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**STUDY OF SEPARATELY LAUNCHED
MULTI-USE SPACE ELECTRICAL
POWER SYSTEM**

MIDTERM TECHNICAL BRIEFING

JANUARY 11, 1968

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ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC

**MISSILE AND SPACE DIVISION
KING OF PRUSSIA PARK
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**STUDY OF SEPARATELY LAUNCHED
MULTI-USE SPACE ELECTRICAL
POWER SYSTEM**

MIDTERM TECHNICAL BRIEFING

JANUARY 11, 1968

PREPARED FOR

**NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
POWER SYSTEMS SECTION
POWER GENERATION BRANCH
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS 77058**

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INTRODUCTION

The Separately Launched Power Module Program was initiated in August of 1967. The purpose of the program was to determine the technical feasibility of a separately launched power module including a radioisotope heat source and to provide technical data which might be used in the evaluation of the advantages and disadvantages of this approach.

This document contains the presentation material used in the oral mid-term briefing given in Houston, Jan. 11, 1968. Each slide is accompanied by a brief description of the key points contained therein.

ADVANCED NUCLEAR SYSTEMS OPERATION

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PROGRAM OBJECTIVES

TO PROVIDE DATA ON A SEPARATELY
LAUNCHED RADIOISOTOPE POWER MODULE TO:

- . EVALUATE MULTI-USE CAPABILITY
- . DETERMINE ADVANTAGES AND DISADVANTAGES

ECONOMIC

OPERATIONAL

SAFETY

The system requirements have been identified by the customer as the performance necessary from a separately launched power module which might be operational in the time period identified. The thermal input is at start of life. The electrical output is the minimum acceptable regulated output from the alternator. The electrical power is split 50 per cent 400 Hz AC and 50 per cent 28 volt DC.

The subsystems, including the PM, for each of the manned spacecraft should be qualified and available for test and integration one year prior to launch.

ADVANCED NUCLEAR SYSTEMS OPERATION

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SYSTEM REQUIREMENTS

ELECTRICAL OUTPUT	5Kw
THERMAL INPUT	25Kw
MISSION LIFE	2 YEARS

SELF-CONTAINED POWER MODULE TO BE USED WITH:

<u>SPACECRAFT</u>	<u>LAUNCH DATE</u>	<u>ORBIT</u>
A. SATURN V SINGLE LAUNCH ZERO "G"	FIRST QUARTER '75	260 NM 28.5 TO 50 ⁰
B. DUMBELL ARTIFICIAL "G"	FIRST QUARTER '75	260 NM 28.5 TO 50 ⁰
C. MARS FLYBY	1980	_____
D. AAP CLUSTER II	1973	260 NM 28.5 TO 50 ⁰

BLANK

SYSTEM CONSTRAINTS

- . Pu 238
- . ORNL FUEL CAPSULE
- . HE CONTAINMENT
- . $T_{\text{MAX}} = 2000^{\circ}\text{F}$ STEADY STATE
- . $T_{\text{MAX}} = 2500^{\circ}\text{F}$ TRANSIENT FOR 30 MIN
- . RADIATION DOSE RATE < 25 REM/YEAR
- . TWO REPLACEABLE POWER GENERATION UNITS
- . HEAT TRANSFER BY RADIATION FROM HEAT SOURCE TO HSHX

The components of a Separately Launched Power Module are shown here. Capability for re-entry recovery is provided by the re-entry vehicle and the controlled intact re-entry (CIR) package. The CIR package and the re-entry vehicle form the total re-entry system (RS). The heat source heat exchangers and power generation units form replaceable units which are mounted on guide rails for ease of handling. The transfer of heat from the fuel capsules to the HSHX is by radiation.

ADVANCED NUCLEAR SYSTEMS OPERATION

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POWER MODULE CONFIGURATION

The power module can dock with any of a number of manned spacecraft. It is shown here docked with a 248,000 lb. zero "g" space station which was studied by Boeing for NASA-MSC. The 6 Kw of electrical power and 3 Kw of environmental control and life support thermal energy supplied by the PM would be typical of the power requirements of a space station which might be flown in the middle to late 1970's. This space station has a 6 man crew and is designed to perform both solar and earth pointing experiments.

ADVANCED NUCLEAR SYSTEMS OPERATION

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PM DOCKED TO MANNED SPACECRAFT

One of the first of the study tasks was to determine whether the proposed Atlas-Agena launch vehicle had sufficient weight capability to launch the PM into the desired orbits.

A launch into a 28.5 degree inclination orbit is a due east launch from Cape Kennedy and therefore the easiest and most logical orbit for early manned spacecraft to fly. A 260 n.m. 50° inclination is the other extreme anticipated for earth surveying missions.

A launch over water was chosen since it provided an added margin of safety for launching of the radioisotope heat source and because it was a nominal launch approximately midway between the 28.5 and 50° inclination orbits which formed a logical parking orbit for calculation of available payload weights.

REQUIREMENTS:

- . PAYLOAD CAPABILITY TO LAUNCH PM TO 260 N.M.
28.5 AND 50° INCLINATION.
- . LAUNCH OVER WATER
- . LAUNCH FROM CAPE KENNEDY

The probability of land impact following an abort during the ascent phase can be minimized by a trajectory which does not pass over the islands of the West Indies, the coast of South America or the southern coast of Africa. Although no constraints have been established which preclude passage over land, a launch azimuth of approximately 114 degrees would not intersect any of these land masses.

EFFECT OF LAUNCH AZIMUTH ON INSTANTANEOUS IMPACT POINTS

NOMINAL TITAN III-C BOOST TRAJECTORY FROM ETR TO 100 N.M.

In addition to the requirement of placing the PM in a parking orbit, it is also necessary to provide sufficient ΔV to raise the altitude to 260 n.m. and the inclination from the 36° corresponding to a 114°E launch azimuth to the maximum inclination requirement of 50° . Choice of a velocity excess of 7200 fps provides some flexibility in the selection of available launch windows.

MINIMUM INITIAL RENDEZVOUS REQUIREMENTS

The payload capability of three typical launch vehicles is shown. With a PM weight in excess of 8000 lbs., it is apparent that the only vehicle which appears attractive is the Titan IIIC.

LAUNCH VEHICLE PAYLOAD CAPABILITIES

The initial rendezvous phase includes altitude change requirements resulting from the use of an initial parking orbit, plane change requirements resulting from the launch azimuth constraint imposed on the Power Module, and any phasing corrections resulting from the actual time of launch. The chart summarizes the more significant characteristics of a number of initial rendezvous options that can be used to achieve Power Module - Space Station separation distances of the order of 50 n.m.

The direct ascent approach involves a direct injection into the terminal Space Station orbit without the use of an initial parking orbit. The required plane change is carried out immediately after orbit injection. A direct ascent launch can only be used when the Power Module orbit injection point intersects the Space Station orbit and when the Station is at the injection point. This double requirement will result in an extremely small launch window and in launch opportunities days to weeks apart.

The requirement for the orbit injection point to intersect the Space Station orbit can be eliminated by the use of an initial parking orbit and a Hohmann or Modified Hohmann transfer involving a 180 degree transfer angle to the terminal orbit. In the Hohmann transfer, the required plane change is split and carried out in conjunction with the initial and terminal impulses required for the altitude transfer. The Modified Hohmann involves a third impulse for the total plane change at the node of the parking orbit (with respect to the Space Station orbit). Some improvement in launch window can be achieved by either of these techniques but at the expense of substantial stay-times in the parking orbit.

The launch window can be increased to the order of several hours per day by the use of transfer angles other than 180° (60° to 300°). The dog-leg transfer is the counterpart of the Modified Hohmann and the intersecting plane transfer that of the Hohmann. In either case, the stay time in the parking orbit and the transfer time to the terminal orbit are optimized to minimize the total propulsion requirement. The subsequent studies have been based upon the intersecting plane approach in order to minimize the total number of transtage burns required for the rendezvous operation.

INITIAL RENDEZVOUS OPTIONS

	DIRECT ASCENT	HOHMANN TRANSFER	MODIFIED HOHMANN	DOG-LEG TRANSFER	INTERSECTING PLANE TRANSFER
LAUNCH WINDOW	VS	S	S	M	M
PARKING ORBIT	NO	YES	YES	YES	YES
ALTITUDE TRANSFER REQUIREMENTS	N	O	O	M	M
PLANE CHANGE REQUIREMENTS	M	M	O	O	M
NO. IMPULSES (AFTER ORBIT INJECTION)	1	2	3	3	2

NOTE: N - NONE
 VS - VERY SMALL
 S - SMALL
 O - OPTIMUM
 M - MODERATE

The characteristics of a typical intersecting plane transfer trajectory are illustrated with respect to an inertial reference frame. The Power Module launch time must be within 0 to 4 hours (24 to 28, 48 to 52, etc.) of the Space Station launch time in order to obtain a node (with respect to the equator) within 0 to 30 degrees West of the Space Station node. The initial impulse changes the parking orbit to the illustrated transfer orbit and provides a part of the overall plane change requirement. The second impulse is applied where the transfer orbit intersects the terminal orbit to complete the plane change requirement and to convert the transfer orbit into the desired terminal orbit. Note that the sum of the individual plane change requirements is greater than the actual plane change since neither of the impulses occur at the intersection of the parking and terminal orbits.

TYPICAL INITIAL RENDEZVOUS TRANSFER TRAJECTORY

The total velocity requirements for achieving an initial rendezvous with the Space Station at a 28.5 degree orbit at 260 n.m. are illustrated. Data are included for Power Module launch time of 0 to 4.5 hours and stay-times in the 100 n.m.-30 degree parking orbit of 0 to 30 minutes with respect to a Space Launch time at 0 hours. These data represent the results of over 10,000 transfer trajectory calculations which have been generated in order to select the envelope of optimum orbit transfer times. Note that the optimum transfer times are of the order of 25 to 40 minutes.

Four discrete launch windows are indicated for a maximum initial rendezvous capability of 7200 fps. There are two relatively narrow windows at 1.7 and 3.4 hours and two somewhat broader windows centered at 2.5 and 4.0 hours. These windows can be expected to be repeated with minor changes every 24 hours.

A minimum velocity requirement of 3800 fps was achieved at 2.45 hours. This is 450 fps higher than the ideal minimum velocity of 3350 fps and is a result of the use of the intersecting plane approach for the initial rendezvous. Conversely, the use of the three impulse dog-leg maneuver should be capable of recovering most of this 450 fps penalty as a result of utilizing an optimum plane change impulse at the node.

28.5° INCLINATION LAUNCH WINDOW

Similar data are illustrated for rendezvous with the Space Station at a 50 degree orbit at 260 n.m. Data are included for Power Module launch times of 0 to 3.5 hours.

The more severe plane change requirements associated with the 50 degree rendezvous have resulted in an increase in the velocity requirements of the order of 3000 fps. Only two discrete launch opportunities are available with initial rendezvous requirements below the assumed maximum capability of 7200 fps. These include a narrow window at 1.6 hours and a broader window at 2.3 hours.

The minimum velocity requirement of 7050 fps, in this case, represents an 850 fps penalty over the ideal minimum velocity which might be recovered by the use of a three impulse dog-leg transfer.

50° INCLINATION LAUNCH WINDOW

The launch windows compatible with a maximum initial rendezvous capability of 7200 fps are summarized in the Table for terminal orbits of both 28.5 and 50 degrees. Note that a total launch window per day of 100 minutes was obtained for the 28.5 degree orbit and a window of 20 minutes per day for the 50 degree orbit. Both of these windows are believed to be ample for use with the Titan IIIC launch vehicle.

POWER MODULE LAUNCH WINDOW

SPACE STATION			
ORBIT INCLINATION	DEG.	28.5	50.0
ORBIT ALTITUDE	N.M.	260	260
LAUNCH TIME	HR.	0	0
POWER MODULE			
PARKING ORBIT INCLINATION	DEG.	36.0	36.0
PARKING ORBIT ALTITUDE	N.M.	100	100
INITIAL RENDEZVOUS VELOCITY	FPS.	7200	7200
LAUNCH TIME	HRS.	1:38-1:44 2:02-2:54 3:17-3:26 3:44-4:17	1:42-1:44 2:09-2:27
TOTAL LAUNCH WINDOW/DAY	MIN.	100	20

The terminal rendezvous phase is used to reduce the separation distance between the Space Station and the Power Module from 50 n.m. to the order of 1 n.m. and to provide a vernier control on the relative closing velocity. The chart illustrates the three terminal rendezvous options that have been investigated.

The first approach utilizes the main engine of the Titan transtage to initiate the terminal rendezvous from a nominal separation distance of 50 n.m. and no closing velocity. The main engine is fired for a minimum pulse width of 5 to 6 seconds to achieve a closing velocity of 180 to 190 fps. The transtage attitude control system is used to provide vernier correction. The Power Module is then allowed to coast for about 28 minutes until the separation distance has been reduced to about 1 n.m. The main engine is fired for a second minimum pulse and the attitude control system used to eliminate any residual closing velocity.

The second approach differs from the first in that the transtage attitude control system is used in place of the main engine to provide the initial closing velocity and remove the velocity at the end of the terminal rendezvous. The propellant capacity of the attitude control system will limit the closing velocity to the order of 15 fps and will result in a closing time of about 2 hrs. 50 min.

The third approach involves a modified initial rendezvous maneuver which has been biased to cut off at a closing velocity of 196 ± 36 fps at the nominal separation distance of 50 n.m. This serves to reduce the initial rendezvous requirement and to eliminate the initial firing of the transtage main engine. It does, however, result in a somewhat greater initial vernier correction requirement and in a 47 minute (half orbit) closing time. Note that the longer closing time arises from the fact that the initial rendezvous bias velocity is primarily an out-of-plane error and, consequently, requires a half orbit to re-cross the Space Station orbit.

TERMINAL RENDEZVOUS OPTIONS

The maneuver characteristics of each of the terminal rendezvous options are illustrated in the Figure. Radial and transverse separation distances are plotted as a function of the circumferential separation distance. Note that the positive direction of the X-axis opposes the direction of motion of the Space Station and that the positive direction of the Y-axis is radially outward from the earth.

The terminal rendezvous maneuver with the transtage main engine is initiated from a point 50 n.m. behind the Space Station. The initial closing impulse causes the Power Module to drop 13 n.m. below the Space Station orbit and to re-cross the orbit at a point 1 n.m. behind the Space Station. The second impulse then cancels the closing velocity. The terminal rendezvous maneuver with the transtage attitude control system is also initiated 50 n.m. behind the Space Station. The Power Module drops 5.5 miles below the Space Station orbit and then crosses the Space Station orbit at a separation of 24 n.m. after one orbit. During the second orbit, the Power Module again drops 5.5 miles and re-crosses the orbit at a separation of 1 n.m. There is no transverse (out-of-plane) separation in either of these maneuvers.

The biased initial rendezvous maneuver must be initiated from a point 50 n.m. ahead of the Space Station in order to utilize the residual radially outward velocity component from the initial rendezvous maneuver. It will also have a somewhat larger initial transverse velocity component from the plane change associated with the initial rendezvous. The transverse separation increases to a maximum distance of 32 n.m. at an altitude 12.5 n.m. above the Space Station orbit during the terminal rendezvous. Both the radial and transverse separations are reduced to zero at a circumferential separation of 1 n.m.

TERMINAL RENDEZVOUS MANEUVERS

TRANSTAGE ATTITUDE CONTROL SYSTEM

BIASED INITIAL RENDEZVOUS

Velocity and propellant requirements are summarized for each of the terminal rendezvous options investigated. The transtage main engine approach has been based upon a minimum main engine impulse of 184 fps for each burn which required 550 lbs. of main engine propellant. Attitude control system propellant requirements include propellant for the 15 fps of vernier correction and, in addition, the propellant requirements for five burns of the main engine (for injection into the parking orbit, initial and terminal burns for the initial rendezvous, and initial and terminal burns for the terminal rendezvous) for propellant settling and vehicle alignment.

The attitude control system approach eliminates the main engine propellant requirement and the associated attitude control system requirements for the last two main engine burns. The increased attitude control system translation requirements, however, result in a total ACS propellant requirement of 104 lbs. Note that this is only 10 lbs. less than the capacity of the transtage ACS propellant tank.

The biased rendezvous approach combines the best features of each of the prior approaches. The single main engine impulse required has been included as a part of the nominal initial rendezvous requirement. Attitude control system propellant requirements are based upon four main engine burns and a terminal vernier requirement of 10 fps. and have, therefore, been reduced to 86 lbs. Finally, the closing time of 47 minutes is roughly comparable to the 28 minutes requirement for the main engine approach. The biased rendezvous approach has, therefore, been selected for the nominal vehicle.

TERMINAL RENDEZVOUS REQUIREMENTS

RENDEZVOUS MODE	TRANSTAGE MAIN ENGINE	ATTITUDE CONTROL SYSTEM	BIASED INITIAL RENDEZVOUS
INITIAL RENDEZVOUS VELOCITY-FPS.	7200	7200	7004±36
TERMINAL RENDEZVOUS VELOCITY-FPS.			
INITIAL IMPULSE - MAIN ENGINE ATT. CONT. SYS.	184±5	15±.5	
VERNIER CORRECTION	5	.5	
TERMINAL IMPULSES - MAIN ENGINES ATT. CONT. SYS.	184±5	15±.5	196±36
VERNIER CORRECTION	10	1	10
TOTAL VELOCITY REQUIREMENT	393	32.5	10*
PROPELLANT REQUIREMENTS - LBS.			
MAIN ENGINE	550	—	—
ATTITUDE CONTROL SYSTEM	91	104	86
TOTAL PROPELLANT	641	104	86
CLOSING TIME	28 MIN.	2 HRS. 49 MIN.	47 MIN.

*WITH RESPECT TO NOMINAL 7200 FPS. INITIAL RENDEZVOUS REQUIREMENT

Having defined the trajectory from launch to terminal rendezvous, it is possible to define the available payload weight for the PM.

Current plans and development to increase the payload capability of the Titan IIIC should provide additional margin for vehicles to be flown in the 1973 to 1980 time period.

All of the payload data is based on that provided in the reference given below.

Titan IIIA and IIIC Standard Space Launch System (SSLS) Definition for Payload Contractors, SSD-CR-65-18 Rev 2, June 1966.

AVAILABLE PAYLOAD WEIGHT SUMMARY

	LBS.
PARKING ORBIT - 100 N.M. AT 36°	
NOMINAL PAYLOAD	24,800
PAYLOAD SHROUD ALLOWANCE	260
ACTUAL PAYLOAD	24,540
INITIAL RENDEZVOUS - 50 N.M. SEPARATION	
TRANSTAGE MAIN ENGINE PROPELLANT	15,440
TRANSTAGE ACS PROPELLANT	39
PAYLOAD	9,061
TERMINAL RENDEZVOUS - 1 N.M. SEPARATION	
TRANSTAGE ACS PROPELLANT	75
AVAILABLE POWER MODULE SEPARATION WEIGHT	8,986

Evaluation of various docking schemes requires consideration of the entire mission profile. The following charts indicate a multitude of possibilities and identify those attributes utilized in selecting a technique for docking.

Several docking options are available, depending on the launch sequence adopted and the equipment capabilities that will exist at the time of the proposed mission. U. S. docking experiments thus far have employed a passive vehicle equipped with an attitude control and an active spacecraft controlled by an astronaut on board. A recent Russian mission has demonstrated the feasibility of automatic unmanned docking. Docking simulation experiments have shown that no serious docking problems are encountered in an arrangement whereby an astronaut aboard one spacecraft controls, by means of an RF link, a second spacecraft equipped with an attitude control and propulsion system.

Regardless of the procedure used, it is essential that the docking operation be highly reliable and that it be performed with minimum hazard to the crew and the orbiting equipment. Docking equipment flexibility is desirable to permit the substitution of alternate docking procedures should the prime mode fail.

Five docking sequences that have received serious consideration are shown. Analysis of these five sequences indicates that there are four actual docking situations that can occur:

- Case I. Docking of the Logistics Spacecraft to the Space Station. This is an operation not directly associated with the Power Module and is therefore not considered further.
- Case II. Docking of the Power Module to the Space Station by means of a ground command initiated automated docking system, with attitude control and propulsion system on board the Power Module.
- Case III. Docking of the Power Module to the Space Station by means of commands carried through an RF link from the Space Station to the Power Module (attitude control and propulsion systems on board the power module)
- Case IV. Docking of the Power Module to the Space Station by first docking it to the Logistics Spacecraft and then employing the Logistics Spacecraft as a space tug to rendezvous and dock the Power Module to the Space Station.

LAUNCH, RENDEZVOUS AND DOCKING OPTIONS

SS — SPACE STATION

LSC — LOGISTICS SPACECRAFT

PM — POWER MODULE

Concerning ourselves only with the docking options of the PM for the moment, it is noted that these are summarized on the left hand side of the chart and that, in fact, there are only three options. The sources of attitude control and propulsion for each are shown on the right hand side of the chart.

Advantages and disadvantages of Cases II, III, and IV are presented in the following charts. The final choice of docking maneuvers is made after considering the characteristics of these docking alternatives along with other facets of the mission that are discussed in connection with the SLPM mission profile.

The use of a CSM as a tug is shown on the first chart.

DOCKING OPTIONS

MANNED DOCKING IMPLICATIONS

ADVANTAGES

- NO PM PROPULSION
- NO LAUNCH VEHICLE MODIFICATIONS
- LOWEST WEIGHT PM
- LOWEST COST PM

DISADVANTAGES

- ONE MORE SEPARATION AND DOCKING
- LOGISTICS S/C MODIFICATIONS
- HIGHEST FUEL USAGE
- ASTRONAUT VISION LIMITED
- LSC MUST BE LAUNCHED BEFORE PM CAN BE DOCKED
- SECOND DOCKING RING REQUIRED
- MEN COMMITTED BEFORE PM POWER AVAILABLE

LSC PROPULSION

MANNED DOCKING IMPLICATIONS

ADVANTAGES

- PM WEIGHT < AUTOMATIC CASE
- COMPLETE ASTRONAUT CONTROL
- ONLY ONE DOCKING MANEUVER
- GOOD ASTRONAUT VISION

DISADVANTAGES

- PM CANNOT BE DOCKED UNTIL AFTER CREW IS ON SPACE STATION
- CREW IS EXPOSED TO DOCKING MALFUNCTION HAZARDS
- CREW COMMITTED BEFORE PM POWER IS AVAILABLE

PM PROPULSION

UNMANNED DOCKING IMPLICATIONS

ADVANTAGES

- SIMPLEST PM
- NO CREW DANGER
- ONE DOCKING MANEUVER

DISADVANTAGES

- REQUIRES TRANSTAGE MODIFICATION
- INVERTED LAUNCH POSITION
- EASE OF R/V ABORT REDUCED
- TWO DOCKING RINGS
- NO BACKUP AVAILABLE

TRANSTAGE PROPULSION

UNMANNED DOCKING IMPLICATIONS

POWER MODULE PROPULSION

ADVANTAGES

- COMPLETELY SELF-CONTAINED PM
- MINIMUM DOCKING PROPELLANT
- NO CREW DANGER
- ONE DOCKING MANEUVER
- PERMITS MANNED AS BACKUP
- NO LAUNCH VEHICLE MODS

DISADVANTAGES

- HEAVIEST PM
- MOST COSTLY PM
- MOST COMPLEX PM

The chart summarizes the individual maneuvers required for the docking operation. It is assumed that the docking operation begins at a point 1 n.m. from the Space Station and at essentially no closing velocity. The Titan transtage is separated from the Power Module away from the Space Station and the Power Module yawed 180° to point the laser sensor at the target. After laser lock-on has been achieved, two of the yaw engines and two of the pitch engines are fired in a translation mode to achieve a 2.5 fps closing velocity. During the 40 minute closing phase, the thrustors are limit cycled to maintain an orientation of ± 0.2 degree about each axis. The other pitch and yaw engines are then fired to cancel the closing velocity at a separation of 5 to 10 feet from the space station. The thrustors are then used as required to realign each axis, to eliminate the final separation distance, and to carry out the actual docking.

DOCKING MANEUVER

INITIAL CONDITIONS

RANGE	1 N.M.
RANGE RATE	0
ROLL SEPARATION RATE	.1 DEG/SEC.
PITCH & YAW SEPARATION RATES	.04 DEG/SEC.
SEPARATION POSITION ERRORS	.2 DEG/SEC.

OPERATION	TIME MIN.	TRANSLATION			CONTROL IMPULSE LB. SEC.
		V FPS.	IMPULSE LB. SEC.	PULSE SEC.	
YAW 180°	0	0	0	0	28
FIRE 4 ENGINES TO START CLOSURE	6	2.5	775	14	0
LIMIT CYCLE OPERATION DURING COAST	6	0	0	0	252
FIRE 4 ENGINES TO STOP CLOSURE	47	-2.5	775	14	0
VERNIER CORRECTION	49	2.5	775	14	0
REALIGN AXES (12 ONE AXIS ROTATIONS)	49	0	0	0	133
DOCK TO SPACE STATION	59				
TOTALS	59		2325		413

The requirements for the docking system have been summarized. Four nozzles have been assumed for each axis in order to permit vehicle translations in either direction during the final vernier correction phase and to provide balanced couples for each axis rotation. These would, in all probability, involve two 4-nozzle clusters mounted on each end of the yaw axis and two 2-nozzle clusters mounted on each end of the pitch axis. System weights have been calculated on the basis of a 50% total impulse margin and for systems utilizing either Freon 14 or Hydrazine propellants. Weight and power requirements for a laser sensor which would be used to control the operation have been included. These numbers have been based upon a device which has been under development by NASA-MSFC for the past few years. It is anticipated that a comparable system will be operational when needed.

DOCKING REQUIREMENTS

CONTROL PARAMETERS		UNITS	ROLL	AXIS PITCH	YAW
CONTROL TORQUE		FT.-LBS.	52	140	140
NUMBER OF NOZZLES			2+ 2-	2+ 2-	2+ 2-
ANGULAR ACCELERATION		DEG./SEC. ²	±1.0	±1.0	±1.0
NOZZLE THRUST		LBS.	5.2	14.0	14.0
MOMENT ARM TO C.G.		FT.	5	5	5
LIMIT CYCLE RATE		DEG./SEC.	.025	.025	.025
			POWER MODULE		SPACE STATION
PROPULSION SYSTEM	TOTAL IMPULSE (INCLUDING 50% MARGIN)		4110 LB.-SEC.		
	POWER REQUIREMENTS		30 WATTS PLUS 120 WATTS PEAK FOR TWO 14 SEC. PULSES		
	PROPELLANT		FREON 14	HYDRAZINE	
	PROPULSION & CONTROL SYSTEM WT.		136	36	
LASER SENSOR	POWER REQUIREMENTS		33 WATTS		24 WATTS
	SYSTEM WEIGHT		46 LBS.		14 LBS.
TOTAL DOCKING SYSTEM WEIGHT			182 LBS.	82 LBS.	14 LBS.

The next three figures depict the selected mission profile pictorially; the following figure presents it diagrammatically. The mission profile selection is based on a number of judgements that can only be subjective at this date; consequently the profile is subject to change as developments unfold in such vital technologies as automated docking, for instance.

The two events that largely determine the mission profile are the launch sequence and the docking technique employed. The advantages and disadvantages of various docking techniques have been discussed previously.

There are six possible launch sequences:

1. Space Station, Power Module, Logistics Spacecraft
2. Space Station, Logistics Spacecraft, Power Module
3. Power Module, Space Station, Logistics Spacecraft
4. Power Module, Logistics Spacecraft, Space Station
5. Logistics Spacecraft, Space Station, Power Module
6. Logistics Spacecraft, Power Module, Space Station

Sequences 4, 5, and 6 each require an LSC launch prior to the SS launch. There appears to be no merit in this procedure, and many reasons to avoid it; these sequences are therefore eliminated.

MISSION PROFILE (SS AND PM LAUNCH)

SS CHECKOUT

PM CHECKOUT & START UP

JETTISON TRANSTAGE

DOCK PM TO SS

The next question to consider is whether the SS launch should precede the PM launch, or vice versa. Arguments favoring each procedure are:

SS preceding PM

1. Safety of General Populace. The potential hazard to the general populace is relatively high during a PM launch; relatively low during a SS launch. It therefore appears desirable to commit the PM only after SS orbit and operation has been assured.
2. Cost. Production and launch costs of the SS will be much higher than those of the PM. The practicality of providing a backup SS is questionable.
3. Launch Vehicle Characteristics. The Titan IIIC is relatively simple, is designed for launch within 2 seconds of a pre-selected time, and can be subjected to 30-day holds any time prior to T-195 minutes.
4. Payload Characteristics. Since the SS is larger and more complex than the PM, the likelihood of launch delays appears greater. Such delays would be undesirable if the PM were already orbiting.
5. Backup launch capability for the PM can be more readily furnished than can backup launch capability for the SS (follows from items 2 and 3 above).
6. A propellant savings can be effected by launching the SS/Saturn V first - estimated at 2000 pounds equivalent payload.

MISSION PROFILE (LSC LAUNCH)

LSC CHECKOUT

DOCK AND CREW TRANSFER

LOGISTICS
SPACECRAFT
(MANNED)

BLANK

MISSION PROFILE (RECOVERY)

SHUT DOWN PM - SEPARATE R/E SYSTEM

SHUT DOWN SS/SEPARATE LSC

CIR

AIR RECOVERY

SPLASH DOWN
AND RECOVERY

Combining launch sequence 1 and the preferred automatic docking mode, the tentative selection of mission profile is shown diagrammatically in the facing figure. In this profile, the first launch is that of the SS. Following checkout of the SS by information telemetered to ground stations, the PM launch is effected at $T + 2$ days (T being SS liftoff). The PM is placed in an initial rendezvous position with the SS and is started up by ground station command. PM performance is monitored via telemetry for three days: at the end of this period the transtage is jettisoned and the PM is docked to the SS in a ground command initiated automated sequence. When the PM-SS dock has been verified, the LSC launch will be executed at $T + 5\frac{1}{2}$.

Following the docking of the LSC to the SS, crew transfer, and initial checkout of the SS by the crew, all PM electrical and fluid connections to the SS are made up by the crew. The PM is then given a final check out before being placed "on the line" at $T + 7\frac{1}{2}$.

Alternate mission profiles are shown in Sequences 2, 3, 4, and 5. Note that the Sequence #2 mission profile could be employed if automated docking equipment and procedures are not developed as a part of the U. S. space effort. Although each of the mission profiles examined has its particular set of advantages and disadvantages, all appear capable of execution.

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

MISSION PROFILE BLOCK DIAGRAM (SEQUENCE NO. 1)

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

MISSION PROFILE BLOCK DIAGRAM (SEQUENCE NO. 2)

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

MISSION PROFILE BLOCK DIAGRAM (SEQUENCE NO. 3)

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

MISSION PROFILE BLOCK DIAGRAM (SEQUENCE NO.4)

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

MISSION PROFILE BLOCK DIAGRAM (SEQUENCE NO. 5)

One of the first tasks of the study was to evaluate the heat source design which resulted from a study by Oak Ridge National Laboratory (ORNL) for NASA-Lewis*.

Past experience has shown that the evaluation and design of an isotopic power source is based primarily on the satisfaction of safety criteria, rather than operational considerations. The accompanying figure summarizes typical mission phases and the possible credible accident or failure modes which might occur. An evaluation of the Oak Ridge National Laboratory heat source design is based upon the ability to fully contain the PuO_2 fuel under such accident or system failure conditions. The major results of this investigation are shown on the following pages.

*See "Brayton-Cycle Radioisotope Heat-Source Design Study Phase II (Preliminary Design) Report", ORNL-TM-1829

FLOW CHART - ACCIDENT MODES

The chart opposite outlines the Oak Ridge fuel capsule configuration. 156 capsules are required to make up a heat source. The capsule design is intended to fully contain the fuel and generated helium for a period of ten half-lives or more under all credible conditions. The maximum design surface temperature (steady state) of the fuel capsules at any time during the mission is not to exceed 2000°F (2500°F in the case of transients).

The inner layer of the capsule is an annular tungsten cup that holds the fuel. This is contained within a stress-bearing envelope of tantalum-base alloy T-111, which, in turn, is clad in a corrosion protection shell of platinum. A diffusion barrier layer of thoria is used between the platinum and T-111.

The length-to-diameter ratio (L/D) of 4 was chosen primarily on consideration of impact survival. The design of the T-111 primary containment capsule was based on the criterion that creep would not exceed 1% in 5 years.

ORNL FUEL CAPSULE

ORNL-DWG 66-12488R

The components that make up the ORNL heat source system include:

1. the re-entry protective shell
2. fuel plate and fuel capsules
3. re-entry body support ring
4. heat exchanger assembly
5. main support structure
6. LiH shield

A layout of the re-entry shell, fuel plate and capsules is shown in the accompanying chart. The fuel capsules are contained in Cb-1% Zr tubes which are attached to the surface of the fuel plate. The fuel plate assembly is made up of two Cb-1% Zr disks approximately 50 inches in diameter. Rapid temperature excursions of the fuel capsules are prevented by the heat storage capacity of pressed beryllium oxide contained between discontinuous radial ribs that separate the two fuel plate disks.

Heat is transmitted, principally by radiation, from the surface of the capsules to the tubular holders and then, to the working fluid heat exchanger. The disk diameter of the heat exchanger is the same as that of the fuel plate and is located 5 inches from the tubular holders.

Linde superinsulation, a multifoil type of insulation, surrounds the heat exchanger fuel block arrangement. Calculated heat losses are less than 5%.

The re-entry shell is a 60° sphere cone with a 15-inch nose. The ablation heat protection material has a uniform thickness of 2.66 inches and is bonded to a 1-inch thick honeycomb aluminum substructure. The chosen design configuration yields a hypersonic ballistic coefficient ($W/C_D A$) of 40 psf and a terminal $W/C_D A$ of 78 psf. The latter limits impact velocities on the order of 250 fps.

ORNL HEAT SOURCE

ORNL-DWG 66-11770R

Submersion in deep water of the ORNL fuel capsule was evaluated for both rupture and buckling. A summary of these analyses is shown in the accompanying chart. The thick (0.175) T-111 shell eliminates the possibility of failure due to water submersion as indicated by the required pressure depths.

SUBMERSION OF THE ORNL CAPSULE

CAPSULE RUPTURE

@ 130°F, CAPSULE RUPTURE STRENGTH $\approx 143,000 \text{ #/IN}^2$
DEPTH REQUIRED FOR CAPSULE RUPTURE = 72,000 FT.

CAPSULE BUCKLING

DEPTH REQUIRED FOR CAPSULE BUCKLING = 170,000 FT.
MAXIMUM OCEAN DEPTH $\sim 35,000 \text{ FT.}$

CONCLUSION: SUBMERSION IS NOT A PROBLEM

The ORNL capsule survival capability was evaluated for an earth burial situation. The chart indicates the predicted capsule surface temperature for two common earth-borne materials as a function of burial depth-to-capsule diameter parameter. In the lowest conductivity material (Dry Fused Silica Sand) to be commonly found, the surface temperature never approaches the 2000°F limit.

It should be pointed out, however, that burial of an entire heat source including the re-entry vehicle results in overtemperature conditions for most burial situations. Capsule surface temperatures of approximately 1000°F would be reached only in the case where the re-entry vehicle lands nose down and the capsules are fully exposed to convective currents. Should the re-entry vehicle land inverted i.e., capsules buried, surface temperatures of 3000°F or greater would be reached.

INDIVIDUAL FUEL CAPSULE BURIAL

The Titan IIIC launch vehicle consists of hypergolic core stages and two strap-on solid propellant booster stages. The response of an individual fuel capsule to a fireball resulting from a launch explosion has been evaluated for each stage separately since the resulting fireball from the solid stage represents a much worse environment, albeit at a lower probability of occurrence.

The incident heat flux arising from the Titan IIIC liquid propellants decreases from a maximum of $400 \text{ btu-sec}^{-1}\text{-ft}^{-2}$, at time zero, to $13 \text{ btu-sec}^{-1}\text{-ft}^{-2}$ over a period of six seconds. The $13 \text{ btu-sec}^{-1}\text{-ft}^{-2}$ level characterizes the afterfire which exists for a period of about 30 minutes.

The chart shows the temperature characteristics of the ORNL capsule immersed in a Titan IIIB fireball.* Initial capsule temperature of 1400°F is a conservative indication of the pre-launch cooling environment. Although the platinum temperature rises above the 2000°F limit, it does not reach the melting point of 3215°F . The tungsten inner liner reaches 1980°F as a result of the residual 1827°F fire.

*The Titan IIIB includes the Titan stages 1 & 2 and the Agena. The Agena propellant weight is approximately equivalent to that of the Transtage.

CAPSULE RESPONSE TO FIREBALL

The response of the heat source/re-entry vehicle assembly to a Titan IIIB (hypergolic core only) fireball is shown in the opposite chart. The capsule tubular holder (Cb-1%Zr) reaches a maximum temperature of 3250°F while the front portion of the capsules sees a 2200°F temperature for approximately 4 seconds. This transient above the 2000°F limit would be allowable according to the ground rules.

HEAT SOURCE RESPONSE TO FIREBALL

- ORNL HEAT SOURCE
- TITAN III B LAUNCH VEHICLE

Submersion of the heat source assembly or individual fuel capsules in the fireball resulting from an explosion and combustion of the total propellant aboard the Titan IIIC constitutes the most severe thermal environment associated with launch abort accidents.

Neither the heat source nor the individual fuel capsules will survive direct exposure to burning solid propellant; an environment characterized by a constant heat flux of $580 \text{ BTU-sec}^{-1}\text{-ft}^{-2}$ for three minutes. This time period assumes stage zero thrust termination or accident induced grain casing failure and resulting exposure of the burning solid propellant grain to atmospheric pressure, causing its burning time to increase from a normal 110 seconds to 180 seconds.

The probability of heat source assembly or individual fuel capsule exposure to direct contact with the burning solid propellant is very low as illustrated on the facing table. The probability that a launch abort of the Titan IIIC will result in destruction of the strap-on zero stage solids, leading to large grain segments of burning solid propellants on the ground is 0.0007. The probability that the heat source assembly or individual fuel capsules will be in contact with the burning solid propellant is obtained by reducing the basic accident probability by the probability that the heat source assembly or an individual fuel capsule will be located on or near to a burning solid propellant grain segment. This is a combination of the probability that both the malfunction detection system and the abort system do not work, and the probability that the heat source assembly falls on the burning solid.

RESPONSE TO FIREBALL

SOLID ROCKET BURN

TEMPERATURE

5450°F

DURATION

180 SEC.

PROBABILITY OF OCCURRENCE

1. PROBABILITY OF SOLID MALFUNCTION LEADING TO SOLID PROPELLANT FIRE*	0.0007
2. PROBABILITY OF MALFUNCTION DETECTION NOT WORKING	0.0001
3. PROBABILITY OF ABORT SYSTEM NOT WORKING (I.E. NO SEPARATION)	0.0001
4. PROBABILITY THAT THE HEAT SOURCE ASSEMBLY CONTACTS THE BURNING SOLID PROPELLANT	0.50 (EST)

PROBABILITY OF OCCURRENCE $P(4)P(1) [P(2) + P(3)] = 7 \times 10^{-8}$

* TAKEN AT 10% OF THE PROBABILITY OF LAND IMPACT DUE TO SOLID PROPELLANT (STAGE 0) MALFUNCTION

Re-entry survival of a bare, unprotected ORNL fuel capsule has been investigated. The top chart on the opposite page shows the stagnation heat flux history resulting from an earth orbital decay re-entry trajectory. The two capsule attitudes of interest include the side-on (maximum projected frontal area) and the random tumble and spin case.

The temperature response of the ORNL capsule to the re-entry heating pulse is shown in the lower left hand chart for an assumed side-on re-entry attitude. After 1530 seconds of flight, at an altitude of approximately 281,000 feet, the platinum cladding begins to melt with subsequent failure of the capsule.

Capsule temperature history for a random tumble re-entry is shown in the lower right hand chart. Here again the platinum coating melts leading to failure.

RE-ENTRY HEATING

Three generic modes in which fuel capsules can experience impact are illustrated on the opposite chart. In the case where the capsule separates from the heat source assembly, terminal velocities up to 550 fps can be expected. From past experience this has been shown to be beyond the survival capability of the capsule. In the second case, the orientation of the plate will significantly affect the terminal velocities. These could range from 240 fps for a plate perpendicular to the flow to 900 fps for a plate aligned to the flow. The third case illustrates the condition of integral fuel capsule/re-entry vehicle impact. Impact velocities can be limited to approximately 250 fps based upon the re-entry vehicle maintaining a small angle of attack.

MODES IN WHICH FUEL CAPSULES OF CAN EXPERIENCE IMPACT

Past experience on the SNAP-27 fuel capsule program indicates that the capsules have a good chance of surviving impact if the plastic strain energy is equal to, or greater than, the kinetic energy of the impacting system. Tests have shown that failure by rupture occurs when the opposite is true. A preliminary evaluation was conducted to determine the inherent energy capacities of the various impacting system configurations.

On the chart are shown (1) the available plastic strain energy assuming no deformation of the capsules, (2) the strain energy assuming the fuel capsules deform, and (3) the kinetic energy at impact.

Only in the case of the integral impacting system is there sufficient strain energy to absorb impacting loads. It is clear that the assembly must be designed to preclude the occurrence of modes 1 or 2.

IMPACT

	BARE CAPSULE 1	COMPLETE HEAT SOURCE PANEL 2	INTACT RE-ENTRY VEHICLE 3
PLASTIC STRAIN ENERGY AVAILABLE WITH NO DEFORMATION OF CAPSULE (FT-LB)	0	.935 x 10 ⁶	2.04 x 10 ⁶
PLASTIC STRAIN ENERGY AVAILABLE WITH DEFORMATION OF CAPSULE (FT-LB)	.37 x 10 ⁴	1.81 x 10 ⁶	3.85 x 10 ⁶
KINETIC ENERGY OF IMPACTING SYSTEM (FT-LB)	1.9 x 10 ⁴	20.86 x 10 ⁶	1.82 x 10 ⁶

Corrosion of the T-111 alloy, leading ultimately to fuel capsule failure (possibly explosive failure) and subsequent fuel release, can occur if the platinum oxidation protection coating is penetrated. Penetration of the platinum coating by shrapnel arising from a launch vehicle explosion is identified as one of the more probable modes, requiring that a blast shield be located between the heat source assembly and the launch vehicle.

Corrosion of the fuel capsule could also be accomplished by liquid plutonium metal which may be formed by reduction of the initial PuO_2 fuel form as a result of oxygen gas pressure loss from within the fuel capsule. Fuel capsule failure and the subsequent release of the more toxic, higher volatility plutonium metal is more hazardous than the release of the PuO_2 fuel form. The reduction of PuO_2 to plutonium metal plus some plutonium sesquioxide in a sealed tantalum crucible has been experimentally verified at 1900°C^* . At this temperature, the free metal is produced in approximately 25 hours due to oxygen diffusion through the wall of the tantalum crucible. It is expected that this reaction would also occur at 1080°C (2000°F) although the time required to produce the free plutonium metal at these lower temperatures remains to be defined. A change in fuel capsule material to inhibit the loss of oxygen by either diffusion or oxidation of the fuel capsule materials would prevent the formation of plutonium metal. For example, at 1200°C in the presence of oxygen, PuO_2 reduces to PuO_{2-x} , where x varies from 0.4 to 0.6, but further reduction of the oxide is prevented by continuous re-oxidation to the higher state in the presence of oxygen. Considerable long term, high temperature testing will be required to fully evaluate the stability of PuO_2 with the various refractory metal fuel capsule materials.

*OHSE, R. W. and Ciani, V., "Evaporation Behavior and High Temperature Thermal Analysis of Substoichiometric Plutonium Oxide in the Composition Range from $\text{O/Pu} = 1.51$ to 2.00 ", European Institute for Transuranium Elements, Karlsruhe, Germany, September 1967.

CORROSION OF T111 CAN OCCUR IF THE Pt COATING IS PENETRATED

POSSIBLE METHODS OF PENETRATION ARE:

SHRAPNEL

DEFORMATION OF FUEL CAPSULE SUPPORT STRUCTURE

STRESS CORROSION RESULTING FROM SEA WATER IMMERSION

SUB-STOICHIOMETRIC PuO_2 WOULD BE EXPECTED TO BE PRODUCED IF OXYGEN

IS LOST FROM WITHIN THE FUEL CAPSULE

IDENTIFIED OXYGEN LOSS MODES

FUEL CAPSULE MATERIAL OXIDATION

OXYGEN DIFFUSION THROUGH TANTALUM

CONSIDERABLE TESTING WOULD BE REQUIRED TO DETERMINE LONG-TERM

STABILITY AT ELEVATED TEMPERATURES

The re-entry protection shell must serve two functions since the bare capsule cannot survive re-entry heating nor is it likely to survive impact. Past experience has shown that attempts to provide individual re-entry protection for fuel capsules operating at 2500°F and below have not been successful. Higher operating temperatures allow greater heat rejection and consequently the opportunity to design a capsule with re-entry heat protection material encapsulating it. It has been shown that even if all of the plastic strain energy of the capsule were available for energy absorption, survival of the bare capsule upon impact would be unlikely.

With the extremely low thermal conductivity of the Linde superinsulation in air, the chances of capsule survival are remote if the capsule comes to rest upside down. If the multifoil insulation were allowed to rapidly oxidize or otherwise disintegrate then perhaps capsule temperatures could be limited to 2000°F.

If the platinum cladding is lost for any reason, the T-111 capsule would be subject to rapid oxidation. At 2200°F tantalum oxidizes at the rate of approximately 20 mils/hr, which leaves only hours of safe containment.

The possibility of reduction of the initial PuO₂ yielding liquid plutonium metal is a corrosion problem which may lead to fuel capsule failure and the subsequent release of the toxic, highly volatile plutonium metal.

It might be pointed out that the ORNL heat source heat exchanger is not designed as a replaceable unit nor was there a requirement to do so as in the present study. However, it has been found to be an attractive feature and should be considered a design goal.

One feature of the ORNL heat source design which appears to be unattractive is the method of tying in the central bolt and fuel plate to the re-entry shell. The large central bolt "floats" in a cavity within the superinsulation. Although this tends to minimize the problem of thermal leakage it forces all loads through the insulation.

R/V MUST REMAIN INTACT FOR FUEL CAPSULE TO SURVIVE

- 1) RE-ENTRY
- 2) IMPACT (MAY BE MARGINAL)

FUEL CAPSULE WILL MELT IF THE R/V COMES TO REST UPSIDE DOWN

CORROSION OF T-111 CAPSULE CAN OCCUR RAPIDLY IF 20 MIL PT COATING IS
PENETRATED

POSSIBLE REDUCTION OF PuO_2 TO Pu

HSX CAN NOT BE REPLACED

LOAD TRANSMISSION THROUGH SUPER-INSULATION IS NOT DESIRABLE

Evaluation of the response of the ORNL fuel capsule and heat source design provided a set of basic guidelines for the heat source design in the PM study. These guidelines were reviewed with the NASA-MSD and became the ground rules for the design for the rest of the study. These were very important since the design of the heat source and its attendant safety problems form the heart of the power module design.

The ORNL fuel capsule was chosen as the basic element of the heat source design by NASA-MSD so that the results of this study could be evaluated with the results of a study being conducted by NASA-Lewis on an Isotope Re-entry Vehicle and with the results of a fuel capsule contract being conducted for the AEC. The selection of this fuel capsule and the results previously presented defined the second and third requirements.

It was deemed prudent to minimize the probability of an accident with the radio-isotope by providing controlled re-entry and recovery for the re-entry vehicle. Also, recovery of the multi-million dollar heat source and the possibility of reuse of the source provides an economic incentive.

The need to provide for survival if the R/V should come to rest upside down is based again on the study of the ORNL heat source in which this was defined to be a relatively probable failure mode.

Inclusion of a maximum amount of crush-up material is necessary in order to minimize the probability of deforming the ORNL fuel capsule thereby endangering its ability to withstand creep stresses for an extended period of time, i.e., many half-lives of the Pu-238 fuel.

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

GROUND RULES

ORNL FUEL CAPSULE

RETAIN WITHIN R/V

CIR PACKAGE & RECOVERY EQUIPMENT

DESIGN TO SURVIVE EVEN IF CIR & RECOVERY EQUIPMENT DON'T WORK

BACKFACE PROTECTION

SHOCK ABSORBING

ORBITAL DECAY

SUBMERSION

The seven design conditions used in the evaluation of the reference heat source are shown in the accompanying figure. Nominal deboost would be provided by the retro rocket in the CIR package which gives a nominal ΔV of 750 ft/sec after the re-entry system has been oriented in the proper direction and spun up for stabilization. The 750 ft/sec gives an input dispersion of approximately 110 n.m.

The decay trajectory might result from malfunction of the CIR and is representative of decay from the nominal 260 n.m. orbit. Decay from such an orbit would require several years for the re-entry system alone. Such a decay could theoretically cause either a forward or a backward re-entry.

The other trajectories would only occur if the transtage guidance system incorrectly oriented the PM and then fired such that all of the remaining fuel were utilized to propel the system earthward. The two reference conditions shown are those for the maximum velocity addition which would yield the maximum soak time and for the maximum pitch down and resulting re-entry angle which would yield the maximum heating rate. The probability of these occurrences is very small and has been estimated to be of the order of 3×10^{-12} .

FLIGHT MECHANICS

	<u>V</u>	<u>γ</u>	<u>REMARKS</u>
NORMAL	25,730	3.2	
DECAY	25,710	0	
ΔV OF 7000 FT/SEC	24,610	16	MAX $\gamma, \beta = 70^\circ$
ΔV OF 7000 FT/SEC	31,700	5	MAX $U_\infty, \beta = 140^\circ$

DECAY AND BOOSTER MALFUNCTION COULD LEAD TO EITHER FORWARD OR BACKWARD RE-ENTRY.

The heat source assembly is designed to hold up to 162 ORNL fuel capsules in a flat array across the back of the sphere-cone re-entry vehicle. In order to protect the capsules if the R/V re-enters backward, they are mounted behind a refractory metal re-entry shield. This metal shield is one of two Columbium alloy ribbed disks which are welded together to form a sandwich panel. The fuel capsules are set in trays or boats and inserted into the panel from the side. The side is then closed with an angle ring to hold the capsule boats in place. The boats are presently indicated as made of BeO but substitution of ZrO, Al₂O₃ or a metal foam is being considered to avoid the toxicity problems associated with handling and cutting BeO.

The sandwich panel or heat source plate is attached to its R/V mounting ring by eight hollow pins. The panel and its mountings are designed for the worst case 42.9 g re-entry loads. The eight pins are aligned on radii to accommodate thermal expansion and contraction of the heat source plate relative to its mounting.

The mounting ring also holds the multifoil insulation blanket which runs across the back of the heat source plate. This insulation is made of molybdenum metal so that it will oxidize away after the heat source returns to earth and thereby help keep fuel capsule temperatures below safe limits, especially if the R/V comes to rest backface down on the ground.

The entire heat source assembly has been designed to permit check fit assembly and disassembly without the use of live fuel capsules and to minimize the complexity and length of operations required for loading the live fuel capsules.

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

HEAT SOURCE ASSEMBLY

A summary of the re-entry heating environments and resulting R/V performance is tabulated on the facing chart.

Note that maximum heat shield recession is 0.105 in. of a total shield of 0.75 in. of three dimensional graphite fibrous material.

The maximum temperatures shown for backward re-entry are those of the aft columbium alloy plate. The plate does not exceed the allowable fuel capsule transient temperature of 2500°F for the orbital decay case, however it does exceed the allowable temperature for the cases of maximum re-entry path angle and maximum free stream velocity with the 7000 ft/sec addition of ΔV .

RE-ENTRY HEATING

TRAJECTORY	$\dot{q}_C$ MAX.* (BTU/SEC-FT ²)	SHIELD THICKNESS (IN)	SHIELD MAX. TEMP.		MAX. ΔT ACROSS SHIELD (°R)	SHIELD TOTAL RECESSION (IN)
			FRONT FACE (°R)	BACK FACE (°R)		
ORBITAL DEBOOST	120	.75	3961	752	3384	.075
ORBITAL DECAY	76	.75	3540	1296	2531	.105
MAX. γ_e WITH AV = 7000 FPS	255	.75	4684	561	4174	.036
MAX. U_∞ WITH AV = 7000 FPS	212	.75	4621	667	4102	.088

* STAGNATION POINT HEATING FOR $R_N = 19.824$ IN.

BACKWARD RE-ENTRY

TRAJECTORY	$\dot{q}_C$ MAX.** (BTU/SEC-FT ²)	AFT COVER THICKNESS (IN)	AFT COVER MAX. TEMPERATURE (°R)
ORBITAL DECAY	30	.125	2915
		.06	2922
MAX. γ_e WITH $\Delta V = 7000$ FPS	99	.125	3618
		.06	3857
MAX. U_∞ WITH $\Delta V = 7000$ FPS	85	.125	3802
		.06	3812

** HEATING AT $S/S_c = .8$ LOCATION ON AFT COVER

The shield backface temperature is shown as a function of the thickness of 3DGFM in the accompanying chart. The structural shell of the shield is made of RENE-41 and would not be expected to be adversely affected by any temperature less than 2000°F.

However, high temperatures such as those shown for the decay orbit with small thicknesses of 3DGFM impose a stringent requirement on any crushup material which must be designed to absorb energy shortly after the maximum temperature was achieved and yet must be designed to vaporize or in other manner remove itself should the R/V land upside down on the earth after a forward re-entry.

The current design has employed aluminum crushup but development of a resin system which could survive decay re-entry and then would vaporize should the R/V land upside down would be quite desirable. Tests of the aluminum crushup and development of the resin system would be necessary for the development of a heat source with this fuel capsule.

MAXIMUM BACKFACE TEMPERATURE

MAX BACKFACE TEMPERATURE - 3DGFM SHIELD STAGNATION POINT

The temperature response of the columbium plate is shown to be a very rapid response in which the aft plate exceeds 2500°F for less than 20 seconds. The impulse in fact is so brief that the fuel capsule temperature never exceeds 2500°F for this case. We may therefore conclude that the design is adequate for this condition.

BACKWARD RE-ENTRY TEMPERATURE

As in the previous plot, the response for even the maximum velocity small path-angle case is quite sharp. The period during which the aft plate exceeds 2500°F is less than 100 seconds.

In this case, the outer surface of the fuel capsule does exceed 2500°F for less than one minute but the temperature is considerably below the platinum melt temperature, and therefore does not seem to be sufficient reason for a design change when the likelihood of this condition is investigated.

BACKWARD RE-ENTRY TEMPERATURE

MAX. U_{∞} WITH $\Delta V = 7000$ FPS

BACK PLATE THICKNESS = .125" Cb

$S/S_c = .8$

One possible adverse environment to which the fuel capsule might be subjected occurs when the R/V comes to rest upside down on the earth's surface. Such an orientation would be quite stable and provision must be made for protection of the fuel capsule.

This figure illustrates the temperature response of the fuel capsule if the R/V were upside down, the crushup material had vaporized away, the molybdenum superinsulation was intact and infiltrated by air and the 3DGFM heat shield was intact. It assumes that insufficient oxygen is available to oxidize either the Mo superinsulation or the 3DGFM.

The steady state temperature of the fuel capsule is slightly less than 2000°F. Since complete oxygen starvation is unlikely, this analysis provides one conservative approach for evaluating the upside down condition.

HEAT SOURCE TEMPERATURE

- NO MOLY OR GRAPHITE REACTION HEAT (OXYGEN STARVATION)
- CRUSH-UP DEPLETED
- $k_{\text{EARTH}} = 0.5 \text{ BTU/HR FT } ^\circ\text{F}$
- $(Wc_p)_{\text{HEAT SOURCE}} = 100 \text{ BTU/}^\circ\text{F}$
- 25 Kw GENERATED IN HEAT SOURCE
- IMPACT AFTER FORWARD RE-ENTRY

Another limiting condition would be that in which complete molybdenum oxidation occurs but oxidation of the 3DGFM has not perceptibly begun, its oxidation process being much slower. Oxidation of Mo is an exothermic reaction, and it is therefore of interest to determine the significance of the thermal addition of this reaction.

In these calculations a conservative assumption was made that the air which feeds the oxidation does not extract any heat through a $WC_p \Delta T$ type of convective cooling.

Even with this assumption, the temperature of the fuel capsules does not exceed 2500°F and the nominal 2000°F limit on the capsule is exceeded for less than one-half hour.

HEAT SOURCE TEMPERATURE

A heat generation reaction rate was developed for the case of twice the required air for stoichiometric combustion of molybdenum.

This information was then used to calculate the temperature increase resulting from the exothermic Mo reaction, taking into account the convective cooling resulting from heating the air from 100°F to the reaction temperature. A conservative initial temperature of 2300°F was chosen to compensate for any possible inconsistencies in the steady state fuel capsule temperatures prior to initiation of the reaction.

Note that the temperature increase is now less than 100°F and occurs in approximately 5 minutes.

HEAD SOURCE TEMPERATURE

RATE OF REACTION &
HEAT GENERATION RATE

HEAT SOURCE PLATE
TEMPERATURE VS. TIME

This figure shows that should the heat source become buried the fuel capsules would melt. The only apparent solution to this problem would be to use a fuel capsule which has a melting temperature greater than that of soil; i.e., greater than $\sim 2800^{\circ}\text{F}$.

HEAT SOURCE TEMPERATURE

Aerodynamic investigations have defined the maximum dynamic loads to be encountered by the R/V during re-entry. The peak axial load occurs at the steepest re-entry path angle for forward re-entry and yields a load of approximately 43 g's. The re-entry vehicle was conservatively designed to handle the 43g load even though such a loading has previously been shown to be very unlikely.

MAX. DYNAMIC LOADS DURING RE-ENTRY

<u>ATTITUDE</u>	<u>TYPE OF RE-ENTRY</u>	<u>SECONDS FROM 400 KFT</u>	<u>G_x</u>	<u>q_{∞} PSF</u>
FORWARD	DECAY	4346	7.36	246.5
	DE-BOOST	166*	8.61	286.5
	7000 FPS, MAX. V	146	8.89	297.4
	7000 FPS, MAX. γ	43	42.9	1434.0
BACKWARD	DECAY	4152	7.31	203.1
	7000 FPS, MAX. V	141	9.20	255.4
	7000 FPS, MAX. γ	43	42.0	1167.0

*TIME FROM 350 KFT

The figure shows 23 of the possible vehicle loading conditions. The apparent critical conditions may be summarized as launch abort, burnout and axial vibration, burnout and lateral vibration, normal re-entry, re-entry with a 7000 fps ΔV , water impact and land impact.

The launch abort condition defines the acceptable loads to be applied by the abort rockets in getting the re-entry system away from the rest of the power module.

The Rene 41 liner was sized based on the 14.7 psi pressure differential created on the conical frustum and 30.6 psi pressure differential at the nose created by the 43g re-entry axial loads of condition N. The 0.10 inch thick plate has a 26 percent margin of safety at 800°F.

Overpressure resulting from an explosion of the transtage with 10 percent yield requires the addition of ring stiffeners to break the single bay of the frustum into approximately 10 smaller bays. This assumes that preservation of the integrity of the liner is required to survive free fall after the pad abort overpressures are encountered.

The columbium FS-85 back plate was sized to withstand the 43g loads of backward entry condition N. Bending stresses in the plate outer fibers were defined as the critical mode of failure. Transient stress analysis of backward re-entry is required to verify the adequacy of the plate thickness.

The truss that attaches the CIR package to the vehicle supports was found to be critical for and was sized for the combined conditions F and I. Buckling analysis shows that a combined margin of safety of 73 percent was available with .028 inch wall 7/8 inch diameter aluminum tubing.

An aluminum honeycomb was sized to absorb sufficient energy for the fuel capsules to not be deformed by impact nose down. This was accomplished by holding stress levels below the elastic limit of the capsule materials.

DESIGN LIMIT LOADS ON SLPM - TITAN IIIC

LTR.	LOAD CONDITION DESCRIPTION	LIMIT LOAD FACTORS	
		Gx	Gn
A	GROUND HANDLING	3	3
B	LAUNCH ABORT	20	20
	MISCELLANEOUS:		
C	MARMAN CLAMP	LATER	
D	BOLT PRELOAD	LATER	
E	ZERO STAGE BURNOUT	3.9*	2**
F	FIRST STAGE BURNOUT	5.2*	2**
G	SECOND STAGE BURNOUT	4.0*	2**
H	AXIAL VIBRATION	10*	-
I	LATERAL VIBRATION	-	10*
J	DOCKING	NEGLIGIBLE	
	SEPARATIONS		
K	R/S TO PM	< 1g	4g
L	DE-BOOST	1g	
M	NORMAL RE-ENTRY	10	3**
N	RE-ENTRY WITH 7000 FPS Δ VEL.	43	4**
O	NORMAL RE-ENTRY, G _N MAX.	8	5**
P	FIRE PARACHUTE EJECTION PISTONS	N/A	N/A
Q	PARACHUTE DEPLOYMENT	6.2	2**
R	AIR SNATCH	25	4**
S	WATER IMPACT	35	5**
T	LAND IMPACT (FREE FALL)	800	0
U	SUPERORBITAL RE-ENTRY	LATER	LATER
V	COMPONENT MOUNT VIBRATION	30	30
W	SPIN VIA ROCKETS ON CIR		1/2

* THE MARTIN COMPANY DOCUMENT NO. SSD-CR-65-18 (REV. 2) TITAN III A & C STD. SPACE LAUNCH SYSTEM (SSLS) DEFINITION FOR PAYLOAD CONTRACTORS, JUNE 1966

** ESTIMATED LATERAL LOADS

The re-entry system (RS) is composed of the controlled intact re-entry (CIR) package and the re-entry vehicle (RV) plus the associated trusses, fasteners, separation devices and abort mechanisms required to separate the RS from the power module and the CIR pack from the RV.

The structural components of the re-entry system have been discussed in the thermal and dynamic sections and will not be enumerated again here. The separation of the RS from the PM will be by a V band with hot wire tension links. The RS will be ejected from the PM by two small (140 lb-sec) rockets. They will yield a separation velocity of 3.5 ft/sec.

The re-entry vehicle contains a parachute, flotation balloon, including beacon transmitter, antennas and flashing lights, gas bottles for flotation balloon inflation, a drogue chute, "g" switch for activation and a programmer for decoding signals from the "g" switch and the command system. The electrical power for these components is supplied by two automatically activated Ag Zn batteries which are connected in a redundant mode. If both are operational, they will provide power to the components for 72 hours should the RV not be recovered in the air but the backup flotation system was operational.

The air recovery system, the TT&C subsystem and the CIR package are discussed on following pages.

RE-ENTRY SYSTEM ASSEMBLY

The controlled intact re-entry package includes the attitude control subsystem for sensing the horizon and orienting the RS to the proper re-entry path angle prior to retro, the tracking, telemetry and command subsystem, the propulsion subsystem, the electrical power and distribution subsystem and the separation subsystem.

The attitude control subsystem is composed of an infrared horizon sensor, a multi-axis rate gyro, a magnetometer with programmer and the spin, despin and retro propulsion systems to de-orbit the RS upon receipt of the proper command. The attitude control system utilized is a cold gas system which has been based on the system developed for the Biosatellite program.

The propulsion system is composed of two spin and despin rockets mounted on the circumference of the CIR plus a retro rocket. The spin rockets are MARC-7D4 rockets which have been flight qualified for the Mark 6 RV program and provide a total impulse of 489 lb-sec with a burn time of 1.035 sec. The retro rocket, an Aerojet SVM-1, provides a ΔV of 750 ft/sec with a total impulse of 47,500 lb-sec.

The electrical power and distribution system for the CIR includes a silver oxide-cadmium rechargeable battery and an automatically activated silver oxide-zinc battery. The Ag-Cd battery will be used for readiness checks of the CIR and will be recharged from the PM electrical power supply.

The Ag-Zn battery will be activated prior to the de-orbit mode. Either battery is capable of meeting the total power requirements of the CIR.

After the CIR system has performed its de-orbiting functions, it will be separated from the RV prior to re-entry by a lanyard actuated pin activating hot wire tension links. The ejection will be accomplished by helical coil springs, one located at each of the tie down struts. These are designed to give a separation velocity of 3.5 ft/sec.

CONTROLLED INTACT RE-ENTRY SYSTEM

The requirements for the CIR Telemetry, Tracking and Command (TT&C) subsystem are:

1. To provide status information and command capability for the power module from launch until the PM-CIR separate.
2. To provide status and event monitoring, command and sequence control, and tracking during the PM-CIR separation, de-orbit, and CIR-RV separation phases.

The components included in the subsystem, except for the antennas are shown in the accompanying diagram. The three functions have been divided such that each function is supplied by a separate and distinct RF carrier. The command function is provided by a 400 MHz command receiver followed by a command decoder. The command decoder provides real time command capability, while the programmer supplies the stored commands required for the de-orbit sequence.

The telemetry system utilizes an S-Band transmitter. The sensor data is processed in the signal data converter. Some information is wide-banded and sent back in analog form, the remaining data is converted to digital form in the PCM multicoder. This multicoder also combines its data with that of the remote PCM multicoder. The remote PCM multicoder is located in the Power Module (PM) in order to reduce the number of lines which are required to cross the CIR-PM interface. The analog and digital information are combined in the baseband assemblies to obtain the composite signal for transmission.

The tracking function is performed by a C-Band radar system with an active target. As shown in the figure, a C-Band transponder provides the tracking capability.

The antennas for the three systems are all located on a stub which is aligned with the roll axis of the CIR as shown in the CIR drawing.

The figure on the opposing page and the following figure describe the effects of neutron and gamma irradiation of components within the CIR package and the re-entry vehicle. The individual classes of components within the CIR and the RV have been grouped by subsystem and have been rated in terms of anticipated degradation into three categories.

The three levels are threshold, moderate, and severe. In general, a threshold level denotes that range of accumulated radiation dose where specific effects begin to occur in the piece parts and/or materials and which would likely require some consideration in component design to insure proper subsystem operation. Moderate effects on the other hand, generally denotes that range of radiation dose which would result in significant degradation of component performance and would require special design considerations for components to operate. State-of-the-art radiation hardening design practices would be adequate to design for this radiation dose, however. The severe effects levels would seriously impair component operation, and, in many cases, new design approaches would be required. Considerable efforts would be required in order to design components to operate in this region.

As can be seen, combining the component sensitivities with the anticipated two year neutron doses for the CIR Subsystem results in several components experiencing threshold level damage effects. Thus, some consideration of these effects in component design would be required. This can be readily accomplished, for example, by additional piece part de-rating and circuit operating margins. For the threshold levels indicated here, it is anticipated that this subsystem could be adequately radiation hardened without a significant power or weight penalty.

The Recovery Subsystem, on the other hand, is much closer to the radioisotope power source, and, consequently receives a significantly higher neutron dose than the CIR Subsystem. As can be seen in the figure, this results in the induced radiation effects being considerably more pronounced for the Recovery Subsystem with the most significant effects being associated with the semiconductor piece parts. Although the anticipated magnitude of these effects are such that it is well within the state-of-the-art to design the subsystem to operate properly in this environment, some increase in the power requirements for the subsystem will likely result.

NEUTRON EFFECTS SUMMARY

CIR SYSTEM

RECOVERY SUBSYSTEM

The effects of the gamma ray component of the radiation environment are minimal for the CIR subsystem, being slightly below the threshold levels in all cases. For the Recovery Subsystem, the anticipated gamma dose is such that some effect may be experienced in certain semiconductor devices. It is expected that these potential effects can be easily overcome with the proper selection of semiconductor devices.

IONIZATION EFFECTS SUMMARY

CIR SYSTEM

ELECTRONICS
FREQUENCY CONTROL CRYSTALS
TRANSMISSION LINES

ELECTRONICS
IR SCANNER, GYRO, MAGNETOMETER
PYROTECHNICS & ROCKETS
REGULATOR VALVE, & FILTER

ELECTRONICS
BATTERIES

TT&C S/S

ATTITUDE
CONTROL S/S

EP&D S/S

2 YR. RTG DOSE

RECOVERY SUBSYSTEM

ELECTRONICS
PYROTECHNICS
ORGANIC SEALS
BATTERIES
PARACHUTE & BALLOON

RECOVERY S/S

2 YR. RTG DOSE

THRESHOLD EFFECTS
MODERATE EFFECTS

10^3

10^4

10^5

10^6

GAMMA DOSE, RAD (c)

Current air recovery equipment and techniques, demonstrated on such programs as Discoverer, A-95, TE, TT and Biosatellite, provide capability for very reliable recovery of vehicles up to 300 lbs. in weight using canopies of from 12 to 36 feet in diameter. This technique is not directly amenable to a vehicle the size of the design RV since a chute sized to be compatible with a velocity of 25 ft/sec at 10,000 ft altitude becomes too large to be compatible with the available air recovery gear. Therefore, a target canopy must be provided in addition to the main canopy.

The four systems which were evaluated for the current design are shown in the accompanying figure. They were:

Tandem Parachute System - A small reinforced pickup chute is attached to the apex of the main canopy by means of an axial tow line. This configuration includes a Ringslot target canopy which trails a Ringsail main canopy.

Conical Extension System - A 10 percent Extended Skirt canopy design incorporates on its crown an extension in the form of a reinforced conical frustum constructed of solid cloth supported by the main canopy air pressure.

Modified Ringsail System - Similar to the conical extension configuration except that the conical frustum and canopy follows the sail design of a ringsail parachute instead of utilizing a solid cloth construction.

Annular Parachute System - The pickup target consists of a ringslot canopy attached to the cross vent lines of the main canopy with a multiple suspension system. This system has been selected for the SLPB base line design.

MAIN PARACHUTE CONFIGURATIONS

MODIFIED RINGSAIL
PARACHUTE SYSTEM

ANNULAR PARACHUTE SYSTEM

TANDEM PARACHUTE SYSTEM

CONICAL EXTENSION
PARACHUTE SYSTEM

One of the major reasons for the selection of the annular configuration is shown in the figure. The greater drag efficiency of this design results in a smaller weight and volume for the canopy for a given vehicle weight. The other major reason is that development tests are currently being performed with this type of canopy for use with payload weights up to 2000 lbs.

RECOVERY PARACHUTE COMPARISON

For optimum thermal efficiency the heat source and the HSHX are encased in a heavily insulated heat plenum. Provision must be made for removal of isotope heat when the PGU's are shut down. Two cooling modes are involved: alternate or standby cooling for controlled shutdown and as a backup, emergency cooling. The ORNL fuel capsule is designed for normal operation up to 2000°F and, under emergency conditions, up to 2500°F for no more than 30 minutes. The two cooling modes are therefore identified with the two temperature levels.

The alternate cooling mechanism, which maintains normal operational capsule temperatures, must be capable of repeated controlled operations since it must be used during PGU shutdown periods and turned off for PGU startup. Since the PGU's are launched in a shutdown condition, alternate cooling should be provided to avoid the necessity of turning it on during or immediately after powered flight.

The emergency cooling mechanism is the final thermal protection for the fuel capsules in the PM configuration and, as such, must be as reliable as possible.

The ground cooling mechanism must cool the heat source surface to less than 360°F to prevent ignition of booster fuel vapors which might reach it, to minimize personnel thermal hazards and to protect refractory metals used in the heat source from oxidation.

ALTERNATE COOLING

MAINTAIN FUEL CAPSULE TEMPERATURE $< 2000^{\circ}\text{F}$ WHEN PGU NOT OPERATING

RELIABLE

MAINTAINABLE

REPEATABLE USE

LAUNCH OPEN

EMERGENCY COOLING

MAINTAIN FUEL CAPSULE TEMPERATURE $< 2500^{\circ}\text{F}$ (< 30 MIN.) IF ALTERNATE COOLING FAILS

VERY RELIABLE

AUTOMATIC

INDEPENDENT OF NORMAL POWER SUPPLIES

OPERATE ONCE

GROUND COOLING

COOL HEAT SOURCE SURFACE TO $< 360^{\circ}\text{F}$

MINIMUM IMPACT ON PM DESIGN

MINIMUM IMPACT ON LAUNCH FACILITIES

In the reference design two swinging shutters are provided in the side walls of the heat plenum. With only one of the shutters open the heat leak to space is sufficient to maintain the fuel capsule temperature below 2000°F. The shutters are hinged at one end and controlled by a push-pull electromechanical actuator attached to the stiff outer wall of the shutter and an extension of the hinge base. The actuator draws power from the batteries carried on the PM and is controlled by a thermostatic switch located on the inner wall of the heat plenum. To keep the shutters open during launch and to prevent arcing at the switch contacts during powered flight vibrations, the actuator circuit would be held open by a separate disabling switch. The actuator-hinge assembly is accessible for repair or replacement.

For emergency cooling a thermal battery is mounted on the inside wall of the plenum. When fuel capsule temperature approaches 2500°F the eutectic salt electrolyte in the battery melts and provides a current pulse to fire gas generators in the shutter hinge mount, blowing the shutter and hinge assembly off the heat plenum.

HEAT SOURCE COOLING DESIGN

SWING MOTION SHUTTER

1 Swing Motion Shutter

Pro . Simple mechanism
 . Light
 . Easily removed in an emergency
 . Easily supported for open launch
 . Low power requirement for actuator

Con . Moderate heat leak penalty
 . Adds some height to heat plenum

3 Slide Motion of PGU & HX Chamber

Pro . Low heat leak penalty

Con . Requires extensible electric and
 hydraulic connections
 . High power requirement for actuator
 . Heavy actuators and supports
 . Difficult to support PGU, HSHX and
 plenum in launch environment
 . Critical fit at the insulation
 interface
 . Addition of vulnerable insulation
 shell to PGU package
 . Sliding of heat plenum across
 backface of R/V

2 Slide Motion Shutter

Pro . Relatively light
 . Relatively simple mechanism
 . Easily supported for open launch
 . Low power requirement for actuator

Con . Moderate heat leak penalty
 . Adds some height to heat plenum
 . More difficult to remove for repair

4 R/V Deployment

Pro . No heat leak penalty
 . Simple mechanism

Con . Difficult to support R/V in
 launch environment
 . Requires extensible electrical
 connections
 . High power requirement for actuator
 . Heavy actuators and supports

COOLING SYSTEMS

With the heat source and heat source heat exchangers encased in such a large insulated enclosure, heat leakage is a major problem. The table opposite gives the breakdown of the heat leaks for an 80 layer multifoil insulation system and for a 160 layer system. The heat plenum layout is such that multifoil insulation can be used only on the back disks behind the heat source and the HSHX's and on the gas ducts (pipe header) connecting the HSHX's to the PGU's. It should be noted that it is not desirable to use 160 layer multifoil on the disk behind the heat source since this layer of insulation controls the fuel capsule temperature if the R/V comes to rest backface down after it returns to earth.

HEAT LEAK SUMMARY

<u>LOCATION</u>	HEAT LEAK IN WATTS	
	<u>80 LAYER MULTIFOIL</u>	<u>160 LAYER MULTIFOIL</u>
HARD STRUCTURE (R/V)	40	40
BACK DISK (R/V)	365	182
BACK EDGE	125	125
3 DGFM (0.5 IN. WIDE)	135	135
BOTTOM DISK UNDER HSHX	377	188
SIDE OF PLENUM (NO PENETRATION)	850	850
SIDE PENETRATION	120	120
PIPE HEADER	200*	200
SHUTTERS	200	200
	<hr/>	<hr/>
	2412	2042
*160 LAYERS		

The heat, mass and pressure balance for the Brayton cycle in the reference design is shown in schematic diagram opposite. For the Power Module study the Brayton Rotating Unit (turbine-alternator-compressor), the recuperator and waste heat exchanger design and operating parameters are fixed. Design variation is allowed for the heat source heat exchanger and the radiators provided critical cycle conditions are maintained.

REFERENCE BRU CONDITIONS

The radiator area and efficiency is controlled by the DC-200 film temperature drop in the coolant tubes. The problem of reducing the DC-200 film temperature drop cannot be solved by forcing the coolant through small tubes at high Reynolds numbers since the pumping pressure drop is limited to about 25 psi by available power (about 50 watts). The problem can be solved by departing from round tube design and providing finned ducts which offer a large amount of heat transfer area. Although fluid flow is in the laminar regime the heat transfer coefficient in the ducts is kept modestly high by reducing the hydraulic diameter which can be as small as 0.020 inches (0.010 inch spacing) for this design. Small hydraulic diameters do not result in excessive pressure drops since the cross sectional flow area in this design is much greater than that provided by one or two round tubes.

Using the DC-200 finned duct geometry shown opposite, the radiator is designed as a single load-bearing shell with two separate DC-200 duct systems sharing the shell as fin area. The table lists the design data for the radiator sections.

RADIATOR DESIGN

TEMPERATURES (°F)

SINK TEMPERATURE	-5	-5
RADIATOR, EFFECTIVE	140	85
@ COOLANT INLET	254	122
@ COOLANT EXIT	61	62
COOLANT, INLET	275	127
EXIT	65	65
PRESSURE LOSS, PSI	13.7	5.9

DIMENSIONS

DIAMETER, FT.	9	9
AREA, FT. ²	415	120
FIN THICKNESS, IN.	.0625	.058
CHANNELS, NUMBER	84	84
SLOTS/CHANNEL	10	10
SLOT WIDTH x HT.	.01 x .2	.01 x .16
WEIGHT, LBS.	605	149

Isotope heat is transferred to the Brayton cycle gas via the HSHX. Two of these heat exchangers share the space opposite the heat source in the insulated heat plenum. By dividing the HSHX into four long strips and intermeshing the two units like the fingers of joined hands, both units are given identical view of the heat source area. The plane of the HSHX's and the heat source plane are separated by a 9.5 inch gap to provide a better radiative view and thereby minimize temperature gradients across the heat source surface.

As shown in the detail opposite the HSHX fingers or strips are made up of a rectangular panel section fed by a single inlet tube located beneath it. This geometry minimizes the gas film temperature drop without excessive pressure loss. The pressure loss for this design was calculated to be 3.23 percent, which meets cycle requirements.

The HSHX's, which are made of TD Nickel are supported at the manifold end by attachment to the Power Generating Unit. When two units are installed, pawls on the ends of each section engage sockets on the opposite unit for support. Sliding fit of the pawls allows thermal expansion and contraction.

HEAT SOURCE HEAT EXCHANGER

HSHX WGT. 207 LBS

HEAT SOURCE TEMP. 1750°F

GAS INLET TEMP. 1200°F

GAS OUTLET TEMP. 1600°F

HEAT EXCHANGER TOP AREA 1077 IN.²

The PGU consists of a full set of power conversion equipment, excluding the DC-200 radiator system, made up as a replaceable module. As shown opposite, the PGU carries its own HSHX cantilevered above such that it enters the heat plenum when the PGU is installed in its bay in the Power Module structure. The PGU package weighs a little more than twelve hundred pounds and carries with it gas storage and management systems as well as the voltage regulator and exciter and the speed regulator and controller. The parasitic load radiator is shown mounted on the side of the PGU structure. This component might be relocated to the outside face of the PGU to preclude thermal interaction with other gear such as the attitude control system equipment.

POWER GENERATING UNIT

The power module is a source of thermal energy for the manned spacecraft which is in addition to the electrical power being supplied. This thermal energy which is used for environmental control and life support on the manned spacecraft is of particular interest for space stations expected to be used in the mid-70's which have regenerative life support equipment aboard.

The requirements postulated for the zero "g" spacecraft were:

<u>User Subsystem</u>	<u>Heat Required (watts)</u>	<u>Temperature OF</u>
CO ₂ Concentrator	2000	250
Waste water recovery, solid waste management	100	120

The EC/LS heat is extracted from the radiator coolant in the EC/LS heat exchanger located upstream of the primary radiator. The available EC/LS heat is matched to the radiator coolant flow rate that will maintain the Brayton Cycle gas temperatures at their design point value. This will allow more than the required amount of heat to be drawn from the EC/LS heat exchanger by the spacecraft as shown in the accompanying chart.

It is probable that no separate radiator fluid flow rate control system will be required in conjunction with EC/LS heat removal because the DC-200 fluid viscosity increases with decreasing temperature. The decrease in the average DC-200 temperature in the space radiator, caused by the EC/LS heat removal, increases its viscosity and therefore decreases its flow rate. This sequence is required to maintain the gas side temperatures at the design point value. At 3.4 kw(t) EC/LS heat removal (87 percent of design point radiator flow rate), the increase in DC-200 viscosity will prevent further EC/LS heat removal unless a lower viscosity fluid is substituted for the DC-200 radiator coolant.

GENERAL ELECTRIC

MISSILE AND SPACE DIVISION

ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM

In the reference design a central structural core, shown opposite, is provided. This structure holds the PGU's, the radiation and blast shield, the heat plenum and heat source cooling shutters and the attitude control system for the PM. It is the structural link between the radiator and the Re-entry System.

The attitude control system components can be installed in the lower bays of the intermediate structure, flanking the PGU bays. In this location, the attitude control nozzles are located at the approximate center of gravity of the Power Module.

INTERMEDIATE STRUCTURE AND ATTITUDE CONTROL

PM INTERMEDIATE STRUCTURE

ATTITUDE
CONTROL SYSTEM

A blast shield is considered necessary to protect the heat source and its re-entry vehicle from the shrapnel of a booster explosion. While the blast protection analysis is underway, a shield thickness of 3/8 inch of stainless steel has been assumed. Since the radiation shield, of stainless steel canned lithium hydride, is of equivalent structural density the blast shield becomes a peripheral sheet surrounding the radiation shield.

The radiation shield was sized and its weight calculated for a single Power Module docked to the side of the Boeing Zero-G Space Station where typical separation distances between manned areas and the heat source are on the order of 30 feet. Calculations were made for two grades of isotope fuel with the following source radiation:

<u>1968 Fuel</u>	<u>Dose Rate from 25 KW @ 10 ft.</u>
neutrons	3280 Rem/yr.
decay gamma (soft)	185 Rem/yr.
Tl-208 gamma (hard)	185 Rem/yr.
<u>Fuel Made with Depleted Oxygen and Reduced in Pu-236</u>	.
neutrons	436 Rem/yr.
decay gamma (soft)	180 Rem/yr.
Tl-208 gamma (hard)	90 Rem/yr.

Shield thickness was based on a design dose rate of 25 Rem/yr (23 Rem/yr. for manned space station areas and 2 Rem/yr. for EVA). The figure opposite shows the total shield weight variation possible for the two fuels where the percentage of Li^6 in the lithium hydride is enriched over normal level. The shield weight carried in current weight estimates for the Power Module is that which corresponds to 1968 Fuel with > 70% Li^6 .

RADIATION AND BLAST SHIELDING

To meet the payload limitations of the Titan IIIC, the total Power Module weight should not exceed 9020 lbs. The table opposite indicates a margin of approximately six percent to accommodate weight increase as component design and mission requirements are developed.

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

POWER MODULE WEIGHT

PGU's (2 REQUIRED)	2438
RE-ENTRY SYSTEM	2735
HEAT PLENUM	250
RADIATION AND BLAST SHIELD	1180
RADIATOR	754
BOOSTER AND DOCKING ADAPTER	75
COOLING SYSTEM COMPONENTS	110
ATTITUDE CONTROL AND PROPULSION	200
ELECTRIC POWER AND DISTRIBUTION	180
EC/LS HEAT EXCHANGERS (2 REQUIRED)	175
SUPPORT STRUCTURE	400
	<hr/>
	8497

PGU WEIGHT

TURBOMACHINERY	141
HSHX	207
RECUPERATOR	200
WASTE HEAT EXCHANGER	160
DUCTING	35
STRUCTURE, LATCHES, DISCONNECTS, ETC.	80
INSULATION	211
GAS MANAGEMENT SYSTEM	90
VOLTAGE REGULATOR AND EXCITER	15
SIGNAL CONDITIONER	15
SPEED REGULATOR	65

1219

RE-ENTRY SYSTEM WEIGHT

HEAT PROTECTION	347
AEROSHELL AND STRUCTURE	398
HEAT SOURCE ASSEMBLY	996
RECOVERY ELECTRONICS	42
FLOTATION EQUIPMENT	27
(TOTAL R/V SUSPENDED WEIGHT)	(1810)
PARACHUTE SYSTEM	102
(TOTAL RE-ENTRY WEIGHT)	(1912)
CIR	441
SUPPORT STRUCTURE AND SEPARATION SYSTEM	118
ABORT ROCKETS	264
	<hr/>
	2735

The SLPM ground flow plan permits standard operations throughout yet minimizes interference due to nuclear radiation. All heat source assembly and installation work is performed in the warm cell while the Power Module Subassembly (PMS) is installed and tested on the Titan IIIC core in the Vertical Integration Building (VIB). The Re-entry Systems Subassembly (RSS) containing the heat source is then installed on the PMS atop the Titan IIIC launch vehicle emplaced on the pad. Launch pad operations continue through countdown and liftoff. The Integrate-Transfer-Launch (ITL) facilities of the Titans serve excellently, providing ideal conditions for both the Power Module Subassembly installation in the VIB and for the Re-entry System Subassembly installation on the pad.

Fabrication of both Subassemblies begins 125 days or more prior to launch. The PMS and the RSS are treated as two independent spacecraft until T-70, at which time the PMS and RSS are mated to form the Power Module Assembly (PMA) which is subjected to a systems test involving the use of a simulated heat source. Upon successful conclusion of this test, the PMS and RSS are de-mated and prepared for shipment to the launch site.

The PMS is required at the launch site at T-50, 25 days ahead of the RSS, which can be stored in its shipping container or subjected to additional acceptance testing just prior to shipment. The PMS, upon arrival at the launch site is inspected and tested, and on T-31 is installed on the Titan IIIC core vehicle in the VIB. Tests of the PM are then conducted along with standard launch vehicle operations.

At T-25, the RSS arrives at the launch site and is subjected to receiving inspection and system testing. Meanwhile, at any reasonable time prior to T-17, the fuel capsules arrive at the launch site to be stored in the warm cell and installed in the RVA (as described in the section on fuel handling techniques). At T-17, the strap-on Stage 0 rockets are installed, and at T-14, the launch system moves to the pad for final launch preparations. These include installing the RSS on the PMS, squib installation, flight fairing installation, joint booster-payload tests, and fueling of the liquid stages of the Titan IIIC.

ADVANCED NUCLEAR SYSTEMS OPERATION

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION

SLPM GROUND FLOW PLAN

FACTORY OPERATIONS

LAUNCH SITE OPERATIONS

KEY

RSS RE-ENTRY SYSTEM SUBASSEMBLY
PMS POWER MODULE SUBASSEMBLY
PMA POWER MODULE ASSEMBLY
V B VERTICAL INTEGRATION ON BOARDING
SLPM SEPARATE LAUNCHED POWER MODULE

* THE FOLLOWING LAUNCH COUNTDOWN HOLDS CAN BE ACCOMMODATED

• 30 DAYS 95 MIN
• 6 HOURS 45 MIN
• 1 HOUR MIN

TIME BEFORE LAUNCH SHOWN IN DAYS
EXCEPT WHEN NOTED OTHERWISE

The radiation emanating from the heat source is of sufficiently low level that all normal operations involving the installation of fuel capsules and handling of the completely assembled heat source can be accomplished by manual operations or by using relatively simple tools as an aid. The estimated dose rate is 400 Mr/hr at one meter from the edge of the source; this means that operator or technician stay time in the proximity of the source will be limited, but can be extended by use of long handled tools and/or portable shielding. The heat source must be sub-cooled by a flow of inert gas (to prevent damage to the columbium heat source plate) from the time the first fuel capsule is introduced, and this cooling to an estimated 300°F should ameliorate operator discomfort due to thermal radiation.

The fuel capsule storage and installation work will be conducted in a "warm cell" constructed at the launch site for that purpose. The cell provides suitable conditions for storage including a controlled atmosphere, effluent filtering system, and radiation monitoring equipment. It must be large enough to accommodate the Re-entry System Sub-assembly and its handling gear in addition to all other equipment such as the fuel capsule storage casks, portable shielding, tools and heat source plate parts and assemblies. The warm cell does not include shielding walls, windows and remote manipulator equipment such as are found in standard hot cells.

The fuel capsule shipping and storage cask can be designed to carry from 5 to 15 capsules and will attenuate radiation to acceptable levels as called out in appropriate ICC regulations. Cooling is accomplished passively, largely by convection. Fuel capsule temperature will not exceed 300°F. The estimated weight of a typical shipping cask is 1500 lbs.

FUEL CAPSULE SHIPPING AND STORAGE CASK

Fuel capsule handling operations begin with the removal of the capsules from the cask. The inert gas cooling is applied to the heat source plate, and the capsules are placed in beryllium oxide "boats", using a long handled tool as shown. The "boats" with the fuel capsules are then inserted into the fuel plate and locked in position by means of a circumferential locking ring. The assembly of the ring and other hardware is accomplished manually, with the technician's exposure time closely monitored and limited. The completely assembled heat source can be stored indefinitely, provided that an adequate supply of cooling gas is furnished at all times.

FUEL CAPSULE INSTALLATION

Upon completion of its launch site receiving inspection and tests, the Re-entry System Subassembly is transported to the warm cell, and the heat source assembly is installed in the Re-entry System Assembly as shown. This again is basically a manual operation, using handling equipment as required.

HEAT SOURCE - RE-ENTRY VEHICLE MATING

The Re-entry System Subassembly with the installed heat source is now ready for installation on the Power Module Subassembly, and this operation is accomplished by transporting the Re-entry System Subassembly to the launch pad (at T-11 days) where the Titan IIIC with the Power Module Assembly atop has been emplaced since T-14. The presence of the heat source in the payload will have little or no effect on subsequent pad operations other than those which must be performed in the immediate vicinity of the source. A radiation monitoring and health physics program is required however, and provision must be made to supply cooling gas to the heat source at all times prior to lift off.

Launch countdown holds present no problems, and if a pre-launch mission abort should occur, the fuel handling steps can be reversed to any desired stage of assembly.

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RE-ENTRY SYSTEM - POWER MODULE MATING

Depicted here is the SLPM docked and delivering power to the Saturn V Single Launch Space Station, the reference station for the SLPM study. This Space Station accommodates a crew of 6 in a nominal 260 n.m. orbit at an inclination between $28\frac{1}{2}$ and 50° . It is 33 feet in diameter, weighs 248,000 pounds, and is oriented with one end facing the earth at all times. The power requirement of the station as determined in the Boeing study* is 8 KW average, but for the purposes of the SLPM study, it has been considered to be 5 KW average. The PM is self-contained with a startup system, attitude control, and auxiliary power. It delivers regulated 1200 Hz, 3 phase AC to power conditioning equipment that is located on board the Space Station. In addition, the PM provides the Space Station with approximately 3 KW(th) of heat for EC/LS process work. This power is delivered by a liquid loop.

The SLPM is docked to the Space Station by means of a standard Apollo docking adapter (this hardware may not be consistent with automated docking systems that will be developed later). The drogue is mounted on the base of the PM and the probe is mounted on the periphery of the Space Station in exactly the same manner as are the Logistics Spacecraft docking adapters. All electrical and liquid loop connectors linking the PM and the Space Station are located within the confines of the docking ring. These connections are made up by a crew member (in a space suit, but inside the station) as part of the Space Station check out sequence that follows crew transfer from the Logistics Spacecraft to the station.

An effort has been made to minimize dynamic and thermal interactions between the PM and the SS by placing the PM near the SS center of gravity and by orienting PM radiators and load flattening resistor panels to present small surface areas to the SS. Heat transfer calculations have indicated that the effect of the PM and SS radiators on each other is negligible. The neutron radiation emanating from the isotope source is attenuated by a lithium hydride shield that is an integral part of the PM and by the separation distance ($27\frac{1}{2}$ feet minimum) so that the total neutron and gamma dose delivered to an occupant of any part of the Space Station is 23 REM over a one year period. An additional 2 REM has been reserved for exposures that may occur during periods of EVA required for in-orbit maintenance and repair. The low radiation level has no effect on any Space Station equipment.

*Saturn V Single Launch Space Station and Observatory Facility Study Contract
(NAS9-6816) 1967

ADVANCED NUCLEAR SYSTEMS OPERATION

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SLPM/SATURN V SINGLE LAUNCH SPACE STATION

The artificial gravity space station/SLPM arrangement is shown in the accompanying figure. The occupied area of the space station is a spacecraft generally similar to the zero-g reference design. It is connected to a spent SII stage by cabling, and the entire system spins at 4.8 rpm about a center 40 feet from the spacecraft center of gravity. This produces an artificial 0.2 g.

The PM-Spacecraft configuration is identical to the reference space station design. The 0.2 g condition should have no significant effect on PM operation. However, it would seriously impair in-orbit maintenance and repair operations to the point that de-spinning of the station would probably be necessary if any work of this kind were to be undertaken.

Docking of the PM to the Spacecraft would presumably occur prior to Space Station spin-up. Thermal and nuclear radiation effects and interactions would be the same as those of the reference Space Station configuration. The dynamics would obviously be different, but the effect of the PM on the system is not expected to be significant.

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SLPM/ARTIFICIAL - G SPACE STATION

The AAP-2 Cluster configuration with the SLPM is shown in the figure. The cluster consists of a CSM, an SIV-B orbital workshop (OWS), a multiple docking adapter (MDA) and airlock module (AM), and the power module (PM). The station is in a 260 n.m. earth orbit, and has an assumed average power requirement of 5 KW. As shown here, the PM is docked to the end docking port of the MDA while the CSM occupies one of the radially located ports. In this arrangement, the isotope source is approximately 65 feet from the nearest point on the experiment deck of the OWS, and 75 feet from the crew living quarters. This appreciable separation distance plus the corresponding reduction in subtended angle required to shield the 21-foot diameter OWS would result in a major reduction in SLPM shielding thickness, diameter, and overall weight. Thermal interactions between the PM and the remainder of the AAP cluster should be slight.

An alternate configuration (not shown) would have the positions of the CSM and the PM reversed. This is not so attractive because the separation distance from source to OWS experiment deck is about 45 feet. Also, it is difficult to physically interpose the shield between the source and the OWS without either increasing the SLPM diameter or using a shield that is moved into position following docking. Thermal interactions between the PM and other cluster components would be small.

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SLPM/AAP-2 CLUSTER

Shown is a Manned-Mars Flyby Spacecraft powered by two SLPM's. The spacecraft has power requirements averaging 8 KW while in the vicinity of Mars; for this reason and to preserve a symmetry of the spacecraft (required to maintain control during periods of propulsion system operation), two PM's are employed. The additional unit also provides redundancy desirable on a mission of this length.

The earth departure stage is likely to provide peak acceleration loads of from 1/3 to 1 g; this may cause some minor redesign of PM structure (particularly in the docking adapter region) but no serious problems are anticipated due to these induced forces.

A mission abort during or just after Earth departure stage firing would result in the separation of the propulsion module/earth entry module assembly from the rest of the spacecraft for return and re-entry. The mission module, with the PM's attached would go into a high earth orbit or an escape trajectory depending upon the time of abort. Likewise, as the spacecraft approaches completion of a normal mission, the mission module and PM's would be separated prior to re-entry of the Earth entry module, and would go into a solar orbit.

The Power Module radiation shield would require extensive redesign from the reference case for three reasons:

- . Two PM's will double the dose rate, assuming separation distance to be constant.
- . The allowable dose rate may be much smaller because of the 2 year mission commitment and the possibility of as much as 100 REM from solar flares and cosmic rays.
- . The smaller shadow required.

The net result of the above factors in terms of shielding size and weight has not yet been established.

SLPM/MANNED MARS MODULE

The major problem areas are summarized on the accompanying chart. In all but two cases these problems result from the fuel capsule not being able to adequately demonstrate survival. Item 2 represents an extremely severe thermal environment which no present day or near state-of-the-art capsule design could survive. However, the predicted probability of occurrence of this accident is < 1 chance in 10 million and would not seem to warrant designed protection.

Ground rules established for the present study call for the use of the ORNL isotope fuel capsule within the basic heat source concept developed by GE. The fuel capsule is well defined and provides an excellent basis for the preliminary heat source design. However, the temperature limits placed on the capsule severely penalize the usefulness of this design.

In conducting an objective study of the feasibility of a separately launched, isotope fueled, power module, it is vital that each component represent a fully compliant design satisfying all operational and safety criteria. Recent studies have shown that fuel capsules which would allow larger margins between operating and emergency temperatures have the potential of solving the majority of problems listed. It is therefore recommended that a task be undertaken to evaluate such an improved fuel capsule.

Item 7 lists power module weight as a problem area. It should be pointed out that the total system weight is within the limitations set by the Titan III C launch vehicle. However, nearly 700 lbs is represented by the radiation shielding alone and this is predicted on the use of highly enriched LiH_x . If this were not a feasible approach the use of other shielding materials could potentially cause an overweight problem.

Problem area 5 poses a rather difficult design challenge. The material selected for impact energy absorption within the re-entry shell must be able to withstand an operational environment of 600-700°F. Reduction of this temperature during re-entry can easily be accomplished by providing additional insulation at the backface of the re-entry shell. However, should an upside-down landing occur the added insulation would create an energy dissipation barrier and excessively high capsule temperatures. Several materials have been studied but a feasible solution has not been demonstrated analytically or experimentally.

PROBLEM AREAS

1. OVERTEMPERATURE DURING SUPERORBITAL RE-ENTRY BACKWARDS WITH ORNL FUEL CAPSULE
2. FALLING INTO THE TITAN IIIC SOLID FIRE AFTER EXPLOSION
3. BURIAL OF THE ENTIRE HEAT SOURCE
4. REDUCTION OF PuO_2 BY T-111 CAPSULE
5. CRUSHUP MATERIAL SELECTION WITH ORNL CAPSULE
6. BACKWARD IMPACT WITH ORNL CAPSULE
7. WEIGHT OF THE PM

The remaining tasks on the Power Module study are identified in the schedule shown. The present plan for evaluating the maximum power capability of the reference PM is to determine the payload launch capability anticipated for the Titan IIIC in the 1975 launch period and to then determine the maximum power output of a PM constrained by the weight limit.

The alternative payload capabilities of each of the four vehicles noted will be evaluated. For example, the Atlas-Agena might be able to launch a re-entry system alone into a 260 n.m. orbit. Such a launch would not meet the requirements of a power module design but it might have safety advantages when compared to an integral launch concept.

The preparation of a development plan, costs and critical item list is anticipated to be one of the most important outputs of the post midterm endeavors.

The final review is now tentatively scheduled for April.

FUTURE PLANS

TASKS

MAX. POWER CAPABILITY OF REFERENCE PM
DOCUMENT CAPABILITY OF VARIOUS LAUNCH VEHICLES
ATLAS-AGENA
ATLAS-CENTAUR
TITAN IIIB
TAT AGENA
PREPARE A DEVELOPMENT PLAN
PREPARE COST AND CRITICAL ITEM LIST
EVALUATE ADVANTAGES AND DISADVANTAGES OF SLPM

FINAL REVIEW

JAN.	FEB.	MAR.	APR.

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