foreshortens the impact point by approximately 2800 statute miles.

As defined earlier, any failure after the critical 760 seconds will result in a short orbit condition unless the final circularizing maneuver is successful. Successful controlled reentry from the elliptical short orbit could result in a deep ocean desposition. However, the short orbit condition may last from less than one complete orbit up to orbits measured in years. For a short orbit of one or more revolutions, the desired impact point and time to deorbit can be established and optimized by range safety personnel at the time of the incident. For short orbits of one revolution or less, preprogrammed deorbit times must be established as a function of "velocity" or time after liftoff, and must be capable of automatic updating as the flight progresses on or off the nominal established trajectory. For these aborts the impact area may not be optimum based on a total world scale. For example, on a polar orbit the IIP may be on Saudi Arabia at the instant of abort. Successful reentry would deposit the system in the Somali Basin north of Madagascar. This desposition, although meeting the 6000-foot criterion, is not optimum compared with the mid-Atlantic or Pacific Ocean, where the waters are deeper and diffusion slower.

C. HAZARD AREAS

1. Launch Pad Fireballs

During a launch pad accident which yields an explosion from uncontrolled fuel ignition, four distinct phases of fireball development have been identified. These are:

- (1) Fireball growth
- (2) Fireball rise and dissipation
- (3) Fireball products rise and dissipation
- (4) Residual ground fire.

The main hazard parameters associated with this event are heat transfer, overpressures and fragmentation velocities which might result in encapsulation meltdown, oxidation and/or rupture from overpressures or impact.

The most significant and severe phase of either accident is the initial fireball and its subsequent growth. During this phase, the fuel

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encapsulation could be subjected to high heat flux and associated temperatures. Shock overpressures at the payload area do not appear to be excessive (~100 psi) and encapsulation integrity can be assured.

The environments for a typical fireball, described in Appendix Z, were based primarily on half-scale Titan II destruct tests (Refs. V-4, V-5, and V-6), Vanguard, Navaho, Thor, Atlas and Titan missile accidents (Ref. V-7), and various hypergolic and LOX/RP tests (Refs. V-8, V-9, and V-10). Table V-4 describes the parameters used to analyze the thermal integrity of the RTG/RB and individual fuel capsules. However, new environmental data are being generated by Project Pyro at Sandia and should be periodically reviewed for updating the CRONUS analyses in subsequent phase studies.

TABLE V-4

Fireball Thermal Environment

Time duration of environment	$T = 0.23W^{1/3*}$
Temperature of fireball during time, T	3000° F
Heat flux of fireball during time, T	80 Btu/ft ² -sec
Fireball emissivity	1.0
Fireball liftoff time	$T = \frac{W^{1/3}}{22*}$

*T = time (sec), W = total weight of fuel (lb)

The results of the fireball thermal analyses are summarized in Fig. V-7 and are discussed in Appendix C. Two models were investigated, the RTG/RB and an individual fuel capsule. In each analysis the models were assumed to be directly exposed to the environment of the fireball as defined in Table V-4. This is a conservative assumption since the RTG/RB will probably be surrounded by the launch vehicle shroud, initially at least, and the individual fuel capsules will be protected by the RTG/RB radiator and CIR structure which are designed to withstand the predicted overpressures.

The RTG/RB is expected to be breached within 10 seconds of exposure to the fireball. This breach occurs at Node 2 (Fig. V-7) which is the thinnest section of the radiator, and is backed up by insulation which inhibits internal heat transfer. Initial temperatures for the RTG/RB model were established by the steady-state operation analysis.

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Fig. V-7. Temperature of Generator Components in Fireball

Fig. V-8. Pad Abort Fireball Liftoff Time

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The fuel capsule was also exposed to the fireball environment although the system is designed to contain the capsules during an explosive force. Integrity of the encapsulation was shown to exist up to 20 seconds of fireball exposure. Melting point of the Haynes-25 encapsulation is nominally from 2425° to 2570° F. After 20 seconds in the fireball, the encapsulation reaches a temperature of 2334° F.

The time the components actually spend in the fireball is best described in a probabilistic manner, and efforts toward this end are currently being developed at Sandia Corporation for Project Pyro. At this time, however, a technique is not available. One useful criterion presently available to establish limits for exposure times is the fireball liftoff time as presented in Fig. V-8 (Ref. V-11). For the Titan IIC fuel inventory of 3×10^5 pounds, the liftoff time for the fireball is three seconds. If one assumes the RTG/RB has fallen to the ground or been blown free of the main vehicle structure, the time spent in the fireball environment will be shorter. Even if the RTG/RB remains attached to the main booster and settles downward through the fireball, the total time within the environment should be less than five seconds.

In conclusion, the fuel capsules will remain intact even if directly exposed to the fireball environment. However, additional effort is required to better define the environment and provide probabilistic definition of the different modes by which an RTG or fuel capsule can be exposed to the environment.

As described in Appendices E & G, the RTG/RB and the capsules will maintain structural integrity under conditions of anticipated overpressures.

Dispersion of fuel at the launch site should, therefore, be in the form of individual capsules. A direct radiation hazard exists, but since the area is posted and trained recovery teams are available, no personnel radiation damage is anticipated in the event of an accident.

There is a remote possibility that a capsule could be punctured or torn by another piece of flying debris. In this instance (the possibilities of which are considered too nebulous to define; less than 10^{-6}) the fuel could be exposed. Both fuels (Sr-90 and Pu-238) are normally inert, i. e., they will not oxidize, sublime or dissolve and as a result should be available for recovery, albeit with more caution and effort required on the part of the recovery team.

2. Shallow Water Deposition

An analysis of the release of one megacurie of soluble strontium-90 fuel form was conducted under the Strontium Fuel Development Program

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Fig. V-9. Concentration at Point of Shallow Water Release for Soluble Sr-90 Fuel

Fig. V-10. Maximum Concentrations of Shallow Water Release for Insoluble Sr-90 Fuel

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(Ref. V-3). The concentration at a given release point as a function of time was determined for an instantaneous release of the total inventory as shown in Fig. V-9. Within 20 days the concentration at the release point had reduced to the recommended maximum permissible concentration (MPC) level of 3.4×10^{-5} curie/cubic meter.

For a highly insoluble fuel such as strontium-titanate, the situation is quite different. Zigman and associates have made a careful study of the solubility rate of this material for typical fuel pellets on the order of one by one inch cylinders (Ref. V-12). They found a solubility monotonically decreasing with time. It could be expressed as:

$$R = 105 (t + 6)^{-1} \mu\text{gm/cm}^2\text{-day} \quad (\text{V-1})$$

where t is expressed in days. If the total inventory of CRONUS were to be exposed to the ocean water, only 570 curies/day would be released during the first 10 days, and would reduce to 2 curies/day at 10 years. Such release rates would not produce detectable concentrations over areas of appreciable size.

Based on a diffusion velocity of 500 meters/day, a thickness of diffusion layer of 50 meters and a horizontal diffusion angle of $3\pi/2$ radians, the concentration at any time should not exceed those shown in Fig. V-10 as a function of distance from the source.

In Ref. V-3, dose calculations were performed to determine the maximum dose a person might receive by human consumption of Sr-90 contaminated fish. The calculations were based on an average shallow water concentration of 2×10^{-7} curie/cubic meter for five years, a fish concentration factor of 20 (ratio of concentration in fish to concentration in water) and an average individual consumption of fish of 14.5 gm/day. The California coast was the location used. If this amount of fish consumption was maintained for five years, the 70-year integrated dose would be 1 rem.

Although a shallow water release does not yield an injurious dose, the number of people receiving the dose is greater than 10^7 . The genetic consequence of contamination to this many people cannot be defined.

Ingestion of fish or other marine life that has come into equilibrium with plutonium-contaminated water is also the potential hazard from this type of release. Some radio-nuclides are concentrated in the bodies of fish by metabolic processes, but since Pu-238 is not a normal constituent of sea or fresh water, the concentration factor for fish is not known.

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The National Academy of Science-National Research Council committee that investigated waste disposal in the Pacific Ocean selected a factor of 1000. A fish in equilibrium with sea or fresh water which exhibits a MPCC, would represent a body concentration of $10^{-7} \mu\text{Ci/gm}$. Although the fish body disposition is not homogeneous and will be more pronounced in the bones and liver, homogeneous disposition is assumed for conservatism. The time for the fish to reach an equilibrium with the water will vary with age and species.

An average person eats approximately 14.5 grams of fish a day. Again, postulating water at a MPCC, the daily intake in the case of Pu-238 would be about $1.5 \times 10^{-6} \mu\text{Ci}$. The body burden thus acquired is calculated by the equation:

$$q = \frac{fp}{\lambda} (1 - e^{-\lambda t}) \quad (\text{V-2})$$

where:

- q = body burden acquired in critical organs, μCi
- f = fraction of ingested activity that goes to the critical organ, bone
- p = daily intake, μCi
- λ = effective decay constant (day^{-1})
- t = ingestion period (days)

In the case of plutonium, a fish diet for a year would result in an acquired body burden of $1.6 \times 10^{-8} \mu\text{Ci}$. This can be compared to the maximum permissible body burden for a radiation worker of $4 \times 10^{-2} \mu\text{Ci}$.

* MPCC - Max. Permissible Continuous Concentration ($10^{-10} \mu\text{Ci/gm}$)

3. Deep Ocean Disposal

Much of the earth's surface is covered with water, and there is a high probability that the fuel from an isotopic generator would be deposited in the deep ocean. The fuel would not be recoverable and would eventually be released to the ocean water.

Deep ocean water is understood to be water at depths greater than 6000 feet. It is assumed fuel released in deep water will not diffuse into the upper layers nor will fish transport the contaminant either by their movement or through the organisms on which the fish feed. Oceanographers estimate that an average of 300 years is required for deep

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ocean water to appear at the surface. The process is cyclic and some deep ocean water is continuously appearing at the surface.

A committee has recently studied the disposal of radioactive wastes on the Pacific Coast (Ref. V-13). The zone in which fish and other marine life are found extends to a depth of about 2.2 kilometers. This committee has calculated the maximum permissible continuous concentration for strontium-90 from waste disposal to be 3.4×10^{-5} curies/cubic meter. Thus, for the solution of a generator loading of one megacurie, dilution by about 3×10^{10} cubic meters would achieve this concentration.

Some idea of the capacity of the oceans for absorbing radioactive wastes can be obtained from Table V-5.

TABLE V-5
Areas and Depths of the Oceans

<u>Ocean</u>	<u>Area (km²)</u>	<u>Average Depth (km)</u>
Pacific	16.6×10^7	4.3
Atlantic	8.2×10^7	3.9
Indian	7.3×10^7	4.0
Arctic	1.4×10^7	1.2

From these data, it can be shown that the ocean volume below 2.2 kilometers is about 6.2×10^{17} cubic meters. This is about seven orders of magnitude greater than required for the dilution of a megacurie of strontium-90.

A somewhat similar method of estimating the disposal capacity of the deep ocean is referred to in another National Academy of Science-National Research Council Publication (Ref. V-14). A group of United Kingdom and United States scientists assumed that fission products deposited on the ocean floor would drift and disperse for at least 10 years before surfacing, at which time the contaminated volume would be a disc about two kilometers thick and 70 kilometers in diameter, which would subsequently be dispersed throughout the surface layer. They further estimated that repeated deposits of one megacurie of strontium-90 made at the rate of 10 per year would result in an average concentration of strontium-90 of not over 10^{-8} curies/cubic meter in the mixed layer, equivalent to 25 picocuries/gram of calcium.

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The postulated disc has a volume of 7.7×10^{12} cubic meters; so, if a megacurie is dissolved therein, the average concentration at the end of 10 years becomes 1.0×10^{-7} curies per cubic meter. As this deep water comes gradually to the surface, there would be further dilution in the mixed surface layer above the thermocline, where diffusion is relatively good. This mixed layer varies in thickness at different places and seasons but is estimated to have an average thickness of 75 meters. Over the principal oceans, it consequently has a volume of 2.4×10^{16} cubic meters.

For the purposes of this study, based upon the foregoing arguments, it is concluded that deposition of the fuel in deep ocean does not represent a significant hazard.

4. Random Reentry

Random reentry from short orbit is expected to occur with a probability of 7.8×10^{-4} (Table V-1) through failure of the retro system during an unsuccessful attainment of the design orbit. The probability for the burnup condition represents the failure probability of the retro system (0.17) just as the random reentry did, but the time of abort which leads to burnup occurs earlier during ascent than the continental IIP and as a result yields a higher probability of occurrence, 8.2×10^{-3} . The product of 1.4×10^{-3} represents a greater probability for being in a burnup condition than a short orbit condition.

a. Reentry heating and impact

The RTG/RB is designed to be aerodynamically stable and is capable of surviving reentry heating. The dynamics or stability of the reentry body and the reentry heatup are described in Chapters III and VIII. Only accidents of an extreme nature, such as a third stage ignition explosion, could conceivably lead to a condition of body burnup. Even then, it is not clear whether the capsules will be exposed early enough to receive sufficient heat to release the fuel since CRONUS is designed to withstand explosion overpressures. The fuel capsules must be ejected into the reentry environment at altitudes greater than 250,000 feet, in order that they release fuel during descent.

Accidentally initiated explosions, excessive tumble rates and heat shield damage are obvious precursors to a burnup condition. However, the actual identification of component malfunctions which lead to these major conditions are numerous and difficult to describe numerically without a thorough reliability analysis of a given design. A design malfunction analysis of this magnitude is beyond the scope of this feasibility study.

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It is felt that the 10^{-7} criterion is feasible. As shown in Table V-1 a probability of 7.1×10^{-5} is the constraint on component malfunction leading to a burnup condition. Because of multiplicities of failures required to yield the burnup condition, such as malfunction detection system (MDS) failure, followed by a structural failure with subsequent orientation yielding abnormal ablation patterns, the 7.1×10^{-5} failure probability does not appear excessive. Obviously much more analysis is required in this area.

Table V-6 lists the impact velocities for various incoming portions of CRONUS. Based on the results of typical capsule impact studies, only the end-on, individual capsule impact has a velocity great enough to jeopardize intact integrity. Integrity for the other velocities is considered within state of the art for Haynes encapsulation.

Further enhancement of impact integrity has been gained through the total containment design of the RTG/RB. During intact impact, the RTG/RB will sustain the first impulse and internal capsules will be subsequently cushioned by about 3.5 inches of Min-K insulation.

TABLE V-6

Impact Velocities

Reentry body without CIR package

Stable	170 fps
Tumbling	202 fps

Reentry body with CIR package (rocket motors empty)

Stable	310 fps
Tumbling	277 fps

Heat source (intact)

End on	276 fps
Side on	205 fps
Tumbling	253 fps

Fuel capsule (individual)

End on	585 fps
Side on	147 fps
Tumbling	218 fps

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b. Geographical considerations

Figure V-11 illustrates the relative proportions of surface area in each altitude zone of the earth. The total land area is 29% of the world's surface, but 10% of the land is glaciated. Therefore a randomly re-entering body has a 0.26 probability of impacting on populated land. Also, 11.5% of the world's surface is between sea level and 6000-foot ocean depth. This, then, is the probability of a shallow water deposition from a random short orbit reentry.

The difference in land mass to water ratio between a polar and equatorial orbit is slight and is not differentiated in this feasibility report. Figure V-12 compares the fraction of land for both orbits in terms of population densities. Figures V-13 and V-14 illustrate the world's land and population density by latitude degrees.

c. Potential hazards

The predominant hazard from reentry is the direct radiation from the fuel. For the strontium system, the resultant dose rates are quite significant. The dose rate from the plutonium system is a factor of 10^5 less than the strontium system.

In determining an injurious dose for the intact direct radiation condition, a value of 100 rem was established as the injurious dose. This is the dose which has been estimated to reduce an individual's lifetime by a small percentage but is not sufficient to cause immediate physical impairment.

The calculations for direct radiation from the intact system are presented in Section E and constitute the major argument for meeting the 10^{-6} criterion of personal injury.

Shallow water depositions result in eventual contamination of many people, none of which are expected to receive dosages in the order of 100 rem. It is concluded no one will receive an injurious dose from an unrecovered shallow water deposition, but the number of people involved makes this hazard considerable.

Atmospheric dispersion of the fuel through reentry burnup is also undesirable from the viewpoint of the numbers of people exposed to radiative contamination. The number of people who could possibly receive airborne contamination equal the world's population for a complete vaporization or small particle dispersion during burnup. Large particulate burnup still yields receptors in the order of 10^8 persons. Consequences of dispersed fuels must be evaluated in subsequent phases of this study since reentry ablation experiments have not been

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Fig. V-11. Earth Surface Areas

Fig. V-12. Polar and Equatorial Orbit Populations

Fig. V-13. World's Land Distribution

Fig. V-14. World's Population Distribution

performed sufficiently to describe particulate products from either fuel. The final particulate distributions are the crux of the resultant hazard evaluations.

5. Transportation

a. Discussion

This section describes probable transportation accidents, their probability of occurrence, and the hazards resulting from these accidents.

For the proposed configuration, there is no requirement to transport a fueled RTG. It is recognized that as a result of recovery operations following an abort or an early reentry, transport of a fueled RTG may be necessary. This is considered a special case associated with recovery and will not be covered here, as directed by the contract statement of work.

For the CRONUS concept, the isotopic inventory will be transported with the fueled capsules in the proper position in the heat source assembly. When transported, the heat source will consist of an annular ring of Min-K material with 36 capsules equally spaced on the periphery. The ring will be placed in a shipping container that acts to shield the radiation and provides the necessary cooling. The structural adequacy of the container will be proven for the necessary conditions. Although the total shipping weight for the heat source and container will be in the neighborhood of 10,000 pounds, this would not preclude transportation by air. Practically speaking, the CRONUS heat source assembly will probably always be shipped by surface transport which includes highway, rail or water modes.

Impact, fire and burial (earth or water) are the three basic hazard occasions that can result from a transport incident. The degree of hazard and the probability of occurrence will usually vary in accordance with the transportation mode for which the incident is postulated. For instance, the probability of impact rupture should be significantly less if the carrier is a barge on the water than if it is a truck on the highway.

(1) Resultant hazards

Although design and procedural safeguards are implemented to preclude or attenuate conceivable hazards, it is necessary to consider the kind and extent of possible hazards should there be a release of all or part of the fuel inventory on or into the ground, into the water, or in the air. In the following discussion, evaluation of the results of releases is intended to reflect the most severe of conditions. Many of the

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conditions discussed are of no more than academic concern since it is firmly believed that the proposed system has been designed to eliminate the possibility of their occurring.

(a) Direct radiation

The shipping container is of a thickness sufficient to ensure that external radiation levels are acceptable for handling and transportation. This is not meant to infer that there is no hazard from external radiation while the heat source is in the shipping container. If one or more individuals is unknowingly being exposed to the radiative effects of the contained heat source over an extended length of time, it could be biologically damaging. This could only happen if possession and/or control of the loaded container is lost. The presumption is made that should such a situation occur, a concerted effort at recovery will be made, and the period of time for which control is lost is not extensive.

(b) Atmospheric dispersion

For CRONUS, it is difficult to attach credibility and meaning to an atmospheric dispersion condition during transport. The design and structural characteristics of the shipping container are such that it would require a cataclysmic incident of some sort to result in an atmospheric release and/or dispersion. Typical of this category of incident might be a violent midair explosion or collision, or a surface conflagration. If these conditions are postulated, a maximum theoretical atmospheric release might be analyzed based on dispersion data reported by Guthrie and Nichols (Ref. V-15). This assumes vaporization of the fuels. Additional studies would have to be performed to show this to be a credible and appropriate consideration.

Previous similar analyses and data reviews suggest that for a credible release of vaporized fuel, relatively few people (possibly less than 20, based on average population density) will receive a dose greater than the recommended maximum body burden of $0.04 \mu\text{Ci}$, even if an air accident occurs over highly populated continents. For a maximum theoretical, atmospheric fuel release, approximately 10^6 persons can be expected to receive a body burden above $0.04 \mu\text{Ci}$. However, the physical characteristics of the candidate fuel forms (strontium-titanate and plutonium oxide) are such that the temperatures required for vaporization upon release are very high and have an extremely low probability of occurring with a transportation incident.

(c) Surface resuspension

Resuspension/analysis of insoluble strontium fuels (Ref. V-3) have shown that for a one megacurie release the maximum 70-year dose from a single release event would not exceed 10 rem. Since this is not an

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injurious dose the 10^{-6} criterion is not jeopardized by strontium resuspension following a fuel release subsequent to impact.

Little has been reported on the post-release disposition of plutonium or its oxide. Outdoor tests with plutonium rods (Refs. V-16 and V-17) determined that most of the airborne particles settled out within 200 yards of the release point. In reporting data from British tests (Ref. V-18), Stewart concludes that no significant inhalation hazard would exist at greater than 200 yards, from burning several kilograms of plutonium. A Hanford incident (Ref. V-19), involving a fire in a plutonium area, yielded a release of about 225 μCi . No significant contamination was detected on the ground beyond 200 yards. It appears that the actual transportation of plutonium dioxide created during ignition or meltdown does not conform to atmospheric particulate-transport equations with regard to either range of dispersion. Stewart has defined a resuspension factor of $1 \times 10^{-6} \text{ m}^{-1}$ which indicates that exclusion of resuspension from the atmospheric dispersion evaluation does not significantly affect the expected hazard conclusion.

(d) Subsurface migration

It can be postulated that the heat source could be thrown into a crevice and induce a backfill or landslide that would cover the capsules with earth. Consequently, a fuel release might occur from incompatibility effects, from capsule meltdown after burial, or as a result of capsule mechanical destruction upon impact. The possibility of a near-surface release, coupled with usual surface mixtures of heterogeneous soil and rocks, along with significant surface water movement, suggests that the movement of fuel to surface waters be analyzed as well as the subsurface movement through soil.

A major consideration for the hazard associated with a subsurface release is the resultant fuel form available for possible transport. It is improbable that the original fuel form will be released, but rather that complex compounds will be formed with other heat source materials. In any event, the ion exchange capacity of the soil will significantly affect the migration of the nuclide.

The ion exchange capacity simply means the ability of a particle to exchange cations or anions for free ions in the surrounding medium. This ion exchange capacity is derived by electrostatic attraction or by actual stoichiometric chemical reaction and, for isotopes such as Sr-90 and Pu-238, involves exchangeable ions such as Na^+ , Ca^{++} , Mg^{++} , K^+ , etc. Typically, the movement of the isotope will be most influenced not by ion exchange, but by the mass flow effect of rain water. If ground or surface waters lead to an underground aquifer or watershed, it is probable that at least some of the isotope would

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eventually reach a water supply. Dilution by dispersion and ion retention processes is likely to attenuate any hazard occasions. Nonetheless, it should be recognized that any form of fuel release having an associated probability of human assimilation can constitute a health hazard.

b. Impact

The shipping container is not overly bulky, but it is extremely heavy for its size. The container is mounted on a skid to facilitate handling, and the skid is equipped with tiedown hardware. The container loaded weight and the requirement for positive cooling provisions indicate that transportation by highway would be in a van-type vehicle. Transportation by rail would be in a standard cooled freight car. Finally, transportation by water, in order to provide cooling in the simplest way, would probably consist of the van or trailer being stowed on the ship. Of the three modes of travel, the most severe credible accident, from the viewpoint of impact, will result from a trucking incident.

A collision can be postulated for the carrier vehicle at a highway speed of 70 mph (103 fps). (Collision probabilities will be developed later.) The shipping container and the heat source within would not see this maximum velocity, since the inherent energy absorption characteristics of the vehicle and the container tiedown hardware would attenuate the impact effect of the collision. Regardless, the shipping container will be designed for a maximum impact velocity to assure that the containment integrity of the fuel capsules will not be compromised by an impact event.

c. Fires

It is possible that fires can result from a variety of transportation incidents including the collision event mentioned previously. Gasoline is the combustible which would result in the hottest flame. Conflicting recommendations exist for values of the burning temperature of a gasoline fire to be used in accident fires analysis. The National Fire and Protection Association recommends that a temperature of 1700° F may be realized from a gasoline fire, while Title 10 of the Code of Federal Regulations, Part 20 suggests a temperature of only 1475° F for evaluation purposes. Often a conservative value of 1850° F is used. The CRONUS shipping container and heat source assembly will be qualified with an analysis using either the maximum value or all three values.

d. Water burial

Should an incident occur when the heat source assembly is being transported by water, or over or near a body of water, the container

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and/or the heat source assembly could easily be submerged. Burial in water is not in itself a harmful event in most circumstances. It is possible that the incident would happen in the winter when a very hot capsule would be exposed to very cold water, but the Haynes-25 material should maintain integrity since this is the way it is normally heat treated during fabrication. Essentially, the cold water would act to cool the capsule to keep it from overheating. Corrosion would be a problem if the fuel capsules are not recovered. However, inland bodies of water are not of depths which would preclude recovery within a reasonable length of time.

Since launches from ETR and WTR do not require transporting the heat source assembly beyond the borders of the continental United States and therefore not over ocean regions, ocean burial is not considered credible prior to launch. Both launch and postlaunch ocean burial events are discussed in Section C. 3

e. Earth burial

If the shipping container, with the heat source assembly, were to become buried in soil, it is unlikely that this would be a direct result of an earth impact. Although the frontal loading factor (weight/area, lb/ft²) for the shipping container is approximately 12, the earth-impact velocity would be such that only a fraction of the container, if any, would penetrate the earth. It is more likely that the container would be thrown free of the vehicle, land in a crevice, and start a small slide or landfall that covers it completely, or nearly so. For this condition it is possible that the lack of cooling may result in capsule melt if the capsules are not recovered in time.

f. Occurrence probabilities

(1) Highway accidents

There are numerous highway accidents documented each year. Accident compilations seldom note the relative severity of accidents with the exception of those involving fatalities. For this reason, fatality frequency is interpreted as a value approximation for highway incidents which would result in an undue exposure to the general populace. By using the accident probability as a base, the likelihood of the characteristic consequences (impact, fire and burial) can be determined. An attempt is made to keep all of the following approximations conservative.

Information presented in the World Almanac indicates that there are approximately 2.3×10^{-8} fatalities per passenger mile. This yields a

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serious accident probability of 2.3×10^{-5} , based on a distance of 3000 miles, and the assumption that three persons are involved in each fatal accident.

The post-incident location of the heat source is important. Since transport by highway would require a closed van, it would not be unreasonable to assume that for nearly all events the heat source will remain with the van. Inasmuch as this would eliminate the possibility of burial, less probable but more demanding container dispositions will be assumed.

If the shipping container remains within the van for only 75% of the severe accidents, and for the other 25% it is thrown free, it is obviously subjected to burial in earth or water following 25% of the incidents. Burial resulting from earth penetration is extremely unlikely as was discussed earlier. Therefore, ruling out landslides and avalanches, for burial to occur the container must be thrown into a crevice or into water in addition to being thrown from the van. This probability is approximated at 5%. Therefore, the total probability of burial occurring was assessed as 2.9×10^{-7} . Finally, if a fire is assumed for all of the events when the container remains within the van, the probability of exposure to a gasoline fire as a result of a highway accident is about 1.7×10^{-5} . As a maximum, assume that some sort of impact can occur within the van for every incident. Therefore, this event would have the same probability as the accident, 2.3×10^{-5} . These values along with those to follow are tabulated in Table V-1.

These probabilities infer that accident frequency is synonymous with hazard frequency. Such is not the case, but it is a conservative approach. For greater accuracy, design and procedural safeguards should be taken into account. The material and fabrication of the container provide for excellent secondary containment if the capsules are breached. Thus, even for a severe condition, where release might be considered imminent, it is highly unlikely that it would occur coincident with the accident. Some amount of time would be available to plan and organize a recovery effort that precludes a release.

(2) Railroad accidents

The line of reasoning in this evaluation is essentially the same as that used for highway accidents. The severe incident (fatality) rate for railroads is approximately 0.5×10^{-9} . Based on a 3000-mile trip, the probability of an accident is 1.5×10^{-6} . If the same distribution of results is assumed for highway incidents, the probability for exposure to fire is about 1.1×10^{-6} . As before, the impact probability is the

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same as the accident probability, 1.5×10^{-6} . Design and procedural safeguards were not considered in this evaluation, but the same reservations would apply.

(3) Air accident

Data collected by the Civil Aeronautics Board and presented as a tabulation of "Accidents Versus Total Flying Time in 1963," indicate that the frequency of serious air accident is about 3.2×10^{-5} per hour of flight time, for commercial flights in general. The accident rate for military transports is comparable. From this information the following air accident probabilities were determined for air accidents that culminate in a crash:

Propeller plane (average velocity = 350 mph):

9.1×10^{-5} /1000 miles

Jet plane (average velocity = 600 mph):

5.3×10^{-5} /1000 miles.

For air incidents that culminate in a crash, it is most likely that the heat source will remain with the vehicle throughout the event. Consequently, it may be subjected to all three characteristic hazards (impact, fire and burial). If the incident occurs over water, only some form of impact and/or water burial is likely to result. If a mid-air explosion which throws the generator free is postulated, the ground impact velocity coupled with the container frontal loading factor could easily result in earth penetration and burial.

To summarize, based on a 3000-mile trip, the probability of severe accident for each transportation mode is:

Highway	2.3×10^{-5}
Railroad	1.5×10^{-6}
Air, jet	1.6×10^{-4}
Air, propeller	2.7×10^{-4}

Obviously rail transportation offers the least probability of impact type collisions.

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The tabulated values of K_{eff} relate to neutron multiplication with critical flux distributions. Neutron behavior was analyzed with the DTK code (LA 2595) which computes numerical solutions to Carlson's multi-group one dimensional SN equations. Cross sections were specified for 16 fast groups and one thermal group using a Martin-modified version of the GAM-I code and library (GA 2451). These included values for σ_a , $\nu\sigma_f$, in-group and down-group scattering. The scanty experimental data for Pu-238 were incorporated with the scattering matrix of Pu-240 to generate a cross section file for Pu-238. In-group values for Pu-238 including capture, fission and total scattering cross sections and ν were obtained from LASL data. The only exceptions were the use of data relevant between 0.1 mev and 1.5 mev as reported by Butler and Sjoblom of Argonne (AED-Con. -63-048-54).

No critical mass of this isotope fuel form or of Pu-238 alone has yet been assembled, and experimental cross sections are most meager in scope, so some uncertainty must be accepted with these calculations. However, the methods and data herein described result in critical mass estimates slightly below those of L. L. Carter of Hanford Laboratories and W. D. Stubbins of the University of Cincinnati, and so may be taken as conservative in the range of the present state of the art of analysis.

D. SHIELDING

1. Strontium

An analysis of the CRONUS design was performed to determine the radiation environment produced by the base heat source, the heat source installed in the RTG/RB and the heat source installed in the shipping cask.

Maximum gamma dose rates around the installed generator are given in Table V-9. Figure V-15 gives the location of the points in relation to the generator. The beryllium radiator provides negligible shielding for the heat source, therefore, the gamma dose rates of Table V-9 are also applicable for the heat source. Neutron fluxes and approximate dose rates in mrem/hr at various points around the installed generator are given in Table V-10. The locations of the points are shown in Fig. V-15. There is no neutron flux for the heat source alone. Examination of these results shows that the neutron dose rates are negligible compared to the gamma dose rates.

The cask in which the fuel array is to be shipped and stored includes a shield of uranium molybdenum alloy. Dose rates at the shield surface, at the surface of the protective cage and at one meter from the center of the capsule array versus uranium thickness are shown in Figs. V-16 and V-17. Figure V-16 gives the dose rates to the side of the cask along

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Fig. V-15. Dose Point Locations of Installed Generator

Fig. V-16. Gamma Dose Rates at Side of CRONUS Shipping Cask (Sr-90)

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Fig. V-17. Gamma Dose Rates at Top and Bottom of CRONUS Shipping Cask (Sr-90)

Fig. V-18. Cross Section for the Reaction $\text{Be}^9 (\nu, n) \text{Be}^8$

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the horizontal centerline, and Fig. V-17 shows the dose rates for the top and bottom of the cask along the vertical centerline. If ICC regulation of 10 mr/hr at one meter from the center of the radio-activity is to be met, 3.6 inches of uranium alloy are required on the sides and 3.45 inches are required on the top and bottom of the shield.

TABLE V-9
Dose Rates for Installed Generator (Sr-90)
Gamma Dose Rates (rem/hr)

<u>Horizontal Centerline</u>		<u>Vertical Centerline</u>	
<u>A (ft)</u>	<u>Dose Rate</u>	<u>H (ft)</u>	<u>Dose Rate</u>
0	8.73 (+4)	1	9.02 (+3)
2	5.41 (+3)	2	1.98 (+3)
3	2.21 (+3)	4	2.82 (+2)
6	8.51 (+2)	6	1.02 (+2)
12	1.29 (+2)	12	2.46 (+1)
24	3.22 (+1)	24	6.04
50	7.65	50	1.37

NOTE: $Z.ZZ (+Y) = Z.ZZ \times 10^Y$. See Fig. V-15 for dose point locations.

TABLE V-10
Neutron Fluxes and Approximate Dose Rates
for Installed Generator (Sr-90)

<u>Horizontal Centerline</u>			<u>Vertical Centerline</u>		
<u>A (ft)</u>	<u>Flux (neut/cm²/sec)</u>	<u>Approx. Dose Rate (mrem/hr)</u>	<u>H (ft)</u>	<u>Flux (neut/cm²/sec)</u>	<u>Approx. Dose Rate (mrem/hr)</u>
2	5.09 (+3)	727	1.5	3.25 (+3)	464
3	1.88 (+3)	268	2	2.40 (+3)	343
6	4.24 (+2)	60.5	3	1.34 (+3)	191
12	1.03 (+2)	14.7			
24	2.56 (+1)	3.6			
50	5.88	0.84			

Approximate center of retrorocket 1.54 (+3) 220

NOTE: $Z.ZZ (+Y) = Z.ZZ \times 10^Y$
See Fig. V-15 for dose point locations.

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All analytical results were obtained by use of the generalized shielding code developed at the Martin Company. This code integrates dose rates from an array of point sources and uses buildup factors to approximate the contribution of scattered radiation to the total dose rate. General equations for surfaces of the second degree are used to define the shape of the source and shields, thus allowing asymmetric shields to be evaluated readily. The source distribution may be uniform or non-uniform throughout the source volume. Maximums of 400 source points and 10 initial source energies may be accommodated. Total gamma attenuation coefficients are used for gamma ray dose rates and fast removal cross sections are used to obtain neutron dose rates.

Path lengths through the various materials, along a line joining a point source and the dose point, are found and used to calculate relaxation lengths and buildup for each of the materials between these two points. The individual relaxation lengths are added to obtain the total relaxation length. Buildup along the individual path segments is defined as the infinite medium buildup factor minus one. The total buildup factor along the path from the source to dose point is assumed to be one plus the sum of the individual buildups. Infinite media buildup factors are approximated by the sum of two exponentials. The direct energy flux at the dose point is evaluated for each source energy and source position and converted to dose rates by the appropriate flux-to-dose conversion factor.

The spectral distribution of the bremsstrahlung gamma radiation used in the analysis is given in Table V-11 along with the flux-to-dose conversion factor used.

TABLE V-11
Energy Grouping and Abundances of Gamma
Radiation from Sr-Y-90 (Ref. V-20)

<u>E (mev)</u>	<u>$\bar{E}_j$ (mev)</u>	<u>$F_j \frac{\gamma}{(\text{dis})}$</u>	<u>$R_j \frac{\text{mrem/hr}}{(\gamma/\text{cm}^2 \text{ sec})}$</u>
0.0 to 0.22	0.11	7.37×10^{-2}	1.72×10^{-4}
0.22 to 0.44	0.33	5.72×10^{-3}	6.71×10^{-4}
0.44 to 0.66	0.55	2.22×10^{-3}	1.36×10^{-3}
0.66 to 0.88	0.77	1.03×10^{-3}	1.57×10^{-3}
0.88 to 1.10	0.99	4.90×10^{-4}	1.92×10^{-3}
1.10 to 1.32	1.21	2.18×10^{-4}	2.22×10^{-3}
1.32 to 1.54	1.43	9.00×10^{-5}	2.50×10^{-3}
1.54 to 1.76	1.65	2.97×10^{-5}	2.78×10^{-3}
1.76 to 1.98	1.87	6.80×10^{-6}	3.05×10^{-3}
1.98 to 2.20	2.09	7.32×10^{-7}	3.31×10^{-3}

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Fig. V-19. Neutron Generation in Beryllium Radiator Resulting from the $\text{Be}^9 (\nu, n) \text{Be}^8$ Reaction with Sr-Y-90 Bremsstrahlung Greater than 1.666 Mev

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a. Neutron production

When gamma rays greater than 1.666 mev impinge upon beryllium neutrons, they are released according to the reaction $\text{Be-9 } (\gamma, n) \text{ Be-8}$. The cross section for this occurrence, given in Fig. V-18, was taken from Ref. V-21. Gamma fluxes (energies >1.666 mev) at the inner face of the beryllium radiator were found by the generalized shielding code. Neutron source strength was calculated by the following relation:

$$N = (\phi) (\sigma) (NA) \quad (V-3)$$

where

N = neutrons/cm²

ϕ = gammas/cm²

σ = cross section for reaction (cm²/atom)

NA = atoms/cm²

$$= \rho \times \frac{AN}{A} \times t$$

ρ = density (gm/cm³)

AN = Avogadro's number (6.023×10^{23})

A = atomic weight

t = thickness of radiator (cm)

The calculated values of N along the length of the radiator are plotted in Fig. V-19. Neutron source strengths for the Martin Generalized Shielding Code were obtained by dividing the radiator surface into 64 approximately equal areas and multiplying these areas by an average neutron production for the area. Neutron fluxes found at various points around the generator are listed in Table V-10. The actual dose rates will be lower. This is caused by treating all neutrons as fast neutrons (i.e., having energies greater than 1 mev) and using a conversion factor of 7 neutrons/cm²/sec-per millirem/hr. In actuality, the neutrons range from about 1 mev down to 24 kev. The information available in Ref. V-21 is not sufficient to estimate the actual energies.

2. Plutonium

An analysis was made to determine both neutron and gamma dose rates from a plutonium oxide source for a range of thermal powers from 2 to 12 kilowatts. The dose rates were calculated parametrically since the final plutonium system may not be optimized for that fuel but may merely be plutonium substituted for strontium in the strontium capsules. In this manner a range of dose rates is established which can be used regardless of the final plutonium loading. The present design is 5.39 kilowatts(t) of power for the optimum plutonium system.

A power density of 4.51 watts/cm³ of oxide and a specific power of 0.0345 watts/curie were used for plutonium oxide along with the following gamma spectrum and conversion factors:

Fig. V-20. Neutron Dose Rates

Fig. V-21. Gamma Dose Rates

<u>E (mev)</u>	<u>$\gamma/\text{sec/gram Pu-238}$</u>	<u>$\left(\frac{\text{mrem/hour}}{\gamma/\text{cm}^2\text{-sec}}\right)$</u>
0.099	4.97×10^7	1.54×10^{-4}
0.75	3.13×10^5	1.54×10^{-3}
0.875	1.24×10^5	1.74×10^{-3}
1.75	3.71×10^3	2.58×10^{-3}

The neutron dose rates were based on a production rate of 2.5×10^4 neutrons/sec-gm of Pu-238. This is the maximum generation rate measured by Mound Laboratory (Ref. V-22) using samples of PuO_2 .

The neutrons were assumed to be fast neutrons of approximately 1.0 mev with a flux-to-dose rate conversion factor of $7.0 \text{ neutrons/cm}^2\text{-sec}$ to equal to 1.0 mrem/hr.

Equations used in determining dose rates for the source configuration were explained in the previous section on strontium.

Parametric carpet plots of both the neutron and gamma dose rates as a function of thermal power of the source and distance are shown in Figs. V-20 and V-21. The attenuation factors of various shield materials as a function of shield thickness are shown for gammas by Fig. V-22 and for neutrons by Fig. V-23. The neutron attenuation factors are based on fast removal cross sections. These curves can be used with the dose rate curves to determine the thickness of shielding required to reduce the dose rate by any factor from zero to about 10^5 .

An example of how one might use these curves is as follows:

Given :

(1) Thermal power	6 kw
(2) Distance from source to point of interest	2 ft
(3) Maximum allowable dose rate	200 mrem/hr
(0.2 rem/hr)

The neutron dose rate at two feet for a six kilowatt source is about 6.2×10^2 mrem/hr and the gamma dose rate is 5.1×10^1 mrem/hr from Figs. V-20 and V-21. Let the gamma dose rate contribute 5% of the 200 mrem/hr or 10 mrem/hr. Therefore, the gamma attenuation factor required is:

$$\frac{5.1 \times 10^1}{1 \times 10^1} = 5.1$$

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Fig. V-22. Attenuation Factors for Gamma Radiation

Fig. V-23. Attenuation Factors for Neutron Radiation

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From Fig. V-22 it is seen that an attenuation of 5.1 requires 0.8 inch of lead.

The neutron attenuation factor required is:

$$\frac{6.2 \times 10^2}{1.9 \times 10^2} = 3.26$$

From Fig. V-23 one finds that the 0.8 inch of lead used as a gamma shield provides a neutron attenuation factor of 1.3. The additional attenuation required is:

$$\frac{3.26}{1.3} = 2.50$$

E. PDN ANALYSIS

The PDN analysis presented in this section was developed to define the probability of receiving an injurious dose from direct radiation as a result of a random land deposition of the entire RTG/RB. The probabilities of occurrence for this situation were presented in Section B, and the total analysis summarized in Table V-1.

1. General Approach

The uncontrolled intact reentry case results in deposition of the isotopic source within some population density. In order to determine the dose received by a person, first consider the probability of N persons (P_N) being within a circle of radius, r , from the source for a population density ρ . A Poisson function may be used to evaluate P_N :

$$P_N = \frac{(\pi r^2 \rho)^N}{N!} e^{-\pi r^2 \rho} \quad (V-4)$$

The dose (D) received by any person with a fixed center of motion* within a circle of radius r from the source is a function of the dose rate (d) from the source and the time (t) spent within the area, or:

$$D(r) = t_R (\rho) d(r) \quad (V-5)$$

*In previous probabilistic dose determinations (Ref. V-3), a roving receptor definition was incorporated in the analysis in the form of a Gaussian description of a person's motion around a fixed center of motion. The effect of this consideration upon ultimate dose received by an individual was not substantial due primarily to the low sigma values assigned to the function. An individual's motion has not been applied to this analysis; however, it is desirable to do so. When a more realistic roving receptor model is developed, it will be used in the PDN calculations.

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Where t_R is the time for discovery/recovery of the source and is intuitively a function of the population density, i.e., the larger the population density, the faster the source will be discovered and recovered. The $d(r)$ is simply the dose rate at any given distance, r .

To evaluate the effect of population on recovery time, the following hypothesis is established. Define A_o as the area of source recognition, i.e., the area within which one can identify the source (Fig. V-24). The probability of someone ($P_{N>0}$) being in that area is

$$P_{N>0} = 1 - e^{-A_o \rho} \quad (V-6)$$

and for $A_o \rho$ suitably small

$$P_{N>0} \approx A_o \rho \quad (V-7)$$

ρ = population density

A_o = area of discovery/recovery

$d(r)$ = dose rate at any distance, r

D = dose received by an individual at distance r for time t_R

t_R = time for discovery/recovery

Fig. V-24. PDN Analytical Model

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Now, assuming $(P_{N>0}) (t_R)$ is invariant, it can be concluded that the time for recovery is

$$t_R = C/\rho \quad (V-8)$$

where C can be established by a reasonable assessment of a person's capability to recognize and identify an attractive (noncommon place) object of a given size.

$D(r)$, Eq(V-5), may now be represented by

$$D(r) = C/\rho d(r) \quad (V-9)$$

The dose rate (d) can be approximated as :

$$d(r) \approx \frac{d_o}{\pi r^2} \quad (V-10)$$

which merely states that the dose rate is inversely proportional to the square of the distance from the source, except for the region within a source diameter surrounding the source and the region exceeding a thousand feet from the source.

Combining Eqs (V-9) and (V-10) results in:

$$\pi r^2 \rho = \frac{d_o}{d(r)} \times \frac{C d(r)}{D(r)} = \frac{C d_o}{D(r)} \quad (V-11)$$

which is the argument of the Poisson function, and Eq (V-4) becomes:

$$P_N = \frac{\left(\frac{C d_o}{D(r)}\right)^N}{N!} e^{-\frac{C d_o}{D(r)}} \quad (V-12)$$

2. Strontium-Fueled CRONUS

To assess the probability of being exposed to an injurious dose from the strontium-fueled RTG/RB, C, d_o and D must be defined numerically.

For the CRONUS system, it has been postulated that an individual within 200 feet of the RTG/RB (A_o in Fig. V-24) will discover the vehicle on sight and, within an hour, recovery will have been effected, either by removal of the system or by the establishment of exclusion boundaries.

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Under these conditions

$$C = t_R \rho = 8 \times 10^{-6} \text{ hr/ft}^2$$

d_o can be evaluated from Eq (V-10) and the data contained in the strontium shielding section.

$$d_o = \pi d(r) r^2 = \pi \left(129 \frac{\text{rem}}{\text{hr}}\right) (12 \text{ ft})^2 = 5.85 \times 10^4 \text{ rem-ft}^2/\text{hr}$$

D, the injurious dose to one individual is defined as 100 rem as discussed earlier in Section D. The probability of one person receiving a dose of at least 100 rem can now be assessed.

$$P_N = \frac{\left(\frac{Cd_o}{D}\right)^N}{N!} e^{-\frac{Cd_o}{D}} = 0.0047$$

where $N = 1$, $D = 100 \text{ rem}$, $C = 8 \times 10^{-6} \text{ hr/ft}^2$ and $d_o = 5.85 \times 10^4 \text{ rem-ft}^2/\text{hr}$.

3. Plutonium System

The PDN analysis of the plutonium-fueled RTG/RB differs only in the dose rate an individual receives at any given distance from the source. From the data contained in the plutonium shielding section,

$$d_o = \pi d(r) r^2 = \pi (0.081) (6)^2 = 9.15 \text{ rem-ft}^2/\text{hr}$$

for a loading of 6420 kilowatts of plutonium oxide.

Therefore,

$$P_N(D) = \frac{\left[\frac{(8 \times 10^{-6})(9.15)}{100}\right]^1}{1!} e^{-\frac{(8 \times 10^{-6})(9.15)}{100}} = 7.3 \times 10^{-8}$$

Thus, the probability of an individual receiving a dose of 100 rem or more is 7.3×10^{-8} for the plutonium system. Quite simply, if one would continually remain within five to six feet of the RTG/RB, it would take 1000 hours for the individual to accumulate a dose of 100 rem.

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VI. RELIABILITY

A. SUMMARY

The reliability analysis for this feasibility study is oriented toward meeting safety and operational reliability requirements. Although the reliability analyses in many cases cross system boundaries, to maintain consistency of presentation throughout this chapter, the reliability technical discussion is divided as follows:

- (1) Radioisotope thermoelectric generator (RTG) system including the voltage regulator
- (2) Controlled intact reentry (CIR) and reentry body (RB)
- (3) Ground handling equipment (GHE).

The evaluation of reliability of each of these subsystems is discussed in detail in Sections B, C and D, respectively, following this summary.

RTG

The primary reliability goals for the RTG (including the voltage regulator) is to achieve a 99% probability of providing at least 250 watts of power at 28 volts dc $\pm$ 10% for a period of one year, and a probability of 95% of providing this minimum electrical power requirement for five years. The reliability analysis performed during this study shows the actual reliability to be 99.79% for the first year of operation and 97.97% for five years of operation. The voltage regulator is the major contributor to the unreliability for the five year mission. The reliability values are based on an availability of the RTG of 1.0 at lift-off; i. e., checkout procedures and monitoring of critical systems during the prelaunch phase eliminate launch of a malfunctioning or out-of-tolerance RTG.

CIR/RB

Reliability goals for the controlled intact reentry (CIR) system are safety oriented with respect to population radiation injuries from a ground impact and dispersion of radioactive material in the earth's atmosphere during reentry from a short orbit or abort. These goals are dependent upon the CIR system plus reentry body reliability, probability of land impact, population distribution

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and launch system reliability. Since this involves both the proper operation of the CIR as well as the structural survival and proper operation of the reentry body portion of the RTG/RB, this reliability portion is called the "CIR/RB" reliability. The minimum permissible CIR/RB reliability required in order to meet the safety requirement has been shown to be 83% (Chapter V). The CIR/RB analysis shows that the actual reliability is at least 99.31%. Consequently, the desired safety goal is exceeded. The analysis is based on initiation of CIR/RB abort as soon as it is determined that a long life orbit has not been attained (approximately one hour after lift-off). Permitting a system to remain exposed to the unique environments of this mission for extended periods after separation of CRONUS from the spacecraft reduces the probability of a successful reentry. The probability of fuel dispersal in the atmosphere is believed to fall within the established criteria since only the extremely unlikely occurrence of a major structural failure will result in burn-up of the CRONUS during re-entry. It is felt that the safety goal will be exceeded even if CIR operations were attempted 30 days after launch.

GHE

The reliability analyses for ground handling equipment (GHE) during the feasibility study are directed toward two goals:

- (1) Determining the feasibility of designing and developing fuel ground handling equipment that could assure the probability of less than 1×10^{-6} of releasing radioactive fuel from the fuel capsules during a ground handling operation.
- (2) Assure design compatibility for providing a reliable and efficient fueling system that can be integrated with the launch facilities and countdown procedures.

The reliability analysis of the GHE for this feasibility study was confined to establishing failure modes and assuring that handling equipment would provide adequate protection against environments that might exceed design conditions of the airborne systems. The probability of transportation accidents and the possible outcomes are discussed in Chapter V.

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B. RADIOISOTOPIC THERMOELECTRIC GENERATOR SYSTEM

1. Reliability Goals and Failure Criteria

The stated reliability goals for the generator system are (a) that the probability shall be at least .99 that the RTG will survive shipping, handling, and launch environments, and will operate properly for one year; and (b) that the probability shall be at least .95, or alternately .75, that the RTG will operate properly for five years.

As used above, "proper operation" of the generator system is assumed to be delivery of at least 250 watts electrical power, at a voltage of $28V \pm 10\%$, when used with a nominal load impedance. Since the spacecraft load may vary, conceivably even be as low as zero, a voltage regulator subsystem is provided. The RTG reliability includes the reliability of the voltage regulator. The RTG requirement, therefore, can be considered to be to deliver up to 250W power, at $28V \pm 10\%$. The voltage must be maintained within tolerances for all loads between zero and the maximum allowed (250W). There is, therefore, only a power failure criteria. If the available power drops below 250W, the system is assumed to fail (even though there is no guarantee that the S/C would call on the full capability of the RTG).

2. Environments and Checkout Procedures

It became obvious during this study that the mechanical and thermal environments seen by the generator prior to launch should be less severe than those seen during launch and subsequent operation. All of the prelaunch environments, including factory acceptance testing, shipping, storage, spacecraft integration, integration testing, erection at the launch site, fuel loading, and checkout, will be simulated using the best techniques available, and the generator system will be designed and tested to survive these environments.

It also became obvious that the prelaunch environments are in some ways less predictable than those occurring after liftoff. For example, inadvertent bumping or dropping of the generator in the handling or transportation phases can produce shock loads of higher magnitude than those seen in a normal launch.

In order to minimize the probability of exceeding design loads, special handling, installation, and test procedures will be generated for use with the generator system. In addition, the approach was taken to check out the system prior to launch by means of flight acceptance testing, S/C integration testing,

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preflight checkout, and thermal and performance monitoring after fueling.

In this way, an availability of 1.0 will be assured at time of liftoff, and the system will have a predictable probability of success.

3. Mathematical Model and Reliability Block Diagram

A failure of the generator system could be caused by a failure of any of several independent components, each of which plays a vital role in the production or regulation of electrical power. If the failures of these components are assumed to be independent of each other, the reliability of the system may be expressed as the product of the reliabilities of the respective components.

The resulting mathematical model and reliability block diagram are shown in Fig. VI-1. The results of the reliability analysis are shown for missions of one and five years' duration.

4. RTG Analysis and Discussion

A discussion of each of the power-producing components represented in the math model is given. Of these, the thermoelectric subsystem is discussed in the most detail since both catastrophic (sudden) failures and performance (gradual) failures are treated therein.

a. Thermoelectric subsystem

The thermoelectric subsystem consists of 33 modules which are connected electrically in series. Within each module, 12 thermoelectric couples are connected in a series-parallel arrangement as shown in Fig. VI-2.

Definition of Symbols Used in Analysis

λ	Failure rate of element	t	Time
ϵ	Ratio of open to total failure rate	X	$N \lambda t$
N	Number of elements in chain, number of couples N/2	P_i	Probability of being in state i
K	Maximum number of allowable couples that have failed	$P_{i,j}$	Transition Probability from state i to state j

$$R_{RTG} = R_{T/E} \times R_{ELEC} \times R_{VR} \times R_{HS} \times R_{INS} \times R_{RAD} \times R_{STR}$$

$$R_{RTG} = .9979$$

For 1 Year

$$R_{RTG} = .979$$

For 5 years

Fig. VI-1. RTG Reliability for One and Five Years

Fig. VI-2. T/E Module Having 12 Couples Connected in Series-Parallel

This arrangement provides redundancy since failure of a single couple will not cause total failure of the generator system. If a single couple fails by open circuit, its current will be conducted by the adjacent parallel couple, whose net power output will now be reduced to zero, and the system output power will be reduced approximately 0.5%. For a given couple failure rate, the loss of power due to failed couples is predictable with time.

If the generator output dropped below 250 watts due to individual couple failures, this would be called a performance failure. If two adjacent and parallel couples failed by open circuit, the system output would drop to zero, and this would be called a catastrophic failure.

An analysis was made to determine the number of extra couples that should be provided to offset the predicted power loss due to individual couple failures. The reliability with respect to both performance and catastrophic failures has

been analyzed as follows.

The analysis first computed the probability of losing K couples, and the probability of a "catastrophic" failure. It was shown that the latter was essentially independent of K provided only that the number of spare couples was more than 3. In this way, the performance failures and the catastrophic failures were separated.

The probability of having sufficient performance was computed if it is known that K couples were lost. This probability times the probability of losing K couples was summed over all values of K, (0, 1,) in order to arrive at the probability of delivering the minimum amount of power.

In order to analyze the probability of catastrophic failures and the probability of losing K couples, the following assumptions are made:

- (1) All couples are statistically independent.
- (2) A couple fails in one of two modes--"open" or "shorted." Once failed, it never changes to another state.
- (3) Each couple has a constant failure rate.
- (4) The probability that a pair (2 adjacent couples) that has failed due to the failure of either couple, will be converted to an open failure is given by $\int_0^t \lambda dt$ (see page VI-4 for definition). This approximation essentially assumes that a failed pair can be considered to consist of one good and one bad couple (statistically independent, assumption #1).
- (5) Initially, all couples are good.

A transition matrix approach was used to develop a series of differential equations for the probability of the system being in a given state. The states are as follows:

<u>State</u>	<u>Probability</u>	<u>Definition</u>
1	P_1	No pairs (or elements) failed
2	P_2	One pair failed

<u>State</u>	<u>Probability</u>	<u>Definition</u>
3	P_3	Two pairs failed
M	P_M	M-1 pairs failed
$K + 1^*$	$P_{K + 1}$	K pairs failed
$K + 2$	$P_{K + 2}$	Failure, i. e. , at least $K + 1$ pairs failed or in at least one pair both elements failed "open"

*NOTE: K is maximum number of allowed pair failures

We shall define P_{ij} as the transition probability of transferring from state i to state j in time dt.

If we ignore simultaneous double failures (a procedure which sounds serious, but is completely correct), then only three different transitions need be considered.

$P_{i,i}$	System remains in state i
$P_{i,i+1}$	One element fails in good pairs
$P_{i,K+2}$	One element fails in bad pairs, both the new and the previous failures being "open"

Defining $X = N \int \lambda dt = N \lambda t$ and defining the operator $\frac{d}{dx}$ we can easily show that:

$$P'_1 + P_1 = 0$$

$$P'_{i+1} + P_{i+1} \left[1 - \frac{i}{N} (2 - \epsilon^2) \right] = P_i \left[1 - 2 \frac{(i-1)}{N} \right] \quad i = 1, 2, \dots, K$$

$$P'_{K+2} = \left(1 - \frac{2K}{N} \right) P_{K+1} + \frac{\epsilon^2}{N} \sum_{i=1}^{K+1} (i-1) P_i$$

The initial conditions are

$$P_1(0) = 1$$

$$P_i(0) = 0 \quad i = 2, \dots, K+2$$

We can readily solve this set of equations as follows:

$$P_1 = e^{-x}$$
$$P_{i+1}(x) = \int_0^x \left[e^{-\left[1 - \frac{i}{N}(\epsilon^2)\right](x-\xi)} \right] \left(1 - \frac{(i-1)}{N}\right) P_i(\xi) d\xi$$
$$P_{K+2}(x) = \int_0^x \left(1 - \frac{2K}{N}\right) P_{KH}(\xi) d\xi + \frac{\epsilon^2}{N} \sum_{i=1}^{K+1} (i-1) \int_0^x P_i(\xi) d\xi$$

Since $\frac{2K}{N} \ll 1$, we can obtain the following useful formula:

$$P_{K+2} = \left[1 - e^{-x} \sum_{j=0}^K \frac{(x^j/j!)}{j!} \right] + (\epsilon^2/N) \sum_{i=1}^K i \left(1 - e^{-x} \sum_{j=0}^i \frac{(x^j/j!)}{j!} \right)$$

Of course, the probability of losing exactly M couples is given by the Poisson distribution,

$$\frac{x^m}{m!} e^{-x}$$

We can now write

$$R_{T/E} = R_C R_P$$

Where:

$R_{T/E}$ = Reliability of Thermoelectrics

R_C = Probability of no catastrophic failure

R_P = Probability of no performance failure

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Let us first examine R_C . From the above equation, for all practical purposes, we can write

$$R_C = 1 - \frac{\epsilon^2}{N} \sum_{i=1}^K i (1 - e^{-x} \sum_{j=0}^i \frac{x^j}{j!})$$

For reasonable values of x (x has the order of magnitude of 1 or less), only the first two or three terms of the above sum (in the dummy variable i) are of any importance.

The evaluation of R_P is slightly more complicated than the determination of R_C . The performance may be below acceptable limits because of either enough couples failing, or because of a degradation in the performance per couple. We shall assume that each individual couple's performance is a stochastic variable with a mean value given by the "nominal" value (time dependent, of course) $1\sigma_C$ variation of $Y\%$ of its mean value. The form of the distribution function is immaterial since for the large number of couples (~ 400), the sum of the individual couples will approach a gaussian distribution with a mean value equal to the sum of the means, and a 1σ variation equal to $\frac{Y}{\sqrt{N}}\%$. This result, of course, assumes that the individual couples are additive and statistically independent.

This deviation has been measured based on long-term operation of four six-couple modules (No. 's MTA 4A-9, MTA 7-10, MTA 8A-9, and MTA 8A-18) at RCA. Data were extrapolated to five years' operation, and the deviation between modules was seen to increase with time.

The above modules are similar in material and configuration, but are not identical in that their fabrication processes and test conditions are different. Still, they display relatively small variations in performance as shown in Table VI-1. For generator systems having 396 couples, their standard deviation (σ) in output power is less than 1% at 5 years based on the above module data. For modules having identical processes and test conditions, the variations between modules (and systems) is expected to be reduced.

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Operating Time (hours)	1 σ Couple Deviation Y, %	1 σ Module Deviation $= \frac{Y}{\sqrt{6}}, \%$	1 σ System Deviation $= \frac{Y}{\sqrt{396}}, \%$
1000	6.95	2.84	.35
8760 (1 year)	13.5	5.51	.68
43800 (5 years)	18.0	7.36	.90

Table VI-1. Standard deviation in output power between couples, modules, and generators, based on long-term operation of four six-couple modules at RCA.

If we know that we have N_A pairs active, at some time, we can compute the probability of having enough performance by computing $\text{erf} \left(\frac{2 \Delta N}{\sqrt{N} \sigma_c} \right)$

where ΔN is the "excess" number of pairs left, assuming nominal performance. Let

N_S = Number of Spare Pairs

K = Number of Pairs Lost

N = Total Number of Couples (twice the number of initial pairs)

Then

$$R_P(K) = \text{erf} \left(\frac{2N_S - 2K}{\sqrt{N - 2K} \sigma_c} \right)$$

Where $R_P(K)$ is the probability of having sufficient performance if K pairs have been lost (and if no catastrophic malfunctions have occurred).

$$R_P = \sum_{K=0}^{N \text{ or } \infty} \frac{X^K}{K!} e^{-X} \text{erf} \left(\frac{2N_S - 2K}{\sqrt{N - 2K} \sigma_c} \right)$$

Both R_C and R_P are a function of the failure rate. As can readily be seen, for a few spares (~ 6 spare pairs), the probability of failure is dominated by R_C for low failure rates and is controlled by R_P for high failure rates.

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The dependency upon $R_{T/E}$ upon the number of spare pairs and failure rate is shown in Fig. VI-3. As the number of spares is increased, the reliability increases until it reaches essentially a plateau. This upper limit is controlled by R_C . R_C varies approximately as $1 - \frac{\epsilon^2 X^2}{2N}$ for reasonable values of X .

How many spare pairs should be used? Three different answers can be given, depending upon the criteria used.

CRITERIA 1: Meet the required reliability goals with "measured" failure rate of $3.5 \times 10^{-5}/1000$ hours.

No more than 4 spare pairs would be required to meet a 95% overall goal for 5 years operation (99% for the thermoelectrics). One spare pair would suffice for a 75% overall reliability goal. (78% for the thermoelectrics)

CRITERIA 2: Meet the required reliability goal, if possible, for the worst conceivable failure rate.

If we do not "know" the unit failure rates, but suspect that the actual failure rate may be much higher than $3.5 \times 10^{-5}/1000$ hours, then approximately 6 spare pairs would be the maximum useful number of spare pairs required to meet the 95% overall reliability goal.

CRITERIA 3: Cost optimize the reliability/weight trade-off ratios.

Assuming that 1 spare pair will cost approximately 3 lb. of RTG/CIR weight and assuming that a typical Titan III payload would weigh about 20×10^3 , we can estimate a trade-off ratio:

$$\frac{\delta R_{T/E}}{\delta \text{ Spare Pair}} = \frac{3}{2 \times 10^4} = 1.5 \times 10^{-4}$$

This trade-off ratio is the slope of the $R_{T/E}$ vs Number of Spare Pair Curve at which the number of spares is an optimum. It can easily be seen from Fig. VI-3 that, for failure rates of $3.5 \times 10^{-5}/1000$ hours, or greater, the required number of spares is that quantity where the reliability curves flatten out. Consequently, about 6 spare pairs should be used with the

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Fig. VI-3. RTG Reliability Versus Number of Spare Pairs

anticipated failure rate. Approximately 7 and 8 pair should be used if the actual failure rate is twice or three times the anticipated ($7 \times 10^{-5}/1000$ hours and $10.5 \times 10^{-5}/1000$ hours respectively).

Since all criteria seem to yield approximately the same number of spare pairs, we would conclude that about 6 spares would be the proper design criteria. For 6 spares, the RTG reliability (exclusive of the voltage regulator) is shown in Table VI-2 for the presently expected thermoelectric failure rate and for a "conservative" failure rate double the present value.

<u>No. of Years</u>	<u>No. of Spare Pairs</u>	<u>Failure Rate</u>	<u>R_C</u>	<u>R_P</u>	<u>R_{T/E}</u>
1	6	.0035 (best estimate)	.999982	1.0	.999982
5	6	.0035 (best estimate)	.99948	.99996	.99942
1	6	.0075 (approx. double best estimate)	.999912	1	.999912
5	6		.9977	.998	.9957

Table VI-2. Thermoelectrics Subsystem Reliability

The estimate of the failure rate (.0035%/1000 hours) is based upon past and current testing.

b. Electrical connectors

The electrical connector reliability block considers all internal electrical connections between couples and modules, through connector, and the spring load self-disconnectors associated with power lead separation during CIR. Martin has tested these spring loaded quick disconnects for SNAP 19 application. The failure modes of interest to the RTG mission are inadvertent disconnect and individual wire connection failure. Failure rate data on connectors of this type are presented in MIL-HBK-217A and are consistent with SNAP 19 test data.

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Since a failure of T/E module interconnections is catastrophic in nature, and approximately 34 such interconnections are required, positive interconnection techniques are vital. If deemed necessary, redundant interconnections shall be employed to increase the reliability.

Recent developments in connector and connection techniques show promise in decreasing the failure rates on this type hardware. A major factor in assuring high reliability of electrical connections is requiring the use of qualified processes and craftsmen who are skill certified in those processes.

c. Voltage regulator

The voltage regulating subsystem of the RTG is described in detail in Chapter III. Basically, the system consists of six shunt regulator blocks connected in parallel across the generator power leads as shown in Fig. VI-4. Each block is capable of dissipating up to 60 watts of power not consumed by the S/C. The circuit for control of this power dissipation by one block is shown in Fig. VI-5. In general, only one block will operate if the excess power is less than 60 watts; two blocks will be operating when up to 120 watts must be dissipated.

The regulating system is designed to be capable of dissipating all of the power (360 watts BOL) from the generator in the event that the spacecraft system is shut off completely. If one were to assume that the Nimbus B power profile would never go below 200 watts, then only three regulator blocks would be required and the reliability of the RTG system enhanced.

An analysis of the regulator circuit reveals that the critical failure mode is a short circuit in any one of the components shown in Table VI-3 and denoted by an asterisk. This failure mode results in 60 watts of power being dissipated by R_S. A failure of one of these blocks in this mode during the first year would not result in mission failure for that period since the BOL power output of the RTG is 360 watts and 350 watts at the end of one year. Thus, the loss of 60 watts would not decrease the power available to the spacecraft below 250 watts. In the evaluation of the probability of success for a five-year mission, it can be readily seen that failure of any one block in the shorted mode will result in mission failure (i.e., less than 250 watts to the S/C bus). Examination of the power output versus mission time curve, shown in Appendix R, reveals that one regulator block failure would not result in mission failure until T = three years (RTG power output below 310 watts; thus less than 250 watts to S/C bus).

The reliability predictions for voltage regulator success for a one-and a five-year mission, based on the critical failure mode, are presented in the following.

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R_{S1} , R_{S2} , R_{S3} , R_{S4} , R_{S5} , and R_{S6} each control up to 60 watts of power.

Fig. VI-4. Voltage Regulator Block Diagram

Fig. VI-5. Schematic Diagram of Typical Voltage Regulator Block

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<u>Part No.</u>	<u>Piece Part</u>	<u>Type</u>	<u>Failure Rate</u>
R ₁ *	Resistor	RNR Metal Film MIL-R-55182	MIL-HDBK-217A .000000019
R ₂ *	Resistor	RNR Metal Film MIL-R-55182	.000000019
R ₃	Resistor	RNR Metal Film MIL-R-55182	
R ₄	Resistor	RNR Metal Film MIL-R-55182	
R ₅ *	Resistor	RNR Metal Film MIL-R-55182	.000000019
R ₆	Resistor		
R ₇	Resistor	RNR Metal Film MIL-R-55182	.000000019
R ₈ *	Resistor	Precision Wire Wound MIL-R-26	.000000009
R ₉ *	Thermistor		.000000002
R ₁₀ *	Resistor	Precision Wire Wound MIL-R-26	.000000009
R _S *	Resistor	Precision Wire Wound MIL-R-26	.000000009
Q ₁	Transistor	NPN	
Q ₂ *	Transistor	NPN	.000000015**
Q ₃ *	Transistor	PNP Silicon	.000000015**
Q ₄ *	Transistor	NPN Silicon	.000000015**

Table VI-3

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Table VI-3 (Cont'd)

<u>Part No.</u>	<u>Piece Part</u>	<u>Type</u>	<u>Failure Rate</u>
Q ₅ *	Transistor	PNP Silicon	.000000015**
Q _S	Transistor	NPN Silicon	.000000015**
CRI	Zener Diode		

* Fail Short Cause Loss of Voltage Regulator Block

** Based on Minuteman Failure Rates

* An asterisk indicates a piece-part whose failure in a short circuit mode could cause that regulator block to continuously dissipate 60 watts.

Fail short failure mode for one year

A parts-count reliability analysis technique was employed using failure rates and operating factors based on MIL-HDBK-217A. Table VI-3 presents the list of piece-parts making up a single generator block. Failure rates were based on operating in approximately a 40°C environment and operating at .2 of the rated value for each part. Transistor reliabilities are based on minuteman failure rates since these are representative of state-of-the-art. The analysis is based on the most critical failure mode of the six regulator blocks failing in a short circuit mode, thereby dissipating 60 watts of power and preventing the RTG system from providing a full 250 watts output.

The probability of a resistor failing in a short circuit mode is considerably less than failing in an open mode. For the purpose of this analysis, it is assumed that 5% of the resistor failures and 50% of the transistor failures would be a short circuit mode.

Summarizing the failure rates and adjusting failure rates as noted, the reliability of each component may be derived as follows:

<u>Component</u>		<u>Failure Rate</u>
$R_1 = (.000000019)(.05)$	=	.00000000095
$R_2 = (.000000019)(.05)$	=	.00000000095
$R_5 = (.000000019)(.05)$	=	.00000000095
$R_8 = (.0000000098)(.05)$	=	.00000000049
$R_9 = (.00000002)(.05)$	=	.000000001
$R_{10} = (.0000000098)(.05)$	=	.00000000049
$R_S = (.0000000098)(.05)$	=	.00000000049
$Q_2 = (.000000015)(.5)$	=	.00000000075
$Q_3 = (.000000015)(.5)$	=	.00000000075
$Q_4 = (.000000015)(.5)$	=	.00000000075
$Q_5 = (.000000015)(.5)$	=	.00000000075
$Q_S = (.000000015)(.5)$	=	<u>.00000000075</u>
Failure Rate for 1 Voltage Regulator Block		.00000005182

Reliability of 1 block for 1 year = R_{BL}

$$\lambda t = (.00000005182)(8760) = .00045$$

$$R_{BL} = e^{-\lambda t} = .99955$$

Reliability of 1 block for 5 years = R_{BL5}

$$\lambda t = (.00000005182)(8760)(5) = .00225$$

$$R_{BL5} = e^{-\lambda t} = .99775$$

Reliability of 6 blocks for 5 years

$$\lambda t = (6)(5)(8760)(.00000005182) = .01362$$

$$R_{6BL5} = .98638$$

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During the first year failure of a single block in the short circuit mode would reduce the power output to 300 watts which still meets specified power requirements. The probability of one and only one block failure during the first year may be calculated by using the first two terms of the binomial equation.

$$R_{VR} = R_{BL}^6 + 6R_{BL}^5 Q_{BL}$$

$$Q_{BL} = 1 - R_{BL}$$

$$R_{VR} = .9973 + .00269 = .99999$$

The reliability of the voltage regulator to regulate the specified power for 1 year is predicted to be 99.99%.

Fail short failure mode for five years

None of the voltage regulator blocks may fail in the short circuit mode during the five year period if EOL power of 250 watts is to be attained. The probability of the six blocks of the voltage regulator to operate for five years is .98638.

The failure of any one of the regulator blocks in an open circuit mode would not cause a mission failure as long as the S/C systems are turned on. This failure mode does not drain power from the RTG, but it does eliminate the ability to dissipate 60 watts of power if required to do so. Since each block controls up to 60 watts of power, then any three functioning blocks can control the BOL power (360 watts) over and above the S/C power requirements given in the mission power profile for the Nimbus B (200 watts minimum). Near the end of the five year mission, any one block could control the power above the S/C requirements.

The six block voltage regulator may provide redundancy by switching to a dummy load when the S/C electrical load is turned off. Switching may be accomplished by a double throw switch or relay performing the function of switching off the S/C and switching to the load. The variation in S/C load would then require three blocks, permitting the remaining three blocks to be in parallel.

It is felt that a more efficient and reliable voltage regulator will be designed for a specific mission. System effectiveness will be optimized when the actual power profile is defined.

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d. Heat Source

The basic function of the heat source is to supply heat to the thermoelectric subsystem. Once the heat source is loaded with radioisotope fuel and installed in the RTG, there is nothing short of structural failure and a gross change in the RTG configuration that could prevent the decay heat from being transferred to the surrounding RTG components. The reliability of the heat source has therefore been taken as 1.0 for one year of operation.

For longer periods of operation, it can be argued that volatilization of chromium from the fuel capsule surfaces could cause an electrical short circuit in the thermoelectric subsystem. A detailed fuel capsule coating development program will be effected to ensure that this does not happen. The heat source reliability for a five-year period has been assumed to be .99999.

e. Thermal Insulation

The primary function of the thermal insulation is to minimize heat loss from the heat source, and to direct the heat flow through the thermoelectric elements. In addition, it is used in the heat source as a structural member to support the fuel capsules. Any cracking, shipping, or long-term change in its thermal properties would tend to increase the parasitic heat losses and reduce the performance of the generator system.

The insulations used in the reference design are an advanced Min-K and low-density Micro-Quartz, both of which will resist temperatures in excess of the highest anticipated heat source temperature. No long-term degradation is therefore expected.

With respect to cracking or chipping of the structural insulation, the selected design includes a fuel capsule retaining ring to spread compressive loads over a larger area of insulation so as to reduce its stresses to an acceptable level. The Micro-Quartz insulation rings above and below the thermoelectric modules are wrapped in Astro-Quartz cloth to provide abrasion resistance during installation of the heat source. In addition, all insulation components will undergo development and subsystem qualification testing to ensure that the components will withstand their operational environment.

The reliability of the thermal insulation has been apportioned at .999 for a one-year mission. Since an additional four years in the zero-g space environment constitutes no additional hazard to the insulation, the same .999 probability of success for a five-year mission was assumed.

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f. Radiator

The radiator dissipates waste heat to the space environment by conduction through the beryllium metal and by radiation from its high-emissivity coating. Its operation is passive in nature and depends only on the properties of the radiator material and of its coating.

The only way in which failure can occur is by degradation of the coating through aging, volatilization, radiation damage, or micrometeoroid bombardment. The beryllium metal may also be damaged by micrometeoroid penetration. A complete evaluation of the extent of damage from such strikes cannot be performed until the detailed specifications and characteristics of the radiator material are defined and its physical properties are known.

This analysis has assumed that the probability of damage to the radiator is equivalent to the probability of a strike by a micrometeoroid. The approach used to calculate the probability of a micrometeoroid strike is presented in Appendix EE.

g. Structure and mounting

From an examination of the generator design, it can be seen that it contains very few structural members. The upper and lower insulation rings and the radiator serve as structural members, as well as serving their major function. The major structures not discussed in another section are the mounting structure for the heat source, and the radiator-to-Min-K struts. These structures shall be subjected to margin-of-safety tests to assure a .99999 probability of not failing in the environment they will encounter while being launched into orbit.

Martin has used this form of testing on the man-rated Gemini program with a high degree of success. It should be noted that failures of certain struts and mounting structures do not constitute a catastrophic failure. Since, through testing, a probability of .99999 can be assured against any failure, a value of 1.0 for structural reliability of the RTG has been assumed.

5. RTG Reliability Prediction

Using the reliability figures presented, the math model was exercised to determine RTG reliability. Figure VI-3 summarizes the data and shows the feasibility of attaining a reliability of:

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- (1) .9979 probability of providing not less than 250 watts for one year.
- (2) .979 probability of providing not less than 250 watts for five years. This includes the reliability of the voltage regulator. If the latter's failures were not included, then the 5 year reliability is about .993. Consequently, the stated RTG reliability goals can and will be met.

6. RTG failure mode analysis

The failure mode analysis for the RTG is shown in Table VI-4.

TABLE VI-4

RTG Failure Modes

Thermocouples

<u>Failure Modes</u>	<u>System Effect</u>	<u>Mission Effect</u>
Couple failure by bond separation	Effective loss of two couples, open circuit	No effect; extra couples compensate for failure
Couple failure by element failure	Effective loss of two couples, open circuit	No effect; extra couples compensate for failure
Failure of adjacent parallel couples	Loss of generator power	Loss of mission; probability of occurrence .00052 for five years
Conductor strap failure in open condition	Loss of two effective couples	No effect; extra couples provided
Short circuit between adjacent elements of hot shoes	Short circuit	50% of time loss of 2 couples; 25% of time no loss; 25% of time loss of 0 to 2 modules (avg - 1)

TABLE VI-4 (Cont'd)

<u>Failure Modes</u>	<u>System Effect</u>	<u>Mission Effect</u>
Short circuit between conductors and radiators	Short 1st occurrence Short 2nd occurrence	No loss Loss of 0 to 100% of RTG (avg = 50%)
Short circuit between pins of power connector	Loss of all power	Loss of mission; Lower heat source temperature
Open circuit at pins of power connector	Loss of all power	Higher heat source temperature. Loss of mission.
Short circuit between pin and case	1st occurrence 2nd occurrence	No loss Loss of power and lower heat source temperature
Open circuit between modules	Loss of generator power	Loss of mission Higher heat source temperature

It is assumed that the beryllium radiator provides protection against micro-meteoroid impact for the fuel capsules and internal couple structure.

Heat Loss

Melt of capsule*	Loss of generator power	Fuel release. Possible injury to people
Weld failure in capsule casing*	Loss of fuel	Fuel release. Possible injury to people

Radiator

Structural failure**	May cause loss of module	May cause loss of generator power
----------------------	--------------------------	-----------------------------------

TABLE VI-4 (Cont'd)

<u>Failure Modes</u>	<u>System Effect</u>	<u>Mission Effect</u>
<u>Insulation</u>		
Structural failure**	Possible loss of support for fuel capsule	Possible degradation of generator power; capsule still retained in generator
<u>Electrical Connections</u>		
Open circuit modules to connector	No immediate effect	Capsules still retained by bearing rings
Short circuit modules to connector	Same as above	Same as above
Failure of connector to separate on reentry***	Effect undetermined	Effect undetermined
Open circuit RTG to voltage regulator to S/C connector	Loss of power to S/C	Loss of mission
Short circuit RTG to voltage regulator to S/C connector	Same as above	Same as above
Latches, locking in heat source	Displacement of heat source	Generator output low; released if another catastrophe occurs
Latching ring for GHE connection fails in latched position	Ground failure, no effect after launch	Cannot readily remove heat source if launch hold occurs

* Hazards possible only as result of previous catastrophe.
** Structural failures may result from a previous catastrophe.
*** Considered an "incredible" event since both relative motion and aerodynamic heating would separate systems.

C. CONTROLLED INTACT REENTRY AND REENTRY BODY (CIR/RB)

1. Mission Reliability Requirements

All recently initiated development programs for new radioisotope power supplies require that these systems satisfy certain established safety criteria associated with earth orbit and interplanetary applications. Two reliability goals in force specifically for the 250 watt generator are:

- (1) The probability of causing radiation injury to people in a small area as a result of either an inadvertent or planned reentry of the radioisotope package into the earth's atmosphere shall be less than 10^{-6} .
- (2) The probability of a wide dispersion of nuclear fuel in the atmosphere shall be less than 10^{-7} .

The CIR/RB system design has been markedly influenced by these two goals. The CIR/RB operational requirements and design characteristics have been evolved and synthesized to minimize the hazard. In fact, the design has been so oriented that, even if a malfunction occurs that may jeopardize the first criteria, there would still be only a very small chance of dispersing nuclear fuel in the atmosphere. Several steps have been taken to assure a high "reliability". As the first step, the choices of the operating sequence were pointed to maximizing the CIR/RB reliability for a reasonable period after liftoff. Second, the CIR/RB has been mechanized, packaged and oriented for spacecraft attachment such that no active control of the CIR/RB is required prior to or during a mission abort. Third, provision has been made to utilize the launch vehicle and spacecraft malfunction detection system for rapid automatic initiation of CRONUS separation in the event of an incipient, mission aborting malfunction. Fourth, the CIR separation and deorbit system functions are performed with well proven types of equipment and straight forward operational techniques. Fifth, the CIR subsystems have been chosen and/or designed to utilize inherently reliable hardware for the heavier components and extensive redundancy for the somewhat less reliable light-weight components. The R/B falls in the same category with the application of the thoroughly tested and flight proven Gemini-shaped configuration. Finally, the single required ground command deorbit signal provides that last assurance of positive control over reentry and impact, with a minimum complexity in the airborne systems.

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The purpose of this section is to illustrate the detailed analytic process followed, provide the results of the associated analysis and indicate the degree of success in achieving the stated reliability goals, through application of the above reliability features. Consideration of the first reliability goal necessarily involves factors other than the reliability of the CIR/RB system hardware. These factors include both the causes leading to an abort or a short orbit mission and the probability of a land impact which might expose people to radiation. The contribution of each of these factors to the reliability goal has been established separately and is described in Chapter V. Based on this analysis, the minimum acceptable probability of a successful controlled intact reentry is 83%. The CIR/RB reliability analysis shows the actual reliability attainable under the previously defined reference conditions to be at least 99.31%.

The reliability evaluation involving the second reliability goal (atmospheric dispersion) requires more data than is available for this feasibility study. This latter evaluation will be performed as the detail design of the CIR/RB system is developed and failure modes can be statistically analyzed, however, it is expected that only the highly unlikely failure of primary structure will contribute to dispersal of the fuel within the atmosphere. This preliminary analysis indicates that no credible set of malfunctions seem to exist which will widely disperse nuclear fuel in the atmosphere.

2. Operational Sequence

A fundamental decision has been made that the Strontium system would exercise the recall capability only if necessary, and that this recall would be initiated as soon as possible. This means that, if an acceptable orbit is achieved (600 n.m. circuits) the strontium system will be left in orbit. If an acceptable orbit is not achieved, then recall will be quickly initiated, either by the automatic MDS system or by ground command.

The operation sequence of the CIR + RB starts, therefore, when a command to initiate the separation sequence should be issued. The reliability analysis, however, starts from liftoff, since the time and environment prior to activation may induce malfunctions in the airborne system. For the sake of ease of analyses, the failure probability is computed as if the complete ascent trajectory, from liftoff, ascent, through coast and final injection into the 600 n.m. circular orbit had taken place.

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The command to initiate the separation sequence of the RTG/RB-CIR from the spacecraft may be generated by either the MDS or by ground command control at any time after ejection of the payload fairing. The MDS will initiate separation commands in the event of an impending catastrophic or out-of-tolerance condition in the booster. For example, the MDS will detect overrates caused by loss of attitude control and to detect impending propellant tank rupture because of overpressure. The MDS system developed for and utilized in the initial Titan III flights has been used for this study. This system should be modified for compatibility with CRONUS applications:

- (1) Addition of an attitude sensing system to signal abort if the launch vehicle deviates more than three degrees from its normal flight path, thus precluding the possibility of aborting from an undesirable attitude.
- (2) Deletion of the separation system firing time delay circuit which allows the crew to prepare for separation in the Gemini application. (Such a delay would result in excess deviation from planned flight; thus is undesirable.)

Ground commands will initiate separation if the desired orbit is not attained, or if a slow malfunction, not detectable by the MDS, should occur. These ground commands can also backup the MDS system in a wide class of malfunctions.

Although an abort command may be initiated by the MDS, this initiates only the spinup and separation systems. All deorbit signals will be initiated from ground control. By initiating deorbit signals from the ground, a wider latitude in the selection of impact areas will be obtained; i.e., in the event that a land impact might occur if deorbit were to immediately follow separation, deorbit can be delayed until an ocean impact area is available. However, deorbit should be initiated as soon as possible after separation to reduce the time-dependent probability of failure of other critical systems and to minimize angular thrust vector deviations due to precession of the spinning CIR/RB. Provisions have been made for five orbits recall capability.

The reliability analysis for the short orbit mission is based on a mission duration of 4012 seconds. This implies that a decision to abort the mission (separate and deorbit the CIR/RB) will be made,

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if necessary, in this period which includes 3832 seconds between ignition of the launch vehicle first stage and circular orbit injection by the transfer stage, and 180 seconds to ascertain that the correct circular orbit has been attained. There appears to be no valid reason to return a Sr-90 fueled RTG to earth once a long lifetime orbit has been established.

For any malfunction or flight deviation resulting in an abort during the launch phase--up to the point of separation of the aerodynamic fairing, the CIR/RB systems cannot be initiated. Thus, the CIR/RB would come down with the booster stages and the payload. If mission abort is required after fairing separation, the CIR/RB is separated from the spacecraft immediately. The CIR/RB deorbit phase command would then be ground command initiated (if necessary) to assure reentry impact into deep ocean areas.

While it is desirable to also initiate deorbit of the CIR/RB within the 4012 second period, the requirements for adequate ground-to-CIR communications and tracking access at a time consistent with the achievement of a satisfactory impact area, may necessitate deorbit signal delay periods of up to several orbits. The effect of this delay will be to slightly increase the probability of failure of a critical deorbit system component prior to delivery of the deorbit signal. Of course, the decrease in the CIR/RB reliability still will be only negligible. The requirement to extend the deorbit decision period beyond the 3 minutes following completion of the final transtage burn will be determined as a result of detailed evaluation of the ground TTC net and abort trajectories during Phase I.

Despinning the RTG/RB after firing of the deorbit rockets is accomplished by two despin rockets which will reduce rotation of the RB to approximately zero revolutions per minute, permitting rapid aerodynamic orientation and stabilization of the RB during initial reentry. The solid propellant rocket-igniter systems are highly reliable; therefore, the probability of one despin rocket failing is extremely small. However, should one rocket fail to ignite, the RB will still impact within an acceptable dispersion pattern.

3. System Description

The airborne system is comprised of two systems, a reentry body which contains the RTG and the controlled intact reentry system (CIR).

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These two systems are latched together during normal spacecraft operation and also provide a protected passage and structural support for RTG electrical power connections; otherwise, they are independent. The RTG reliability already has been discussed (Section B).

a. Reentry Body

The Gemini-shaped RB is comprised of the RTG radiator, the aft cap and forward elastomeric silicone ablative heat shield. The asymmetric configuration provides orientation of the RB with the heat shield forward upon reentering the atmosphere. Unless a major structural failure occurs to change the aerodynamic characteristics of the RB, the body is self-stabilizing on reentry and will adequately recover from very high tumbling rates. The reliability of the reentry body is a function of its structural integrity; therefore, it is inherently reliable within the limits of the operational environment and conditions for which it was designed and qualified. The C-band beacon and battery in the aft cap of the RB is only a tracking aid; therefore, it does not contribute to the achievement of the reliability goals per se.

b. Controlled Intact Reentry Subsystems

The CIR subsystems and components whose malfunction may jeopardize the safety of the system are as follows:

- Support structure--main frame
- Spinup system--spinup bearing, two solid rocket spinup motors, and an ordnance separation device
- Ejection system--solid rocket motor for lateral acceleration, bearings and track
- Time delay (sequencing) devices
- Deorbit propulsion system--four solid rocket motors
- Electrical circuits between components of the CIR system
- Despin and deorbit separation system--latch fingers and separation band, a release clamp (ordnance) and two solid rocket despin motors
- Command System--two independent systems
- Secondary electrical power system--(redundant since RTG supplies primary power)

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The component reliability of ordnance system components is based on vendor data and correlated data from the Navy's reliability data reference source FARADA (Failure Rate Data). Reliability data for electronic and electrical components was obtained from FARADA, MIL Handbook-217, and from data developed during the Titan III and Gemini programs.

Solid propellant rocket data (no igniter) is based on the high reliability repeatedly demonstrated by operational solid propellant rockets over the last 20 years. In view of the over-design aspects of the reference configuration rockets, in both safety factors and the use of state-of-the-art technology, and because of the extensive qualification and operational experience with these rockets (or rockets of almost identical size and configuration), it is believed that the basic rocket reliability of 99.99% is realistic--it is emphasized that this reliability does not include the rocket igniter. Redundancy is provided for components critical to the separation and deorbit functions; e.g., command receivers and ordnance devices such as squibs, delay switches, wiring, clamps, battery and initiators.

In addition, operation of only one of two spin-up rockets is essential to successful mission abort and the two despin rockets are essential only under the most adverse reentry conditions.

The reliability data for mechanical and structural components is based on Martin experience with similar systems.

4. Reliability Analysis

A reliability block diagram and math model of the overall CIR/RB system is given in Figure VI-6. Since the RB is designed to withstand environmental conditions for all kinematically feasible reentry conditions no environmental excursion need be considered. For this study, the structural reliability of both the RB and CIR are included in the Structures block of Figure VI-6. The individual major subsystem block reliabilities shown in Fig. VI-6 were computed using the individual subsystem block diagrams and math models shown in Figure VI-7 through Figure VI-10. The results of the RTG reliability evaluation (see Section B) have been combined with the battery system reliability and included in the individual CIR subsystem evaluation models. The probability of a controlled reentry is calculated for each of the command methods (MDS) and ground command). The reliability of the Titan III MDS system is .9994.

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$$R_{CIR} = R_{ST} \times R_{EL} \times R_{PO} \times R_{SEP} \times R_{DEO} \times R_{EN}$$

+ Ground Command

++ MDS Command

* Based on probability of either despin rocket firing resulting in a wider landing dispersion pattern

** Based on probability that both despin rockets fire

1. R_{CIR} = .9935 Probability of ground command reentry - both despin motors fire
2. R_{CIR} = .9942 Probability of ground command reentry - one despin motor fires
3. R_{CIR} = .9931 Probability of MDS command reentry - both despin motors fire
4. R_{CIR} = .9939 Probability of MDS command reentry - one despin motor fires

Figure VI-6. Probability of Controlled Intact Reentry

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It can be seen from Figure VI-6 that, if an abort condition occurs after fairing release, the minimum reliability of the controlled intact reentry system (including the separation commands and the RB) is 0.9931. While a large number of redundant items contribute to this high reliability, the bulk of these redundancies involve very low weight items. Therefore, although some excessive redundancy may exist, it is believed that a studied campaign to reduce redundancy will have little noticeable effect on overall weight reduction. The rechargeable battery is perhaps the only component of interest with respect to elimination.

Again, it is noted that the high probability of successful controlled intact reentry, refers only to the primary goal that the probability of injury to the population be less than 10^{-6} . Unlike the CIR, the RB is effectively a dead, inert structure which must survive large dynamic loads, high heating rates and large total heat loads during reentry, and high impact loads at splash down. Thus, it is believed that the requirement for a low probability of dispersal of the fuel in the atmosphere will be satisfied through the wide range of abort conditions for which the CRONUS RB was designed. For example, the RB is designed for the extreme reentry conditions imposed by either a decaying orbit (max. total heat) or a deorbit from apogee of the 600 n.m. orbit (max. peak heating rate). The RB can also survive initial reentry spin rates as high as 600 degrees/second. On these bases, the probability of a major structural failure of the RB during reentry is expected to be less than 10^{-7} .

The controlled intact reentry reliability values reflect success based on the probability that critical equipment necessary to this operation works when called upon. Though failures of this critical equipment do occur, controlled intact entry may still be possible and/or non-dispersal of fuel in the atmosphere may still be assured. Some CIR in flight failure modes are shown in Table VI-5. The probability of each of these failure modes occurring will be made in subsequent studying phases. Since some of these failure modes are critical only if another failure occurs, they are credible only in the event of a double failure and will be evaluated accordingly. Each mode of failure will be evaluated in order to provide criteria for establishing deorbit decisions and guidelines for the range safety officer.

The preflight failure modes in Table VI-6 may cause catastrophic failure if not detected prior to launch. The reliability evaluation of these failure probabilities will be investigated in depth during development Phases I and II to provide redundancy or failure-safe circuits to preclude a preflight failure that could contribute to a hazardous situation.

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TABLE VI-5

CIR IN-FLIGHT FAILURE MODES

<u>Failure Mode</u>	<u>System Effect</u>	<u>Mission Effect</u>
	MDS*	
Failure to Sense Tank Overpressure	Booster Failure	Uncontrolled reentry. Overpressures and/or fragmentation may damage reentry systems (if hit by heavy fragments, and CIR protection fails)
Failure to sense impending structural breakup	1. Booster Failure and explosion 2. Booster Failure (w/o explosion)	1. Uncontrolled reentry Damage to RB is possible. 2. Uncontrolled reentry
Failure to Sense Tumbling	Delay in Initiating Abort	Possible uncontrolled reentry
	<u>Spin Up System</u>	
Pin Puller	Failure to fire or disconnect	Will be sheared but time to clear rails and separation velocity will be slightly longer and lower, respectively.
	Premature Ignition Signal	CIR free to roll off rails, Mission prematurely aborted.

*The MDS is backed up by a ground command capabilities. The ensuing lists of malfunctions assume that neither systems has operated.

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TABLE VI-5 (Cont'd)

<u>Failure Mode</u>	<u>System Effect</u>	<u>Mission Effect</u>
Spin Rockets	<ol style="list-style-type: none"> Both fail to ignite CIR/RB not spin stabilized One fails to ignite Partial spin up 	<ol style="list-style-type: none"> CIR/RB unstable and de-orbit thrust vector misaligned--uncontrolled reentry potential* burnup No ill effects unless thrust misalignment and/or tip-off rates unusually large. In that case, possible uncontrolled reentry.
Spin Bearing	Partial spin up (only after track)	No ill effects unless thrust misalignment and/or tip-off rates very large. In that case, possible uncontrolled reentry.
	<u>Ejection System</u>	
Separation Rocket	<ol style="list-style-type: none"> Non-ignition CIR/RB will not separate Premature Ignition Spin up rockets ignite and separation occurs 	<ol style="list-style-type: none"> Uncontrolled reentry potential burnup* Premature mission abort
Bearings	Binding in separation track prevent or hinder spin up and/or separation	Uncontrolled reentry potential burnup

*By the term "potential burnup" it is meant that serious structure damage to the RB is possible if other highly improbable malfunctions occur.

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TABLE VI-5 (Cont'd)

<u>Failure Mode</u>	<u>System Effect</u>	<u>Mission Effect</u>
Separation guide rails	<ol style="list-style-type: none"> 1. Binding, prevents separation 2. Crippling, causing change in orientation 3. High Friction 	<ol style="list-style-type: none"> 1. Uncontrolled reentry 2. Impact area dispersion 3. No noticeable effect
<u>Deorbit Propulsion System</u>		
Retro Rockets	<ol style="list-style-type: none"> 1. All fail to ignite R/B not despun; R/B CIR decay normally 2. One fails to ignite Potential insufficient vector impulse and probable tumbling 3. Premature <ol style="list-style-type: none"> a. Termination of mission if prior to separation b. Normal abort if after separation 	<ol style="list-style-type: none"> 1. Uncontrolled reentry 2. Possible uncontrolled reentry, potential burnup <ol style="list-style-type: none"> a. Uncontrolled reentry & potential damage to RB b. Uncontrolled reentry
Despin Rockets	R/B reenters with maximum spin rate or tumbling	Impact dispersion increases
<u>Deorbit Separation System</u>		
Separation clamps (2)	CIR/RB both reenter Redundant clamp bolts Fail to separate on retainer band	Separation initiates automatically at atmospheric entry. Remote degree of damage to heat shield.

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TABLE VI-5 (Cont'd)

<u>Failure Mode</u>	<u>Systems Effect</u>	<u>Mission Effect</u>
Band Bolts	<p>Premature firing of retainer CIR/RB separate</p> <ol style="list-style-type: none"> 1. During ascent 2. During separation 3. After separation, prior to deorbit 4. During deorbit 5. After deorbit 	<ol style="list-style-type: none"> 1. Loss of mission, uncontrolled reentry 2. Potential burnup, if system damaged by L/V. Uncontrolled reentry 3. Uncontrolled reentry 4. Potential uncontrolled reentry 5. Increase vs. dispersions area
	<u>Structural</u>	
Retro Rocket Support	Loss of number of retro rockets affected by failure	If two adjacent supports fail and failure not detected before retrofire potential tumbling and burnup
Spin rocket support	Partial Spinup	<p>No ill effect unless thrust misalignment and/or tip-off rates unusually large. In that case, possible uncontrolled reentry.</p> <p>Possible damage to CIR and RB if thrusting motor tears loose.</p>

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TABLE VI-5 (Cont'd)

<u>Failure Mode</u>	<u>Systems Effect</u>	<u>Mission Effect</u>
Despin rocket support	Maybe spin stabilized at reentry	Increased dispersions possible.
Locking pin (bearing carriage positioning pin)	Premature separation	Mission failure
Carriage bearing support	Possible inability to separate or incorrect orientation	No control of reentry area
Fairing door actuator or actuator arm	Forward door open during reentry	Burnup on reentry
Attachment fingers between CIR and RTG/RB	Fingers containing instrumentation and power connectors do not separate	Deorbit package may not separate until start of aero heaters. Then OK
<u>Command Receivers (2)</u>		
Signal (non receipt of failure to process)	Separation cannot be initiated	Uncontrolled reentry
Signal Erroneous	Separation	Loss of Mission
Short	<u>Power Supply</u>	
Short Circuit, Power Busses	No power to activate abort system	Uncontrolled reentry Potential burnup
RTG Failure and battery failure	No power to activate abort system	Uncontrolled reentry Potential burnup

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TABLE VI-6

CIR PREFLIGHT FAILURE MODES

<u>Failure Mode</u>	<u>System Effect</u>	<u>Mission Effect</u>
Fairing door not latched, false indication	Flight begins with door unlatched	Door may open in flight and cause vehicle breakup
All latches on fuel block loading door do not latch, correct indication	Takeoff delayed	Vehicle must be defueled for repair
Failure of door actuator to operate	Takeoff delayed	Vehicle must be defueled for repair
RB fuel block loading door unlatched, false indication	Door open in flight possibility	Burnup on reentry no aerodynamic stabilization
Link length does not permit door to latch	Takeoff delayed	Vehicle must be defueled for repair
Link does not detach from door	Fairing cannot completely separate	May cause vehicle breakup
RB door unlatched, correct indication	Takeoff delayed	Vehicle must be defueled for repair

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$R_{ANT} = .9998$ $R_{RCV1} = R_{RCV2} = .9978$ $R_{RTG} = .999999$ $R_{LVS} = .99994$ $R_{MDS} = .9994$
 $R_{BATT} = .99967$ $R_{FS} = .999989$ $R_{SS} = .9999$

Ground Command

$$R_{PGC} = R_{ANT} \times (1 - (1 - R_{RCV})^2) \times 1 - (1 - ((R_{RTG}) \times (1 - ((1 - R_{FS}) (1 - R_{SS}))^2))) (1 - ((R_{BATT}) (R_{LVS}) \times (1 - ((1 - R_{FS}) (1 - R_{SS}))^2))) = .999749$$

MDS Command

$$R_{MDSC} = R_{MDS} \times (1 - ((R_{RTG}) \times (1 - (1 - R_{FS}) (1 - R_{SS}))^2)) (1 - ((R_{BATT}) (R_{LVS}) \times (1 - ((1 - R_{FS}) (1 - R_{SS}))^2))) = .999398$$

Figure VI-7. Probability of Initiating Command to Reenter

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$$R_{SEP} = (R_{IC})^4 \times (1 - (1 - R_{PC})^2) \times (R_{PP}) \times (1 - (1 - R_{SQ})^2)^3 \times (R_{IG})^3 \times (R_{SR}) \times (R_{SUR})^2$$

$$R_{IC} = .9997$$

$$R_{PC} = .999$$

$$R_{SQ} = .999$$

$$R_{PP} = .9999$$

$$R_{IG} = .999999$$

$$R_{SUR} = .9999$$

Figure VI-8 Probability of Separation and Stabilizing R/B (R_{SEP})

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$$R_{DEO} = (1 - ((R_{PT}) \times (1 - (1 - R_{SS})^2))^2) \times (R_{CI})^4 \times (1 - (1 - R_{SQ})^2)^4 \times (R_{IG})^4 \times (R_{DER})^4$$

$$R_{PT} = .999999$$

$$R_{SS} = .9999$$

$$R_{CI} = .9997$$

$$R_{SQ} = .999$$

$$R_{IG} = .999999$$

$$R_{DER} = .9999$$

Figure VI-9. Probability of Deorbit

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$$R_{EN} = (1 - (1 - R_{PS})^2) \times (R_{CI})^2 \times (1 - (1 - R_{SQ})^2)^2 \times (R_{IG})^2 \times (R_{DE})^* \times (1 - (1 - R_{PS2})^2) \times (1 - (R_{CI}) \times (1 - (1 - R_{SQ1})^2) \times (R_{SB}))^2$$

$(R_{DE})^* = (R_{DE})^2$ for probability both despin rockets fire
 $= (R_{DE})$ for probability one despin rocket fires

$R_{SQ} = .999$ $R_{IG} = .999999$ $R_{SB} = .999$ $R_{PS} = .999$ $R_{PS2} = .999$ $R_{CI} = .9997$

Figure VI-10. Probability of Stabilized Reentry

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D. GROUND HANDLING EQUIPMENT

1. Reliability Goals

The reliability analyses for this feasibility study are directed toward two separate goals:

- (1) a safety goal to ascertain the feasibility of designing and developing a fuel ground handling system that would have a probability of less than 10^{-6} of releasing radioactive fuel resulting from failure of the GHE.
- (2) an operational goal to assure design compatibility for providing a reliable and efficient fueling system that can be integrated with the launch facilities and count-down procedures.

2. Transportation Reliability

In the conduct of this analysis, various operations were examined and the most severe credible accidents were investigated in each case. The analysis performed for accidents related to shipping the fuel (within its shielded shipping container) by truck, railroad and air transportation are presented in Chapter V. The general conclusions were that the probability of fuel release during any one shipment of the heat source by truck and rail would be less than 1×10^{-6} . This was based on fuel capsule strength, probability of an accident and the probability that such an accident would be severe enough to cause fuel capsule rupture or melt-down.

3. Reliability Analysis Techniques

The ground handling equipment employed at the launch site for installing the fuel block prior to launch consists of the storage and transport cask and its actuating subsystems, the fuel handling equipment including components such as the fuel handling machine consisting of the fuel probe, compliance mechanism, drive motors, structure, winches, pneumatic lines, etc. The equipment has redundancy and fail safe features incorporated that improve the probability of a successful fueling-defueling operation. This preliminary GHE analysis considers only the probability of the GHE functioning as designed and does not include predictions

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of the probabilities of human error or other launch pad facilities malfunctions. The fueling and defueling procedures will be documented in detail and personnel responsible for the fueling-defueling operation will be certified after a thorough indoctrination in the operation of fuel handling equipment to minimize this probability of human error.

A method of analysis for the launch site fueling and defueling is based on a Markov chain process. This method assumes that there are a sequence of steps or trials with the following properties. The outcome of each step is one of a definite number of possible outcomes. It is also assumed that the probability of any outcome for any given step is not necessarily independent of outcomes of previous steps but depends on the outcome of the immediately preceding step. It is assumed that the process begins in a particular state, then we have enough information to determine the diagram or tree for the ground handling process and can calculate probabilities of states relating to the overall sequence of the steps. Therefore, beginning with the initial state, R_0 i.e. the GHE has been checked out and the fuel block is in position in the cask, then either the fueling is performed successfully by the primary system or the primary system fails with some probability associated with each outcome. The apportionment of these outcomes to the next step is called a transition probability. If the primary system fails during fueling, fueling will be discontinued. Recovery may be completed by the secondary or redundant functions. The fail safe devices are provided and secondary or backup systems are used to return the fuel block to the cask. This method will be further developed in Phase I to establish a matrix of these states so various combinations may be evaluated. Utilizing this method of analysis, the attainment of the safety goal of one chance in a million (10^{-6}) of releasing fuel during a fueling or defueling operation appears feasible.

4. Reliability of Fueling and Defueling Performance

Reliability performance is an important facet in demonstrating the suitability of these fuel forms for spacecraft power application. The integration of new operations within an established launch countdown sequence represents a major work effort. When that new sequence involves radioactive materials, and the health physics controls that accompany it, the task is more complex. Acceptance of the revised procedures will be influenced by:

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- (1) The degree of failure-free operation that can be assured during RTG fueling (and defueling in event of abort).
- (2) The simplicity of operation.
- (3) The extent of radiation exposure anticipated.

The reference designs for the GHE and RTG were analyzed from this viewpoint.

Use of a remotely operated system eliminates operator of radiation exposure and the shielding requirements. A training program will qualify the operators in the operation of the remote fueling. Human factors design considerations in console layout, sensory aids and safety interlocks will minimize malfunctions at this man/machine interface. The degree of reliability at this interface is continually improved by periodic operator requalification, on-the-job training, and simulated fueling and defueling operations.

Due to the lack of adequate definition of all of the specific hardware (and thus their failure rates), a quantitative reliability prediction was not performed on the subsystem, except to illustrate the technique as illustrated in Figure VI-11. However, a failure mode and effects analysis was performed (see Table VI-7) and a qualitative analysis established the following:

- (1) Use of a single heat source assembly simplifies fueling, or defueling if required.
- (2) Adequate provisions for safe pickup and release of the fuel by the GHE are present.
- (3) Adequate provisions to accommodate relative motion between the launch tower and launch vehicle have been made. The capability of this system to function in event of loss of the air-bearing air supply will be demonstrated.

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- (4) Redundant and fail-safe features present malfunction recovery capabilities.
- (5) With the capability of simulating the fueling sequence during the readiness checks, the equipment will have a correspondingly higher probability of successful operation during the actual fueling operation.
- (6) A refurbishment kit for the launch site fueling equipment is necessary to return the equipment to a "like-new" state after each launch. This refurbishment is accomplished simultaneously with the refurbishment of the UT.

It is generally concluded that through proper safety margin and parts qualification testing, a system can be designed, fabricated and demonstrated to meet the goals set forth.

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TABLE VI-7

Ground Handling Equipment Failure Mode Analysis

<u>Failure Modes</u>	<u>System Effect</u>	<u>Outcome</u>
Failure of cask lid actuator on launch pad	Remote handling impossible	Flight delay until repair is made. No hazard.
Failure of ground power source (electrical)	Loss of all electrical systems	Heat source may be stalled between cask and RTG. No hazard. Delay until repairs made.
Failure of FHM motor on GHE boom	Impossible to position FHM	Flight delay until repair made. Possible radiation exposure problems to personnel if heat source cannot be removed remotely.
Release of fuel block between cask and RTG	Activation of the probe latches by a short circuit	Heat source could fall to ground. Net could be provided under boom.
Failure of boom cable*	Structural support	Hinged joint provides redundant structure. Boom could not be retracted.
Fall of cask and fuel block from tower *	Damage to cask and possibly fuel block	Tests will demonstrate reliability. Net may also be provided below boom.
Failure of alignment devices	Fuel block may be damaged	May damage generator insulation and prevent installation. Secondary sensors may be required to detect misalignment before lowering fuel block.
Failure of latches to release in cask	Fuel block cannot be installed in RTG	Load sensors will detect overload and stop hoisting before damage is done. Only double failure would cause damage.
Boom winch fails to retract boom	Boom would strike vehicle on liftoff if not detected	Delay of liftoff. Position sensors and remote TV will detect malfunction
Failure of air breathing compliance device	Possible damage to insulation on insulation of fuel block	Could prevent latching. Latching sensors would also have to fail for potential hazard.
Failure of source of pneumatic pressure	GHE systems inoperative	Fail safe devices will prevent release of fuel block
Failure of latches to hold fuel block*	Drop fuel block	Depending on interim position may drop to foot of launch tower or strike umbilicals. Devices may be installed to prevent free fall to ground.

*Asterisk denotes credible failures that might result in fuel release unless adequate precautions are taken.

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VII. DESIGN EVOLUTION

This chapter presents the logic in arriving at the selected design. In order for the reader to appreciate the reasoning behind CRONUS, he must employ the chart found in the envelope inside the back cover. This chart shows the design decisions made in arriving at the selected design. Alternates (dashed lines) are shown and also those decisions wherein other systems of CRONUS have an input (dashed line + arrow). The number in the circle beside the arrowhead indicates that a discussion of that decision appears in this chapter against the same number.

A fundamental decision was made to separate the power generation and reentry functions into two parts: the RTG/RB and the CIR. This decision was based upon the presumption that it represented an efficient division of functional responsibilities. Only those systems intimately needed for the generation of the raw electric power would be directly exposed to the thermal environment. Therefore, it would be possible to use much off-the-shelf hardware without the need for thermal protection devices.

Since the reentry body would now be stripped of all but the essentials, the weight of the heat shield and the aerodynamic heating of the radiator would also be a minimum. (The question "what constitutes the essentials of the RTG?" is answered subsequently.) Thus, the fundamental decision would save weight in the reentry body.

There was a temptation to also extend the weight saving activities to the portions of the CIR which need not be deorbited. An example of such a component is the telemetry package. However, it was felt that the weight saved in the deorbit rockets by partial staging (about 22% of the weight carried but not deorbited) would not be compensated for by the added complexity and loss of reliability. Notice that this argument does not really apply to the one extra separation sequence (RTG/RB from CIR) since that sequence should have a very high inherent reliability, and is further backed up by the tendency of automatic separation if a joined CIR/RB enters the high aerodynamic heating region.

This fundamental separation of CIR and RTG/RB functions might be revised if there is a need for pinpoint control of maneuvers of the reentry body after deorbit. However, since no such need has arisen, it is felt that this fundamental decision is correct.

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A. RADIOISOTOPE THERMOELECTRIC GENERATOR

101--Heat Transfer Between Heat Source and Thermoelectrics

The most basic decision that was made in the heat source design was to have a thermal radiation interface between the heat source and the thermoelectric subsystem. An interface is required at that point to permit the RTG to be fueled prior to launch. Other types of interfaces considered were solid and gaseous conduction, either of which would have made the fueling operation more complex, and would have lowered mission reliability. Solid conduction could be attained by inserting a smooth fuel block between spring-loaded module hot plates. This would require high contact pressures, and the thermoelectric subsystem would require large hot plates to exert the required contact pressure against the fuel block. Gaseous conduction would require that a conductive gas (such as argon) be sealed within the RTG during the remote fueling operation, and this has not been proved to be feasible.

The selection of radiative heat transfer allows the use of bare fuel capsules, instead of a fuel block, to radiate heat directly to the thermoelectric hot shoes. This is compatible with the selection of SiGe couples.

102--Heat Transfer Between Thermoelectrics and Radiator

The use of heat pipes was rejected early in the study because they had not been proved for long-term operation in space. Metallic conduction through a radiator fin was selected instead because it is static in nature and relies only on the basic properties of the radiator material.

103--Single Heat Source Selection

A combination of ground handling considerations and the selection of the Gemini reentry shape played the key role in the selection of the heat source configuration. The objective was a heat source that could be installed through the top of the reentry body. From a fueling standpoint, handling only a single heat source is preferable to handling two or more items. Consequently, a single cylindrical heat source was selected for its ease of assembly and handling, and for its symmetry of mass and heat flux about the central RTG/RB axis.

104--Small Fuel Capsules Versus Large Capsules

A safety objective that was established was to limit the probability of personnel exposure by reducing the types of hazards available and the number of persons subjected to these hazards. As a result, the primary self-imposed constraint for the heat source design has been containment of the fuel under normal operation and all probable accident conditions. This demands fuel capsule structural integrity under the effects of oxidation, corrosion, impact, pressure, thermal shock and heat blast.

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It is expected that the reentry body shell (radiator, heat shield, and aft cap) will withstand the direct effects of a launch pad abort and explosion, such as shock overpressure and fireball heating. However, there are secondary effects which may breach the integrity of the shell structure, such as reinforcement of the shock overpressure by multiple reflections, or by shrapnel penetration. These secondary effects are random in nature and cannot be adequately analyzed at this time. Until better definition can be achieved, the approach to design individual fuel capsules resistant to earth impact at their own terminal velocity was taken. In this way, the safety objective will be met under the most severe accident conditions.

This decision dictated the use of multiple fuel capsules rather than a single large one. An impact analysis was performed which showed that the encapsulant weight fraction is independent of capsule size as long as capsule configuration and impact velocity are held constant. It can further be shown that the terminal velocity of a free-falling capsule is proportional to the square root of its diameter as long as its density and geometrical configuration are held constant, and it remains in the same aerodynamic flow regime. This being true, smaller fuel capsules will have lower free-fall terminal velocities than larger fuel capsules, which will permit proportionately less encapsulant material to be used. This means smaller fuel capsules can be built lighter than large fuel capsules if these impact criteria are applied.

There are obvious limitations to shrinking the fuel capsule size. One is the selection of workable material thicknesses and gages. Another is that if smaller capsules could be made with thinner walls, at some point they become subject to rupture by crushing or penetration rather than by impact.

A definite limitation was established due to geometrical considerations. It was desirable to have the heat source as a single assembly for handling purposes, yet a large envelope area was desired to minimize fuel capsule operating temperatures. Once the heat source configuration was established (single heat source assembly in a cylindrical geometry), the fuel capsule length was set at 13.12 inches to provide a heat source envelope area of 6.15 ft^2 for heat transfer purposes. In considering the fuel volume required, the heat source diameter of 21.5 inches and the capsule length (including skirts) of 13.12 inches, it was found that the maximum number of capsules that could be placed in a single circular array was between 37 and 38. The number of capsules was selected to be 36, which set the outer capsule diameter at approximately 1.5 inches and allowed a slight clearance between adjacent capsules. The resulting L/D ratio for the capsule is approximately 7.7 (not counting the skirt at each end), which is somewhat above the range (4 to 6) that is normally considered to be optimum for impact

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resistance. However, neither terminal velocity nor impact resistance are sensitive to small deviations above that range, and the reduction of impact resistance is felt to be minor.

The use of two circular arrays of capsules, one above the other, was considered in order to reduce the L/D ratio. This was found to increase total capsule weight and to require an additional structural member between the two arrays. Because of the interaction between the upper and lower arrays, an impact analysis of the complete re-entry body would be difficult to perform and a low level of confidence in the calculations would result. The single array of capsules having a higher L/D was therefore retained as the more suitable of the two concepts.

105--Fuel Capsule Geometry Selection

Spherical and cylindrical fuel capsule configurations were considered. Spherical capsules have the potential advantage of requiring less encapsulant material per pound of fuel to resist impact and internal and external pressure. They are currently being studied, but fabrication techniques have not been developed and impact data are not yet available. The cylindrical shape was selected because experience and test data exist for this configuration.

106--Fuel Capsule Material Selection

Several high temperature materials were considered for encapsulation of the radioisotope fuel. Among these was Haynes-25, a solution heat-treated cobalt-base alloy; TD (thoria dispersed) nickel, a dispersion strengthened alloy, and D-31 niobium (columbium) alloy, which is representative of a growing family of refractory alloys. Precipitation-hardening alloys such as Rene 41 can achieve high strength at elevated temperatures to 1800° F, but they were rejected because of the difficulty of heat treating welded capsules at the fueling facility.

a. Impact resistance

The impact resistances of Haynes-25, TD nickel, and D-31 alloy were calculated based on published mechanical properties (Refs. VII-1, VII-2, VII-3). The results are shown in Fig. VII-1 as a plot of specific energy absorption, u_a , with respect to temperature for the three materials.

As shown, u_a is higher for Haynes-25 than for the other materials between room temperature and 1800° F. TD nickel has an especially low ductility (6% elongation at 1600° F). As a result, a Haynes-25 capsule designed for impact resistance will be lighter in weight than one made of either TD nickel or D-31 alloy.

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Fig. VII-1. Specific Energy Absorption for Haynes-25, TD Nickel and D-31 Alloy

Fig. VII-2. Encapsulant Weight Fraction as Function of Median Failure Velocity

b. Weight comparison

A weight comparison has been made for a fuel capsule temperature of 1600° F, which is representative of a launch abort situation. The comparison has been based on the encapsulant weight fraction, w_c , as a function of the median failure velocity, V_{50} , as is illustrated in Fig. VII-2. For example, if V_{50} is selected to be 250 fps based on terminal velocity considerations, then w_c for a Haynes-25 capsule is 0.35, and that for a D-31 capsule is 0.58. This gives a total fuel capsule weight of 3.12 pounds for that using Haynes-25, and 4.84 pounds for D-31.

c. Chemical properties

(1) Chemical composition

<u>Haynes-25 Alloy</u>							
<u>Si (%)</u>	<u>Cr (%)</u>	<u>Ni (%)</u>	<u>W (%)</u>	<u>Mn (%)</u>	<u>Fe (%)</u>	<u>C (%)</u>	<u>Co (%)</u>
0.10 to 0.20	19 to 21	9 to 11	14 to 16	1 to 2	2.5 to 3	0.05 to 0.15	Balance

<u>TD Nickel</u>	
<u>Thorium Oxide (%)</u>	<u>Balance</u>
2	Nickel

<u>D-31 Niobium Alloy</u>		
<u>Mo (%)</u>	<u>Ti (%)</u>	<u>Nb (%)</u>
11	11	Balance

(2) Melting point (°F)

D-31 niobium alloy	4100
TD nickel	2650
Haynes-25 alloy	2570 (max)

~~CONFIDENTIAL~~(3) Density (lb/in.³)

D-31 niobium alloy	0.292
TD nickel	0.322
Haynes-25 alloy	0.330

(4) Thermal conductivity (Btu/ft²/in./hr/°F at 70° F)

TD nickel	600
D-31 niobium alloy	300
Haynes-25 alloy	65

(5) Thermal expansion (in./in.°F) (70° to 1000° F)

TD nickel	8.5×10^{-6}
Haynes-25 alloy	8.3×10^{-6}
D-31 niobium alloy	2.95×10^{-6}

(6) Elongation and tensile and rupture strength

	<u>Elongation (% , 2 in.)</u>		<u>Maximum Temperature</u>	
	<u>Room Temperature</u>	<u>1200° F on Pad (air)</u>	<u>1875° F Orbit (vacuum)</u>	<u>2100° F Reentry (air)</u>
Haynes-25 alloy	64	35	38	21
D-31 niobium alloy	10	10.5	30	70
TD nickel	22	10	5	3
	<u>Tensile Strength (psi ultimate)</u>			
Haynes-25 alloy	145,000	103,000	28,000	10,500
D-31 niobium alloy	75,000	58,000	25,000	20,000
TD nickel	80,000	30,000	20,000	16,000

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Rupture Strength (1000 hr stress-rupture (psi))

	Room Temperature	Maximum Temperature		
		1200° F on Pad (air)	1875° F Orbit (vacuum)	2100° F Reentry (air)
TD nickel	--	20,000	6,000	4,000
D-31 niobium alloy	--	41,000	10,000	5,000
Haynes-25 alloy	--	40,000	3,500	1,000

(7) Crystal structure

TD nickel	Simple cubic
D-31 niobium alloy	Base centered cubic
Haynes-25 alloy	Face centered cubic

(8) Electrical resistivity ($\mu\Omega$ -cm) (70° F)

TD nickel	7.6
D-31 niobium alloy	~20.0
Haynes-25 alloy	88.0

(9) Oxidation resistance

Haynes-25 alloy. The oxidation resistance of Haynes-25 alloy is good up to 1900° F.

TD nickel. At 2000° F in static air, TD nickel has a specific weight gain of 10.5 mg/cm² after 100 hours. The TD nickel oxidizes at about the same rate as pure nickel (between 1800° and 2400° F) and follows a parabolic rate law over this temperature interval. The scale formed in air is relatively adherent to the base metal and is not subject to severe spalling when subjected to thermal cycling.

D-31 niobium alloy. Although the alloying elements of titanium and molybdenum give D-31 about four times more oxidation resistance than niobium alone, the alloy still must be coated for protection at higher temperatures. One disadvantage of niobium base alloys is that without superior coatings, the niobium is subjected to internal oxidation under elevated temperature conditions.

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(10) Vacuum resistance

D-31 niobium alloy. Under orbital conditions (vacuum) at 1875° F, volatilization of D-31 constituents is not expected to be a problem. Titanium is the only constituent which may undergo noticeable sublimation. Test data are not available.

TD nickel. At present, volatilization data for TD nickel at 1875° F are not available. However, it is known that unalloyed nickel will vaporize in a vacuum at a considerably lower temperature than 1875° F. At this date, available coatings for TD nickel are still in the developmental stage.

Haynes-25 alloy. It has been established that in the high vacuum encountered in space, Haynes-25 alloy will be subjected to vaporization losses of certain alloying constituents, notably chromium. An experiment conducted at Martin determined the rate of vaporization of unoxidized Haynes-25 alloy at 1750° F (1×10^{-5} torr vacuum) (Ref. VII-4). The investigation was carried out to predict the amount of the SNAP 17A fuel capsule that would be lost due to vaporization in a five-year mission. The resulting data are tabulated.

<u>Haynes-25 Alloy</u>	<u>Time of Run (hr)</u>	<u>Loss (gm/cm²/hr)</u>	<u>Predicted Five-Year Loss (gm/cm²)</u>	<u>Estimated Penetration (in.)</u>
1st run	30	0.0000183	0.80	0.033
2nd run	30	0.0000044	0.19	0.008
3rd run	64	0.0000025	0.11	0.005

The same specimen was used in all three runs, indicating that chromium with its higher vapor pressure is being preferentially vaporized in relation to other less volatile alloy constituents.

Three potentially promising coatings have been developed by General Electric for Haynes-25 alloy and are:

- (1) Iron titanate (plasma arc sprayed)
- (2) Calcium titanate (plasma arc sprayed)
- (3) Iron titanate (60%) and lead bisilicate (40%).

A comparison of these three coatings, as reported by General Electric, follows (Ref. VII-5).

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Iron titanate was plasma arc sprayed on Haynes-25 alloy at a thickness of 2.5 mils. The coating was then tested for emittance after 90 hours at 1200° F in a vacuum of 10^{-7} torr. The emittance results are tabulated.

	<u>Time (hr)</u>	<u>Temperature (°F)</u>	<u>Total Emittance</u>
Initial	0	1204	0.907
Final	90	1201	0.898

The initial and final emittance measurements are so close as to be within the range of experimental error.

Calcium titanate was plasma arc sprayed on Haynes-25 alloy with a thickness of 2.5 mils. Total hemispherical emittance values were the same after zero and 66-1/2 hours as shown in the tabulation, although the coating had darkened in appearance at temperatures slightly over 1400° F.

	<u>Time (hr)</u>	<u>Temperature (°F)</u>	<u>Total Emittance</u>
Initial	0	1405	0.902
Final	66-1/2	1423	0.901

Pratt and Whitney also checked total emittance measurements of plasma-sprayed calcium titanate coated on Haynes-25 alloy. The results follow.

	<u>Time (hr)</u>	<u>Temperature (°F)</u>	<u>Total Emittance</u>
Initial	0	1208	0.735
Final	42-1/2	1390	0.814

Pratt and Whitney's work illustrates that the emittance of calcium titanate gave a definite increase with increased temperature.

Iron titanate 60%-lead bisilicate 40% was applied to Haynes-25 alloy and initial emittance measurements of 0.763 were recorded at 1208° F. As the temperature gradually increased to 1502° F, the total hemispherical emittance after 119-1/2 hours was 0.907

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	<u>Time (hr)</u>	<u>Temperature (°F)</u>	<u>Total Emittance</u>
Initial	0	1208	0.763
Final	119-1/2	1502	0.907

These tests represent effective coating results on Haynes-25 alloy. Under the long expected five-year life of the 250-watt(e) RTG, it is not unreasonable to believe that either the calcium titanate coating with lead bisilicate or the calcium titanate alone on Haynes-25 would remain protective. Nevertheless, further long time, high temperature coating life data are needed. It should be added that besides protecting chromium from volatilization, such coatings also prevent other alloying constituents such as nickel and cobalt from vaporizing. Vaporization rates for the major Haynes-25 alloying constituents are tabulated for the uncoated alloy exposed to a partial vacuum (constant pumpdown) at 1750° K.

<u>Element</u>	<u>Vaporization Rate (gm/cm²-hr)</u>
Chromium	1.7×10^{-5}
Nickel	0.080×10^{-5}
Cobalt	0.015×10^{-5}
Tungsten	None observed

In final summary, the data on coatings recorded here are favorable to Haynes-25 alloy vacuum (space) condition resistance. (GE subjected these coatings to severe thermal cycling tests before releasing them for emissivity tests.)

d. Summary and conclusions

The encapsulant weight fraction, w_c , and the absorption efficiency factor, K , were established as criteria for impact resistance optimization. An expression including these factors was developed which can be used to determine the effects of encapsulant material, impact temperature and velocity on the capsule design.

This expression was used to compare capsules made of Haynes-25, TD nickel, and D-31 alloy on the basis of system weight. It was determined that the use of D-31 as an impact resistant structure increased the system weight substantially over that using Haynes-25. TD nickel was found to be totally unsuitable for high velocity impact resistance.

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The expression was also used, along with appropriate data from the SNAP 17A test program, to determine the CRONUS fuel capsule wall thicknesses and weights. This difference of 1.62 pounds per fuel capsule would be reflected as a 75-pound difference in the airborne system weight. The difference would be even greater for higher values of V_{50} .

Note that TD nickel is totally unsuitable for high velocity impact resistance. The 250-fps failure velocity selected in the example is substantially higher than the critical velocity for TD nickel (that velocity at which $w_c = 1.0$), which is 139 fps. This means that even a solid bar of TD nickel would rupture if it were heated to 1600° F and propelled at 250 fps to impact with a hard, unyielding surface such as a granite block.

107--Fuel Pellet Production Technique Selection

The fuel pellet production technique was selected on the basis of overall system weight. The most common production technique for fuel capsules of this size is to cold press and sinter the fuel pellets; load them into a secondary capsule or liner; weld the end cap onto the liner; decontaminate; load the liner into the primary capsule; weld and decontaminate again. The use of secondary encapsulation is mainly for decontamination purposes. If cold pressed and sintered fuel pellets were loaded into the primary encapsulation without a liner, it would be unreasonably costly and time consuming to scrub the capsule weld zone to meet the Bureau of Licensing contamination requirement of $0.05 \mu\text{c}/100 \text{ cm}^2$.

It has been suggested by ORNL personnel* that hot pressed pellets using a platinum foil separator would be lighter than cold pressed pellets in a liner. Hot pressing is usually used to produce larger pellet sizes (2.5-in. diameters and up). Here the platinum separator is used as a handling aid and to prevent contamination of the weld zone in the primary capsule.

In comparing the weights of the two concepts, it was found that the platinum separator weighs 60% less than a comparable refractory liner having a minimum gage thickness of 0.020 inch. Since the hot pressed fuel is typically 10% denser than cold pressed and sintered fuel, the primary encapsulation can be made smaller in diameter. This was partially accounted for in the comparison by using the same fuel density, but assuming that the platinum separator around the hot pressed pellets occupied zero space.

*In a conversation between D. O'Brien of Martin Nuclear and Messrs. Lamb and Schaich of Oak Ridge National Laboratory, 9/14/66.

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The results of the weight comparison are tabulated.

<u>Item</u>	<u>Fuel Weights (lb)</u>		<u>Reduction (%)</u>
	<u>Cold Pressed and Sintered</u>	<u>Selected System Using Hot Pressed</u>	
Fuel capsule	4.69	4.01	14.5
Heat source	206.2	180.4	12.5
RTG	358.6	329.0	8.3
Airborne system	765.5	724.0	5.4

108--Heat Source Thermal Insulation Selection

Various types of thermal insulation were considered for use in the heat source. The end rings inside the support caps are structural in nature. They support the weight of the fuel capsules during the launch environment, staging, orbital maneuvers, separation and deorbiting and reentry. During a 50 g axial acceleration, the compressive stress in the insulation was calculated to be 30 psi at the support cap interface and 60 psi at the interface with the capsule retaining band. Previous experience with Min-K 2000 (20 lb/ft³ density) shows its approximate upper limit in compressive stress to be 100 psi, which is adequate for the stresses calculated.

However, the temperature limitation for Min-K 2000 is approximately 1400° F in a structural application. Above this temperature the insulation will warp, crack and sinter. The interface temperature between the capsule retaining band and the insulation is approximately 1750° F at beginning of life.

Two approaches could be taken which would allow the excellent insulating properties of Min-K 2000 to be used. One approach would be to insert foamed or corrugated ceramic inserts between the capsule retaining band and the Min-K 2000 to spread the compressive forces over a larger area of the Min-K and to lower the interface temperature between the ceramic and fibrous insulations to below 1400° F. The other approach is to utilize the advanced fibrous insulation which is presently being developed and which will have an operating temperature limitation in excess of 1800° F.

The ceramic insulation approach has previously been proposed for use in SNAP 17A (Ref. VII-6). Two types of ceramic insulation have been investigated during this study, but thermal conductivity data are

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not available for either product in a vacuum environment.* The weight penalty associated with the ceramic insulation approach plus the probable (although unknown) increased conductance over the fibrous insulation makes this method the least desirable of the two. The second approach, which uses only advanced fibrous insulation, was selected because of the performance increase that can be attained and because of its high probability of success.

109--Thermoelectric Element Selection

The study presented in Appendix L was performed to compare a SiGe versus a PbTe SrTiO₃ fueled generator at the 250-watt(e) level based on the following indexes: weight, size, efficiency. A right circular cylinder with six fins attached to a 10-inch diameter housing was used as the model for both systems. A radiative interface was assumed to exist for both systems between the module and the heat source in order to simplify remote fueling requirements with Sr-90.

The following conclusions were reached:

- (1) On a weight basis, the SiGe generator at 239 pounds is approximately 100 pounds lighter than the 400° F cold junction PbTe generator (System B) and approximately 150 pounds lighter than the 500° F cold junction PbTe generator (System A).
- (2) On an overall efficiency basis, the SiGe system is essentially equal to System B and one-half percent better (absolute) than System A.
- (3) On a size basis (all diameters are the same), the SiGe generator is superior with its length of 41 inches as compared to lengths of 58 and 77 inches on PbTe Systems A and B, respectively.

Based on these conclusions, SiGe was selected as the thermoelectric material to be used in this study.

110--Comparison of Air-Vac and Flexi-Mod T/E Modules

Two SiGe module configurations have been developed by Radio Corporation of America's Electronic Components and Devices Group in

*THERMACOMB Foam Ceramic, a product of American Lava Corporation, Chattanooga, Tenn., and CERCOR Corrugated Glass Ceramic Material, a product of Corning Glass Works, Corning, N.Y.

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Fig. VII-3a. SNAP 17A THERMOELECTRIC MODULE

Fig. VII-3b. Air-Vac Thermoelectric Couple

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Harrison, New Jersey, and were available for use in the design. The first of these is the bonded Air-Vac module, so named because of its ability to operate in either air or vacuum without a surrounding canister. Modules of this type were proposed for use in SNAP 17A in 1964 (Ref. VII-6) and were subsequently built for testing under that program. One of these, shown in Fig. VII-3, was bonded directly to an aluminum support plate and its attached radiator fin. Thermal insulation has been omitted from the pictured module for clarity. An operational module would be packed with fibrous insulation to minimize parasitic heat loss.

A cross section of an Air-Vac couple and its cold end bond stack features a large hot shoe made of a silicon-molybdenum alloy and bonded directly to the SiGe elements. Each element is, in turn, bonded to a cold shoe of tungsten which has a thermal expansion coefficient approximately equal to that of SiGe. In a subsequent operation, the couple is bonded to a cold end bond stack which is designed to provide an aluminum electrical conductor strap, an alumina insulator, and thermal expansion compensators which are compatible with both the tungsten cold shoe and the aluminum support plate.

The other type of SiGe module is the more recently developed Flexi-Mod. The basic couple is the same as before, with a large SiMo hot shoe bonded to the SiGe N and P elements, each of which is bonded to a tungsten cold shoe. The main difference lies in the fact that the Flexi-Mod couple is not bonded to a support plate, but rather is bonded to a threaded iron stud which can be mechanically fastened to a support plate of almost any material, configuration or thickness.

Other differences exist in the bond stack. A nickel-plated copper conductor strap is used for its high electrical conductivity and good bonding properties. The bond stack shown between the conductor strap and the tungsten cold shoe is primarily for thermal expansion compensation, whereas that between the conductor strap and the iron stud is primarily for electrical insulation and mechanical support.

The module itself is made as a series of couples attached only through their electrical conductor strap; hence, the name Flexi-Mod. It is flexible in another sense in that the module can be assembled to an RTG housing or support plate having almost any desired configuration. This is achieved by inserting the iron studs and their mating soft aluminum buttons into prealigned tapered holes in the housing and installing nuts and washers over the studs outside the housing. The nuts are tightened to a predetermined torque in order to deform the aluminum buttons and provide intimate mechanical contact between the stud and the RTG housing. A temperature drop of approximately 9° F has been measured between the iron stud and an aluminum housing 0.25 inch thick, for a power density comparable to that of the CRONUS RTG.

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FIG. VII-4. MOUNTED FLEXI-MOD ASSEMBLY

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This is a small part of the entire cold end ΔT , which was measured at approximately 45° F by RCA. The use of Flexi-Mod yields the lightest RTG weight and affords the most direct approach to assembly, installation and repair of the thermoelectric subsystem.

A mounted Flexi-Mod assembly is shown in Fig. VII-4. This particular unit was developed for SNAPOODLE, and has characteristically small hot shoes which indicate the presence of a high heat flux. In the CRONUS RTG, with low power density Sr-90 fuel, the use of oversized hot shoes is required. The module overall length of 10.5 inches is within the current manufacturing limits held by RCA. The hot shoe size of 1.27 by 1.64 inches is somewhat larger than has ever been produced by RCA, but it is anticipated that SiMo ingots of this size can be developed with no difficulty. Because the Air-Vac module has been developed for joining to aluminum (and not beryllium) and because the radiator configuration is that of a conical shell, the Flexi-Mod concept was selected.

111--Thermoelectric Module Insulation

The first decision on module insulation was to select a fibrous insulation over reflective foil. The primary objection to a foil system is that there is no attractive technique for fabricating and assembling a SiGe module within a sealed foil container. The container would be required to prevent oxidation during air operation of the module because of the high temperatures involved.

In addition, element insulation sleeves would be required to eliminate electrical shorting through the foil insulation. This heat loss, along with that through the sealed container, offsets the thermal advantage of foil insulation systems. Support of the foil systems during high g loads is also a problem. It is possible that these objections could be overcome with suitable development (use of a palladium foil whose nonoxidizing characteristics would eliminate the sealed container, etc.), but the cost would be large and in the end it is not obvious that the thermal efficiency would equal or exceed that of the anticipated fibrous thermal efficiency of 89%.

Presently, the only attractive fibrous high temperature structural insulation for use in the module is Dyna-Quartz. This insulation is capable of operating in air or vacuum at temperatures in excess of 2000° F. Dyna-Quartz has been selected as a backup insulation only, however, since a high temperature (1900° F) formulation of Min-K is currently being developed under contract to the USAEC and is expected to be ready by next year with properties equal to or superior to Min-K 2000. Between the edges of the hot shoes and in other crevices, Micro-Quartz will be used with compression to a density of 12 pcf. Micro-Quartz was selected over a number of other insulations because of its low thermal conductivity and the fact that an insulating technique has been developed for SiGe modules.

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Fig. VII-5. Thermal Conductivity of Beryllium Versus Temperature

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112--Radiator Selection

The only means of rejecting heat to a space environment is by radiation from the radiator surface. The required surface area was calculated to be 24 ft^2 in order to maintain the desired thermoelectric cold junction temperature. Since the total module area comprises only 7 ft^2 , a suitable method had to be found to transport heat to the balance of the radiator area not covered by the thermoelectric modules.

Beryllium was selected for the radiator material because of its unique physical and mechanical properties. Its most significant physical properties are its high thermal conductivity (Fig. VII-5) for efficient radiator operation, its high specific heat ($0.45 \text{ Btu/lb-}^\circ\text{F}$) for low temperature excursions during reentry, and its low density (0.067 lb/in.^3) for low component weight. As for mechanical properties, beryllium has appreciable tensile strength at reentry temperatures of 1200° to 1500° F , as shown in Figs. VII-6 and VII-7.

It would have been possible to design a composite radiator made of aluminum for high thermal conductivity for normal operation having a superalloy structural lining to resist dynamic loads and high temperature during reentry. This approach was rejected because of the higher weight for the composite structure, the difficulty of fabricating such a component, and the unknown reentry characteristics of a body in which part of the material (aluminum radiator fin) is brought above its melting temperature to be stripped away by the airstream. It was established that an all-beryllium radiator could be made which is lighter, more resistant to external pressure and capable of surviving reentry without an auxiliary superalloy structure.

113--Voltage Regulation Technique

It has been shown in Chapter III that an unregulated RTG configuration can, when fueled with Sr-90, maintain the desired voltage range of $28 \text{ volts} \pm 10\%$ over the entire life if it sees a constant-impedance load. This may be achieved by designing the thermoelectric subsystem to produce 30.8 volts (the upper limit) at beginning of life, then permitting the voltage to decrease due to the fuel decay and couple degradation to approximately 25.6 volts at the end of five years, which is above the lower limit. With a Pu-238 fueled system, the fuel excess over a five-year period is only 4% as compared with 13.2% for Sr-90, so its output voltage would remain within an even closer tolerance.

In reality, however, no spacecraft can provide a constant-impedance load for a five-year period. For example, the Nimbus B load profile varies nominally from a low of 213 watts during Earth night to 390 watts during simultaneous IRLS and spacecraft interrogation.

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Fig. VII-6. Typical Tensile Properties of Hot Pressed Beryllium Block

Fig. VII-7. Tensile Properties of Cross-Rolled Beryllium Sheet

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It is conceivable that the spacecraft bus could be open circuited, either due to a malfunction or to ground command shutdown, in which case an unregulated RTG would produce approximately 60 volts at its output terminal. If the malfunction corrected itself, or if equipments were turned on by ground command after a power shutdown, the resulting high voltage surge would be detrimental to the equipments.

A practical generator system will therefore include some form of voltage regulation across the power supply bus, either between the RTG and spacecraft interface or internal to the spacecraft.

There are several ways in which this could be done. The most promising ones are: regulation for constant output power and voltage regulation.

Constant-power regulation feasibility has been demonstrated on the SNAP 21A undersea generator program (Ref. VII-7). This concept has also been studied in more detail as part of the SNAP 27 program (Ref. VII-8). One of its characteristics is that it dissipates all power produced above the specified (250-watt) power level. The selected RTG design can produce up to 360 watts at BOL with a constant-impedance load. If this power were available, additional functions could be performed by the spacecraft early in the mission. With a constant-power regulator, this additional power capability would be lost.

Another disadvantage with a constant-power regulator is that it depresses the system output voltage at BOL in order to increase the RTG current. The resulting voltage rise from beginning to end of life may be within the specified ($\pm 10\%$) voltage range, but it means that individually regulated equipment aboard the spacecraft will be dissipating power (in individual voltage regulators) at the end of life when power is at a premium, rather than at beginning of life. As a practical consideration, individually regulated equipment will utilize power less efficiently from a constant-power system than from a constant-voltage system.

Because of these considerations, a constant-voltage regulator has been included as part of the selected generator system.

B. REENTRY BODY

151--Aerodynamic Braking Versus Propulsion Braking

The atmosphere will slow down a reentering body until the terminal velocity is reached. Rather than depend upon the atmosphere, it is possible to "fly" the reentry body through the atmosphere by using a propulsion system. There would be considerable advantage in such an

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approach. However, for this technique to be competitive in weight, it is necessary for the reentry propulsion system to have a specific impulse in excess of 3000 seconds. For the thrust levels required (an order of magnitude of 100 pounds), such propulsion systems are fundamentally feasible, but are considerably beyond the present state of the art. Consequently, an aerodynamic braking mode was selected.

152--Elimination of Tumbling Reentry Choice

A study was made to determine if a random tumbling radiation cooled reentry concept was suitable for the RTG/RB. This concept is being pursued for SNAP 29. However, there are large differences between SNAP 29 and the RTG/RB in the required reentry conditions and in the weight of the heat source. Both of these factors affect the suitability of a radiative heat rejection scheme for the RTG/RB.

In order to study this problem, the approximate requirement on the size and ballistic coefficient of a reentering flat plate was studied parametrically. The explicit variables are:

- (1) W = weight of flat plate
- (2) D = plate length
- (3) A = area of plate
- (4) γ_e = entry flight path angle (at orbital velocity)
- (5) $\dot{q}_w$ = allowable heating rate on thin walls.

a. Assumptions

It was assumed that the flat plate is cooled by radiation, that the thin material of the plate is temperature limited, and consequently, that the geometric requirements of the flat plate are governed by the maximum encountered aerodynamic heating rate. It was also assumed that the reentry trajectories for this flat plate will act as if it had a constant ballistic coefficient with no lift. The reentry velocity was orbital, and only fairly shallow entry angles (0 to 15 deg) were considered).

b. Configuration

The reentry shape was assumed to be a thin square with each side having a length, D . The aerodynamic characteristics of this vehicle were analyzed for two cases:

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- (1) The flat plate enters the atmosphere at a right angle to the velocity vector.
- (2) The flat plate is tumbling rapidly during reentry.

The hypersonic drag coefficient, C_D , in continuum flow for these two cases is:

(1) Stable, $C_{D_s} = 1.8$

$$\begin{aligned}
 (2) \text{ Tumbling, } C_{D_T} &= C_{D_{\text{ref}}} \left\{ \int_0^{90^\circ} \cos^3 \alpha \, d\alpha + \frac{A_2}{A_1} \int_0^{90^\circ} \sin^3 \alpha \, d\alpha \right\} / \int_0^{90^\circ} d\alpha \\
 &= 0.425 C_{D_{\text{ref}}} \left\{ 1 + \frac{A_2}{A_1} \right\}
 \end{aligned}$$

where

$$C_{D_{\text{ref}}} = C_{D_s} = 1.8$$

$$A_1 = D^2, \text{ reference area}$$

$$A_2 = tD.$$

Assuming $t = 0.1D$, then

$$C_{D_T} = 0.425 \times 1.8 \left\{ 1 + \frac{0.1D^2}{D^2} \right\} = 0.425 \times 1.8 \times 1.1 = 0.841$$

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c. Aerodynamic heating on a flat plate

The aerodynamic heating distribution on a flat plate (front or back surfaces only) can be nondimensionalized with respect to the stagnation point heating rate on a sphere with a diameter equal to the characteristic length of the plate. The heating rate at the center of the plate (Ref. VII-9) is given by:

$$(\dot{q}_{\text{stag}})_{\text{flat plate}} = 0.5 (\dot{q}_{\text{stag}})_{\text{hemisphere}}$$

while the heating rate at the corner edge (assuming laminar flow) (Ref. VII-10) is given by:

$$(\dot{q}_{\text{edge}} / \dot{q}_{\text{stag}})_{\text{flat plate}} = 2.0$$

This is a theoretical value calculated by Stine and Wanlass in Ref. VII-11. Lee's value from his paper (Ref. VII-12) for the heating rate at the edge is 1.4. From a more recent report (Ref. VII-13), the value at the edge of a flat-faced body is between 1.4 and 1.6 by extrapolating the data presented. The last data point from this document was 86% of the flat-face radius from the center of the face. Therefore, a more realistic value for this ratio would appear to be:

$$(\dot{q}_{\text{edge}} / \dot{q}_{\text{stag}})_{\text{flat plate}} = 1.4$$

If this plate was tumbling, the effective heating rate would be decreased to about 60% of the stable value while the surface was exposed to the flow. Since one-half the time the face of the plate was located away from the flow, then the heating rate for a tumbling body would be decreased to about 30% of the stable value. The effective heating rates at the center and edge of a flat plate for stable and tumbling orientation were:

$$\begin{aligned} [(\dot{q}_{\text{stag}})_{\text{flat plate}}]_{\text{stable}} &= 0.5 (\dot{q}_{\text{stag}})_{\text{hemisphere}} \\ [(\dot{q}_{\text{edge}})_{\text{flat plate}}]_{\text{stable}} &= 0.5 \times 1.4 (\dot{q}_{\text{stag}})_{\text{hemisphere}} = \\ &= 0.7 (\dot{q}_{\text{stag}})_{\text{hemisphere}} \\ [(\dot{q}_{\text{stag}})_{\text{flat plate}}]_{\text{tumbling}} &= \frac{0.5 \times 0.584}{2} (\dot{q}_{\text{stag}})_{\text{hemisphere}} \\ &= 0.146 (\dot{q}_{\text{stag}})_{\text{hemisphere}} \\ [(\dot{q}_{\text{edge}})_{\text{flat plate}}]_{\text{tumbling}} &= \frac{0.7 \times 0.584}{2} (\dot{q}_{\text{stag}})_{\text{hemisphere}} \\ &= 0.204 (\dot{q}_{\text{stag}})_{\text{hemisphere}} \end{aligned}$$

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The 0.584 factor from Ref. VII-10 was the average value of stagnation heating rate over yaw angles from 0 to 90 degrees. Therefore, for a tumbling body, this factor must be reduced by 50%.

d. Aerodynamic heating parameter

The aerodynamic heating parameter for maximum stagnation heating rate and total stagnation heat are presented in Figs. VII-8 and VII-9, respectively, as a function of entry angle (γ_e) at 400,000 feet.

The velocity at this altitude was orbital (25,690 fps). From Fig. VII-8, it can be seen that the heating rate will more than double when the entry angle changes from 0.1 to -10 degrees. The total heat in Fig. VII-9 is reduced by approximately 25% by increasing this entry angle from -0.1 to 10 degrees. The first figure will be used in the next section to determine the required $(W/A)/D$ ratio as a function of entry angle, flat plate orientation and allowable heating rate at two locations on the body.

e. Allowable heating rate

The parameter $(W/A)(1/D)$ is plotted in Figs. VII-10 through VII-14 as a function of allowable heating rate at the center and edge of a stable and tumbling flat plate for various reentry angles, γ_e . At the edge, the ratio of local-to-stagnation heating rates on a flat plate was assumed to be 1.4. In Figs. VII-10 and VII-13 ($\gamma_e = -0.1$ and -10 deg, respectively), an additional two curves are presented for this ratio equal to 2.0.

It can be concluded from Figs. VII-10 through VII-14 that the following are true:

- (1) For a given allowable heating rate, the minimum acceptable $(W/A)(1/D)$ parameter will be increased when:
 - (a) The orientation is changed from stable to tumbling
 - (b) The heating rate ratio at the edge changed from 2.0 to 1.4
 - (c) The reentry angle (γ_e) at 400,000 feet becomes smaller
 - (d) When this allowable heating rate is applied at the center instead of the edge.
- (2) The allowable heating rate increases as the parameter $(W/A)(1/D)$ increases for a given orientation and location on the flat plate.

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$(\dot{q}_s)_{\max}$ = maximum stagnation heating rate on blunt body (Btu/ft²-sec)

R = nose or characteristic radius of reentry vehicle (ft)

$W/C_D A$ = ballistic coefficient (lb/ft²)

W = weight of reentry body (lb)

C_D = hypersonic drag coefficient

A = aerodynamic reference area (ft²)

Entry conditions:

h_E = 400,000 ft

V_E = 25,690-fps circular velocity

γ_E = entry flight path angle at 400,000 ft

V_C = circular velocity

$(\dot{q}_s)_{\max}$ based on continuum flow regime with laminar boundary layer

Fig. VII-8. Maximum Stagnation Heating Rate Versus Entry Flight Path Angle

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Fig. VII-9. Total Stagnation Heat Versus Entry Flight Path Angle

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Fig. VII-10. $(W/A)(1/D)$ Parameter Versus Allowable Heating Rate at the Center and Edge of a Stable and Tumbling Flat Plate for a Reentry Angle of -0.1 Degree

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Fig. VII-11. $(W/A) (1/D)$ Parameter Versus Allowable Heating Rate at the Center and Edge of a Stable and Tumbling Flat Plate for a Reentry Angle of -2 Degrees

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Fig. VII-12. $(W/A) (1/D)$ Parameter Versus Allowable Heating Rate at the Center and Edge of a Stable and Tumbling Flat Plate for a Reentry Angle of -5 Degrees

Fig. VII-13. $(W/A)(1/D)$ Parameter Versus Allowable Heating Rate at the Center and Edge of a Stable and Tumbling Flat Plate for a Reentry Angle of -10 Degrees

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Fig. VII-14. $(W/A) (1/D)$ Parameter Versus Allowable Heating Rate at the Center and Edge of a Stable and Tumbling Flat Plate for a Reentry Angle of -15 Degrees

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For example, if Haynes-25 is the material of the flat plate, the maximum cold wall heating rate for an emissivity of 0.95 is 18 Btu/ft²-sec, assuming radiation from one side. The internal heating rate was not considered. The (W/A) (1/D), if this heating rate is applied at the center for $\gamma_e = -2$ and -10 degrees, is:

In stable flight:

$$[(W/A) (1/D)] \gamma_e = -2^\circ = 1.3 \text{ lb/ft}^3$$

$$[(W/A) (1/D)] \gamma_e = -10^\circ = 0.34 \text{ lb/ft}^3$$

In tumbling flight:

$$[(W/A) (1/D)] \gamma_e = -2^\circ = 6.8 \text{ lb/ft}^3$$

$$[(W/A) (1/D)] \gamma_e = -10^\circ = 1.8 \text{ lb/ft}^3$$

If radiation is considered from both sides, the maximum cold wall heating rate is 36 Btu/ft²-sec for an emissivity of 0.95. The (W/A) (1/D) for this heating rate applied at the center for the same two entry angles is:

In stable flight:

$$[(W/A) (1/D)] \gamma_e = -2^\circ = 5 \text{ lb/ft}^3$$

$$[(W/A) (1/D)] \gamma_e = -10^\circ = 1.4 \text{ lb/ft}^3$$

In tumbling flight:

$$[(W/A) (1/D)] \gamma_e = -2^\circ = 27 \text{ lb/ft}^3$$

$$[(W/A) (1/D)] \gamma_e = -10^\circ = 7.3 \text{ lb/ft}^3$$

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atmosphere. These estimates were confirmed during the study (Appendices JJ and MM).

154--Established Reentry Shape Versus New Shape

Once it was established that an attitude dependent reentry body was required for the RTG/RB, it was necessary to choose the proper aerodynamic shape.

A basic decision was made that the aerodynamic shape of the reentry body should be one which has already been flight tested. The reason for this choice is that a brand new shape (such as a finned body) would require considerable development testing (as well as analysis) before its feasibility with regard to the RTG/RB could be established. This would require several years and several millions of dollars. Such an effort could be justified if a new shape were required, or if a new shape had overriding advantages. However, presently available shapes are suitable for the RTG/RB. Martin experience on several reentry studies has been that although a new shape might have some weight or functional advantages, the actual new benefits of a new shape would not be overriding. Consequently, the basic shape of an existing reentry vehicle was considered adequate for the RTG/RB.

155--Ballistic Body Versus Lifting Body

Two types of hypersonic aerodynamic bodies are possible: those designed to fly at medium or high L/D ratios (lifting bodies), and those designed to have no lift during entry (ballistic). Since the maneuvering capabilities of the lifting bodies are not necessary and since such devices would require undesirable attitude control and guidance subsystems, a ballistic shape was selected.

156--Selection of Gemini-Type Reentry Body

Selection of the specific reentry body was based on considerations of weight, volume, envelope, reentry heating rates and fueling. Sections 161 and 162 present the argument for the reentry body payload being the RTG in total and the afterbody of the reentry body serving as the radiator for the RTG during normal operations.

A comparative weight analysis was made to help establish the weight differences between the following potential reentry shapes:

(1) Blunted cone	18-degree slope (Gemini)
(2) Blunted cone	33-degree slope (Apollo shape)

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$$\text{RATIO } \frac{W}{C_D A} = 1.000$$

$$\frac{q_W}{q_{W \text{ REF}}} = 1.0$$

(q_W HEATING RATE AT WALL)

WEIGHT = 100%

$$\text{RATIO } \frac{W}{C_D A} = 0.646$$

$$\frac{q_W}{q_{W \text{ REF}}} < 1$$

WEIGHT = 105%

$$\text{RATIO } \frac{W}{C_D A} = 1.592$$

$$\frac{q_W}{q_{W \text{ REF}}} = 3.1$$

WEIGHT = 108%

$$\text{RATIO } \frac{W}{C_D A} = 1.254$$

$$\frac{q_W}{q_{W \text{ REF}}} = 2.8$$

WEIGHT = 108.5%

$$\text{RATIO } \frac{W}{C_D A} = 0.646$$

$$\frac{q_W}{q_{W \text{ REF}}} = 2.0$$

WEIGHT = 108.7%

$$\text{RATIO } \frac{W}{C_D A} = 0.323$$

$$\frac{q_W}{q_{W \text{ REF}}} = 1.4$$

WEIGHT = 116.5%

$$\text{RATIO } \frac{W}{C_D A} = 0.163$$

$$\frac{q_W}{q_{W \text{ REF}}} = 1.0$$

WEIGHT = 123.8%

$$\text{RATIO } \frac{W}{C_D A} = 2.325$$

WEIGHT = 124.8%

$$\text{RATIO } \frac{W}{C_D A} = 0.815$$

WEIGHT = 130.7%

Fig. VII-15. Reentry Bodies

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- (3) Flared cylinder (five versions, function of wall heating rate) (warhead)
- (4) Cone sphere (two versions: walls used as radiator and back used as radiator).

All reentry configurations are designed for a 30-ft² radiation surface area. The reentry conditions and nondimensionalized heating rates are assumed to be the same for all. Only the weights of the reentry body were computed. No allowances were made for variation in deorbit propulsion system (basically proportioned to the reentry body weight) or for the spacecraft attachment and separation structure. No allowances have been made for GHE weight penalties.

a. Comparison

The various configurations are shown in Fig. VII-15.

The comparative weights are shown in Table VII-1 where it is seen that Configuration 1 is the lightest. The comparison of the various configurations to Configuration 1 is summarized.

Configuration 1 versus 2. The heating rate on the walls of Configuration 2 is less than the heating rate for Configuration 1. However, since both peak radiator temperatures are below the softening point of beryllium, no weight change in the radiator wall is expected. The aft cap of Configuration 1 is heavier than the aft cap of Configuration 2. However, the weight penalty incurred by the 55% larger base area of Configuration 2 causes it to be about 13 pounds heavier.

Configuration 1 versus 3. The basic parameter for Configuration 3 is the aerodynamic heating rate at the wall during reentry. For the type of initial tumbling reentry conditions encountered, it is estimated that the heating rate on the wall of Configuration 3 is twice as high as the comparative heating rate on Configuration 1 (Fig. VII-15). Five different versions of Configuration 3 were, therefore, investigated. (The cylinder diameter, 3.04 ft, is the same as the average diameter of Configuration 1. An increase in this diameter would result in an increased weight for Configuration 3.) The first, 3a, has no flare. The second through fifth configurations have various sized flares, so that $W/C_D AR$ and, consequently, the wall heating rate can be reduced.

If the heating rates on the walls are increased beyond the 1.2 heating rate of the referenced configuration, namely, beyond 1800° R on the walls, then the walls have to be backed up with a heat resistant structure. (The walls could also be made thicker; but on a weight basis, this is approximately the same.) A Haynes-25 structure, 0.020 inch

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TABLE VII-1
Comparative Weights and Configurations*

	1	2	3a	3b	3c	3d	3e	4a	4b
Thermoelectric + heat source, etc.	230	230	230	230	230	230	230	230	260
Radiator	80	80	80	80	80	80	80	120	80
Radiator backup structure	0	0	45	40	30	30	0	0	0
Heat shield structure	12.6	14.6	11.6	9.5	7.1	6.2	5.6	10.6	21
Side wall structure	0	0	0	0	0	0	0	0	30
Heat shield ablator	27.5	42.0	18.7	17	16	15	14	53	75
Side wall ablator	0	0	0	0	0	0	0	30	0
Aft cap structure	5	2	0	11	25.5	56	115	5	5
Miscellaneous	22	22	22	22	22	22	22	22	22
Total	377.1	390.6	407.3	409.5	410.6	439.2	466.6	470.6	493

*These weights, given in pounds, are made on a comparative basis. Consequently, they are not necessarily the same as the final weights.

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at 0.95 lb/ft^2 will suffice until the heating rate is approximately 2.8 times as high as the reference. Then, a refractory structure (1.5 lb/ft^2) is needed.

The penalty for avoiding these weights by reducing $W/C_D AR$ is the flare. The weight of this flare was estimated by assuming a 1.5 lb/ft^2 structure (including ablators). In spite of this low unit weight, Configuration 3a was the lightest of the Class 3 configurations. Due to the 45-pound backup heat shield structure, it is 30 pounds heavier than Configuration 1. Furthermore, since no fuel capsule compatible condition was introduced in this comparison, there may be additional penalties that have to be paid to keep the reentry fuel capsule temperature exclusions within bounds.

Configuration 1 versus 4. The comparison of Configurations 1 and 4 becomes a bit complicated due to the problem of defining Configuration 4. If the radiator area is the outer conical portion of the cone cylinder, the radiator experiences a relatively high aerodynamic heating rate, even at zero angle of attack. The heating rate increases as the cone angle increases. (The limiting case--the cylinder--has already been described.) Obviously, increasing the nose radius as well as increasing the heating rates will have an adverse effect on the weights, as compared to Configuration 3a. An external conical radiator configuration will, therefore, lose on a weight basis.

There are several other options available. For example, the radiator could be placed on the inside of the conical surface. There would be some loss in radiator efficiency, but this might be compensated for by the low radiator heating rate. This concept is shown as Configuration 4a. A second possible configuration is 4b. In this concept, the radiator area covers the rear portion of the cone. The heating rate on the back face is small compared to Configuration 1. Some destabilization device is needed in Configurations 4a and 4b in order to turn the reentry body as it enters the atmosphere backward.

Due to the geometry of the radiator and the heat source, some weight penalties were assigned to the RTG. A 50% increase in the initial radiator area resulted in a 40-pound increase in the radiator weight for Configuration 4a. Configuration 4b would require an additional 5 ft^3 of insulation (6 lb/ft^3). These weights are shown in Table VII-1.

Configuration 4a is considerably heavier than Configuration 1 because of the higher heating rate and frontal area, because of the weight of the walls and because of the heavier RTG. Configuration 4b is still

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heavier, since it has an even larger frontal area than Configuration 4a, and since it requires additional structure to support the side walls.

b. Conclusions

The various configurations are summarized in Table VII-2.

For the reentry conditions required for the RTG/RB, the 18-degree blunted cone configuration is lighter than the other potential candidates.

If the heating requirements on the side wall were relieved (for example, if a high temperature metal rather than beryllium could be used as a radiator), then a blunted cylinder might have a slight weight advantage. If reentry from orbit is required to be higher than approximately 700 nautical miles, then a 33-degree blunted cone configuration might be better. For the present mission, the weight differences between the 18° and the 33-degree blunted cones are rather small.

TABLE VII-2
Configuration Summary

<u>Name</u>	<u>Weight</u>	<u>Advantages</u>	<u>Disadvantages</u>
Gemini (1)	377.1	Weight, volume	Loading
Apollo (2)	390.6	Low height Mission growth	Broad base (mounting) Loading Radiator efficiency
Warhead (3a)	407.3	Loading Narrow base	Capsule temperature rise Double wall construction
Discoverer (4a)	470.6	Loading Low height	Aero stability Weight

157, 158, 159--Heat Shield Selection

The selection of a heat shield system for reentry thermal protection requires application of the projected operational mission requirements and associated environmental factors to the various design alternatives and, subsequently, the selection of the most promising design concept on the basis of reliability, weight, cost and flexibility over the entire

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spectrum of mission performance requirements. The RTG/RB application is based upon a ballistic reentry from a near earth orbit. The predicted thermal environmental factors associated with RTG/RB recovery are outlined in Table VII-3. Initial entry velocities (25,690 fps) and flight path angles are based on an altitude of 400,000 feet. The body nose radius is 6.1 feet and $W/C_D A = 22$ psf. The heating rates are based on laminar flow and a 300° K wall temperature.*

TABLE VII-3
CRONUS Reentry Heating Environment

<u>Condition</u>	Peak $\dot{q}_{stag}$ (Btu/ft ² -sec)	$\dot{q}_{stag} dt$ (Btu/ft ²)
Orbital decay	44.0	10,680
Abort	95	2,550

Consideration has been given to various thermal protection system concepts which might be applicable to the RTG/RB. Results of these studies and the resultant conclusions are discussed.

a. Reradiation thermal protection

The design of a reradiation thermal protection system is based primarily upon the peak heating rates associated with the critical environment since the peak radiation equilibrium temperature is the prime factor in selecting a heat shield material, based on strength or material degradation characteristics.

The use of ceramic materials has been investigated in the past and the brittle fracture susceptibility of this class of materials requires sophisticated design techniques such as mesh reinforcements. Flight test experience exists on this type of design (ASSET program), but it is significant to note that practical applications of this concept have been restricted to compact configurations such as nose caps and leading edge segments. It is suspected that the fracture sensitivities of this material make it unsuitable for an emergency escape device such as the RTG/RB.

*Since the enthalpy of the air is rather high, the heat transfer at near orbital velocities is essentially independent of the wall temperature.

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Fig. VII-16. True Radiation Equilibrium Temperature Based on Cold Wall Heat Rate (q_{cw})

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The use of metallic reradiative heat shields offers a unique design consideration in that for certain materials and temperature ranges, protection must be provided against rapid oxidation of the metallic layer. The relationship between heating rate and surface temperature is defined in Fig. VII-16, which shows the true radiation equilibrium temperature as a function of cold wall heating rate, emissivity and free stream velocity. This carpet plot is based on the relationship:

$$\frac{\dot{q}_{cw}}{\epsilon} = \frac{0.476 \left[\frac{T_{EQ}}{1000} \right]^4}{1 - \frac{13,300 T_{EQ}}{V_{\infty}^2}} \text{ Btu/ft}^2\text{-sec} \quad (\text{VII-1})$$

It is interesting to note that for free stream velocities less than about 5000 fps, the air might cool the fuel capsules since the stagnation temperature is less than the surface temperature of the fuel elements. However, this free stream velocity will be encountered only late in the reentry trajectory. Consequently, valves or other devices must be designed which can discriminate between air above or below Mach 5, and open a passage to the fuel elements only when the air velocity is sufficiently low.

Three candidate materials for metallic reradiative heat shield systems are listed in Table VII-4 with appropriate maximum allowable surface temperatures and the corresponding peak heating rate limits based on Eq (VII-1). From Ref. VII-14, the velocity at peak heating rate is taken to be 0.85 x the initial reentry velocity at 400,000 feet.

TABLE VII-4

Metallic Reradiative Heat Shield Heat Rate Restrictions ($\epsilon = 0.80$)

<u>Material</u>	<u>Temperature Limit (°F)</u>	<u>Allowable Peak $\dot{q}_{stag}$ (Btu/ft²-sec)</u>
Haynes-25	1800	10.6
Columbium	2500	32
Tantalum	3500	105

b. Heat shield concept evaluation

Based on the data in Table VII-4, the applicability of metallic reradiative concepts to the CRONUS reference vehicle can be evaluated.

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Fig. VII-17. Critical Ballistic Reentry Parameters Governing Radiative Heat Shield Applications

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Figure VII-17 defines the limitations of tantalum, columbium and Haynes-25 based on their respective temperature limitations as a function of vehicle ballistic coefficient and initial entry angle. Note that for the abort case, only the use of tantalum appears practical and, for this particular case, marginal. The use of columbium and Haynes-25 would only be possible through substantial reductions in initial entry angle and vehicle ballistic coefficient. The requirement for oxidation resistant coatings on refractory metal heat shields introduces a potential reduction in the overall heat shield system reliability since these coatings are susceptible to damage. An additional problem associated with metallic reradiative heat shields and their heat rate sensitivity is that catastrophic failures may result if the limiting heat rate for a given material is exceeded by even a small margin. Hence, appropriate design safety factors must be incorporated to account for this sensitivity based on a study of those factors which affect peak heating rates and the degree of accuracy with which they can be predicted. This results in a further constriction of the design boundaries within which metallic reradiative systems can be applied. It is seen then that, based on the reference configuration, metallic radiative heat shields produce a marginal design for the abort case. The projected operational temperature range for tantalum may ultimately extend to 4000° F for loading environments that are not severe but this requires some advancement in tantalum and coatings technology. As implied previously, the transition to a ceramic reradiative concept would involve extensive development effort, and the application of such a concept over the large area of the reference configuration blunt forebody would involve critical design and fabrication problems. Similar problems would accompany the application of graphite heat shield concepts.

Ablative heat shields. Ablative heat shield technology has advanced rapidly in recent years to the point where a wide variety of materials are available to suit a particular set of design requirements. Low density charring ablators display a wide range of performance flexibility for reentry vehicles. The two most prominent materials in this class are the elastomeric silicones and low density nylon phenolic. Under external heat input, these materials will pyrolyze and form a low density carbonaceous char layer on the exposed surface. Ablators will protect the substructure by transpiration of the pyrolysis products into the boundary layer, insulation effect due to low thermal conductivity, heat storage and reradiation from the high emissivity char layer. It is interesting to note that, for heating pulses which are characterized by relatively high heat rates for most of the pulse duration, the major portion of heat rejection is by reradiation from the char surface. Since analysis and plasma arc test data have shown that due to the relatively low conductivity of the char, the surface temperature runs close to the radiation equilibrium value, charring ablators can in a sense be considered a modified form of reradiative heat shields. Here, too, it is found that low density charring ablators are heat rate sensitive since

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Fig. VII-18. Comparison of Heat Sink and Ablator Weights

the carbonaceous char layer is subject to surface recession due to combustion (a function of heating rate, pressure, amount of oxygen diffusing through the boundary layer to the char surface and char density). Test data have shown that for the low density elastomeric silicone materials, the heating rate range in which surface recession starts to become excessive runs from approximately 100 to 120 Btu/ft²-sec. Unlike metallic reradiative shields, the heat rate limitations on charring ablators are not quite so absolute and, for specific cases, brief excursions considerably in excess of prescribed limits can be tolerated. The heating rate limitations on the low density nylon phenolic ablators are somewhat higher than those for the elastomeric silicones since the low density nylon phenolic forms a denser char layer. For reentry environments characterized by prolonged exposure to high heating rates, pressures and aerodynamic shear stresses, the high density ablative systems are applied (i. e. carbon, nylon, silica phenolics). Pyrolysis of these materials creates a stronger and denser char layer which is more resistant to degradation induced by combustion and mechanical breakup. Based on the data in Table VII-3, the reentry heating environment of the RTG/RB reference vehicle configuration can be accommodated by the low density charring ablator group.

Heat sink. The use of a heat sink approach for the management of reentry heating represents a relatively primitive solution to the problem. Beryllium is an excellent material for this application due to its low density and high specific heat. Two approaches may be applied here:

- (1) Maximum permissible temperature in the beryllium heat shield is that value at which the material retains an acceptable amount of strength in order to react the induced airloads. This value is taken to be 1200° F.
- (2) The heat sink peak temperature is the melt temperature (2345° F).

A third option is available which permits the beryllium heat sink to go into a melt phase but this approach implies the increased risk of breaking up of the heat sink due to the effects of structural loads on a material in a molten or near molten state and of diminishing cross sectional thickness. Such a breakup could be catastrophic. This would be particularly critical for the reference configuration. Figure VII-18 show the results of an analysis conducted to determine the required unit weights of the beryllium heat sink for the two design cases of 1200° and 2345° F maximum temperatures (incipient melt). Required weights are presented as a function of vehicle $W/C_D AR$. The applied heating environment was based on an orbital decay, initial entry velocity of

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25,690 fps at 400,000 feet. For comparative purposes, the weight requirements for an elastomeric silicone ablative heat shield system are also presented. (The criteria and analytical techniques upon which the ablator weights are based is covered in detail in Appendix O. The weights shown for the 1200° F heat sink case represent the beryllium only since in this case the heat sink provides its own structural support. For the incipient melt heat sink case, the weights represent the required amount of beryllium, a plastic laminate honeycomb supporting structure, and a low density insulation (Micro-Quartz) which is sandwiched between the heat sink and supporting structure. The ablator weights include the elastomeric silicone ablator material, adhesive bond (HT-424) and plastic laminate support structure. For the reference configuration, the W/C_{DAR} is 3.6 lb/ft³. The weight penalty incurred through the use of a heat sink concept is clearly implied. Additional problems which would be related to the use of a beryllium heat shield on the reference configuration are high material costs and fabrication problems associated with a large spherical disc shape of beryllium (i.e., forming, machining, attachments, joints).

Conclusions. In review, the relative applicability of the heat shield concepts to the reference configuration is briefly summarized.

Metallic systems are marginal for heating rates associated with an abort situation. Reliability is compromised through the use of damage sensitive oxidation resistant coatings and extreme heat rate sensitivity of these systems. Fabrication costs (primarily due to coating processes) will run high. Previous experience (Ref. VII-15) indicates that in the range of heating environments associated with the reference vehicle, metallic reradiative heat shields are somewhat heavier than the low density charring ablator systems. Furthermore, optimum weights in metallic reradiative systems are realized in combination with an active water-cooling system (although at the low total heat loads of interest, the weight reduction due to adding an auxiliary cooling system should be quite small). Upgrading the heating rate limitations on the system by employing ceramics or graphites involves extensive design development work which would not be justified for this application.

Low density charring ablators are capable of providing the lightest weight heat shield system for the reference vehicle. Fabrication techniques have been developed and are directly applicable to the reference configuration. Brief excursions above the design heating environment can be tolerated by ablators far more readily than by the metallic reradiators.

The use of heat sink concepts results in the heaviest heat shield system for the reference trajectory. The use of beryllium implies the incurrence of high costs and fabrication and detail design development work. The fabrication and detail design problems are not considered

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to be insurmountable but a commitment to attack these problems would only be justified by an advantage to be gained in heat shield performance. No such advantage is indicated.

On the basis of the previously discussed comparisons, low density charring ablators were selected as the most promising design concept for the RTG/RB reference configuration reentry heat shield design. The following section outlines the results of the detail design studies conducted on the ablative heat shield systems.

c. Ablative heat shield concept

Three basic ablative heat shield designs have been analyzed and are defined in Fig. VII-19. Concepts 1 and 2 consist, respectively, of a low density nylon phenolic charring ablator over a plastic laminate substructure and a low density elastomeric silicone charring ablator (Martin developed ESA 3560 H) supported on a plastic laminate substructure. Concept 3 is an attempt at heat shield weight reduction through the use of a composite heat shield system consisting of ablator and insulation materials. The philosophy here is to replace the substrata of the ablator material, which at these depths act primarily as an insulator, with a more efficient low density insulation material such as Micro-Quartz. In order to establish realistic heat shield thicknesses and weights, it was necessary to develop a set of temperature boundary conditions related to the specific design being analyzed. Hence, for each design concept a set of initial temperature distributions through the heat shield components were established based upon the idealized analytical model shown in Fig. VII-20. This model was used to establish a steady state temperature variation which was assumed to exist in the heat shield at the initiation of reentry. Subsequently, the required ablator thickness was determined in order to constrain critical components of the heat shield/substructure system within acceptable limits. A review of each of the three concepts reveals that the critical temperature for heat shield sizing exists at the adhesive bond line between the heat shield and the substructure. The use of HT-424 adhesive is recommended since it provides maximum strength at temperature for short time exposures when compared with other adhesive systems. Based on the appropriate strength-temperature relationships, a maximum permissible temperature during reentry at the bond line of 800° F has been selected. The analytical model used for the reentry heat shield analysis was identical to that shown in Fig. VII-20 with the spacecraft interface removed. The ablative heat shield analysis was conducted on the Martin T-CAP III Charring Ablator Digital program which is described in Appendix O. Figures VII-21 through VII-23 present pertinent results for the three heat shield concepts. Note that the heat shield design curves are plotted as maximum bond line temperatures versus heat shield thickness. The desirability of increasing the allowable bond line temperature in the interests of heat shield weight reduction is apparent.

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Fig. VII-19. Ablative Heat Shield Design Concepts

Fig. VII-20. Steady-State Operation for Analytical Model

Fig. VII-21. Concept 1 Heat Shield Design Curve

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Fig. VII-22. Concept 2 Heat Shield Design Curve

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Fig. VII-23. Concept 3 Heat Shield Design Curve

Fig. VII-24. Temperature History for Concept 2

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The analysis was conducted with a design factor of 1.15 applied to the predicted heating rates. The thicknesses shown in the design curves were multiplied by a design factor of 1.10 in order to arrive at the design thicknesses and weights which are shown in Table VII-5 for each of the concepts. The weights of the substrate panel and related adhesive bond layers are considered as structure weight and are not included since they are the same for each of the concepts.

TABLE VII-5
Heat Shield Design Comparisons

<u>Concept</u>	<u>Ablator Design Thickness (in.)</u>	<u>Insulation Design Thickness (in.)</u>	<u>Heat Shield Weight (lb)</u>
1	0.595	--	21.7
2	0.738	--	22.4
3	0.385	0.358	14.03

The weight numbers are based upon the following densities:

<u>Material</u>	<u>Density (lb/ft³)</u>
Low density nylon phenolic	36
ESA 3560 H elastomeric silicone	30
Micro-Quartz insulation	6.5

Although data indicate a moderate reduction in the heating environment toward the edge of the heat shield during zero angle of attack, the effect of angle of attack oscillations will tend to level out this variation. It has been assumed that the heat shield thickness is constant over the forebody.

Figure VII-24 shows a temperature-time history for the Concept 2 heat shield. The heat shield design is critical for the orbital decay case and, hence, the peak bond line temperature associated with the abort case ($\gamma_e = -8.0$ deg) will be less than 800° F. For Concept 2, it peaks at 350° F. The calculated char surface recession for the ESA 3560 H material during abort reentry (high heating rates) is negligible at 0.00074 inch. Table VII-3 shows that for the orbital decay reentry case, the total incident cold wall heat input to the heat shield is 10,680 Btu/ft². The ESA 3560 H elastomeric silicone ablator reradiates 4543 Btu/ft² (approximately 40%) from the char surface.

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ABLATIVE HEAT SHIELD/RADIATOR INTERFACE DETAIL FOR
CRONUS REFERENCE CONFIGURATION

Fig. VII-25. Ablative Heat Shield/Radiator Interface Detail for
CRONUS Reference Configuration

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The calculated heat shield weights of Concepts 1 and 2 are essentially equal. It should be mentioned that lower density materials could be applied to Concept 1 (low density nylon phenolic) on the order of 25 lb/ft³, which would then result in a somewhat lighter weight for Concept 1 when compared with Concept 2. The major drawbacks associated with low density nylon phenolic ablators are related to their fabricability and brittleness. Due to the presence of an internal heat source (fuel capsules) and enclosure of the heat shield in the retro structure prior to reentry, subzero temperatures at which ablator brittleness becomes critical will not be realized. However, since with the reference configuration it is possible to introduce loads into the ablator from the retro package, the limited strain capacity of Concept 1 is a potential problem. Concept 3 indicates that a substantial weight saving can be realized by designing a composite heat shield. An added advantage to this approach is that the ablator material has an additional insulation buffer between it and the fuel heat source during steady state operational life. Thus, the temperature in the ablator during steady-state operation is reduced. Based on the analytical model defined in Fig. VII-20, the steady-state ablator temperatures are 285° F for Concept 2 and 185° F for Concept 3. Concept 3 will be somewhat more difficult to fabricate than Concept 2 and will require more development work (fabrication techniques, load transfer capability, optimization of ablator and insulation thicknesses). It is felt, however, that such a design can be generated without significant problems. For the present, since it is the more proven approach, Concept 2 (ESA 3560 H elastomeric silicone ablator) is the recommended design (Fig. VII-25).

160--Aerodynamic Destabilization Method

The reentry body may reenter the atmosphere in a tumbling mode. The hypersonic stabilizing of a blunted cone, at low angles of attack ($\alpha \sim 60$ deg) has been well established. Consequently, for low angles of attack, there is no doubt that the reentry body will be stabilized about the zero angle of attack axis. For large ($\alpha \sim 180$ deg) angles of attack, however, there is some question about the stability of a conical reentry body.

Aerodynamic instability is desirable for this type of entry. The vehicle must be turned toward the zero angle of attack attitude. If the $\alpha = 180$ degrees axis were stable, or only slightly unstable, then a body entering the atmosphere might experience the aerodynamic heating with the main heat protection devices (the heat shield) inoperative, and the radiator and aft cap (which are metal surfaces rather than ablators) absorbing the main aerodynamic heating pulse. This condition would most likely result in a structural breakup of the reentry body. Consequently, there is a desire for destabilizing a rearward reentering body.

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COMPARISON OF DESTABILIZATION SYSTEMS

Figure VII-26. Comparison of Destabilization Systems

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If the reentry body were a pointed cone, then the allowable CG location would be no more than 29 inches aft of the heat shield. This CG location would not present any design problems. However, such a configuration would be quite long (74.7 in. as compared to 59.5 in. of the referenced configuration) and, consequently, would be volumetrically inefficient, heavy and cumbersome to use. To avoid this problem, the narrow end of the cone should be truncated.

This truncation has a tendency to make the reentry body stable about the $\alpha = 180$ -degree point. Even with modest truncation slopes (18 deg) such as the Gemini spacecraft has, the stable nature of a rearward entering body is not changed.

For a hemispherically capped truncated cone (cone half angle of 18 deg), the cone is neutrally stable ($C_{m_\alpha} = 0$) at a point 30% of base diameter forward of the base (maximum diameter). As the center of gravity is moved forward of this point ($X_{CG}/D = 0.3$), the truncated cone will become more stable ($C_{m_\alpha} < 0$). The center of gravity for the RTG/RB is located forward ($X_{CG}/D = 0.33$) of the maximum diameter when the body is in the small-end first orientation ($\alpha = 180$ deg). Therefore, the RTG/RB will be statically stable ($C_{m_\alpha} < 0$) in this small end first orientation if the small end is a hemisphere. The comparison of some of the destabilization systems is shown in Fig. VII-26. Therefore, a suitable aerodynamic destabilization system must be determined.

Because of the desire for a nonzero C_m ($\alpha = 180$ deg) as well as a position C_m about the $C_m = 0$ ($\alpha = 180$ deg) point, it was decided to use a 45-degree cutoff angle on the cone. The angle at 45 degrees was selected on the basis that a large aerodynamic moment would be generated. Further studies are required to optimize this angle.

This concept has the advantage of having no aerodynamic rolling moment due to the destabilization geometry. Furthermore, it is amenable to a Newtonian analysis to determine the hypersonic aerodynamic characteristics.

161, 162--Selection of Reentry Body Payload

The process of selecting the reentry payload (heat source or heat source plus other components) was as follows:

For the heat source approach, two choices are available:

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- (1) Placing the afterbody between the heat source and T/E hot shoes in a permanent fashion
- (2) Moving the afterbody into the same location upon signal.

Choice 1 was eliminated since, having chosen radiative heat transfer between the heat source and the T/E hot shoes, insertion of an interface would essentially double the temperature drop between fuel capsules and hot shoes. This, performancewise, is unacceptable if a temperature drop is taken on the hot shoes. It is also unacceptable to increase the fuel capsule temperature another 150° to 200° F.

Choice 2 was eliminated because of a basic technical judgment that no movement of components be allowed prior to actual separation.

The heat source plus thermoelectrics approach also presents two choices:

- (1) Radiator is attached to afterbody (exterior) and remains on during reentry
- (2) Radiator is ejected prior to separation initiation.

Choice 1 was eliminated because it presents a reentry body with little if any supporting information as to its reentry characteristics. Choice 2 was eliminated for the same reason as in Choice 2 for the heat source only approach.

The heat source plus thermoelectrics plus radiator (RTG), presented three choices. These are:

- (1) The afterbody moves over the RTG and latches onto the heat shield upon signal.
- (2) The afterbody encloses the RTG during all operations.
- (3) The afterbody functions as the radiator.

Again, Choice 1 is eliminated based on the "no movement" criterion. Choice 2 would require heavy fins on the RTG and attachment to the outer afterbody. The afterbody would then act as the outer radiator. This being the case, it was obvious the approach to pursue was to use the afterbody as the radiator with the thermoelectric cold junction attached. Beryllium was chosen as discussed in Section 112.

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C. CONTROLLED INTACT REENTRY SYSTEM

201, 202, 203, 204--CIR Propulsion Selection

To separate, stabilize and deorbit the RB/CIR, large amounts of energy must be imparted to the RB/CIR in a short but finite time and with minimum reactionary effect on the spacecraft. These requirements imply the need for chemical propulsion rather than mechanical reaction. Further, to minimize weight, high energy rocket systems are called for rather than the so-called gas generator systems.

a. Solid rocket selection

The requirement to provide a positive abort capability for the generator system during boost flight imposes certain design and performance requirements such as minimum size, high thrust-short duration, minimum reaction time, and high reliability and safety. Further constraints are imposed by the need for operation in a spatial environment, possibly after extended storage in this environment. Solid propellant rockets best satisfy all of these requirements. Specifically:

- (1) Solid propellants have a high density (on the order of 105 lb/ft³).
- (2) They can be efficiently designed for high thrust-short duration conditions. Thrust levels on the order of thousands of pounds for durations of a fraction of a second are easily achieved.
- (3) Solid rockets have been utilized successfully in space after months of exposure to spatial conditions, and ground tests indicate that ignition (at least five years in space) and resistance to atomic radiation (at least five years in the Van Allen Belt) will be satisfactory.
- (4) The ignition interval (ignition-to-peak thrust) is on the order of 50 to 100 milliseconds.
- (5) The utilization of a composite propellant is emphasized because of its low hazard classification (Class II--fire hazard).

Solid propellant rockets have been in use for over 20 years in a wide spectrum of applications. Their inherent reliability has been proven by tens of thousands of rocket firings. The fact that a wide variety of applications (hence different rocket designs) has been achieved simultaneously provides a large inventory of on-the-shelf solid rockets from which to draw. The simplicity of solid rocket design permits the

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development and qualification (PFRT and flight) of rockets of the size required for this program to be achieved within 18 to 20 months. This simplicity also permits existing rockets to be readily adapted to other applications through modifications which can be achieved at much lower cost and in less time than a new rocket development. Some additional advantages of utilizing an existing or modification of an existing solid rocket are:

- (1) Early availability of the rocket for additional environmental tests, simulated separation tests, etc.
- (2) Assurance of the availability of a reliable, critical item within the program schedule.

Current rocket technology will suffice for the design, development, qualification and operation of a reliable, new solid propellant deorbit/abort rocket for the program.

Thus, while solid propellant rockets appear to be able to satisfy all operational requirements, it is their reliability, availability and cost which offer the most significant advantages over competitive liquid propellant systems for this application. Therefore, on the basis of all the factors, solid propellant rockets have been selected to perform all the abort propulsion functions.

b. Design and performance considerations

While the requirements for the CIR (exclusive of attitude control) can best be met with the use of solid propellant rockets, the choice of design and performance factors, of necessity, must be selective because of reliability, safety, packaging and the rigors of the spatial environment under which they must operate. In particular, the choice of propellant, case material, motor shape and grain design will be influenced by these parameters.

Propellant selection. High performance composite propellants, for example, ammonium perchlorate (oxidizer) polybutadiene acrylic acid (fuel) with or without aluminum additive, are recommended. The specific impulse of formulations in this class of propellants is in the range of 220 to 240 seconds at 1000-psia chamber pressure and a sea level nozzle expansion ratio. These propellants will perform satisfactorily within temperature ranges of 0° to 120° F for polyurethane binder formulations and -40° to 120° F for polybutadiene binder formulations. However, the larger the temperature variation about the nominal temperature on which performance calculations are based, the wider the range in performance variations. The following performance variations, at constant temperature, can be achieved with the size of rocket required for deorbit of the RB.

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Total impulse ±1% (3σ)
Thrust ±2% (3σ)
Burning time ±2% (3σ)

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1. Fairing Hinged Nose Cap	41. Link Release, Pyrotechnic
2. Fairing	42. Pyroswitch
3. RTG/RB	43. Separation spring, RTG/RB to CIR
4. CIR	44. Carriage Stop
5. Spacecraft	45. Heat Shield Back-Up Bulkhead
6. Spacecraft Adapter	46. Heat Shield Protection Pad
7. T-III Transtage	47. Separation Rocket Ejection Lugs
8. RTG Heat Source	48. Separation Rocket Ejection Lugs Stops
9. Hinged RTG/RB	49. Carriage Position Lock, Pyrotechnic
10. Fairing Heat Sink	50. RTG/RB Aft Cap Hinge
11. Fairing Nose Cap Pneumatic Actuator	51. C-Band Beacon Transmitter
12. Nose Cap to Aft Cap Link	52. C-Band Beacon Battery
13. Separation Tracks	53. Thermal Block Gasket
14. Separation Track Struts	67. Radiator/Heat Shield Splice Frame
15. Separation Carriage	68. Radiator
16. Spin Bearing	
17. Separation Rocket & Mount	
18. Deorbit Rocket	
19. Spin-Up Rocket	
20. Despin Rocket	
21. Command & Telemetry	
22. Separation Band RTG/RB to CIR	
23. Latch Fingers	
24. Electrical Disconnect RTG/RB to CIR	
25. Electrical Disconnect (Squib Fired) RTG/RB-CIR to Spacecraft	
26. Subsystems Equipment Mounting Shelf	
27. Indexing Fitting for Fuel Handling Machine (On Fairing)	
28. C-Band Beacon Antenna	
29. Latch For Hinged Aft Cap	
30. Heat Shield, Ablative	
31. Mounting Bulkhead for RTG/RB	
32. Power Conditioner Resistor Box	
33. Power Conditioner Transistor Box	
34. Pyrotechnic System Solid State Switch	
35. Battery	
36. Command & Telemetry Module	
37. Pyrotechnic Safe Arm Units	
38. Low Voltage Relay Switch	
39. Equipment Access Panel	
40. Separation Band Pyrotechnic Release	

On the other hand, the temperature sensitivity factor (π_K) for the recommended propellants lies in the range 0.10 to 0.12%/°F. Thus, for example, thrust will vary over approximately ± 2.5% if the grain ambient temperature cycles ± 20° F about the nominal design temperature. Temperature variations of this magnitude also could significantly affect vehicle stabilization if the deorbit rockets differ in ambient temperature because of thermal shielding variations. For this reason, passive or active thermal control of the rockets may be necessary to maintain the grain temperature cycle within the limits imposed by the reentry abort error analysis or the limiting propellant firing temperature range. At the present time, with the abort period limited to the first nine hours, this step is unnecessary.

High performance propellants (such as beryllium-loaded composites or exotic encapsulated composites and double-base formulations) were not selected because of the unreliability associated with the generally R&D nature of the exotic explosive classed propellants.

Case material. A steel case will be utilized because of its low susceptibility to damage during transit and ground handling.

Motor shape. Rocket grain configurations can take on any number of variations from spherical to long cylinders, and burning can take place radially, longitudinally or both. Longitudinal burning grains are generally long burning, low thrust units and are not expected to be suitable for this application. Although spherical or low length-to-diameter ratio motors are not the most efficient designs for short burning durations, it is expected that packaging considerations will dictate their use for the deorbit application.

Spin simulations. Solid rockets have been spun in several operational systems, notably Vanguard Stage III and current upper stage Scout missions. Testing and flight experience indicate that rocket motors can be designed to perform satisfactorily under spin rates of at least 160 to 180 rpm.

c. Deorbit rocket design evaluation

The rocket designs considered for application to CRONUS adhere to the design considerations discussed in the preceding section. The applicability of existing and modified existing rocket motors has been given prime consideration in this evaluation. The selection of a deorbit rocket motor from among existing solid rockets or those existing solids that would be subject to reasonable modification was limited due to

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Fig. VII-27. CRONUS General Arrangement

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packaging and/or weight considerations. Applicable rockets required the use of multiple motors to achieve the required total impulse.

A ΔV increment of 1200 fps for deorbit of the reference configuration can be met by utilization of four existing Thiokol SARV Mk I rocket motors. Four of these rocket motors weigh a total of 155 pounds; installation of these rockets is shown in Fig. VII-27. This ΔV increment could deorbit the RTG/RB from any point in the ascent trajectory and coast or final orbit. However, to obtain greater control over the accuracy of hitting within the impact area, a deorbit velocity increment of 1500 fps has been selected. There are three approaches to satisfying this latter performance requirement: modify the existing Thiokol SARV Mk IV rocket; design, develop and qualify a new rocket equivalent in performance to the modified Mark IV rocket motor; and design, develop and qualify a new single rocket which will yield the performance equivalent to four SARV Mark IV rocket motors. All of these motors contain a composite solid propellant.

Modification of SARV Mk IV. The Mark IV rocket motor would impart a ΔV increment of 1700 fps to the reference configuration. To reduce this output to the 1500-fps limit, approximately five pounds (12.0%) of the propellant normally loaded into the rocket must be removed. The propellant reduction of five pounds is performed after processing has been completed and simply entails cutting back the aft propellant surface--an established practice. The gross weight of the modified deorbit system is approximately 220 pounds. This weight is on the order of 75 pounds more than that which could be obtained by developing a new rocket motor to replace the modified Mark IV, i.e., four new rocket motors would weigh about the same as the SARV Mk I deorbit system but would provide a 1500 fps ΔV instead of 1200 fps.

Equivalent single rocket. A single rocket which would replace all four deorbit rockets would provide significant advantages. These are:

- (1) The problems imposed by thrust variations between multiple rockets would be eliminated.
- (2) The alignment of a single rocket on the vehicle centerline could be achieved with greater accuracy than four rockets at the radial extremities. The installation process would also be drastically reduced in time and, finally, structural deflections would have less effect on thrust vector alignment.
- (3) Design efficiency, hence mass fraction, could be improved such that the deorbit system weight is anticipated to be under 130 pounds.

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The particular orientation of the thrust vector for deorbit requires that a single rocket utilize at least two nozzles. The skewed nozzle design is expected to result in a 2% loss of impulse over that delivered from a conventional nozzle design. Precedent for this motor design approach has been established by the development of the Apollo tower jettison rocket motor by Thiokol Chemical Corporation.

This concept appears to be rather attractive from both a weight and a performance tolerance point of view. It was not used in the reference configuration because it is not an off-the-shelf item, and would require some development.

205--Despin Technique Solution

At burnout of the deorbit rocket, the airborne system will be spinning. It is shown in Appendix MM that, for a 60-rpm spin rate, the reentry system would survive reentry heating. However, in order to predict the impact location, it is desirable to despin the system. Several such devices are technically feasible. A yo-yo system, similar to that used on the early Explorer satellites, could be used. It would have the advantage of being able to compensate for tolerances in the spin rate. However, its performance in the presence of moderate or high tumbling rates is not known. Consequently, this system was not selected. A despin rocket system, similar to the spinup devices, was selected as being relatively simple, light and within the present state of the art.

206--Separation of RTG/RB and CIR

In order to separate the RTG/RB from the CIR, only a relatively small impulse is required. Due to the structural interface between the two airborne systems, separation band activated springs are adequate. Other systems (propulsion or gas-activated) would be more complex, less reliable and heavier than the springs. An attempt was made to eliminate even the springs, and to use the forced deflection of the heat shield support structure for separation forces. This approach, however, might cause structural and manufacturing difficulties. Consequently, the spring separation concept was selected.

207--Physical Location of CIR-RTG/RB on S/C

The airborne system must be mounted on the payload. Due to the geometric limits of Nimbus, the airborne system will not fit inside Nimbus (barring major redesign of the S/C). The geometric limits of

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the standard fairing prohibit mounting the CIR-RTG/RB on the side of the S/C. Two options are left; CRONUS can be placed above or below the spacecraft. Emergency separation requirements, fueling considerations and heat rejection requirements indicate that mounting the CIR-RTG/RB system on top of Nimbus is the preferred concept.

208, 209, 211, 213 and 214--Selection of Separation Techniques

Several potential arrangements of the airborne system on the S/C and LV were studied before selecting an arrangement. As part of the physical arrangement studies, it was necessary to examine different airborne systems/LV separation schemes.

The separation study considered the following alternatives:

- (1) Independent guidance system versus S/C attitude-dependent system.
- (2) Independent attitude control during free fall, and/or during ΔV application versus spin stabilization.
- (3) Various reentry body attitudes on the S/C.

The following ground rules were observed:

- (1) No separation prior to fairing jettison.
- (2) Separation system must operate with maximum LV longitudinal acceleration.
- (3) Despin required for any spin stabilized RB.
- (4) All systems must fit within a standard fairing. The fairing may, if necessary, be made longer.
- (5) Nimbus satellite to be used.
- (6) The basic RTG/RB and CIR breakdown is used for all concepts.

Six different concepts were examined. Each is discussed.

a. Concept 1

This concept (Fig. VII-28) uses as an independent attitude control and guidance system to orient the RTG/RB after separation. During the deorbit boost, as well as the separation phase, the system is stabilized along the selected direction by the same attitude control system.

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Fig. VII-28. Conceptual Design, Configuration I

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The system is placed in tandem on Nimbus. The CIR system is attached to the spacecraft, and the RTG/RB is placed vertically over the CIR. This arrangement is essentially symmetric about the LV roll axis. There is essentially no interference between the RTG radiator and the CIR or spacecraft.

Separation from the S/C (and LV) is accomplished by longitudinal separation rockets. In order to avoid the path of the LV, the airborne system is turned slightly during separation.

This concept is composed of three major parts, the RTG/RB, the CIR structure and a system of rockets. The bottom of the CIR is attached to the top of the S/C by separation bolts. The separation and deorbit rockets are attached to the CIR structure and surround the top of the S/C. The RTG/RB is attached to the top of the CIR, and is held down against separation springs by a release device. The control electronics, communications, electrical power, attitude electronics and attitude fuel supply are housed in the CIR. The pitch and yaw attitude thrusters are mounted on the outside of the deorbit rocket nozzle throat, and the roll attitude thrusters are mounted on the outside of the CIR structure.

Upon receiving the separation signal, the separation bolts release and the separation rockets ignite, propelling the airborne system away from the S/C.

While these rockets are burning, the attitude control system, through one of the pitch or yaw thrusters, is causing the separation flight path to curve away from the path of the LV-S/C. Upon completion of the separation thrust, the attitude control system orients the CIR-RTG/RB, by means of the attitude thrusters, for deorbit ΔV application and holds it during the application. At the end of deorbit thrust, the device holding the RTG/RB to the CIR is released, the separation springs push the two bodies apart and they reenter independently. The reentry body survives impacting the water and sinking, while the CIR may be destroyed during descent through the atmosphere.

A preliminary operational and weight analysis of this concept indicated that, although the symmetric and compact arrangement of Concept 1 had certain advantages, the separation system had several drawbacks. In particular, the energy added to the vehicle during separation tended to increase the velocity of the RTG/RB. Unfortunately, to deorbit the RTG/RB, the velocity must be decreased. The desire to eliminate this inefficiency, to reduce the weight of the separation system, and to simplify the guidance/control requirements led to Concept 2.

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Fig. VII-29. Conceptual Design, Configuration 2

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b. Concept 2

This concept (Fig. VII-29) uses an independent attitude control system. The orientation of the system during deorbit is controlled by the orientation of the LV at separation. The deorbit propulsion system also provides the thrust for the RTG/RB-LV separation.

Since the velocity of the reentry body should be decreased during deboost, the propulsion system should point to the aft end of the LV. A compromise must be made between the ideal attitude (direction to the rear) and the obvious interference problems. This compromise is shown in Fig. VII-29. A rail separation system assures proper clearances. It is composed of the following major parts: a support or mounting structure, the CIR, the RTG/RB and a deorbit-separation rocket. The support or mounting structure is permanently attached to the top of the S/C. The CIR is attached to the support structure guide rails by rollers and a release device that keeps it in place prior to the separation signal. The deorbit-separation rocket is attached to the CIR structure which also supports the other subsystems equipment. The RTG/RB is mounted in this structure in the same manner as described for Concept 1. The attitude control thrusters are mounted externally on this structure.

At the separation signal, the guide rail release device is activated and the retro-separation rocket is ignited, propelling the airborne system away from the LV-S/C. The deorbit ΔV is achieved during this rocket burn. The attitude control thrusters maintain this rocket thrust vector direction, as determined by the guide rails prior to separation, during its thrust time.

After the deorbit thrust is complete, the device holding the RTG/RB to the CIR is released, the separation springs push the two bodies apart, and they reenter independently.

A preliminary operation and weight analysis of this concept was performed. As expected, it is somewhat lighter than Concept 1. However, the volume inefficiency of this concept (due to the width of the system) is unattractive.

Since the guidance system has been removed from Concept 2, the only use for the attitude control system (thrusters, sensors, etc.) is to stabilize the vehicle during the operation of the deorbit rocket. The need for an attitude control system would be eliminated if a spin stabilization technique were used during deorbit. The resulting design, Concept 3, is shown in Fig. VII-30.

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Fig. VII-30. Conceptual Design, Configuration 3

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c. Concept 3

This concept uses a spin stabilization system for maintaining the attitude of the reentry body during deorbit boost. The basic approach in Concept 3 on S/C orientation dependency is similar to Concept 2. However, the need for a compact body governs the arrangement of the RTG/RB and the propulsion system, as shown in Fig. VII-30.

The concept is composed of the following major parts: a support or mounting structure, a deorbit-separation rocket package and the RTG/RB. The CIR auxiliary equipment is mounted in the support structure. The support structure is permanently attached to the top of the S/C and contains a spin bearing for mounting the RTG/RB and CIR. The deorbit-separation rocket pack is attached by a release device to the RTG/RB over separation springs. The RTG/RB is mounted in the spin bearing and held down with a release device.

At the separation signal the spin rockets are activated, the deorbit rockets are fired, and the RTG/RB is released from the spin bearing. When deorbit thrust is complete, the despin rockets are activated. The deorbit rockets and mount are released after despin from the RTG/RB, the springs push the two bodies apart and they reenter independently.

An analysis of this concept shows that, although there was some increase in the simplicity and compactness of this concept over Concept 2, the weight of the spin bearing and associated support structure resulted in a net weight increase.

d. Concept 4

Each of the previous concepts had some advantages. A new concept was generated which is supposed to incorporate the following features.

<u>Feature</u>	<u>Advantage</u>
Symmetric arrangement	Volumetric efficiencies, fairing compatibility
Spin stabilization	Weight, reliability, safety
ΔV applied opposite flight path	Efficiency plus weight, independence of roll
Rail separation	Positive clearance
Ground commands to fire deorbit rockets	Flexibility, safety

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Fig. VII-31. Conceptual Design, Configuration 4

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The resulting Concept 4 is shown in Fig. VII-31. In order to clear the boosters, the RTG/RB and CIR are displaced laterally. The guide rails orient the body during the RTG/RB-LV separation phase. To stabilize the body during free fall and also for the time when the deorbit rockets are fired, the system is spun up about the longitudinal axis during separation. Command receivers in the CIR permit ground-controlled ignition of the forward firing deorbit rockets, at any time after separation. It is composed of the following major parts: support separation tracks, the CIR structure, four deorbit rockets, four spin-despin rockets, a separation rocket and the RTG/RB.

The support separation tracks are permanently attached to the top of the S/C. Riding in these tracks is a carriage (with track rollers) to which the CIR is attached through a spin bearing. The separation rocket is mounted on the carriage with a release device. The carriage and spin bearing are both held in place by a single point release device. The deorbit rockets and spin-despin rockets are mounted on the CIR structure. The RTG/RB is mounted in the top of the CIR structure and held down against the separation springs by a release device.

At the separation signal, the carriage and spin bearing release device activates, freeing both for action. The separation rocket and spin rockets fire, thus separating and spinning the RTG/CIR simultaneously. The separation rocket is mechanically detached from the carriage as the carriage leaves the separation tracks. When sufficient lateral distance from the S/C and LV has been achieved (a timed interval), the reentry rockets are fired. Just prior to deorbit rocket burnout, the despin rockets are fired, and upon completion of despin, the RTG/RB release device actuates, allowing the separation springs to move the two bodies apart. They then reenter as independent bodies.

This concept was lighter, simpler (more reliable), and more compact than the other three concepts. It was selected as the separation concept for the reference configuration. The weight comparisons of these concepts is shown in Table VII-6.

TABLE VII-6
Separation Concepts Weight Comparison

Concept	RTG/RB	CIR Structure (exclusive of track)	Tracks and/or Support Structure	Propulsion	Electronics and Ordnance	Total
4	411	76	12	167	58	724
1	0	0	-7	+50	+67	+110
2	0	+15	+75	-17	0	+73
3	0	-55	+168	-18	0	+95

1. CIR structure includes frame, RTB/RB attachment and S/C separation subsystem mounting provisions.
2. Tracks and/or support structure includes all parts that do not separate from S/C.
3. Propulsion includes all solid rockets.
4. CIR electronic systems include all onboard operating systems.

Concepts below dotted line show Δ weights.

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210--Selection of Stabilization Technique

The V vector, when used, must be applied in a specified direction for it to be effective in deorbiting the reentry body. This elemental fact brings out three requirements:

- (1) Finding the proper orientation
- (2) Orienting the propulsion system along this direction
- (3) Keeping the system pointed in the correct direction while thrusting.

Notice that these requirements come in two phases--after separation and during deorbit thrust. The three alternate systems which could perform these functions are:

- (1) A spin stabilization system, which depends upon the initial launch vehicle attitude
- (2) A complete independent guidance and control system, active both during free-fall and during deorbit thrust
- (3) A complete independent guidance and control system, active during coast, followed by spin stabilization during deorbit.

With a launch vehicle MDS system, all three concepts are feasible. The primary advantage of Concept 1 is simplicity; the primary disadvantage is operational rigidity. The advantage and disadvantage are reversed for Concept 2. Concept 3 is an attempt to utilize the operational flexibility of Concept 2 with the simplicity and ruggedness of Concept 1 during deorbit.

A comparison of the three systems is shown in Table VII-7 for 14 different combinations. Only those different goals which are applicable to the criteria for CRONUS are shown here. Also shown is the number of applicable criteria where each concept is favored.

	<u>Applicable Criteria</u>	<u>Concepts</u>		
		<u>1</u>	<u>2</u>	<u>3</u>
Safety	3, 4, 5, 6, 8, 9, 13	6	1	3
Reliability	3, 8, 9	3	0	1
Performance	1, 10	2	0	0

TABLE VII-7
Guidance and Control System Comparison

	<u>Concepts</u>		
	<u>1</u>	<u>2</u>	<u>3</u>
1. Weight	Low	High	High
2. Partial mission success	No	Possible	Possible
3. Reliability of system	High	Low	Low
4. Survival, failure in separation	Possible	Very possible	Very possible
5. Survival, failure ac in free-fall	N/A	No	No
6. Survival, failure in deorbit	N/A	No	N/A
7. Impact dispersion	High (~500 n mi)	Low	Medium
8. Survival, engine out	Possible	No	Possible
9. Survival, close blast	Possible	Unlikely	Unlikely
10. Ground checkout	Simple	Less complex	Complex
11. Development cost	Lower	Higher	Higher
12. Independent control	No	Yes	Yes
13. Quick reaction time	Yes	Maybe	Maybe
14. Suitable for recovery	Poor	Good	Medium

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1. Spin stabilized
 2. Independent guidance and control system
 3. Independent guidance and control system plus spin stabilization

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It is clear from this summary that, for the CRONUS criteria, the simplicity of the spin stabilization technique is of overriding importance. It is also clear that with the specified evaluation method, operational flexibility is a worthless virtue. Changes in the ground rules could change this choice. For example, if the CIR-RTG/RB were specifically designed for a recoverable plutonium system, the optimum system would, most likely, be either Concept 2 or 3. However, for the strontium system, Concept 1 is superior and was chosen.

The specific weights, etc., were discussed in Chapter III.

212--Selection of Aluminum as CIR Structure

This selection is discussed in detail in Appendix QQ.

215--Separation Activation Command Technique Selection

Experience, as well as failure analyses, have shown that certain types of launch vehicle and spacecraft malfunctions which result in termination of a mission can occur in less time than it takes to transmit a warning signal to ground command, interpret the signal, transmit an abort signal to the RB/CIR and achieve physical separation and adequate clearance from the launch vehicle and/or spacecraft. For this reason, a decision was made early in the program to incorporate equipment in the CIR for the receipt of launch vehicle and spacecraft malfunction detection signals that would indicate an impending catastrophic failure. The receipt of such a signal would automatically initiate separation of the CIR-RTG/RB from the spacecraft. A catastrophic failure is interpreted in this case as any failure which occurs very rapidly and would result in uncontrolled reentry of the RTG/RB.

By the same token, other types of failures (such as slow drift in the guidance system, or RTG temperature excursions) either may not cause termination of the mission or may not become critical before mission objectives or a stable orbit have been achieved. Under these conditions, it is desirable to permit ground control to make the decision to abort (separate the CIR-RTG/RB from the spacecraft).

For these reasons, both an MDS and ground command abort activation capability was incorporated in the CIR.

216--Deorbit Command Activation Technique Selection

Since separation of the CIR-RTG/RB may have to be performed (either automatically by MDS or by ground command) at a time which is inconvenient for deorbit because of ground impact restrictions, initiation of the deorbit function should be computed. Rather than burden the airborne system with this function, it was decided to leave this decision to ground personnel.

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217--Tracking, Telemetry and Command Selection

Telemetry support is required for the demonstration flight. It also would be very valuable for the first few fueled flights. Consequently, a 256 channel telemetry system is included in the proposed CIR concept.

To assure proper ground operation, it is necessary to know accurately where the RTG/RB-CIR is after spacecraft separation. Although this could be done from launch vehicle data or with skin tracking (if time is not a factor), it was felt that a tracking beacon in the CIR would greatly enhance the operational capability. This feature in either the CIR or the RTG/RB or both may be considered necessary, although not specifically required according to the statement of work.

Some acceptable systems for TTC are, at present, state of the art. Consequently, there seems to be no need to develop a new system.

Among the systems available are the Air Force space-to-ground link subsystem (SGLS) and the NASA unified S-band system (USBS). SGLS is an S-band integrated carrier system planned for many operational vehicles while USBS will be utilized in the Apollo program. Both are of modular construction and, therefore, with little effort, can be modified to restricted requirements. The two systems are compatible on the down-link side. A wide distribution of ground facilities for these units is available. The Apollo net will include five ships and eight aircraft capable of communication with USBS. Tracking coverage for SGLS is equally extensive since the Air Force Satellite Control Facility and the range ships are to be equipped with this unit. An S-band system seems then to be a logical choice for the CIR. No incompatibility is apparent with communication frequencies for Titan IIIC which operates with VHF telemetry, UHF command and C-band tracking. SGLS is, in fact, presently planned for MOL/Titan III.

With regard to satellite communication systems, it may be expected that these will be selected for many future applications. However, CIR data may always be multiplexed with that of the satellite system. Another reason for selecting the S-band is the intent of the military to vacate the VHF band for telemetry by 1970, thus making much of current equipment unsuitable.

D. GROUND HANDLING EQUIPMENT

301--Ground Handling Equipment Payload

In the optimization of a GHE concept, the heat source configuration is a major consideration. Cask weight growth for comparable payload configurations of the nuclear source and RTG, or RTG/RB or RTB/RB-CIR imposes increasing penalties on the GHE and launch site facility.

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The cylindrical cask weight increases approximately as the square of the characteristic mean diameter. Cylindrical cask growth from a 30-inch mean diameter to 60 inches increases the cask overall weight from approximately 8000 to 32,000 pounds. The handling difficulties and umbilical tower modifications are increased by the same factors.

System installation of a configuration that includes the heat source and RTG, RTG/RB or RTG/RB-CIR becomes increasingly more difficult due to the increasing number of mechanical, electrical and structural connections that would have to be accomplished remotely. Hence, the heat source alone has been selected as the GHE payload.

302--Location of GHE

a. Mobile Service Tower (MST)

It is desirable to place the fuel block system on a fixed structure that remains adjacent to the launch vehicle through the launch count-down. Installation of the loading system on a structure such as the MST which is moved away from the launch vehicle during the count-down would necessitate the use of remotely controlled drives and alignment equipment, and remote disconnect and coupling connections for electrical power, GN_2 , air conditioning and other facility interfaced services provided to the MST.

b. Movable tower concept

Figure PP-4 in Appendix PP is not tailored to a specific complex and is a study of a movable tower concept. It consists of a tower structure mounted on a trailer or vehicle. The cask would be installed at the tower base and the fuel block, supported by the cask lid, would be hoisted up the tower. A transition to horizontal motion would take place at the top of the tower where a dolly would pick up the fuel block and carry it out the boom to the RTG.

A Bucyrus Erie type Mark 1-50 tower crane or equivalent was reviewed and its applicability evaluated within this study. A tower is obtainable consistent with the loads to be supported. Modifications for emplacement of the traversing rails for tower, or footings and pads if the tower is to be stationary, are required. Space requirements at the site and access provisions for the positioning and movement of the MST and LV on its transporter influence the moving tower configuration from the viewpoint of mobility and boom reaches.

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Truck-mounted towers employing outriggers for stability have been reviewed but are of questionable overall advantage from a consideration of the deflections and other structural flexibilities inherent in such a system. Fuel block loading into a LV-mounted RTG is made more complex from such a system than from an umbilical tower-mounted fuel handling system with the inherent greater rigidity of a stationary structure. The erection of this structure is a time dependent process estimated similar in span to that necessary to install the palletized selected concept.

The addition of a moving tower at the launch site to support the fuel loading system is indicated to be financially disadvantageous to the loading system. Approximately 130 feet of tower is required to position the fuel loading system at the payload level and the existing structure would not be utilized. This capability exists in the UT with minor adaptive modifications.

c. Umbilical tower

The Titan IIC umbilical tower provides the structure necessary for the support of the selected concept fueling equipment at the proper elevation and orientation relative to the launch vehicle. The umbilical tower is a stationary structure designed to remain in position during launch and is, therefore, continuously available to the nuclear source loading or unloading operation as is dictated by the program of the launch countdown. Use of the umbilical tower for installation of a palletized fuel handling concept avoids a space problem at the launch site and provides a cost avoidance advantage by eliminating approximately 130 feet of erectable moving tower which is not required when the existing umbilical tower is used. Hence, the UT is selected for locating the GHE.

303--Palletized Versus Permanently Installed GHE

Within the contract and included ground rules letter is an indicated interest in a concept identified as a "Moving Tower Concept." This configuration was investigated in detail in Appendix PP, and the Martin Company interpretation of the Customer's desires as expressed by the stated concept is one in which the nuclear GHE is independent of the launch vehicle GHE and facility and which includes within its design a maximum amount of mobility. Mobility is interpreted to mean readily removable from one site, transported and reinstalled on another site without major, permanent type site modifications which would tend to disrupt the existing functions of the launch site equipment and which would require significant and expensive rework to the facility to return it to the as-was condition. The selected concept is palletized so as to facilitate its ability to be erected or de-erected to and from the various sites as required.

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304--Selection of GHE Concept for WTR

Chapter IX describes the Western Test Range. The umbilical tower at WTR used with the Titan IIIB in a typical WTR launch mission is a light structure designed for other use and modified for launch of the Titan-Agena. Modifications to the Titan IIIB umbilical tower, similar to those specified for ETR, are not feasible and if effected, it is estimated these modifications would approximate the cost of an ETR-type umbilical tower.

Review of other facility equipment indicated that the modification necessary to the Titan IIIB MST to accept the selected GHE concept is minimum, requiring automation of the MST positioning capability. The interfaces between the MST and the static complex are few as compared to the interfaces between the Titan IIIC MST and its complex. The penalties incurred in accomplishing the necessary automation to the Titan IIIB MST are less than the efforts and costs to rebuild the UT to a Titan IIIC configuration.

305--Selection of Compliance Mechanism

Many systems, to compensate for the relative motion between the launch vehicle and the tower-supported GHE were reviewed (REF. Appendix PP). The compliance system shown in the selected concept is preferred over schemes using cables or flexible structures to compensate for relative motions. The dynamic properties of the selected concept mechanism are uniform in the station plane of the launch vehicle (the plane of indicated maximum relative motion), analytically predictable and adjustable by design to minimize the influence of the fueling equipment on the launch vehicle structure.

306--GHE Detach Prior to Liftoff Versus Detach at Liftoff

Advantages are indicated in a concept where the fueling equipment remains connected to the launch vehicle until launch vehicle liftoff initiates its disconnect and retraction. These advantages are particularly apparent in a side loading concept, i. e., one which penetrates the launch vehicle side structure through a self-closing access panel. A connection maintained in this scheme eliminates the necessity of accomplishing a remote reconnection of the fueling equipment with the launch vehicle in the event of an aborted launch.

The selected top loading concept prohibits the continuation of the design philosophy of a connection maintained until liftoff due to the dynamics of the launch vehicle and fueling equipment at liftoff. The fuel boom must be retracted to provide launch clearances. The mechanism and power required to accelerate the boom add unnecessary complications to the fueling system configuration. This is also true of the equipment needed for arresting the boom.

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The alternative to the complexities indicated is presented in the discussion on fueling equipment for the selected concept. A satisfactory solution to the problem of recoupling in a nuclear environment is presented in a system which accomplishes this coupling with ease and repeatability. Remote disconnect and reconnection is a normal operational function in the selected concept, thus facilitating a less complex overall system of equipment.

307--Top Loading Versus Side Loading

The study of the 14 idealized concepts point up certain advantages of some methods of loading over others. However, the advantages and disadvantages of top and side loading are not so explicit and definitive as to permit a technical judgment stating that one method is definitely superior to the other based on GHE considerations only. Influences and tradeoffs with the overall system, i.e., the RTG and RB, and the optimization of the system and inputs relative to the LV used have determined the selected directional mode as top loading. In general, the following advantages and disadvantages are inherent in top or side loading schemes.

a. Gravity assist and system symmetry

In a typical top loading scheme, gravity simplifies the configuration of the fueling loading equipment by making feasible a system essentially symmetrically loaded or one which tends to minimize the cantilevered loads on the fuel handling mechanism, thus simplifying the mechanism design. Loads on the cantilevered structural boom will be the same, generally, for either mode of loading, but the loading in the mechanism operating at the LV end of the boom will vary. Rotation of this mechanism supporting the fuel block 90 degrees from the vertical for side loading purposes requires a design consistent with the moments now induced on the mechanism, i.e., fueling probe, compliance mechanism and other mechanisms in the system.

b. Compliance considerations

A comparison of the compliance requirements indicates a desirability for a top loading method. To illustrate the relative complexity of compliance mechanism compensatory motion, the system must be viewed as a top loading scheme supported fuel block looking down into the RTG opening on the LV. A midpoint in this opening (assuming the point in the centerline of the LV) is approximately 130 feet above the ground and allowing this point ± 3 inches excursion, it is seen that this point moves in a spherical plane with a 130-foot radius in a circumscribed area of a 3-inch radius. A section cut through this plane of motion is essentially a flat plane since, as shown in the sketch (exaggerated

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for clarity), h is approximately zero and, therefore, Point O as viewed by the observer moves, for all practical purposes, in a plane. The compliance equipment for a top loading concept then compensates for major motion in the station plane of the LV. Iteration of the process to a side loading consideration will show that an observer, and hence the compliance mechanism, sees in a side opening RTG, a point moving ± 3 inches laterally and essentially zero vertically, in a plane perpendicular to the line of observation and, in addition, this plane is moving ± 3 inches along the line of sight. The design penalties of this additional motion in a second direction are obvious.

c. Cask complexities

The study of the concepts indicate that a top loading method permits the use of a less complex cask and a simple symmetrical motion in the transfer of the fuel block to and from the cask and RTG. Side loading imparts the added complexity in cask design as splitting the cask halves and parting laterally, passing through the cask from bottom to top and then to the RTG or providing the fueling equipment with a 90-degree rotational capability so that the fuel block may be extracted from the cask and rotated 90 degrees for insertion in the LV in a side manner.

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VIII. DESIGN CREDIBILITY

This chapter presents a credibility assessment of the selected CRONUS design. The credibility evaluation consists of examining the design with respect to the following:

- (1) Selection of materials and components that have established credibility based on similar applications and including the same environmental conditions.
- (2) Depth of analyses supporting the design and techniques of analysis applied.
- (3) Identification of problems requiring solution so the design may become established qualified hardware.

The third item is included as early identification of problems improves the chances of solutions being found, and thus increases design credibility.

Analysis summary tables are provided later in this chapter that are summaries of the detailed presentation of techniques and results appearing in the appendices. Neither the analysis summaries nor the appendices should be interpreted as a complete accounting of the analyses performed during the feasibility study. Many of the more straightforward analyses have been omitted, e.g., structural analysis of the AGE.

The chapter is divided into three sections. For convenience the re-entry body credibility and that of the CIR are discussed together. The main sections are:

- (1) Power Generation (RTG)
- (2) Deorbit (CIR/RB)
- (3) AGE.

A. POWER GENERATION (RTG)

The analyses supporting the RTG are summarized in Table VIII-1.

1. Heat Source

The most serious development problem associated with the CRONUS design is that of the heat source. The design requirement on the Haynes-25 fuel capsule imposes an extension from present state-of-art

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TABLE VIII-1
Generator System Analysis Summary

<u>Type</u>	<u>Name of Analysis</u>	<u>Appendix</u>	<u>Results and/or Significance</u>
Thermal	RTG/RB Thermal Analysis	C	Radiator sized and shaped for minimum weight capsules survive reentry heating and Titan fireball
Thermal	Bare Heat Source Temperature Calculation	A	Heat source temperature in air during handling 775° F
Thermal	Steady State Fuel Capsule Temperature	B	Capsule temperature determined at end and beginning of life for normal and open circuit operation
Thermal	Thermal Efficiency Analysis	II	RTG thermal efficiency with advanced high temperature Min-K in module = 89% at end of life overall system efficiency = 4.4%
Thermal	Capsule Surface Temperature Parametric Study	K	Parametric graphs generated for quickly determining capsule temperature as function of area, emissivity and heat flux
Thermal	Sampled Perturbation Theory	VI	Theory of transient solution to generalized heat transfer problem
Thermal	Absorption Factors	N	Theory of absorption factors used in radiative heat transfer
Thermal	Capsule Temperature Time History for Pu-238	X	Open-circuit capsule temperature determined for 300 years
Thermoelectric	SiGe Versus PbTe RTG Comparison	L	SiGe T/E chosen on basis of generator lightness, compactness and equal or better efficiency as compared to PbTe
Thermoelectric	Thermoelectric Analysis	Q	T/E efficiency, No. of couples and element size determined
Thermoelectric	Beginning of-Life Electrical Characteristics	R	Power, hot junction, current and voltage determined as function of time over 3-yr mission
Thermoelectric	Segmented Thermoelectric Subsystem	S	Efficiency gain of 4 to 8% over present unsegmented couple realized using N type InAs GaAs or N type GaSb InSb Break through needed on P leg
Thermoelectric	Cascaded Thermoelectric Subsystem	U	Cascaded RTG significantly heavier at 479 lb compared to 345 lb for noncascaded Only possible advantage of cascaded system is efficiency (3.27% overall vs 4.42%) If the price of PuO ₂ of \$1000/watt(t) valid cascaded system offers savings of \$1 million system Savings would be offset by additional launch costs of heavier cascaded system Theoretical efficiency advantage possibly important due to availability of PuO ₂
Thermoelectric	Derivation of Cascaded Efficiency Expression	W	Formulae derived used in cascaded T/E subsystem analysis
Structural	Fuel Capsule Thermal Shock Analysis	D	Capsule survives thermal shock associated with water quench from operating temperature
Structural	Fuel Capsule Structural Analysis and External Pressure Capability	E	Capsule survives external hydrostatic pressure of ~9400 psi (~21,000 ft) at 70° F and blast overpressure of 4500 psi at 1600° F
Structural	Capsule Impact Analysis	F	Median rupture velocity (V ₅₀) determined as ~300 fps significantly higher than terminal velocity ~220 fps
Structural	Radiator Structural Analysis Blast Overpressure	G	Radiator will survive a shock overpressure of 100 psi
Structural	Internal Capsule Pressure Buildup Analysis Pu-238	F	For 10-yr helium pressure containment with Pu-238 capsules substituted into the strontium reference design wall thickness of 0.2 inch is required
Electronics	Power Conditioning for Peak Loads	V	Optimum power conditioner and power storage subsystem in conjunction with NIMBUS available
Electronics	Electronics Thermal Analysis	X	Electronics temperature determined with Titan fireball

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to higher temperatures. Although the selection of Haynes-25 as the fuel capsule material is considered a good one, it must be recognized that careful attention must be applied to capsule design to take advantage of the inherent advantages of this material.

a. Impact analysis

A fuel capsule must be impact resistant so as to ensure containment of the radioisotope fuel under certain accident conditions. Some credible accidents which may involve impact of the fuel capsules are: roadway accidents during transportation of the fuel capsules; handling accidents which could include dropping of the fuel capsules from the umbilical tower; possible destruction of the launch vehicle and impact of the reentry body at its terminal velocity; and impact of single fuel capsules at their own terminal velocity. Previous experience has shown that if impact against a hard, unyielding surface is predicated, then the latter case--impact of individual capsules--is by far the worst.

More complex accidents can be hypothesized, such as impact with the sharp edge of a steel structural shape, or the impact of a fuel capsule with the entire Titan III behind it. As a practical consideration, however, the fuel capsule is a very hard and tough structure, and the chances of its impacting and simultaneously being hit by an equally hard and tough component of the Titan booster are so small as to be considered incredible.

The basis for the fuel capsule design was therefore taken to be the impact of a single fuel capsule at its free-fall terminal velocity against a stationary rigid body.

(1) Energy considerations

In any discrete impact between a moving and a stationary body, the kinetic energy of the moving body prior to impact will be absorbed in the following ways:

- (1) Elastic deformation of the moving body
- (2) Plastic deformation of the moving body
- (3) Rebound of the moving body
- (4) Temperature rise of the moving body
- (5) A similar set of factors (Items 1 through 4) for the rigid body
- (6) Radiation of sound and thermal energy to the surroundings.

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If it is assumed that the stationary body is perfectly rigid and unyielding (as approximated by a large block of granite), then Item 5 has zero contribution to energy absorption. Experience also shows that the contributions of Items 3, 4, and 6 are minor compared with those of Items 1 and 2, and can be neglected for purposes of a conceptual fuel capsule design. Since Haynes-25 is the primary impact absorbing material in the fuel capsule, it may be stated as an approximation that the kinetic energy of the fuel capsule prior to impact will be absorbed by elastic and plastic deformation of the encapsulant material.

(2) Analytical techniques

The basis for the analysis is shown by two curves in Fig. VIII-1. The curves represent the distribution of terminal velocity and rupture velocity for a large number of test samples. Only a small number of samples will have terminal velocities in excess of their respective rupture velocities as represented by the overlap between the two curves.

Fig. VIII-1. Distribution of Free-Fall Terminal and Rupture Velocities for Impact-Resistant Fuel Capsule

The object of the design is to spread the curves apart so as to minimize the probability of capsule rupture upon impact from a free fall. This may be done by reducing the capsule size or extending its surface, thereby reducing its terminal velocity. It may also be done by using the most impact resistant configuration and materials known.

Various methods have been developed for prediction of fuel capsule impact resistance. Some of the work which is most applicable to the design of cylindrical fuel capsules has been published by Dr. Stoneking and Messrs. Young and Colp of Sandia Corporation (Ref. VIII-1). Together they performed an 810 specimen test phase

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in which the effects of the capsule wall thickness and L/D ratio on the impact integrity of the capsules was investigated for end-on impact. This test was a natural outgrowth of earlier tests performed by Sandia Corporation on the SNAP 9A fuel capsule and of structural analyses performed by Dr. Stoneking.

One of the conclusions was that with other parameters held constant, rupture velocity increases as the thickness of the specimen tube wall increases. This was expected since the primary mode of failure for end-on impact was found to be shear failure of the tube wall.

However, it must be noted that increasing the capsule wall thickness simultaneously increases its weight and its free-fall terminal velocity. It is not clear, therefore, that increasing the capsule wall thickness will always reduce the hazard associated with fuel release. Each fuel capsule design must undergo a separate analysis in order to properly evaluate the hazard involved in its use.

A second conclusion was that with other parameters held constant, rupture velocities decrease as the L/D ratio is increased. This could also be expected from the test conditions, since increasing the L/D ratio of a constant diameter capsule is equivalent to increasing the length, weight and kinetic energy of the capsule.

A third conclusion was that there is no apparent relationship between kinetic energy and rupture velocity. The experiment was not designed to consider the effects of kinetic energy which varies with the tube wall thickness, L/D ratio and impact velocity.

It is felt that in a design program involving a space power system, weight optimization is an important factor, and impact test results should be evaluated on the basis of encapsulant weight and kinetic energy.

This approach was taken by Conway (Ref. VIII-2) after impact tests had been performed on the SNAP 9A and the original SNAP 19 fuel capsules. His method related the impact energy to the encapsulant weight and energy absorption capacity at the impact temperature. After considering the condition of the impacted fuel capsules and their rebound characteristics, he defined an energy absorption density which is equal to the sum of the following factors:

- (1) The area under the elastic portion of the stress-strain curve for the encapsulant material at its test temperature.
- (2) A factor, F , times the area under the plastic portion of the stress-strain curve for the encapsulant material at its test temperature (F to account for the fraction of encapsulant material plastically deformed).

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Knowing the approximate median failure velocity (V_{50}) from the impact test results, the impact energy was equated to the energy absorption density times the volume of encapsulant material. In this way, a plastic deformation fraction of 0.33 was obtained for the original SNAP 19 fuel capsule design.

This method is approximate in nature and relies on empirical results to obtain the factor F . However, it may account for several discrepancies which are not otherwise calculable, such as the approximate nature of the stress-strain curve (which is assumed to be comprised of straight lines joining the origin, the yield point and the point of rupture), the fraction of encapsulant material which is deformed prior to rupture, and the fact that the capsule does not deform in pure one-dimensional tension or compression, but rather in three-dimensional tension, compression and shear.

The importance of this method lies in the fact that once the impact resistance of a single fuel capsule design has been evaluated empirically, the data can be extrapolated for other encapsulant materials, other temperatures or even other capsule configurations having comparable energy absorption efficiencies.

The method used in this study is similar to Conway's but is simplified by the use of an absorption efficiency factor, K . The factor, K , is applied to both elastic and plastic energy absorption and is dependent only on capsule configuration. An encapsulant weight fraction, w_c , is defined as the fraction of total fuel capsule weight that must be allocated to encapsulation. The relationship between w_c and impact velocity, fuel capsule configuration and encapsulant material and temperature is derived in Appendix F, and is given as:

$$w_c = \frac{V^2}{2gK u_a} \quad (\text{VIII-1})$$

where:

V = impact velocity (fps)

g = acceleration due to gravity = 32.2 fps

u_a = specific energy absorption (ft-lb) of encapsulant material.

The adequacy of this expression was assessed by substituting impact test data from the SNAP 17A capsule impact test program. The SNAP 17A capsule was selected because it was tested at relatively high temperatures (nominally 850° to 1600° F), used Haynes-25 encapsulation, contained sintered fuel pellets, and had a similar configuration ($L/D = 5.2$) and terminal velocity (225 fps) to that of the CRONUS fuel capsule.

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The following results were extracted from the Series B impact test data (Ref. VIII-3)

$T = 1600^{\circ} \text{ F}$ (nominal)

$w_c = 0.51 \text{ lb encapsulant/lb total capsule}$ (for 0.080-inch wall)

$V_{50} = 260 \text{ fps}$ (estimate based on capsules 6B through 10B having a 0.080-inch wall). Note these were development tests only, and data presented are insufficient for calculation of V_{50} by standard statistical techniques)

By obtaining $u_a = 3080 \text{ ft-lb/lb}$ (see Chapter VII) and substituting into Eq (VIII-1), the absorption efficiency factor was calculated.

$$K = \frac{V^2}{2g w_c u_a} = \frac{(260)^2}{2(32.2)(0.51) 3080} = 0.67 \quad (\text{VIII-2})$$

This indicates that prior to rupture, the SNAP 17A fuel capsule absorbed a quantity of energy equivalent to deforming 67% of the encapsulant material to ultimate tensile rupture. It was evident from the impacted test sample that less than 33% of the material was deformed to that extent. The SNAP 17A fuel capsule was therefore more efficient in absorbing kinetic energy than could be predicted by the equation using energy absorption of the encapsulant material alone.

An attempt was made to calculate the energy absorption of the fuel pellets, but this was unsuccessful because the boundary conditions between the fuel and its encapsulant are not known during impact.

In the absence of a better analytical technique, the expression for w_c was utilized to account for energy absorption by the fuel. Failures which occurred in the SNAP 17A capsule were located in the heat-affected end cap weld zone, which has lower strength and ductility than the parent material. Since the CRONUS fuel capsule has no end cap welds, it is judged to have an even higher absorption efficiency than SNAP 17A. Its efficiency factor, K , was estimated to be 0.90, which would yield a nominal rupture velocity of 301 fps for the same encapsulant weight fraction $w_c = 0.51$.

(3) Conclusions

The encapsulant weight fraction, w_c , and the absorption efficiency factor, K , were established as criteria for impact resistance optimization. An expression including these factors was developed which can

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be used to determine the effects of encapsulant material, impact temperature and velocity on the capsule design.

The CRONUS capsule characteristics listed herein were based on the use of this expression with appropriate data from the SNAP 17A test program.

b. Development problem

The selection of Haynes-25 for the fuel capsule material has minimized the hazards relating to fuel release and dispersion, and has reduced the heat source weight.

The primary problem associated with the use of Haynes-25 at the designated operating temperatures is the volatilization and release of chromium from the fuel capsule surface. A significant weight loss would be intolerable because the escaping chromium could cause electrical short circuits by redeposition on adjacent thermoelectric surfaces. Also, the remaining fuel capsule material would no longer be Haynes-25, and its performance would not be known.

There are several approaches that could be taken to solve this problem. One of the most promising (currently being evaluated on the SNAP 27 program) is the application of an external coating which would prevent volatilization of the Haynes-25 constituents. Coatings which should be considered include iron titanate, calcium titanate, flame-sprayed Ha-27-Ta, and Al-90. Another approach would be the stabilization of naturally formed chromium oxide for long-term operation in a space environment by means of rare earth additions to the alloy.

The best solution to the problem may consist of a coating system rather than a single coating to achieve the desired material stability and high emissivity.

c. Fabrication and assembly

Standard hot cell fabrication and assembly techniques will be used to load the fuel capsules and assemble the heat source. All metallic and fibrous insulation components, including the fuel capsules and spacers, upper and lower support caps, tube and latch mechanism, threaded ring, lockwire, capsule retainer rings and insulation components will be supplied to the fueling facility prior to the start of the fueling operation. Special jigs and fixtures will also be supplied.

Since the centrally welded fuel capsule is somewhat of a departure from more conventional designs using welded end caps, fueling facility personnel were asked to review the design and comment on its fabricability. It was established that the centrally welded capsule would be

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more difficult to gage for selection of internal spacers since two pellet stacks (one in each end) would have to be measured instead of one. It was also established that the weld jig would have to be more complex in order to grip both ends of the capsule. These modifications are minor in nature, and existing loading and welding techniques will not be perturbed.

It was also established that several jiggling techniques are available to hold the capsules in position to assemble the heat source. The preferred assembly procedure would be:

- (1) Place the upper support cap in the inverted position on a turntable
- (2) Assemble the insulation and capsule retaining ring to the support cap
- (3) Install the fuel capsules one by one rotating the turntable and jiggling the capsules as required
- (4) Install the lower capsule retaining ring, insulation and support cap
- (5) Install the threaded ring by tightening to the desired torque, then lock the ring in place using lockwire.

Handling tools will be provided which duplicate the gripping action (inside the support tube) of the fuel handling machine.

2. Thermoelectric Subsystem

a. Contributing SiGe technology

Considerable experience exists in the field of SiGe thermoelectric energy conversion technology. This ranges from basic research in thermoelectric materials to the large-scale fabrication of thermoelectric power conversion modules and, in the case of fossil fuel generators, the design and construction of complete systems.

These past and present efforts may be divided into the following major categories:

- (1) Thermoelectric materials
- (2) Thermoelectric materials contacting technology
- (3) Thermoelectrics for radioisotope systems

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Fig. VIII-2. Status of Module Operational Life Tests

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- (4) Thermoelectrics for reactor systems
- (5) Thermoelectrics for fossil fuel systems
- (6) General thermoelectric technology.

These items are not discussed further in this report but noteworthy is the extent of electrical, mechanical, environmental and operational testing for all new concepts ranging from small-scale laboratory tests to full-scale module life tests. This procedure has led to devices constructed with extremely high levels of reliability at operating conditions.

b. Life and environmental performance

A significant quantity of life performance data has been accumulated which illustrates the excellent stability and reliability that can be obtained with silicon-germanium systems. The current status (as of 9/30/66) of major life tests on SiGe thermoelectric modules is shown in Fig. VIII-2.

Although differences in the module structures exist between SNAP 10A, advanced SNAP 10A, SNAP 17A and SNAP Poodle designs, much of the data from the life tests can be correlated for credibility concerning the present design. These test data are not presented here but may be found in the appropriate literature on the various programs.

c. Development problems

Based on two-year SNAP 17A data, it appears that the performance of SiGe as proposed in this design should be predictable for the five-year period. Based on SNAP 10A reliability data, it also appears that reliability goals for this system can be achieved.

Some innovations are used in this design such as the large shaped hot shoes and the contact interface between the element terminating studs and the radiators. These are not expected to present difficulty, based on an RCA assessment and source preliminary interface conductance data.

One area that must be further explored concerns the behavior of silicon-germanium alloy in high nuclear radiation fields. According to RCA, gamma radiation effects on silicon-germanium alloy have not been completely studied but dosages in excess of 10^9 roentgens are not anticipated to have harmful effects. This total dose is approximately that which applies to the proposed design.

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d. Fabrication and assembly

The Flexi-Mod construction embodies several unique and practical features particularly adaptable to the RTG concept proposed with its curved supporting beryllium radiator.

The problems of fabricating the module are simplified with Flexi-Mod since it can be maintained in a planar configuration during processing. After fabrication, the flexibility of the Flexi-Mod construction permits its mating to the nonplanar radiator.

The RTG radiator and other support structures are prefabricated as a subassembly. Thermoelectric modules as subassemblies, partially insulated and interconnected electrically, are assembled to the radiator as discussed in the design section. As part of this operation, accomplished at the power system contractor's site, thermal insulation is put in place, all intermodule electrical connections and generator terminations are made and all converter instrumentation is assembled.

3. Thermal Insulation

The selection of advanced high temperature Min-K has been based on the successful development of this material by Johns-Manville under USAEC Contract No. AT(30-1)-3633. If it is not successful, a composite insulation can be supplied which will consist of a layer of foamed or corrugated ceramic insulation adjacent to the capsule retaining rings, and a layer of Min-K 2000 between the ceramic insulation and the support plate. The ceramic insulation will spread the capsule compressive forces over the Min-K 2000 to 1400° F or lower, which is within the allowable operating limits for that material.

4. Radiator

Forming of the conical beryllium shell is the most complex fabrication problem in CRONUS. Beryllium fabrication technology has advanced in recent years to the point where operational structural configurations utilizing this material are currently being used, and further applications are planned. Nonetheless, the low ductility and susceptibility to brittle fracture, which is characteristic of this material, require that the implied fabrication problems be considered in the early stages of design to the extent that they may be mitigated by acceptable design modifications, and that suitable development work be initiated as indicated in order that practical solutions are available at the proper phase in the overall program. Prior experience has indicated that forming, drilling and other machining operations on beryllium may create problems such as crack propagation and delamination.

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There are few licensed beryllium fabricators capable of handling a part of this size. Licensing is required because the material in certain forms is toxic. Persons who breathe air laden with beryllium dust or fumes may contract berylliosis or dermatitis, so that strict adherence to public health regulations is required.

At least two beryllium fabricators* are aware of this radiator configuration and are interested in working with it, and discussions were held with several persons familiar with beryllium fabrication techniques. On the basis of these discussions, three methods will be proposed for fabricating the conical shell, any of which will have a high probability of success.

The first method is to machine the cone from a hot pressed cylindrical billet of beryllium. This would provide a grainy metallic structure having ample mechanical strength in all directions. Because of the expense of the material and the fact that so much of it would be wasted, this method is probably the most expensive of the three. The large expense can be reduced to some extent by starting with a hollow cylindrical billet, and by salvaging the material which is removed.

The second method is by back extruding a full-sized conical shell. Using this method, a billet of beryllium metal would be pressed between male and female conical dies to produce a thin wall conical shell which could subsequently be machined to the desired wall thickness. This method would produce somewhat higher strength due to grain orientation during the extrusion process. It would use the metal more efficiently so that material costs would be lower. The die cost would be high, but this could be spread over the number of radiators produced.

The third method would be to form the cone in segments from cross-rolled beryllium plates, then braze them together to form a continuous structure. A half-lap joint at the adjoining edges would provide adequate strength without complexity of machining. By using cross-rolled material, maximum strength would be obtained in the parent material, and thermal conductivity in the longitudinal and meridional directions would show a slight increase over that of the hot pressed or extruded material.

At present the third method appears to be most economical, whereas the first method can be used with the greatest confidence. A more detailed comparison should be made on the basis of mechanical performance, cost and schedule requirements.

During this study phase, a coating for the beryllium radiator was not identified. It is imperative for normal steady-state operation of

* SOLAR Division, International Harvester Company, San Diego, California and the Beryllium Corporation, Reading, Pennsylvania.

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the radiator in orbit that a high emissive surface be provided to reject heat. Furthermore, this coating must have a low solar absorptive coating to minimize direct and reflected solar input. Under the SNAP 27 program the same problem has existed and a solution appears feasible. Hence, the approach for CRONUS will be to take advantage of the SNAP 27 progress as much as possible and hopefully adopt one of the coatings already being developed and tested.

B. DEORBIT (CIR+RB)

Table VIII-2 presents a summary of analyses performed to support the design of the CIR system and the RB portion of the generator system.

1. Reentry Body

The design of the reentry body is based upon present state-of-the-art techniques and materials. The basic RB concept--no active attitude control during entry into the atmosphere--has been successfully used in other concepts.

For the reference configuration, both analog and digital computations appear to confirm the fact that, for reasonable initial conditions, the designed reentry vehicle will orient itself around the only stable entry attitude. Since this system may have to survive under malfunctioning conditions, it is necessary to investigate the reentry kinematics arising from other than reasonable initial conditions. At the present, the chosen reentry shape is adequate for initial pitch, yaw or roll rates up to 600 deg/sec. The exact behavior at higher rates, or even at lower pitch and roll rates combined with other nonstandard characteristics, must be studied. In particular, an evaluation must be made of the relative importance of high (abnormal) spin rates. If such high rates are of critical importance, then an investigation must be made to determine the aerodynamic properties of the reentry vehicle that are required to compensate for such rates.

At the present time, the aerodynamic properties of the chosen configuration appear adequate. Some question may exist about the asym-

metric nature of the RB, and the nonzero of $\int_0^{360} C_m dx$. In a planar

(three degree of freedom) representation, the net pitching moment has a tendency to spin up the reentry body. However, in a real, six-degree-of-freedom world, the effect of the net pitching moment seems to be quickly compensated by the slow, unavoidable roll or yaw of the reentry body. The amount of allowable net integrated moment must be studied

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TABLE VIII-2
Summary of Analyses

<u>Analysis</u>		<u>Appendix</u>	<u>Conclusions</u>
Reentry Trajectories	Reentry Trajectories	III	Presents ballistic reentry trajectories. Compilation of various worst entry conditions and generation of design constraints.
	Dispersion Analysis	P	Develop analytical and computational techniques for analyzing impact.
	Abort and Impact Area Contours	II	Impact points and 3σ contours for various flight and orbit phases. Operational analysis for tracking and command control.
	ΔV Requirements	LL	Evaluate ΔV requirements on basis of range sensitivity, impact area and ascent trajectories. $\Delta V = 1500$ fps. Selected for design.
	Dynamic Reentry Analysis	MM	Six degree of-freedom analysis of reentry body in atmosphere. Dynamic reentry computed for several reentry conditions and initial tumbling rates. Maximum allowable tumbling rates established (600 deg/sec pitch, 600 deg/sec roll) to permit stabilization in decaying orbit.
Attitude Control	Analytical Study of Tumbling Reentry	AA	Analytical verification of computed results of MM.
	Guidance/Control Characteristics	BB	Design and analysis of S/C independent guidance and attitude control systems. Systems not used due to weight (~90 lb), complexity and lack of need.
	Six Degree of-Freedom Program	CC	Digital six degree of freedom variable mass trajectory program to analyze stability of spin stabilized system during separation, free fall, deorbit and despin.
	Separation and Deorbit Stabilization	KK	Results of digital analysis (CC) applied to separation and deorbit phases. Analysis of dispersions as function of spin rate, thrust misalignments, etc. Analysis of allowable tipoff moments. Demonstration of adequacy of design for 2 total thrust tolerances.
Aerodynamics	Aerodynamic Coefficients and Pressure Distributions	Y	Results of Newtonian analysis and test results. Hyper-onic aerodynamic coefficients as function of total angle of attack and roll angle.
	Reentry Heating Analysis	JJ	Aerodynamic heating distribution as function of body position and angle of attack. Integrated total heat for spectrum of entry conditions as function of peak angle of attack during reentry. Evaluation of reentry heating margin of safety for radiator.
Miscellaneous	Terminal Velocities	NN	Terminal velocities for reentry body with or without attached CIR. Terminal velocity for individual capsule and heat source computed.
	Heat Shield Analysis	O	Description of transient, one dimensional heat transfer program including irreversible chemical and physical changes of ablator. Used to size and select heat shield.
	Micrometeoroid Penetration Probability	II	Estimated probability of penetration as function of material thickness, surface area and time. Used to show no micrometeoroid shields required for reference configuration.
	SGLS Description	II	Performance and description of selected SGLS system.
	Propulsion Analysis	DD	Sizing and design of rocket motors. Demonstrate adaptability of existing hardware.
	Structural Design Studies	QQ	Design conditions for CIR and heat shield structure generated. Phenolic glass honeycomb heat shield substrate chosen on basis of weight and thermal conductivity. CIR frame structures and material concepts compared. Skin frame construction (VI) selected.

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further. In particular, the cause of the asymmetric aerodynamic properties should be examined. The present destabilizing system (45-degree cutoff of the cone) is only the first attempt at providing destabilization moments in case of a rearward, spinning entry. Although the present design is adequate, further work will have to be done to find the optimum geometry for destabilization.

The aerodynamic coefficients which have been presented are based upon tests of similar configurations, and upon computations using Newtonian theory. These results are more than adequate for feasibility studies. For subsequent phases, however, detailed aerodynamic tests must be made upon explicit configurations. Included in these studies must be abnormal situations such as having the CIR attached, opening the cone frustum, etc. Although no unpleasant surprises are anticipated from the required aerodynamic tests, such experiments must be conducted to create actual flight hardware.

A beacon (with power) is a part of the RTG/RB. Since one of its functions is to aid in determining the exact RTG/RB return trajectory and impact point, it must be located in the RB. Unfortunately, the temperature inside the RB is rather high (above 200° F throughout its orbital life). This temperature presents an environmental problem for electronic components. Consequently, special provisions must be made to either protect the beacon from the temperature, or a device must be developed which will withstand the environment.

The present design includes a hinged aft cap. Clearly, this cap must be securely fastened prior to liftoff. Both the locking and unlocking must be done remotely during ground handling. The exact fastening mechanisms and associated circuitry to allow positive verification of the closing (or opening) have still to be devised. No real problem is anticipated in this area.

Abundant test data are available relative to the performance of low density charring elastomeric silicone ablators. However, the effects of exposure to radioisotope radiation on the thermal and mechanical performance capabilities of the ablator are not currently defined. Some data indicate that no real problem will be encountered. It will be necessary to perform a test program to evaluate these effects. In addition, the acceptable steady-state temperature levels in the ablator material for prolonged exposure periods must be similarly determined.

The strength of such structural elements as adhesive bonds and plastic laminates may be reduced as a result of exposure to radioisotopic sources. Radiation exposure dependent strength data for candidate materials of this type will have to be developed and evaluated.

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The mode of structural breakup upon land or water impact must be understood to evaluate the probability of violating the integrity of the fuel capsules. Structural breakup results in attenuation of the impact forces imposed upon the fuel capsules but also presents the possibility of penetration of the fuel capsule by structural fragments. Analytical and experimental studies are prerequisite to a realistic evaluation of this potential problem.

2. CIR

The CIR of the recommended CRONUS concept utilizes standard engineering design and construction techniques, mechanical devices and component state of the art. The fundamental engineering problems are encountered in synthesizing the components and operating characteristics into an efficient, reliable system which is compatible with the space environment, the unique internal environment generated by the RTG and the potential LV + S/C malfunctions which CRONUS must withstand.

a. Operational considerations

At first glance, the airborne system appears to require a relatively large number of operations to complete an abort maneuver; however, the program requirements dictate a minimum number of consecutive operations. These are:

- (1) Structural separation of the CIR-RTG/RB systems from the spacecraft.
- (2) Application of a directional impulse which will cause the RB to reenter the earth's atmosphere.
- (3) Separation of the impulse delivery system prior to reentry of the RB. While this last item is not obvious, sufficient analyses have been performed to show that aerodynamic stability of RB configurations after reentry cannot be achieved within reasonable vehicle gross weight limits if the deorbit system is an integral part of the RB.

Additional functions are related only to the second operation, i.e., assuring proper initial directional delivery of the deorbit impulse and maintaining this directionality during delivery of the impulse. These functions can be accomplished in three ways:

- (1) The incorporation of a guidance and control system, including attitude control propulsion into the CIR-RTG/RB systems.
- (2) Fixed orientation of the CIR-RTG/RB systems with spin stabilization. In this approach, the conditions requiring abort may have narrower limits.

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(3) A combination of Items (1) and (2).

Martin has chosen to recommend a minimum weight vehicle concept utilizing the second approach (fixed orientation). Therefore, two additional operations are required in an abort situation, namely, spin and despin of the airborne system. Both of these latter operations are in common use today in spacecraft which operate in the environment of space. To provide an initial fixed orientation that provides the maximum opportunity for a successful abort over a wide variety of potential abort conditions, the preseparation-spin separation track system concept was developed.

This particular overall abort system design approach will require several specialized engineering solutions.

(1) Spin separation track system

The principal problem associated with the overall separation operation is the ability of the spin and track bearings to meet performance and/or integrity specifications after extended periods under space environmental conditions. Considerable testing and evaluation have been performed on solid lubricants (Ref. VIII-2) for bearings applicable to space systems. Molybdenum disulfide has been shown to be a very effective lubricant in vacuum and under operating conditions similar to those here. However, this area of design remains as one requiring further investigation particularly with respect to suitable bonding materials and the effects of nuclear radiation.

(2) Spin stabilization

To avoid high heating rates during initial reentry, tight impulse tolerances for the spin and despin rockets must be maintained. Also deorbit rocket thrust misalignments which tend to roll the vehicle must be minimized.

Impulse tolerance of better than $\pm 1\%$ can be easily achieved for the spin-despin rockets. A maximum variation in residual spin anticipated from this source is approximately 1 rpm. Special attention must be given to achieving close alignment tolerances in the manufacture of the deorbit rockets (thrust axis to principal axis). Tolerances equivalent to those achieved in the Surveyor program are satisfactory.

In addition, thrust-time variations from rocket to rocket affect the magnitude of the spin rate required to minimize deorbit impulse vector losses and flight path angle deviations (which affect impact point accuracy). Special control of the rocket manufacturing process must be instituted in an attempt to prevent this variation from exceeding 1.50% (3σ) at a given ambient temperature. However, control of this variation

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carries through to the control of propellant temperature in the space environment. Environmental control techniques must be employed to minimize the variation in bulk temperatures between the four deorbit rockets.

(3) Ordnance system

The ordnance system utilizes existing components and techniques in a redundant circuit which has been examined in considerable detail with respect to safety, sequence integrity and reliability. No particular development problems are anticipated; however, the effects of nuclear radiation on ordnance charges must be investigated during the development program. The possibility of deleting the safe arm devices has been discussed. Such simplifications are desirable if they are consistent with the overall system requirement. This study should be conducted in the next phase.

(4) Spacecraft attachment

The attachment device is a straightforward method of holding the CIR-RTG/RB systems to the top of the spacecraft. The separation tracks and their supporting struts are bolted to the spacecraft. The separation carriage, containing the spin bearing to which the CIR-RTG/RB is mechanically connected, slides into the tracks until the carriage seats against the central track stop. The pin puller is engaged with its mating fitting and this fitting adjusted to remove any remaining play in the carriage and spin bearing. The selection of a pin puller of sufficient strength to withstand the imposed load while engaged, is the only area that requires further investigation. Various sizes are available as off-the-shelf hardware.

(5) Airborne system attachment

The attachment fingers-retention band system has been used many times and offers no recognizable development problem.

(6) Abort malfunctions

Because reliable, controlled intact reentry under mission abort circumstances is the essential design commitment in developing a CIR-RTG/RB, and because of the limited number and magnitude of critical conditions from which a successful abort can be reliably achieved, the MDS functions to be monitored by the CIR-RTG/RB must be carefully selected and stringent limits must be placed on the malfunction boundaries and detection time. The Martin Company has had considerable experience with the design, integration and installation of malfunction detection systems on the man-rated Titan III and Gemini

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boosters. However, the safety requirements for the strontium fuel require a somewhat different analysis from that required for man rates. The effects of multiple malfunctions must be analytically traced through the total systems to accurately determine the possibility of exceeding the safety requirements. Consequently, special attention should be given to the establishment of abort conditions and to the signals from the launch vehicle or spacecraft.

3. Environmental Considerations

The CIR-RTG/RB components and materials that are sensitive to gamma and/or neutron radiation (such as the propulsion items) will have to have limitations determined to minimize the weight of protective devices. The limits for electronic gear are fairly well established; however, the limitations of solid propellant have been determined only in a gross, conservative manner. A test program involving long term testing of selected solid propellant formulations with low radiation rates is a necessity if long term missions are to be seriously contemplated. The internal radiation environment and natural shielding for the CRONUS configuration must also be defined so that additional shielding requirements can be minimized.

Long term storage (five years) of solid propellant in a vacuum environment does not appear to be a problem; if necessary, internal pressurization can be applied. In any event, a long term vacuum storage program using production solid propellant rockets should be initiated as soon as the propellant formulations are selected. Bell jars are perfectly suitable for these tests since all the rockets are small than one cubic foot.

4. Hardware Selection

All the ordnance and propulsion components used in the CIR are basically off-the-shelf items. The electronic system is also considered standard hardware. The structure is made from commonly used materials formed in a rather simple design. The only unchecked hardware in the CIR are the spin bearings and the track separation items. Some failures have occurred in space vehicles that attempted to use mechanical links. It is felt that the suggested design will not only overcome these failure problems by the proper tests and design, but it is also felt that there is a basic difference in the approach. The spacecraft system which had mechanical difficulties depended upon small forces to perform the desired motions. In this case, the mechanical connections are overpowered by highly reliable propulsion and ordnance devices. Consequently, only major structural changes would inhibit the successful completion of the mechanical tasks which the CIR must perform.

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The environment in the CIR is well suited for the contained ordnance and electronic items. However, this is not true for the RTG/RB, where there may be a temperature environment problem with the beacon.

A summary of some of the CIR and RTG/RB concepts is presented in Table VIII-3 with an indication of some of the flight systems which have used the explicit features.

TABLE VIII-3

Summary of CIR and RTG/RB Features Correlated with Experience

<u>Feature</u>	RTG/RB	
	<u>Typical Flight Experience</u>	<u>Comments</u>
Blunted cone shape, F**	Manned vehicles	Well tested aerodynamic shape
Destabilization shape, F	*	More than adequate for purpose; may be optimized
Ablative heat shield, H**	Prime	Completely adequate; well within SOA
Free entry, F	Manned vehicles	Well proven concept
Beacon, H	Prime	Environment problem exists
CIR		
Frame structure, H		Most commonly used construction and materials
RTG/RB attachment, D**		Basic concept used in Scout and other spacecraft
RTG/RB separators, D	Vanguard	Can't fail system
S/C separation, D		Rail separation technique used in Titan I and III. Explicit design must be qualified
Spin bearing, H		Well known design, bearing patented. Explicit design must be qualified

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TABLE VIII-3 (continued)

CIR		
<u>Feature</u>	<u>Typical Flight Experience</u>	<u>Comments</u>
Spin stabilization, F	Vanguard	Well known and often used
Ordnance subsystems, HF		Off-the-shelf hardware used in conventional fashion
Separation motor, H		Suggested motor developed for twice thrust needed
Spin motors, H		Off-the-shelf hardware
Retro motors, H		Developed hardware. Additional qualification testing required
Telemetry, H		Off-the-shelf hardware
Command receivers, H		Off-the-shelf hardware
Ground-based deorbit, F	Discoverer	No problem, if suitable ground station is present (functions of type of orbit)

*Similar devices have been used on Gemini

**F = function; H = hardware; D = design

5. Assembly

a. Reentry body

The ablator will be applied to the heat shield supporting structure in the manner and sequence described by the flow diagram (Fig. VIII-3). The aft cap assembly consists of a Haynes-25 frame stabilized skin structure which is assembled in a straightforward manner utilizing standard tooling and quality control techniques.

b. Controlled intact reentry system

The conventional aluminum skin-stringer-frame configuration is also assembled in a straightforward manner utilizing current techniques. The RB support ring with its retention fingers is an integral part of the riveted CIR basic structure as is the bearing assembly at the base of the CIR.

All equipment except line rockets and separation ordnance devices will be installed in the CIR. These include:

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Fig. VIII-3. Heat Shield System Assembly Flow Diagram

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- (1) Ordnance circuitry, switches, delay devices and safe-arm units
- (2) Auxiliary power (battery)
- (3) Track-telemetry-command unit
- (4) Communication programmer.

C. AGE CREDIBILITY

The AGE system of CRONUS is a composite of previous space system equipment and industrial techniques and practices. Examples of the indicating circuits, locking devices, controls and actuators and other mechanisms are found in abundance in modern, automated manufacturing processes.

The analyses supporting the design are for the most part standard analytical techniques. Two of the analyses are not of the usual handbook variety and appear as Appendices GG and OO. Appendix GG illustrates the feasibility of hydrostatic bearing use and estimates the facility gas pressure and flow requirements. The other appendix dealing with AGE (Appendix OO) estimates the relative motion between the umbilical tower and launch vehicle. This defines the excursion requirements in the compliance equipment.

1. Umbilical Tower Mount

Umbilical tower-mounted equipment in a space launch complex is used in a Titan IIIB installation. A tower-mounted boom services the Agena. The Saturn V umbilical tower has several installations of structural booms retracted just before launch, and others initiated by the first movement of the vehicle. The design of the structural support pallet and boom for the CRONUS is consistent with similar equipment used in other installations.

2. Fuel Handling Machine and Cask Mechanisms

Power-operated mechanisms similar to that required for the cask are found in the steel industry in remote loading operations.

Mechanisms are required in the fueling machine such as those for the fueling probe and lock. The drive motor applications and the spring centering assembly of the compliance equipment are used in the food packaging, bottling and other processing industries.

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3. Hydrostatic Bearing

Hydrostatic bearings are commonplace items. A manufacturer of home refrigerators is featuring a gas-bearing principle device to move the unit for cleaning purposes. The exhaust of a household vacuum cleaner is used to provide the gas pressures required.

A hydrodynamic support for Saturn V, consisting of hydrostatic bearings, was designed, fabricated and system tested at the Martin test facility. This device was designed to support a static load of 7.2×10^6 pounds during a ground vibration survey for the Saturn V. The hydrostatic oil bearings designed for this application included six degrees of system freedom. A 100- to 200-pound force was required to move the static load supported by the hydrostatic bearings.

4. Miscellaneous

Many of the proposed components of the system are found in other day to day activities. The applications for the boom winch, drum and brake are too numerous to enumerate. The actuator cylinders, electro-mechanical actuators, air motors, rollers and rack and pinion gears are catalog items used in many applications. The electrical and pneumatic rewind reel is used in gasoline service stations to support the pump hose when runouts for long reaches are required and in the tire pumping equipment for rewind of the pressurization hose and nozzle.

5. Development Problems

There are no development problems with the AGE in the sense of advancing the state of art. The problems that will be encountered are those of normal design. At this time it is impossible to foresee all of these and hence design for them. However, an attempt has been made to identify as many as possible and to find solutions. This has been done in a failure mode analysis.

(a) Fuel handling equipment

The problems anticipated in the use of the fuel handling equipment and possible solutions are given:

- (1) Problem: Failure of cask lid to open.
Solution: Manual repairs can be made to the cask lid actuator since the radiation source has not been exposed.

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- (2) Problem: Failure of cask lid actuator with lid in partial or full open position.
- Solution: Provisions are made to attach a hand crank to the actuator which will permit closure of the cask lid from the platform below the cask. During this operation, the operator is shielded by the cask from the fuel block within the opened cask.
- (3) Problem: Failure of the FHM drive motor to position the FHM over the cask.
- Solution: Close the cask lid and repair FHM drive motor.
- (4) Problem: Failure of fueling probe to extend.
- Solution: Close the cask lid and repair actuator.
- (5) Problem: Failure of fuel block locking balls to engage lock to fuel block.
- Solution: Retract the fueling probe, close cask lid and repair fueling probe lock mechanism.
- (6) Problem: Failure of the fueling probe to retract fuel block from cask.
- Solution: The fueling probe must be removed from the cask for repairs. The fuel block locking balls are released and a redundant fueling probe actuator operated to elevate the probe. The cask lid is closed and the probe actuator repaired.
- (7) Problem: Failure of the FHM drive motor to move the fuel block along the boom to the RTG.
- Solution: Extend the fueling probe, returning the fuel block to the cask. Disengage the fuel block locking balls and retract the fueling probe. Close cask lid using the motion of the lid to displace the FHM along the boom track. Repairs can then be made to the FHM.
- (8) Problem: Failure of the FHM drive motor with FHM and fuel block in any position other than over cask.
- Solution: Actuate redundant cable system attached to FHM which pulls FHM back over cask. Unload fuel block into cask and close lid. Make repairs to FHM.

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- (9) Problem: Inadvertent actuation of the fuel block locking balls causing the fuel block to be released.
- Solution: To keep the problem from occurring, the fueling probe locking mechanism will be designed so it will fail in the locked position with loss of power. Shorting devices will be provided so stray signals will not actuate the release. Safety guards will be provided over a dual set of actuation switches to minimize inadvertent human actuation. If, after all system precautions are taken, the fuel block is dropped, a steel cable safety net, rigged from the umbilical tower to the side of the launch vehicle immediately below the transfer boom, will catch the fuel block. Technicians working in a shielded GFE crane will pick the block from the net and deposit it in a cask on the ground or in a field-expedient shield material for transportation to a remote area and subsequent disposition.
- (10) Problem: Failure of one or both position locator lock pins to actuate.
- Solution: Return fuel block to cask and repair lock pin actuator.
- (11) Problem: Failure of fueling probe to extend fuel block into RTG due to pneumatic line leakage, broken O ring seals, malfunction of solenoid control valve, etc.
- Solution: A redundant mechanical system is provided to extend the fueling probe. The retraction operation consists of releasing the position locator lock pins and returning the FHM to the position over the fuel cask. The redundant actuator drive is employed to extend the fueling probe, lowering the fuel block into the cask. The fuel block locking balls are released and the fueling probe is raised by the primary or redundant retraction mechanism. The cask lid is closed and repairs are made to the fueling probe.
- (12) Problem: Failure of the fuel block locking balls to release.
- Solution: Return fuel block to cask. Operate redundant probe plunger actuator to release fuel block and retract fueling probe. Close cask lid and repair lock mechanism.

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- (13) Problem: Failure of fuel block handling probe to retract due to pneumatic line leakage, broken O ring seal malfunction of solenoid control valve, etc.
- Solution: The philosophy that the fuel block be returned to the cask whenever a malfunction occurs in the fuel loading equipment will be followed even if the fuel block is completely installed in the RTG. The probe will be locked to the fuel block and a redundant actuator consisting of an electrically driven cable will raise the probe. The fuel block can then be returned to the cask and manual repairs made.
- (14) Problem: Failure of position locator lock pins to retract.
- Solution: The fuel block will be removed from the RTG and returned to the cask in the following manner. The fueling probe is extended into the fuel block, the fueling probe locking mechanism actuated, and the fuel block raised into the FHM environmental shroud. The fueling boom will be raised approximately two inches to lift the lock pins from the position index fittings and the FHM retracted to a mid-point on the bottom and the boom returned to a horizontal position. The retracting of the FHM will continue until the position locators engage the position index fittings on the cask. This aligns the fuel block with the cask. The position index fittings on the cask are of sufficient depth that they will accept the position locators even with the lock pins extended. The FHM is held connected to the cask indices by actuation of this fueling machine traversing motor. The fuel block is installed in the cask and repairs are made to the lock pins.
- (15) Problem: Failure of boom winch.
- Solution: Boom retraction is not critical to returning the fuel block to the cask; therefore, in the event of winch malfunction the fuel block can be returned to the cask. Winch repairs or replacement can be made in a normal manner.
- (16) Problem: Electrical failure within fuel block loading unit caused by broken or shorted wires.

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Solution: Each critical function necessary to returning the fuel block to the cask is provided with a redundant means of performing the function.

(17) Problem: Break in pneumatic line.

Solution: In the event of a break in a pneumatic line to a component with a system function critical to returning the fuel block to the cask, a redundant system using electrical or secondary pneumatic power is provided.

(18) Problem: Failure of launch vehicle nose cap latch actuator.

Solution: In the event that the nose cap latch fails before the fuel block is loaded, the fuel block shall be returned to the cask and manual repairs made to the malfunctioning latch. If the fuel block has been installed in the RTG and the nose cap must be reopened because of an aborted launch, alternative means of opening the latches are provided. One method is the incorporation of an explosive device to release the latch mechanism. Another is to provide pin pullers in the latch linkage which are actuated by electrical or pneumatic means.

(19) Problem: Failure of launch vehicle nose cap actuator.

Solution: In the event that the nose cap actuator fails before the fuel block is installed, the fuel block will be returned to the cask and manual repairs made to the actuator. If the fuel block has been installed in the RTG and the nose cap must be reopened because of an aborted launch, redundant means of opening the nose cap are provided in the event of an actuator malfunction.

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IX. IMPROVEMENT POTENTIAL AND DESIGN FLEXIBILITY

As the title implies, this chapter discusses the flexibility and potential improvements of the reference CRONUS design presented in Chapter III. Potential improvements discussed include weight reduction and sensitivity, cascaded and segmented thermoelectric concepts, impact area control, and the use of Transtage as the CIR. Flexibility is discussed with regard to the alternate fuel, Pu-238, alternate launch complex (WTR), and adaptation of the GHE for side loading.

A. POTENTIAL DESIGN IMPROVEMENT CONSIDERATIONS

1. Weight Reduction

a. Introduction

As has been noted, the explicit design called the "reference configuration" has not been subjected to a weight reduction program. Its main purpose was to establish a basis for demonstrating the technical feasibility of the 250W RTG controlled intact recovery concept. This has been accomplished, as presented in this report.

Near the end of the study effort, an investigation was conducted to determine what changes in the initial design parameters (heat source L/D ratio and thermoelectric cold junction temperature) should be made in order to minimize the total system weight. The results of this optimization effort resulted in a marked decrease in the weight and size of the airborne system. Further investigations were conducted to determine the impact of relaxing certain self-imposed requirements upon system weight and volume.

These studies show that major weight reductions, in addition to the improvement due to optimization, are possible.

The purpose of this section is to present and explain the optimization of the reference configuration, to present and explain seven additional designs with varying design criteria, and to discuss additional weight reduction opportunities.

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b. Logic

The basic logic of the various configuration optimizations is shown in Figure IX-1. First, the reference configuration is "optimized." This involves: a) raising the radiator temperature to reduce the radiator area, b) increasing the L/D of the heat source to reduce its weight as well as to permit a reduction in the aft cap and heat shield diameters.

All the safety criteria, material choices, functions, concepts, etc. of the reference configuration are preserved in this optimization. This basic, optimized configuration is labeled Column 2 in Figure IX-1. The main changes in this configuration, as compared to the reference configuration, are due to a reduction in size, since the radiator surface area could be reduced to 12.5 ft² as compared to 24 ft² for the reference design.

Three basic variations were made of the optimized configuration. These variations are:

- (1) Some of the self-imposed safety criteria were relieved. This permitted a substitution of a refractory heat source for Haynes 25; and consequently, the EOL T/E hot junction temperature could be raised 200°F. The resulting radiator area of 9.7 ft² resulted in considerable weight savings. Furthermore, the 1500 fps velocity increment for deorbiting was reduced to 1200 fps. The specific weight of the propulsion system was, therefore, decreased.
- (2) Some of the auxiliary functions and services which are not essential, are eliminated. These functions involve the space tracking capability in the RB, the backup power supply, the airborne telemetry system, and the elimination of most of the safe arm devices from the CIR ordnance subsystem.
- (3) The third category of changes to the weight optimized reference configuration involves the use of advanced materials in the heat shield, structure, and propulsion subsystems of the CIR.

Each of these three changes will be described in more detail later in this chapter.

The airborne system weights for making any one of the above changes are presented in Table IX-1. These three designs are labeled Column 3 through

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Fig. IX-1. Weight Summary--CRONUS Airborne Systems
Weight Improvement Options

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	COLUMN 1	COLUMN 2	COLUMN 3	COLUMN 4	COLUMN 5	COLUMN 6	COLUMN 7	COLUMN 8	COLUMN 9
ITEM	CRONUS AIRBORNE SYSTEM REFERENCE CONFIGURATION NON-OPTIMIZED WEIGHT (LBS)	REFERENCE CONFIGURATION WEIGHT OPTIMIZED (LB)	SELF IMPOSED SAFETY CRITERIA MODIFIED (LB)	ELIMINATION OF FUNCTIONS (LB)	UTILIZATION OF NEW MATERIALS (LB)	ELIMINATION OF FUNCTIONS AND SAFETY CRITERIA MODIFIED (LB)	UTILIZATION OF NEW MATERIALS AND SAFETY CRITERIA MODIFIED (LB)	ELIMINATION OF FUNCTIONS AND NEW MATERIALS UTILIZED (LB)	REF CONFIG WEIGHT OPTIMIZED, SAFETY CRITERIA MOD., FUNCTIONS ELIM & NEW MATERIALS UTILIZED (LB)
RTG/RB	412 2	298 5	251 9	278 6	293 3	232 0	247 5	273 6	228 2
Heat Source	181 6	163 3	127 4	163 3	163 3	127 4	127 4	163 3	127 4
Fuel Capsules	63 8	65 1	53 3	65 1	65 1	53 3	53 3	65 1	53 3
Fuel	7 7	7 9	19 0	7 9	7 9	19 0	19 0	7 9	19 0
Capsule Liners	72 9	74 4	41 5	74 4	74 4	41 5	41 5	74 4	41 5
Outer Capsules									
Capsule Retaining Bands	1 2	1 2	1 2	1 2	1 2	1 2	1 2	1 2	1 2
Capsule Support Rings (2)	2 0	1 4	1 3	1 4	1 4	1 3	1 3	1 4	1 3
Thermal Insulation	24 8	8 4	6 2	8 4	8 4	6 2	6 2	8 4	6 2
Min K Rings (2)	19 2	3 4	4 7	3 4	3 4	4 7	4 7	3 4	4 7
Micro-Quartz Cylinder	5 6	5 0	1 5	5 0	5 0	1 5	1 5	5 0	1 5
Support Caps (2)		3 0	3 0	3 0	3 0	3 0	3 0	3 0	3 0
Central Tube		0 4	0 4	0 4	0 4	0 4	0 4	0 4	0 4
Ball Detent & Latch Mechanism		1 5	1 5	1 5	1 5	1 5	1 5	1 5	1 5
Thermoelectrics									
Hot Shoes	32 0	32 6	26 8	32 6	32 6	26 8	26 8	32 6	26 8
Elements	6 4	6 5	5 5	6 5	6 5	5 5	5 5	6 5	5 5
Cold Shoes	5 6	5 7	4 6	5 7	5 7	4 6	4 6	5 7	4 6
Flux-and Bond, Stack & Hardware	4 6	4 7	3 8	4 7	4 7	3 8	3 8	4 7	3 8
Thermal Insulation	8 8	9 0	7 4	9 0	9 0	7 4	7 4	9 0	7 4
Thermal Insulation (above & below T E)	13 6	4 0	3 1	4 0	4 0	3 1	3 1	4 0	3 1
Upper Ring	4 2	0 8	0 3	0 8	0 8	0 3	0 3	0 8	0 3
Lower Ring	9 4	3 2	2 8	3 2	3 2	2 8	2 8	3 2	2 8
Radiator		30 1	23 3	30 1	30 1	23 3	23 3	30 1	23 3
Mechanical & Structural Items	28 8	8 7	18 7	8 7	8 7	18 7	18 7	8 7	18 7
Support Cone	15 1	5 0	15 0	5 0	5 0	15 0	15 0	5 0	15 0
T Ring	3 7	0 9	0 9	0 9	0 9	0 9	0 9	0 9	0 9
U Ring	4 1	0	0	0	0	0	0	0	0
Latch Ring	1 1	1 1	1 1	1 1	1 1	1 1	1 1	1 1	1 1
Locking Tube Support	0 9	0 9	0 9	0 9	0 9	0 9	0 9	0 9	0 9
Heat Source Guides (8)	2 5	0 3	0 3	0 3	0 3	0 3	0 3	0 3	0 3
Rivets	0 9	0 5	0 5	0 5	0 5	0 5	0 5	0 5	0 5
Electrical Items	2 7	2 7	2 7	0 2	2 7	0 2	2 7	0 2	0 2
Instrumentation (includes Harness)	2 0	2 0	2 0	0	2 0	0	2 0	0	0
Instrumentation Receptacle	0 5	0 5	0 5	0	0 5	0	0 5	0	0
Power Leads and Receptacles	0 2	0 2	0 2	0 2	0 2	0 2	0 2	0 2	0 2
RB	82 0	87 1	49 9	39 7	51 9	32 5	45 5	34 7	27 7
Heat Shield Assembly	52 8	33 9	28 0	33 2	28 7	27 3	23 6	28 2	23 5
Ablator	22 4	17 2	14 7	16 5	12 0	14 0	10 3	11 5	10 2
Adhesive Bond	1 6	0 8	0 6	0 8	0 8	0 6	0 6	0 8	0 6
Substrate Panel	6 5	3 4	2 6	3 4	3 4	2 6	2 6	3 4	2 6
Adhesive Bond (2)	3 4	1 8	1 4	1 8	1 8	1 4	1 4	1 8	1 4
Frame Shell	4 4	2 3	1 8	2 3	2 3	1 8	1 8	2 3	1 8
Splice Frame	5 7	2 1	1 4	2 1	2 1	1 4	1 4	2 1	1 4
Insulation Ring	8 8	6 3	5 5	6 3	6 3	5 5	5 5	6 3	5 5
Aft Cap Structure	12 5	6 5	5 2	6 5	6 5	5 2	5 2	6 5	5 2
Skin	5 9	3 1	2 4	3 1	3 1	2 4	2 4	3 1	2 4
Bulkhead	1 3	0 7	0 6	0 7	0 7	0 6	0 6	0 7	0 6
Frame	4 3	2 2	1 7	2 2	2 2	1 7	1 7	2 2	1 7
Fittings and Miscellaneous	1 0	0 5	0 5	0 5	0 5	0 5	0 5	0 5	0 5
Tracking Beacon	16 7	16 7	16 7	0	16 7	0	16 7	0	0
Transmitter	3 0	3 0	3 0	0	3 0	0	3 0	0	0
Battery	5 0	5 0	5 0	0	5 0	0	5 0	0	0
Antenna	2 0	2 0	2 0	0	2 0	0	2 0	0	0
Mounting	2 0	2 0	2 0	0	2 0	0	2 0	0	0
Electrical Harness	4 7	4 7	4 7	0	4 7	0	4 7	0	0
CIR	313 2	247 7	204 6	199 2	207 7	160 9	174 7	137 7	134 8
Structure	88 0	61 5	54 6	59 2	51 4	52 8	45 7	49 0	44 3
Frame Assembly	52 0	31 4	26 3	29 1	27 5	24 5	23 3	25 3	22 0
Skin	7 8	4 0	3 1	4 0	2 6	3 1	2 0	2 6	2 6
Stringers	6 6	3 4	2 7	3 4	2 2	2 7	1 8	2 2	1 8
Equipment Shelf	4 2	3 0	2 6	1 5	3 0	1 3	2 6	1 5	1 3
Frames	12 1	6 5	5 0	5 9	5 2	4 5	4 0	4 7	3 6
RB Support	9 3	4 8	3 7	4 8	3 8	3 7	3 0	3 8	3 0
Intermediate	2 1	1 1	0 8	1 1	0 9	0 8	0 6	0 9	0 6
Equipment Shelf	0 7	0 6	0 5	0	0 5	0	0 4	0	0
Retro Mount Fittings	3 0	2 1	2 0	2 1	2 1	2 0	2 0	2 1	2 0
Gussets	0 8	0 6	0 5	0 6	0	0 5	0 5	0 6	0 5
Heat Shield Bed	10 5	5 4	4 2	5 4	5 4	4 2	4 2	5 4	4 2
Rubber Pad		1 1	0 7	1 1	0 7	0 7	0 7	1 1	0 7
Honeycomb Panel	8 3	4 3	3 5	4 3	4 3	3 5	3 5	4 3	3 5
RB Hold Fingers (8)	3 0	3 0	3 0	3 0	3 0	3 0	3 0	3 0	3 0
RB Ejection Cartridges (4)	4 0	3 4	3 2	3 2	3 4	3 2	3 2	3 4	3 0
Spin Bearings	6 0	4 3	3 8	4 3	4 3	3 8	3 8	4 3	3 8
Separation Carriage	12 0	8 6	7 5	8 6	6 4	7 5	5 6	8 6	5 0
Separation Tracks	12 0	12 0	12 0	12 0	8 0	12 0	8 0	12 0	8 0
Separation Rocket Mount	6 0	5 2	5 0	5 2	5 2	5 0	5 0	5 2	4 9
Ordinance	16 8	16 8	16 8	12 6	16 8	12 6	16 8	12 6	12 6
Initiation System Units	12 0	12 0	12 0	10 0	12 0	10 0	12 0	10 0	10 0
Solid State Switching	3 0	3 0	3 0	1 5	3 0	1 5	3 0	1 5	1 5
Pyrotechnic Switches (6)	0 6	0 6	0 6	0 6	0 6	0 6	0 6	0 6	0 6
Installation Provisions	1 2	1 2	1 2	0 5	1 2	0 5	1 2	0 5	0 5
Propulsion	167 2	128 2	92 0	112 9	98 3	81 0	71 0	87 6	63 4
Separation Rocket	7 0	6 5	6 4	6 4	6 4	6 5	6 5	6 3	6 2
Spin-Despin Rockets (4)	5 8	4 5	4 4	4 4	4 4	4 5	4 5	4 3	4 2
Deorbit Rockets (4)	154 4	117 2	81 2	102 1	87 5	70 2	60 0	77 0	53 0
Electronics	27 7	27 7	27 7	7 0	27 7	7 0	27 7	7 0	7 0
Multiplexer	2 0	2 0	2 0	***3 0	2 0	3 0	2 0	3 0	3 0
Receiver	3 4	3 4	3 4	***2 0	3 4	2 0	3 4	2 0	2 0
Transmitter	2 6	2 6	2 6	****2 0	2 6	2 0	2 6	2 0	2 0
Power Conditioner (Command)	1 5	1 5	1 5	0	1 5	0	1 5	0	0
Power Conditioner (Transmission)*	2 1	2 1	2 1	0	2 1	0	2 1	0	0
Base Band Assembly	1 3	1 3	1 3	0	1 3	0	1 3	0	0
Telemetry Inputs	8 0	8 0	8 0	0	8 0	0	8 0	0	0
Signal Conditioner	1 5	1 5	1 5	0	1 5	0	1 5	0	0
Command Decoder	2 3	2 3	2 3	0	2 3	0	2 3	0	0
Antennas (3)	3 0	3 0	3 0	0	3 0	0	3 0	0	0
Electrical Power	11 0	11 0	11 0	5 0	11 0	5 0	11 0	5 0	5 0
Battery	3 0	3 0	3 0	0	3 0	0	3 0	0	0
Relay	2 0	2 0	2 0	0	2 0	0	2 0	0	0
Wiring Plugs, etc	5 0	5 0	5 0	5 0	5 0	5 0	5 0	5 0	5 0
Miscellaneous	1 0	1 0	1 0	0	1 0	0	1 0	0	0
Voltage Regulator	2 5	2 5	2 5	2 5	2 5	2 5	2 5	2 5	2 5
TOTAL SYSTEM WEIGHT	725 4	546 2	456 5	477 8	501 0	392 9	422 2	437 3	363 0

* Plus 10 lb. for refractory fuel capsule protection
*** Antenna
** Transmitter/Receiver
**** Mounting

Detail Weight Summary--CRONUS Airborne Systems Weight Improvement Options

TABLE IX-1

Column 5 in Figure IX-1. Three additional designs were generated, utilizing combinations of two of the above three changes. The definition of these designs is also shown in Figure IX-1 (Columns 6 through 8), and the detailed weights are presented in Table IX-1. Finally, all three changes were made to the optimized designs labeled Column 9.

The resulting weight analysis indicates that, by optimizing the basic 24 ft² configuration, the total airborne system weight is reduced to 546 lb. The additional weight savings, due to the above discussed three changes, are:

<u>Changes</u>	<u>Additional Weight Savings (lb)</u>
Safety Criteria	89
Elimination Function	68
New Materials	45

These changes are approximately additive. If all three changes are incorporated, the airborne configuration weighs 363 lbs. (~1 watt/lb BOL power) as shown in Column 9 Table IX-1. The size of the configuration is a function of what changes are incorporated. All the configurations which contain change (1) have a radiator area of 9.7 ft². All the configurations which do not include change (1) have the same area as the optimized reference configuration, 12.5 ft².

c. Optimization (Column 2 design)

The first and primary alteration to the basic design is itemized in Column No. 2 of Table IX-1. In this modification, all of the features of the basic concept have been preserved including subsystems equipment and operation, material and design philosophy. However, certain features of the generator have been size optimized with the result that the system has been significantly reduced in size and weight.

The specific optimizations made here are described in detail in the following paragraphs.

The gross effects of increasing the thermoelectric cold junction temperature are a reduction in thermoelectric efficiency and a reduction in the required radiator area. Reduced efficiency tends to increase the required thermal output

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of the heat source and to increase the size and weight of the thermoelectrics themselves. The reduction in radiator area permits the overall radiator and reentry body to be reduced in size and provides a reduction in the weight of the radiator as well as that of the heat shield, aft cap, and all structural and thermal insulation components. Since radiator area is related to the fourth power of radiator temperature, this weight reduction is large with respect to the weight added in the heat source and thermoelements for small increases in cold junction temperature.

A more subtle effect occurs in the parasitic heat losses. Since raising the cold junction temperature will reduce the temperature gradient through the thermal insulation, the thermal efficiency of the generator will be increased. This effect will partially offset the reduction in thermoelectric efficiency and will tend to reduce the heat source weight.

The combined effect of these factors is to gradually reduce the RTG weight as the cold junction temperature is increased. The minimum weight occurs at a cold junction temperature around 700°F, above which the added heat source and thermoelectric weights dominate and the system weight begins to rise.

It was found that mechanical interference precluded going to a cold junction temperature higher than 605°F. Above that temperature, the smaller body size did not provide adequate internal volume for the heat source or sufficient radial clearance (1") for ease of heat source installation. The cold junction temperature was, therefore, set at 605°F for all subsequent considerations.

When the cold junction temperature is increased, several geometrical changes take place which indicate a higher L/D ratio is desired for the heat source. For instance, reducing the radiator size reduces the diameter of its upper end so that a smaller heat source diameter D must be used to permit installation through the opening. Likewise, since the thermoelectric area was held proportional to the thermal inventory (so as to maintain a constant hot junction temperature), the thermoelectric modules were placed over a greater proportion of the radiator in the longitudinal direction. This required that the heat source length L be increased to provide the same thermal view factor as before. The net change was to increase the L/D ratio of the heat source from 0.9 to 1.5.

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This required that a different fuel capsule configuration and placement be used. This does not affect the total weight of the capsules since the terminal velocity of individual capsules (and therefore, the encapsulant weight fraction w_c) is assumed to remain unchanged. However, a substantial weight reduction was realized in the generator system because of the reduced size of the conical support structure, support caps, and thermal insulation.

A further weight reduction was made by decreasing the thickness of thermal insulation from 3.0" to 1.5" at the ends of the heat source and its support tube. The three-inch thickness was used previously to minimize parasitic thermal losses through the top and bottom of the heat source so as to maximize the system efficiency and made the RTG suitable for use with Pu-238 fuel.

In re-calculating the parasitic heat losses, it was found that the effects of decreasing the insulation thickness were more than compensated for by the reduced insulation area (proportional to radiator area) and reduced temperature gradient from hot to cold junction. The thermal efficiency of the weight-optimized configuration is 92% as opposed to 89% for the reference design. This increase in thermal efficiency nearly compensates for the reduction in thermoelectric efficiency due to the increased cold junction temperature. Overall system efficiency drops from 4.42% to 4.32% which is a relatively minor change.

A fourth factor which had only an implicit effect on system performance was found to provide a significant reduction in radiator area. The radiator efficiency η_R for the reference configuration was only 61% because of the length of the radiator fins above and below the thermoelectrics. In the weight-optimized configuration, the fin length was reduced substantially which improved the fin efficiency. Further, since the proportion of fin area to prime radiator area (thermoelectric module area) was reduced significantly, the radiator efficiency was increased to over 90%.

In summary, the effect of optimizing only the cold junction temperature T_{cold} or the heat source L/D ratio or the insulation thickness is to realize small reductions in generator size and weight. However, by optimizing all three of these factors, a more favorable thermal geometry was achieved which permitted a large reduction to be made. The radiator area is reduced from 24 to 12.5 ft².

The optimization of the generator components, just described, makes it possible to scale down the size of RTG/RB-CIR and this is done so as to preserve the proportional relationship of the primary components of the system.

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Specifically, the linear dimensions (L) of the referenced design are reduced by the square root of the ratio of the new radiator area (A_R') to the referenced design radiator area (A_R), to give the new linear dimension (L'). $L' = L \sqrt{\frac{A_R'}{A_R}}$
or $L' = L \sqrt{\frac{12.5}{24}} = L \cdot .721688$. Certain geometric limits are inherent to this

design and they concern primarily the heat source dimensions, diameter and length, as they relate to the RB.

The heat source diameter must be less than the inside diameter of the small end of the radiator to allow it to pass through during fueling and the length must be less than the distance from the inner face of the heat shield backup structure to the radiator small end plane, to allow for some insertion lead angle, structural support and reasonable view angles to the thermoelectrics. At the same time, the heat source diameter should be as large as possible, as above, to keep the gap between it and the thermoelectrics to a minimum. For this modification to the reference design, the heat source L/D is 1.5 as compared to .9 for the referenced design.

With the exception of the heat source support structure, the structural arrangements of RTG/RB and CIR are the same as the referenced design through reduced in size. The heat source support structure is changed from two concentric reversed conical frustums attached to the radiator, to a single cone frustum support attached to the heat shield backup structure.

Since the Table IX-1 Column No. 2 design is a reduced size version of the referenced Column 1 design, the weights of the structural items are reduced in most cases since they are of smaller size. The reduced weight for each item can be approximated by $W' = W \left(\frac{A_R'}{A_R} \right)^{3/2}$ where W' is the new weight and W

is the original referenced weight for each item. However, since this could well lead to minimum gage problems, the more conservative approach of $W' = W \left(\frac{A_R'}{A_R} \right)$

or approximately $1/2W$ is used.

In this case, as in all design variations, the heat shield weight is computed as a function of the size and weight of the reentry body. The ablator thickness, and hence weight, is computed by the techniques repeated in Chapter VII, Section 157, 158, 159. The weight of the heat shield, backup structure, adhesive bonds, etc. are considered to be proportioned to the surface area.

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The operating subsystems equipment is the same in both designs, therefore, these weights are the same, except for the propulsion units which have been reduced in weight with respect to the reduced mass that they are now expected to accelerate.

The rocket utilized for the reduced gross weight configuration is essentially a scaled down version of the Reference Configuration rocket motor. The principal design characteristics remain unchanged as can be noted in Table IX-2 (Column No. 2). However, the new rocket will utilize the advantages (lighter weight case) of a completely spherical configuration.

d. Column 3 design

Column No. 3 design represents a further weight and size reduction that is realized when the Column No. 2 reductions are augmented by additional reductions associated with "self-imposed safety criteria." The term "self-imposed safety criteria" as used here means that the fuel capsules are always contained within the RB at earth impact. That is the individual capsules never impact independently but are surrounded by the shock attenuating (by impact deformation) reentry body. This was not considered with the Column No. 1 and Column No. 2 designs where the capsules were fabricated of Haynes 25 to preclude their bursting on independent terminal velocity impact. Fuel capsules of refractory metal may be used instead of Haynes 25 if they are protected and since they may be operated at a higher temperature, offer further opportunities for weight and size reduction.

It has been shown (see Appendix G) that the radiator and support structure would withstand the blast overpressure associated with a launch abort during ascent. However, it was felt that under certain conditions reinforcement of the shock wavefront may occur at the radiator surface, and the combined effects of pressure and shrapnel from the blast may strip away the outer structure and thermal insulation and permit the fuel capsules to fall to earth individually. A safety criterion was, therefore, imposed that the fuel capsules be designed to withstand earth impact at their individual terminal velocity.

In working with a smaller radiator and structure, it is apparent that it will be more resistant to blast effects, and that it will be protected to a greater extent by the CIR components. It is, therefore, felt that survival of the blast by a smaller RTG/RB can be guaranteed, and that earth impact of the total body will occur with, at worst, its own terminal velocity.

NOTE: In both designs, the retro rocket mass fraction is .80 and the Isp is 260 sec.

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On this basis, a new design criterion has been established in the interest of weight reduction; that the fuel capsules be capable of surviving earth impact within the RTG/RB and at the terminal velocity of the RTG/RB. In considering the impact energy absorption of the reentry body and of the thermal insulation and any other crushable structure that may be added to the body, an equivalent impact velocity for an individual fuel capsule has been set at 100 ft/sec. This compares with the 300 ft/sec. criterion which was used before.

This drops the energy absorption requirement for the fuel capsule to one ninth of its previous value. The capsule walls would then be sized on the basis of considerations other than impact resistance, such as puncture resistance or fabrication from minimum gage thicknesses. Capsules designed to meet this criterion will still survive any credible accident during handling or transportation.

This change reduces the weight advantage of using Haynes 25 superalloy for the capsule material and allows refractories to be considered for operation at higher temperatures. The problems associated with this approach are (1) that chemical compatibility between the Sr TiO_4 fuel and refractories has not been established at operating temperatures, and (2) refractory materials are susceptible to rapid oxidation in air at elevated temperatures. If these problems can be overcome, then a substantial performance increase can be achieved by operating the heat source at a higher temperature so as to raise the thermoelectric hot junction temperature to its upper limit (1800°F or higher) at beginning of life.

Personnel at RCA were questioned as to what limits in hot junction temperature exist for the SiGe thermoelements. The best data which exist are from a single module which has operated for over 4000 continuous hours in vacuum at a hot shoe temperature of 1020°C or 1868°F. This temperature is measured by a thermocouple located near the center of the hot shoe. Since a nominal temperature drop of 50°F occurs through the hot shoe, the above conditions would yield a hot junction temperature of approximately 1843°F. Volatilization does not appear to be a problem, and there is nothing to indicate that operation at a hot junction temperature of 1800°F cannot be attained.

The approach taken in Column 3 of Table IX-1 was to assume a refractory fuel capsule that is resistant to impact at 100 ft/sec and capable of operating at 2000°F or higher. Crushable material (such as Haynes 25 honeycomb) was added within the RTG/RB to make the heat source withstand an impact of the RTG/RB at the terminal velocity of that body. Since the thermal gradient from the back to front surfaces of the fuel capsules is less of a consideration in this design, and since the capsule terminal velocity is no longer important, the

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capsules were made somewhat larger so as to reduce the heat source envelope. This permitted the reentry body to be scaled down to provide a radiator area of 9.7 ft².

Because of the gain in thermoelectric efficiency (from 5.43% to 6.42%) by operating at the higher hot junction temperature, it was found that the radiator and cold junction temperatures could remain unchanged from those in the design for Column No. 2 of Table IX-1. As noted, the RTG component weights shown in Column 3 are significantly lower than those in Column 2.

The optimization of the generator components described just above makes it possible to scale down the size of the RTG/RB-CIR an additional amount. The scaling factor is derived the same way as for the Column No. 2 design but in this case has the value $L' = L \cdot .63574$. The same type of geometric limits as described for the Column No. 2 design apply to this one also. The L/D for the heat source in this design is 1.4.

The structural arrangement of this version is the same as the Column No. 2 version though reduced in size. The weight scaling factor for the structural members is approximately .404W, where W = the Column No. 1 item weight.

Here again, there is no change to the operating subsystems except for the propulsion units.

In the case of the separation and spin rockets, they are sized for the mass to be accelerated as are the deorbit rockets, however, for this design the deorbit ΔV is reduced from 1500 to 1200 fps, thus allowing a further significant reduction in deorbit rocket weight.

Again, the Reference Configuration deorbit rocket design criteria are retained in going to a still smaller spherical rocket required to provide a smaller ΔV to a lower weight payload. The primary characteristics of this rocket are given in Table IX-2 (Column No. 3).

e. Column 4 design

The Table IX-4 Column No. 4 design represents further possible reductions in weight, to the reductions previously described for the Column No. 2 design, resulting from the elimination of certain functions. The functions reduced or eliminated and the reasoning associated with these changes is discussed below.

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The beacon is in the reference configuration RTG/RB as a convenience to aid in the tracking after deorbit. This is a valuable and important function. Under the present ground rules, however, no safety or performance features would be lost if the beacon were removed, provided that some tracking capability is present in the airborne system. Consequently, the C-band beacon, and associated batteries, may be eliminated from the reentry body.

The reference and the optimized configuration have allowed for CRONUS telemetering inflight instrumentation information. This feature is desirable for operating systems, but is not absolutely required. Not all of the telemetry requirements can be eliminated. For example, certain basic RTG information, such as power output, may be required even for operational flights. However, the spacecraft and/or launch vehicle telemetry system can readily handle these few bits of information without requiring a telemetry system in the airborne power supply.

It is perfectly reasonable, therefore, to consider eliminating the 256 channel telemetry feature of the basic airborne system. With this requirement eliminated, some of the reasons for the choice of the integrated airborne SGLS system in the CIR are eliminated.

Several alternatives exist for weight reduction in the electronics systems. The C-band beacon may be used as a command receiver and decoder for a limited number of commands since telemetry may be reduced drastically after the demonstration flights and those functions actually required (RTG current voltage, temperature) may be monitored by satellite vehicle telemetry. The TTC functions of the SGLS are then covered and this unit may be eliminated. The required commands that the C-band beacon must be able to receive are:

- (1) Arm ordnance at fairing release (an automatic signal backed up by ground command).
- (2) Separation of the CIR.
- (3) Deorbit.
- (4) Transmitter enable (stand-by operation).

All of these commands are given from the ground in real time as a function of predicted impact in a safe area. The C-band beacon can be made to receive these commands. Consequently, this beacon is placed in the CIR, rather than the RB, and the SGLS airborne system is eliminated.

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Ordnance circuit power cartridges and igniters can be designed to provide protection against stray current and RF energy in the same manner as the Command Initiator units (safe-arm devices) used in the reference configuration. In order to retain the type of initiators used in the Reference Configuration, at least one or two of the Command Initiator type units must also be retained because the two types of units are designed as an integral function. However, the replacement(s) for the Command Initiators used in the Reference Configuration must be designed to permit multiple initiator functions. The ordnance system weight can be reduced through elimination of some of the Command Initiators. This additional weight reduction results in a slight reduction in deorbit retro rocket weight, as shown in Table IX-2 Column No. 4.

f. Column 5 design

The design for Column No. 5 of Table IX-1 reflects the improvement that is realized in the Column No. 2 design by substituting new or "exotic" materials for those materials originally considered for the various parts. These substitutions are described in the following paragraphs.

The structural weight of the CIR can be decreased by going to a beryllium honeycomb construction with the core being glass phenolic. The beryllium surface is bonded to the core material and is used primarily for stability for the high over pressure design condition. This type of construction will pose some development problems, primarily in the area of fabrication of the beryllium faces; however, it is believed to be compatible with the CRONUS development schedule.

Three types of insulation were evaluated during the CRONUS study using the Martin T-Cap III charring Ablator Digital program. The proven approach using the Martin developed ESA 3560H elastomeric silicone ablator was applied to the Reference Configuration. The more advanced approach replaces the substrata of the ESA 3560H ablator with an efficient insulation material such as Micro-Quartz. While more development work is required in the areas of fabrication techniques, load transfer capability and optimization of ablator to insulation thickness, it is believed that a heat shield of this type can be generated without significant problems.

Since these changes represent a weight reduction over the Column No. 2 design, the propulsion system weight will obviously be less but since new materials plus reduced margins on the motor cases, etc. are introduced here, even more significant weight reductions are realized. These propulsion improvements are discussed in detail below.

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The previous rocket designs have been along conservative lines in order to guarantee early availability, maximum demonstrated reliability and maximum safety. Relaxation of these constraints to permit the development of a completely new rocket utilizing the best possible state-of-the-art in technology and materials will require more rigorous qualification and testing of components and subsystems; however, the weight savings can be considerable. An evaluation of existing, high performance spherical solid propellant rocket motors was performed in order to determine the magnitude of critical design factors that would represent the best current technology. These design and performance characteristics reflect better chamber and nozzle materials (such as titanium), higher performance (non-exotic) propellants and lower safety factors. The design characteristics applied to the deorbit rocket optimization are based on this evaluation of current rockets (see Table IX-2, Column 5).

g. Columns 6,7,8 & 9 designs

Column Nos. 6,7,8 and 9 of Table IX-1 define the weight reductions that are realized by combining, in the manner shown on the table, the weight saving features introduced in Columns 2 through 5. Column No. 9 has all of the weight reduction features included in it and is, therefore, the lightest weight design possible.

The propulsion system weight varies for each of these designs since the design weights are all different. These propulsion weights are discussed below.

The combination of maximum reduction in the basic RTG/RB-CIR weight, reduction in deorbit velocity requirements and the utilization of best state-of-the-art in solid propellant spherical rocket design and materials application yields a final rocket that weighs approximately one-third that for the Reference Configuration (Table IX-2 Column 9). A specific design study is required in order to specify the exact design characteristics and final weight; i. e., the final system weight for a specific rocket configuration could vary $\pm 5\%$. The rocket weights for the combinations specified by Columns 6,7 and 8 are also shown in Table IX-5.

h. Comparisons of optimized designs

Figure IX-2 illustrates the comparative sizes of the designs described in the preceeding discussion and is self-explanatory.

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Column Number	RTG/RB Gross Weight lbs	Deorbit V ft/sec	Propellant Specific Impulse sec	Mass Fraction %	Volumetric Loading %	Propellant Weight * lbs	Rocket Diameter * inches	Rocket Weight ** lbs
1	725.4	1500	260	80	85	31.3	10	154.4
2	546.2	1500	260	80	85	23.6	9.6	117.2
3	456.5	1200	260	80	85	16.3	8.5	81.2
4	477.8	1500	260	80	85	20.5	9.2	102.1
5	501.0	1500	285	90	92	19.8	8.8	87.5
6	392.9	1200	260	80	85	14.0	8.1	70.2
7	422.2	1200	285	90	90	13.5	7.8	60.0
8	437.3	1500	285	90	91	17.3	8.4	77.0
9	363.0	1200	285	88	90	11.7	7.4	53.0

* Single Rocket - 4 rockets required for deorbit
 ** Total Deorbit Rocket Weight (4)

TABLE IX-2

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Linear scale factor = 0.635
Radiator area = 9.7 ft²
System total weight = 363.0 lb
RTG/RB weight = 228.2 lb
Column 9*
Same size for Columns 3, 6 and 7 weights

Linear scale factor = 0.721
Radiator area = 12.5 ft²
System total weight = 546.2 lb
RTG/RB weight = 298.5 lb
Column 2*
Same size for Columns 4, 5 and 8 weights

Linear scale factor = 1.000
Radiator area = 24 ft²
System total weight = 725.4 lb
RTG/RB weight = 412.2 lb
Column 1*

Column 1*
RB for 24 ft²
radiator

Column 3*
RB for 9.7 ft²
radiator

Column 2*
RB for 12.5 ft²
radiator

RB size comparison
Columns 1, 2 and 3*

*Column numbers refer to Table IX-4 weight statement.
Dimensions are in inches.

Effects of Weight Optimizations on CRONUS RTG/RB-CIR Size

Fig. IX-2.

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Three systems designs emerge from this investigation as representative of basic levels of approach, as follows. The Column No. 1 design is a non-optimized system which shows system feasibility and for which the operational modes and subsystems have been defined.

The Column No. 2 design makes no alteration to the Column No. 1 concept or mode of operation but reflects the results of the optimization of various of the subsystems resulting in a system weight reduction of 179.2 lbs. The Column No. 9 design reflects the ultimate in weight reduction and incorporates some changes in subsystems and operational features that may be considered marginal, while removing 365.4 lbs. from the system weight.

The Column No. 2 design is, therefore, considered to be the best fully capable system which could be designed and built to operate satisfactorily within the present state-of-the-art.

Figure IX-3 shows both the Column No. 1 and Column No. 2 designs mounted on the Nimbus spacecraft in the Titan III launch vehicle fairing.

Table IX-3 compares various characteristics of the three representative size designs.

2. System Weight Sensitivity Study

A system weight sensitivity study of the optimized 546.2 lb. design has been performed. This study shows the gain or loss of weight associated with increasing or decreasing the electrical power output and of the effects of small changes in subsystem weight, area and efficiency.

In the system weight as a function of power level study, the weight-to-power trade-off parameter is dW_T/dP , where

W_T = Total system weight (lb)

P = End-of-life power (watts)

The variation of system specific power with end-of-life power level is shown in Figure IX-4. The variation of total system weight and RTG weight with power, is shown in Figure IX-5.

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Fig. IX-3 Nonoptimized and Optimized Designs in Titan III Launch Fairings

Characteristics ↓	Major Sys. →	C I R			R T G / R B			T O T A L		
	Version →	Col. 1	Col. 2	Col. 9	Col. 1	Col. 2	Col. 9	Col. 1	Col. 2	Col. 9
Weight (lb)		313.2	247.7	131.8	412.2	298.5	228.2	725.4	546.2	363.0
Roll Moment of Inertia (Slug. Ft. ²)		---	---	---	14.3	5.13	X	41.1	X	X
Pitch Moment of Inertia (Slug. Ft. ²)		---	---	---	16.0	6.92	X	63.9	X	X
Yaw Moment of Inertia (Slug. Ft. ²)		---	---	---	16.0	6.94	X	63.9	X	X
C.G. Location (in.) (from H.S. Face & toward apex)		---	---	---	18.0	14.4	12.05	7.5	X	X
Length (in)		31.5	23.65	21.25	59.25	42.60	37.60	84.25	61.42	54.60
Maximum Diameter (in.)		60.0	46.0	39.20	45.15	32.55	28.65	---	---	---
H.S. Curvature Radiator (in)		---	---	---	73.5	53.0	46.60	---	---	---
R.B. Cone Half Angle (Degrees)		---	---	---	18	18	18	---	---	---
Radiator Surface Area (Ft. ²)		---	---	---	24.0	12.5	9.7	---	---	---
Heat Shield Surface Area (Ft. ²)		---	---	---	15.2	7.9	6.15	---	---	---
Aft Cap Surface Area (Ft. ²)		---	---	---	7.7	4.0	3.11	---	---	---
Total Surface Area (Ft. ²)		21.8	11.3	8.82	46.9	24.4	18.96	67.8	35.7	27.78
Volume (Ft. ³)		14.77	5.55	3.8	23.50	8.84	6.04	38.27	13.39	9.83
W/C _p A Hypersonic (RSF)		---	---	---	22.6	30.7	30.3	---	---	---
W/C _p A Subsonic (PSF)		---	---	---	34.7	47.2	46.6	---	---	---
Mass Density (Lb/Ft ³)		20.38	44.60	34.70	17.55	35.20	37.80	18.95	40.80	36.60
Power Watts BOL		---	---	---	360	360	360	360	360	360
Power Watts EOL		---	---	---	250	250	250	250	250	250
Specific Power BOL (Lb/Watt)		---	---	---	1.145	.83	.634	2.01	1.52	1.0
Specific Power, EOL (Lb/Watt)		---	---	---	1.650	1.185	.913	2.90	2.18	1.44
Power Density BOL (Watts/Ft. ³)		---	---	---	15.31	40.7	59.6	9.4	26.9	36.3
Power Density EOL (Watts/Ft. ³)		---	---	---	10.63	28.3	41.4	6.53	18.66	25.45

--- Not Pertinent

X Not Computed

Table IX-3 Summary of Characteristics

The system weight change factor associated with a change in power level is (dW_T/dP) 1.895 lb/watt (for $P = 250$ watts). Furthermore, a reasonable change in the power level will not appreciably change this factor.

It will be noted from Figure IX-4 that the slope of the RTG/RB-CIR curve is still rising, at 500 watts, indicating the optimum system power level is above this point. This same Figure shows that the RTG/RB is about optimum at 250 watts. This feature of the RTG/RB is influenced rather heavily by the assumption made as to the change in thickness of the radiator, as size changes with power. In this study, over the power range investigated, the radiator thickness was assumed constant (.25") which may not be possible, for reasons of structural integrity in sizes required for generators beyond 500 watts.

Sensitivity and trade-off parameters are presented in Table IX-4. The effect of selected changes upon system weights is presented. It is assumed that the RTG efficiency is not affected by any change (except where noted).

It is interesting to see that every pound added to the reentry body requires an additional 0.35 pounds of propulsion, structure etc. When the aerodynamic heating is kept constant (by holding W/D_0AR constant). A further penalty of about 0.2 pounds has to be paid for the larger base diameter.

The basic parameter that is determined in Table IX-4 is:

$$\frac{\delta W_T}{\delta X_i}$$

where X_i is any one of eight parameters. The trade-off ratio (or equivalents) between two different X_i can readily be computed by

$$\frac{dX_i}{dX_j} = \frac{\delta W_T}{\delta X_j} \bigg/ \frac{\delta W_T}{\delta X_i}$$

These results are useful in local optimization in the selection of concepts and in gaining an insight with the system interaction.

3. Cascaded Thermoelectric Subsystem

A thermoelectric cascaded subsystem was examined wherein a SiGe topper rejects heat to a PbTe (2N-3P) package. The details of the study are presented

Fig. IX-4. Specific Power Vs. Total Power

Fig. IX-5. Weight Vs. Electrical Power

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Weight Category	+ 1 lb RTG	+ 1 lb Heat Shield	+ 1 lb Aft Skirt	+ 1 lb Misc.	+ 1 lb (5+6+7)*	+ 1 sec I _{sp}	+ 10% RTG Eff	+ 1 ft ² Radiator Area
RTG	1	0	0	0	0	0	-29	2.5
Heat Shield	0.042	1	0.042	0.042	0	0	-3.4	3.4
Aft Skirt	0	0	1	0	0	0	- .5	0
Miscellaneous	.05	.05	.05	1	0	.02	-1.5	1.5
Communications, electronics power and structure	0	0	0	0	1.0	0	-5.0	5.0
Propulsion	0.254	0.244	0.241	0.254	0.233	-0.45	-9.2	2.76
Net Weight changes	1.346	1.294	1.346	1.346	1.233	0.47	-48.6	15.2
$\frac{dW}{d \text{ parameter}}$	$\frac{1.346}{\text{lb/lb}}$	$\frac{1.294}{\text{lb/lb}}$	$\frac{1.346}{\text{lb/lb}}$	$\frac{1.346}{\text{lb/lb}}$	$\frac{1.233}{\text{lb/lb}}$	$\frac{0.47}{\text{lb/sec}}$	$\frac{-1.5}{\text{lb/w}}$	$\frac{15.2}{\text{lb/ft}^2}$
Parameters held constant:								
Power	Yes	Yes	Yes	Yes	Yes	Yes	No	Yes
W/C _D AR	No	No	No	No	Yes	Yes	No	No
A (radiator)	Yes	Yes	Yes	Yes	Yes	Yes	No	No
P	Yes	Yes	Yes	Yes	Yes	Yes	Yes	Yes

*In structure, etc.of CIR

Table IX-4. Tradeoff Parameters

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in Appendix U. The conclusions reached are:

- (1) On a weight basis, the cascaded RTG is significantly heavier at 463 pounds as compared to 329 pounds for the noncascaded reference design.
- (2) The only possible advantage of the cascaded system will be one of efficiency (5.27% overall vs 4.42%). At beginning of life, these efficiency differences correspond to thermal wattage differences of 1040 watts. If the price of PuO_2 of \$1000/watt(t) is assumed, the cascaded system would offer fuel savings of approximately one million dollars per system. The theoretical efficiency advantage may also be important because of limited PuO_2 availability.

4. Segmented Thermoelectric Subsystem

A detailed discussion of the use of segmented thermoelectrics for improved couple efficiency is given in Appendix S. The current RCA segmented program, funded by the USAEC, is summarized. It appears that with several years development, a relative efficiency gain of 4 to 8% over the present unsegmented couple can be realized by using N-type InAs-GaAs or N-type GaSb-InSb in conjunction with N- and P-type SiGe. Significant improvements beyond this will occur only if new improved P-type segments can be found.

In conclusion, it appears that based on present RCA published results, only a nominal increase will be realized by segmenting the thermoelectrics in the present design. Further breakthroughs are needed to make segmenting attractive for the CRONUS system.

5. Impact Area Control

The entire design philosophy adopted in this feasibility study is one of reliability and unsophistication which, in turn, promotes the greatest confidence in providing nuclear safety at all times. Appendix LL shows that greater impact range control can be gained by selecting a thrust application angle less ($\delta \approx 135^\circ$) than the proposed angle of 180 degrees. This selection of $\delta < 180^\circ$ also would reduce the ΔV requirement from 1500 to 1200 fps. However, in this case, the design benefits to be gained, such as elimination of attitude control systems, all forms of onboard intelligence (except for MDS, and complex mounting, separation and spinup problems, tend to stress further examination of this design approach. A decided advantage of this $\delta = 180^\circ$ selection is the elimination of launch vehicle roll-induced errors which would contribute both to downrange, and most significantly, crossrange errors for $\delta \approx 135^\circ$.

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Furthermore, if δ is fixed at 180 degrees, an inherent flexibility is then realized with regard to abort thrust delays. For example, if an RB abort signal is received at injection and the impact trajectory (along ETR) intersects Africa because of the initial launch azimuth, then a spin-stabilized coast can be initiated at the point of injection. The spin-stabilized coast is continued until the impact point lies beyond the east coast of Africa at which time abort thrust is applied. By initially aligning the spin axis to $\delta = 180^\circ$, a coast of over 60 degrees of arc can be realized because δ will locally vary toward the optimum angle of 135 degrees. This coasting capability along the transfer trajectory would not be available otherwise if δ is initially equal to 135 degrees.

a. Impact point control

Although the application angle and abort ΔV magnitude are the prime control variables to effect an impact from a transfer orbit or orbit, the approach to injection poses an altogether different situation. A certain degree of impact range control can be obtained by shaping the boost trajectory. First, the injection into the transfer orbit can be performed at lower altitudes. The vacuum range from the abort altitude to entry will then be substantially reduced, thereby effecting a reduction in range to impact. Second, if a complete safeguard is desired to guard against a catastrophic situation, e. g., a failure of the insertion stage at apogee, coupled with a no abort-thrust ignition, then shaping the injection to have a high γ will allow positive ocean impact.

The degree of impact range control that can be gained by the first form of trajectory shaping is illustrated in Fig. IX-6. This figure shows range to impact as a function of injection altitude. By injecting at an altitude of 80 nautical miles instead of 100 nautical miles, the range to impact can be reduced from 3690 to 3420 nautical miles--a reduction of 270 nautical miles. If the injection altitude is 90 nautical miles, then the reduction in impact range is 160 nautical miles. This type of range control can become quite important if the impact dispersion pattern from an injection abort lies too close to a land mass.

The second form of trajectory shaping, unlike the first, is costly in terms of payload penalty as will be shown presently. Basically, the technique establishes a transfer orbit at injection which has its vacuum perigee point within the sensible atmosphere. In the event that the circularization maneuver at apogee does not take place and the RB cannot abort, the RB will follow a trajectory that ensures entry and subsequent impact. The ability to select such a trajectory depends on the initial injection conditions (R , V , γ) that are needed to arrive at the first apogee of 600 nautical miles, then impact in an acceptable area.

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Fig. IX-6. Trajectory Shaping for Impact Range Control Injection Condition with Perigee of 600 Nautical Miles

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From Appendix II, it is found that a downrange impact requirement of 180 degrees (liftoff to impact) is necessary to ensure impact in the Indian Ocean for both ETR and WTR launches. This means that from the point of injection, the arc range to impact must be limited to 142.5 degrees. Figure IX-7 shows that an injection flight path of 11.5 degrees and injection velocity of 25,100 fps are required to obtain the single pass transfer trajectory. The 3σ dispersion that can be expected is 178 nautical miles downrange and 20.2 nautical miles cross-range.

The payload capability of the launch vehicles with and without the single pass transfer orbit requirement is summarized.

	<u>Titan IIC</u>	<u>Titan IIIA</u>
Without single pass requirement (lb)	22,000	4500
With requirement (lb)	15,000	2500

Thus, the cost of incorporating this trajectory feature is approximately 30% of the payload.

b. Impact dispersion control

To obtain a greater degree of control in limiting impact dispersions, greater sophistication must be incorporated in the design. This added sophistication takes on the form of a three-axis attitude control system and possibly onboard retro firing control and sequence logic. The need for an attitude control system stems from the fact that the largest dispersions are a result of inaccuracies in aligning and maintaining the retro thrust orientation during firing.

The next largest source of error which causes high dispersion is the thrust cutoff error. This error can amount to 15 fps. It is eliminated by means of an onboard accelerometer and thrust cutoff system.

It is estimated that the system would reduce the downrange dispersion by over 200 nautical miles to a 3σ dispersion of approximately 70 nautical miles. Dispersions due to atmosphere and vehicle uncertainties have not been assessed in this study effort although it is estimated that their contribution would be in the tens of miles. To eliminate atmospheric dispersion and to counteract other tolerances, some lift-to-drag ratio must be provided, together with a control and guidance system.

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Fig. IX-7. Impact Range Control by Trajectory Shaping at Injection

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It is felt that incorporation of most of these systems and features unduly overdesigns a vehicle that needs only to impact in any oceanic area. However, if physical intact retrieval is to be an operational requirement, then some sophistication, namely, an attitude control system or even a flight control system, is mandatory.

6. Transtage as CIR

The orbit injection stage (Titan transtage) contains guidance, control and propulsion capabilities. It might appear attractive to use these features to replace some of the functions of the CIR, particularly if a return from the final orbit is desired.

Unfortunately, this approach is not warranted for either the strontium or plutonium fuel form. Since the LV systems are in the transtage, the various sources of abort conditions during ascent, coast and final injection are caused by the failure of the very system that would be used for the CIR functions. For example, if there is a failure in the LV autopilot, the transtage will not fly correctly. Consequently, there would be a severe reduction in the effective reliability of the CIR during abort.

Furthermore, the transtage cannot perform a rapid unexpected staging function. If such an event were to occur, it is doubtful whether the stages would separate, it is unlikely that the engines could start, and it is questionable that the ascent staging portion of the MDS could be disabled. Therefore, since it is not practical to use the transtage as an emergency escape device, this concept is not acceptable for strontium.

A second alternate exists--use of the transtage for a deorbit stage without real emergency capability. Such an approach might be suitable for plutonium (five-year return capability). This approach is technically feasible. Some other S/C than NIMBUS would have to be considered since it would be necessary to keep the transtage attached to the spacecraft during orbit. Furthermore, the transtage is qualified for only relatively short stays in orbit. The same applies to the Agena (short as in reference to five years in orbit). However, it could be done.

Unfortunately, there will be no real decrease in weight. The reason for this is the necessity of deorbiting at least the empty mass of the transtage (4100 lb). Consequently, even if a NIMBUS satellite (900 lb) were to survive, it would require about 870 pounds of transtage propellant to deorbit the system. If the

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full payload capabilities of the Titan IIC were employed, the amount of transtage propellant required would increase to about 4000 pounds. Either of these amounts are large compared to the weight of the CIR. Consequently, there would be no saving in the effective payload weight penalty of CRONUS by using the transtage.

B. DESIGN FLEXIBILITY

1. Pu-238 Design

The isotope, Pu-238, differs from Sr-90 in five important characteristics:

- (1) Its half life (86 years) is longer than that of Sr-90 (28 years) so that over the same five-year period fuel decay is 4% for the former as compared to 13% for the latter.
- (2) Pu-238 is scarce and costly relative to Sr-90.
- (3) Pu-238 is an α emitter and, as such, generates a considerable quantity of helium in the decay process.
- (4) Pu-238 has a higher energy output per unit volume or weight as compared to Sr-90 compounds.
- (5) Pu-238, as mentioned, is an α emitter and the shielding requirements are minimal relative to Sr-90.

As a result of these differences:

- (1) Less decay (i. e. , less change in operating characteristics from beginning to end of life) permits operation at a higher hot junction temperature and, hence, a higher conversion efficiency at end of life.
- (2) Recovery may be of interest because of scarcity and possibly because of cost.
- (3) The generated helium must be either relieved through a porous plug (presently undeveloped) or contained within the capsule with the latter designed as a high temperature pressure vessel.

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- (4) Pu-238 systems with porous plugs can be lighter than Sr-90 systems. Added features such as recovery, long term (years) reentry capability and helium pressure containment ability will make the Pu-238 system heavier, however.

a. RTG

With Pu-238 fuel, three RTG designs were considered:

- (1) A nonoptimized design with helium pressure relief (porous plug) wherein a Pu-238 heat source is substituted into the reference CRONUS system that was designed for Sr-90.
- (2) Same as Item (1) except capsules are designed to contain helium for a five-year period.
- (3) An optimized RTG (along with an optimized CRONUS) wherein hot junction temperature is upgraded. Because of the higher temperatures, only a pressure relief system was considered.

Each of these three RTG systems is presented in the following.

- (1) Nonoptimized, pressure relief

This RTG represents a system wherein the only two changes in the RTG from the reference Sr-90 concept are that the heat source is different and, as a consequence, some of the structural support items are changed. The radiator and thermoelectrics remain unchanged from the reference Sr-90 RTG. Likewise, operating characteristics are essentially the same. A complete weight summary of the RTG is given in Table IX-5. These weights correspond to the Column 1 design shown in Table IX-1 and reduction of weight can be attained similar to those demonstrated by Columns 2 through 9 of Table IX-1. The effect on the CRONUS is discussed later.

The vented fuel capsule is illustrated in Fig. IX-8. Its size, thermal inventory and external configuration are identical to those of the SrTiO₃ fuel capsule shown in Fig. III-5 and are directly interchangeable in the heat source.

Porous vents are shown in both the outer capsule and the liner to vent the helium gas out of the capsule, yet retain the plutonium dioxide fuel particles. Porous Haynes-25 metal is available with an approximately two-micron pore

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Fig. IX-8 Vented Fuel Capsule for Use with Pu-238

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size in densities of 50% and higher. It is available as either powdered metal or sintered wire or whisker structure. A porous vent using this material has already been designed for use internal to the SNAP 19 fuel capsule to permit helium atoms to flow into the void space within the fuel annulus.

The outer capsule vent is fitted into a 0.31 inch diameter hold, and can be welded, brazed or swaged into place. The liner vent must be brazed or welded into place as it forms an integral part of the liner structure. Experience has shown that porous metals 50% dense or greater can be welded to thin sheet or foil.

Because of the higher power density of Pu-238 fuel, the small annulus indicated is sufficient to contain the required fuel volume. Space within the inner wall of the liner is filled with a Haynes-25 honeycomb structure to prevent the inner liner wall from collapsing upon impact. Experience with SNAP 19 fuel capsule impact tests shows that an unsupported inner wall will collapse upon impact, and will let all the fuel migrate toward the impact end. The honeycomb filler, indicated in Fig. IX-8, will eliminate this tendency and will make the impact characteristics of this fuel capsule more predictable.

(2) Nonoptimized, pressure containment

This RTG differs from the previous one only in that the helium generated is contained within the capsule for five years using an open circuit temperature ($\sim 1800^{\circ}\text{F}$). The stress analysis of the capsule is presented in Appendix T. This approach may be of interest in case development of porous plugs is not fruitful.

A weight summary of this system is given in Table IX-5.

The configuration of the pressure containment capsule is identical to that of the vented capsule except that the outer capsule vent has been eliminated and the capsule wall thickness has been increased to 0.20 inch (constant from end to end). The porous liner vent will permit helium atoms to bleed from the fuel annulus into the void space inside the inner liner wall. This void is also occupied by honeycomb structure which must be fabricated of perforated material to permit helium to permeate the honeycomb.

Haynes-25 material has been used for all components, since high temperature compatibility with plutonium dioxide has not been proved for any other material.

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TABLE IX-5
Pu-238 RTG Weights

	Nonoptimized Pu-238 Design Using Pressure Relief	Nonoptimized Pu-238 Design for Pressure Containment	Optimized Pu-238 Design Using Pressure Relief
Total RTG weight	300.6	394.6	255.0
Heat source subsystem	155.9	237.1	131.0
Fuel capsules (36)	119.9	201.1	100.3
Fuel	35.5	35.5	29.7
Capsule liners and spacers	24.7	24.7	20.7
Outer capsules	59.7	140.9	49.9
Capsule support rings (2)	2.0	2.0	2.0
Thermal insulation	24.8	24.8	20.7
Min-K rings (2)	19.2	19.2	
Micro-Quartz cylinder	5.6	5.6	
Support caps (2)	7.5	7.5	6.3
Central tube	0.2	0.2	0.2
Ball detent and latch mechanism	1.5	1.5	1.5
Thermoelectric subsystem	32.0	32.0	27.8
Hot shoes	6.4	6.4	5.9
Elements	5.6	5.6	4.6
Cold shoes	4.6	4.6	3.8
Flexi-Mod bond stack and hardware	6.6	6.6	5.4
Thermal insulation	8.8	8.8	8.1
Radiator subsystem	112.7	129.5	96.2
Radiator	71.5	71.5	60.1
Insulation, upper ring	4.2	4.2	12.5
Insulation, lower ring	9.4	9.4	
Support cone	13.0	19.8	10.9
T ring	3.2	4.8	2.7
U ring	3.5	5.4	3.0
Latch ring	1.0	1.4	0.8
Locking tube support	0.8	1.2	0.7
Locking tube assembly	0.4	0.6	0.4
Heat source guides (8)	2.2	3.3	1.7
Rivets	0.8	1.2	0.7
Wiring harness	2.0	2.0	2.0
Receptacle	0.7	0.7	0.7

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Fig. IX-9 SiGe Thermoelectric T/E Efficiency Versus Hot Junction Temperature

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(3) Optimized, pressure relief

For a five-year mission with Sr-90, exponential fuel decay requires that a 13% excess in thermal inventory be provided at the beginning of life. With Pu-238 with its longer half life, the decay will only be 4%. Therefore, with plutonium, the possibility exists of operating at higher temperatures at end of life to the point where the beginning-of-life temperature on the fuel capsule is the same as that of the reference Sr-90 system. With this approach, it is possible to operate a plutonium system at end of life at 1540°F on the hot junction (compared to 1450°F with Sr-90 system).

The thermoelectric operating characteristics can be determined from Figs. IX-9, IX-10 and IX-11 for an end-of-life hot junction of 1540°F and a cold junction of 550°F. For 3/4 inch long SiGe elements,

$$\eta_{T/E} = 6.0\%$$

Number of couples = 356 (series-parallel)

P element area per couple = 0.0515 in.² (series-parallel)

N element area per couple = 0.0805 in.² (series-parallel)

The Sr-90 generator has a thermal efficiency of 89%; a Carnot efficiency ratio indicates that the optimized Pu-238 generator will have a thermal efficiency of 88%. The overall efficiency then becomes $\eta_{O/A} = \eta_{T/E} = \eta_{\text{thermal}} \eta_{i^2r} \eta_{\text{regulator}} = (6.0) (8.8) (0.02) (0.995) = 4.82\%$. The end-of-life fuel inventory is

$$q_{eol} = \frac{250 \text{ watts}}{0.0482} = 5190 \text{ watts} = 17,700 \text{ Btu/hr}$$

The beginning-of-life inventory is

$$q_{bol} = q_{eol} = 5190 (1.0392) = 5390 \text{ watts} = 18,400 \text{ Btu/hr}$$

At beginning-of-life, steady state operating temperatures will be essentially the same as those that exist with the Sr-90 RTG. The conditions during reentry and the fireball with the Pu-238 RTG will be close to those with the Sr-90 RTG, although minor differences will be apparent because of differences in heat source mass and fuel properties.

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Fig. IX-10. SiGe Number of Couples Versus Cold Junction Temperature, Series Connected

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Fig. IX-11 SiGe Element Diameter Versus Hot Junction Temperature, Series Connected

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A weight summary of the RTG is given in Table IX-5.

b. CIR/RB

(1) Short-term deorbit

The reference configuration CIR and reentry body can be used with a plutonium heat source (nonoptimized, pressure relief) without modification. The fact that this plutonium RTG is 29 pounds lighter than the strontium RTG does not adversely affect the operation of the reentry systems. The lower weight increases the effective velocity impulse and separation velocity by about 4.3%. These changes are relatively minor for the reentry trajectory and for the dispersion of the velocity vector.

The center of gravity of the reentry body is essentially unchanged, since the Δ weight of the plutonium RTG occurs almost at the present center of mass. The pitch, yaw and roll moments of inertia of the RB would remain essentially unchanged for the same reason. Consequently, there would be no adverse effect on the aerodynamic stability during reentry. The contours of the maximum angle of attack, encountered during reentry and calculated conservatively for the strontium system, will also be applicable. Direct replacement of the strontium fuel capsule with a lighter plutonium system (nonoptimized with pressure relief) will therefore have no significant effect on the reference configuration.

The second plutonium concept (nonoptimized with pressure containment) is 65 pounds heavier than the reference Sr-90 design. This increase in weight diminishes the heat shield margin 8% and the velocity increment 90%. It might therefore appear that a slight redesign may be required with this heavier plutonium system. However, the conservative design of the strontium orientation system will probably permit operation with plutonium, since the latter isotope is less demanding on the reentry flight safety margins. This conclusion, however, must be checked quantitatively.

If the plutonium system is optimized, some nonnuclear weight savings can occur in the RTG/RB and in the CIR. Optimized means a rescaling of the airborne system. With this assumption, it is possible to reduce the CRONUS size and thus the total system weight by 115 pounds (compared to Sr-90 design). This weight saving is due to the reduction in 2 ft² required radiator surface area and the 74-pound decrease in the RTG subsystem (as compared to the Sr-90 design).

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(2) Five-year deorbit

It may be desirable, for particular reasons; to return the Pu-238 system after it has been in orbit for five years. A system designed for reliable return for periods up to five years probably requires some modifications to the reference configuration (Sr-90).

The heat shield could be changed to a heat sink material (i. e., beryllium), since insufficient data are available on ablating materials that have been exposed to five years in an environment of thermal-vacuum and nuclear radiation with the possibility of micrometeoroid damage. The results of long term exposure to these environments on the structural and ablating characteristics are not well known at this time. Loss of significant heat shield material would result in burnup of the heat source on reentry.

The attitude control and guidance system of the S/C would be modified to permit correct orientation of the CIR/RB for separation and deorbit. The Nimbus vehicle requires attitude control throughout the period of its mission; consequently, it would also be available for deorbit orientation. However, since attitude control and guidance systems have experienced reliability problems, a redundant attitude control and guidance would have to be provided. This secondary system would be in a dormant condition until ready for use. Components and systems compatible with the unique environments, or protection from space and nuclear environments, would be required for this backup system. To preclude an uncontrolled reentry, a checkout system would be required to ensure that the redundant system was operating properly prior to initiation of the separation-deorbit sequence. If the redundant attitude control and guidance system and the S/C system were both malfunctioning, then the RTG/RB would not be deorbited.

Initiation of the signal to separate at five years would be by ground command. The signal for initiating deorbit would be a separate signal, also initiated by ground command.

In the event the S/C is other than Nimbus and did not require an attitude control or guidance system, then an attitude control system would be incorporated into the CIR system. This system would be designed to meet the same mission requirements as those defined for a redundant system. Protective shielding would provide an environment to approach shelf life storage. A checkout system would be incorporated to ensure that all systems are operating before initiation of the separation and deorbit sequence.

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The results of these changes in the reference strontium configuration required for a system that will deorbit plutonium (optimized) after a long time in space are shown in Tables IX-6 and IX-7 for two different operating concepts:

Concept I: Five-year return capability for optimized Pu-238 RTG.

Concept II: Concept I plus an added attitude control system for CIR to achieve the required attitude prior to deorbit.

The net weights are as follows (again based on the Column 1 design of Table IX-1):

	<u>Sr-90 Reference Design</u>	<u>Short-Term Deorbit Optimized Pu-238 Design</u>	<u>Concept I</u>	<u>Concept II</u>
RTG/RB	411	328	479	479
CIR	<u>313</u>	<u>281</u>	<u>538</u>	<u>705</u>
Total	724	609	1017	1184

Some of the explicit development problems associated with either system are detailed in the following.

	<u>System I</u>	<u>System II</u>
RB	Long-term beacon	Same
CIR structure	Quality for long stay in space, Determine radiation compatibility	Same Same
Propulsion	Quality for long stay in space, Determine radiation compatibility, Develop micrometeoroid shields*	Same Same Same Develop long stay compatibility attitude control system
Structure	Develop long space environment compatible bearings, track	

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	<u>System I</u>	<u>System II</u>
Electronic	Develop long stay compatibility checkout gear, transmitter, command receiver, etc.	Same
		Develop long stay compatibility autopilot, sensors, guidance system

*This requires determining the actual micrometeoroid flux.

These systems will only deorbit the plutonium fuel. If recovery is desired, several other systems and functions should be added. In particular, a flotation device, a parachute and a location device would be almost mandatory. Furthermore, special provisions should be made to limit the potential dispersion, at impact, of the RTG/RB. Techniques for limiting the dispersion are discussed in Section A-5.

It should also be emphasized that the work "optimized," as used in this study for Pu-238, has merely meant upgrading of RTG operating temperatures because of less fuel decay (Pu versus Sr).

TABLE IX-6

Long Term Concept I

<u>System</u>	<u>Change</u>	<u>Δ Weight (lb)</u>
<u>RTG/RB</u>		
RTG	Change to optimum Pu system	- 69
Heat shield	Change to a Be heat shield	+ 120
Beacon	Make redundant	+ 17
Weight Change RTG/RB		+ 68
<u>Deorbit (CIR)</u>		
Micrometeoroid shield	Add to protect solid motors and ordnance (10 ft ² , 0.4 in. Al)	57

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TABLE IX-6 (Cont'd)

<u>System</u>	<u>Change</u>	<u>Δ Weight (lb)</u>
Radiation shield	Add to protect solid motors, spin motors, separation rocket and auxiliary systems	0 0 0
Bearing	Develop long space environment compatible bearing	20
Track	Develop long space environment compatible track	10
Command receivers and electronics	Make redundant; add orbital checkout	30
Separation rocket	Increase for increased weight	11
Deorbit Propulsion	Increase for increased weight	97
Weight change, deorbit system		<hr/> 225
Total weight change		293

TABLE IX-7

Long Term Concept II

<u>System</u>	<u>Change</u>	<u>Δ Weight (lb)</u>
<u>RTG/RB</u>		
RTG	Change to optimum Pu system	-69
Heat shield	Change to a Be heat shield	120
Beacon	Make redundant	<hr/> 17
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TABLE IX-7 (Cont'd)

<u>System</u>	<u>Change</u>	<u>Δ Weight (lb)</u>
<u>Deorbit</u>		
Micrometeoroid Shield	Add to protect solid motors and ordnance attitude control system	57 29
Radiation shield	Add to protect solid motors, separation rocket and auxiliary system	0 0 20
Bearing, Tracks	Delete	-18
Payload attachment	Add (replaces tracks and bearings)	+12
Separation rocket	Still needed, add for increased weight	+15
Attitude control	Add new system	50
Guidance and autopilot	Add new system (needs some development)	35
Electrical power (batteries)	Add, needed for autopilot and guidance and attitude control system	22
Command receivers and electronics	Make redundant, add orbital checkout	30
Deorbit propulsion	Increase for increased weight	140
Δ Weight, deorbit system		392
Δ Weight, total		460

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2. Ground Handling Equipment--WTR Modifications

The selected concept is applicable for use at WTR. A Titan IIIB complex, PALC 2, Pad 3 (recently redesignated Space Launch Complex (SLC)) at Vandenberg AFB, California (see Fig. IX-12) is similar to a Titan IIIC complex at ETR. A Titan IIIB complex is equipped with a mobile service tower which moves on rails between the launch stand and a stowage position out of the blast area. Travel speed is 36 fpm in a 35-mph wind or 10 fpm in a 50-mph wind. The MST is used to erect the L/V onto the launch stand and also provides work platforms. The platforms have the capacity of 40 psf or 2000 pounds concentrated over an area 2.5 ft^2 . A much smaller fixed structure is provided beside the launch stand for umbilical booms. The umbilical tower structure is approximately 100 ft^2 by 130 feet high (see Fig. IX-13). One corner of the tower is cut out to accept the Agena umbilical boom which folds into the tower at liftoff.

The umbilical tower at WTR was considered for installation of the fuel block loading system. However, inspection of the facility and, specifically the UT at Pad 3, PALC 2 indicates that the tower does not have sufficient strength, stability or overall height to support the cask and fuel block loading system. As an alternative to an UT installation of the equipment, installation of the nuclear GHE on the MST was reviewed with the Martin Company WTR engineering personnel and was concluded to be a feasible approach.

An area on the south face of the MST at payload level was evaluated as providing adequate space and existing structural support for the system installation. Other modifications to the MST are required to automate the structure for remote operations compatibility with the selected fuel handling concept. These modifications are feasible and consist of making remote service-type connections, i.e., electrical, water, GN_2 , etc., and alterations to the MST drive system. Although feasible, these connections will be complex in configuration. System and design tradeoff studies might indicate the desirability of abandoning the idea of an automated MST in preference to the necessary extensive structural modifications to the existing UT or the provisioning of a new UT.

Modifications required at the WTR launch site are as follows:

- (1) Beef-up MST in area of open structure on south elevation TS 139.
- (2) Cut hole through MST weather curtain for boom and FHM to penetrate.

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Fig. IX-12. Titan III Launch Pad at Point Arguello Launch Complex

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Fig. IX-13. North Orientation of Umbilical Tower at PALC 2, Pad 3

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FIG. IX-14. FUEL BLOCK LOADING CONCEPT TOWER BOOM SIDE LOADING

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- (3) Install boom and winch on top of MST on south side to raise fuel loading system and cask to Elevation 139.
- (4) Modify MST drive controls and limit switches for remote operation.
- (5) Provide quick disconnects at base of MST for services needed during fuel block loading operation.
- (6) Installation of pneumatic, electrical, instrumentation and communication lines to the fuel handling installation for remote actuation, control and monitoring of the system.

a. Services and service routing

Although not investigated in detail, it is estimated that the modifications necessary for services and service routing are similar to those identified at ETR (see Chapter III-3).

b. Countdown modifications

The Titan IIIB countdown is less affected by the fuel block loading sequence than is the Titan IIIC countdown.

Within the current Titan IIIB count, all personnel are evacuated from the site at T-60. A programmed hold is not forecast in the Titan IIIB system for fuel block loading. Fuel block loading will commence at T-60, proceeding simultaneously with other L/V check functions, and be completed at T-30. From T-30 until launch, the RTG/RB and CIR would be in a monitor status.

3. Ground Handling Equipment--Side Loading Modification

The side loading fuel block loading system is shown in Fig. IX-14. The side loading concept was adapted from the selected top loading concept. That portion of the system which is on the tower structure (pallet assembly) including the cask is identical to the top loading configuration. The area of change is in the four-foot shorter boom and a revised compliance device which permits a 90-degree rotation of the fueling probe. An additional position locator is required with a latching device at the end of the probe to lock the FHM to the launch vehicle. This probe is mounted on the structure supporting the fueling probe and will maintain alignment between the fueling probe and the RTG during L/V away and clocking. A socket is provided in the L/V fairing with a funnel-shaped lead-in of six inches diameter at the mouth to assist in remote connection. Detail operation of the side loading system is similar to the top loading system. The two vertical index probes engage the cask, aligning the FHM with the fuel block. The

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block is retracted into the nitrogen shroud and the shroud with the fuel block is rotated 90 degrees as the FHM is moved along the boom. The position locator engages and locks onto the L/V and the fueling probe is extended transferring the fuel block to the RTG.

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