SD 71-481-2
APRIL 15, 1971

Phase III Final

MASTER

NUCLEAR SHUTTLE SYSTEM DEFINITION STUDY

PHASE III FINAL REVIEW

Space Division
North American Rockwell

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Space Division
North American Rockwell

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NUCLEAR SHUTTLE SYSTEM DEFINITION STUDY
PHASE III FINAL REVIEW

CONTRACT NAS8-24975

APRIL 15, 1971

Prepared for

George C. Marshall Space Flight Center

By

T. M. Littman
Program Manager

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FOREWORD

This final briefing report, covering the ten-month technical effort from 1 May 1970 to 5 March 1971 on the Phase III Reusable Nuclear Shuttle (RNS) Definition Study, is presented in accordance with the data requirements (MSFC DRL-198, DRL Line Item -2) of Contract NAS8-24975.

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BRIEFING OUTLINE

• INTRODUCTION - T. M. LITTMAN

• MISSION REQUIREMENTS & OPERATIONS - A. R. JUSAK

• SYSTEM CAPABILITY - A. R. JUSAK

• SYSTEM DEFINITION - D. GARCIA

• PROGRAM PLANNING - W. G. ANTYPAS

• COST ANALYSIS - T. M. LITTMAN

• CONCLUSIONS & RECOMMENDATIONS - T. M. LITTMAN

PHASE III STUDY OBJECTIVES

From the initiation of this three-phase study in July 1969, primary concern was to establish requirements and conceptual designs of attractive first-generation nuclear flight stages with ultimate growth to meet the needs of manned planetary exploration. Initial efforts were concentrated on expendable concepts. During this past year, the focus has moved to fully reusable systems in support of NASA's integrated program plan (IPP) with cost minimization being a prime driver. Interorbital flight between earth and moon represents the primary mission for the purpose of establishing requirements and comparing alternate design concepts.

The Phase III effort, which began in May 1970, had the specific objectives indicated on the facing chart. In keeping with its Phase A conceptual definition status, the effort emphasized synthesis of design alternatives, using quantitative trade studies at the system and subsystem level to the maximum extent practical. Concurrently, requirements of the lunar shuttle and other potential mission classes were defined as well as performance data for concept comparisons and to aid NASA in mission planning. Integrated program planning supplied development schedules, costs, and requirements for manufacturing, testing, facilities, safety, reliability, quality assurance, and other support activities necessary to define a realistic program.

Other pertinent objectives include support of the SNPO NERVA program by conducting specialized analytical studies and defining engine-stage interface requirements as well as identifying physical and functional interfaces of the RNS with other IPP elements.

It is the specific purpose of this briefing to review the rationale and design/analysis process which led to the selection of the recommended RNS design concept, its subsystems, and operating mode.

PHASE III STUDY OBJECTIVES

BASIC GOAL

- DEFINE ATTRACTIVE RNS CONCEPTS CAPABLE OF PROVIDING LOW COST MANNED / UNMANNED SPACE TRANSPORTATION IN EARLY 1980's WITH PROGRESSIVE GROWTH TO MEET LATER MANNED PLANETARY APPLICATIONS

SPECIFIC OBJECTIVES

- COMPLETE PRELIMINARY TECHNICAL DEFINITION OF RNS CONCEPTS
DESIGN CRITERIA & SYSTEM SPECIFICATION
ENGINEERING DRAWINGS & FUNCTIONAL SCHEMATICS
SYSTEM / SUBSYSTEM TRADE-OFFS
- PROVIDE RNS MISSION / PERFORMANCE DATA
- ESTABLISH INTEGRATED PROGRAM REQUIREMENTS / PLANS, SCHEDULES, & COSTS
- SUPPORT RELATED NERVA STUDIES
- IDENTIFY INTERFACES WITH OTHER PLANNED NASA SPACE SYSTEM ELEMENTS

PHASE III RNS STUDY TASKS

In accordance with the contract work statement, nine technical tasks were performed. These are indicated on the facing chart along with the approximate distribution of direct technical effort and the significant outputs. As can be seen, the major emphasis was on operational requirements (pre-launch, earth-launch, orbital, mission, interface, safety and logistics) and system definition (subsystems, reliability, maintainability, radiation environment, thermodynamics, and integrated system optimization). The ten-month technical effort for Phase III was initiated in May 1970.

PHASE III RNS STUDY TASKS

ASSUMPTIONS & GUIDELINES

The major Phase III RNS study ground rules are listed on the facing chart. Initial RNS concepts reflect the anticipated 1974 state-of-the-art with limited consideration given to incorporation of more advanced technology later in the program. The RNS is to be designed for space operations over a three-year period using maintenance in low earth orbit as required to compensate for design lifetime limitations.

The NR study was restricted to RNS designs employing a 33-foot diameter main propellant tank with launch to orbit by the INT-21. This booster (designated the "MSFC Baseline" in the concurrent INT-21 study) uses standard J-2 engines but is structurally strengthened compared to the design used earlier in the RNS study. The space shuttle is assumed to deliver all other supplies including payloads, replacement NERVA's, propellants, etc.

As in the Phase II study, the operational program covered a ten-year span. However, rather than using a specific mission model, flight requirements were considered parametrically over the range 2-10 per year. Program planning studies considered use of government and contractor facilities wherever practical based on projected availability and applicability. The indicated flight test and initial operating capability (IOC) dates were to be considered as guidelines rather than hard targets. Of prime importance throughout the study was the achievement of lowest possible non-recurring and recurring costs without compromising key milestones and objectives. An important corollary was to assess approaches where major cost reductions may be possible.

ASSUMPTIONS & GUIDELINES

- 1974 TECHNOLOGY FOR INITIAL MAN-RATED RNS—DESIGN FOR LUNAR SHUTTLE
- RNS DESIGN LIFETIME UP TO 3 YR + EARTH ORBITAL MAINTENANCE
- 33-FOOT DIAMETER MAIN PROPELLANT TANK—NOMINAL 300K LB LH₂
- NERVA FULL FLOW CYCLE ENGINE— $I_{sp} = 825$ SEC, $T = 75K$ LB
- RNS LAUNCH TO ORBIT BY MSFC BASELINE INT-21 WITH STD J-2 ENGINES
- SPACE SHUTTLE DELIVERS OTHER ITEMS
- FIRST FLT TEST MID-¹⁹⁷⁹~~1970~~ + IOC IN 1981
- 10-YR OPERATIONAL PROGRAM, 2-10 FLTS/YR
- MAXIMUM PRACTICAL USE OF EXISTING FACILITIES & EQUIPMENT
- MINIMIZE DEVELOPMENT & OPERATIONAL COSTS WHILE MEETING:

KEY MISSION OBJECTIVES

MAJOR PROGRAM MILESTONES

PHASE III RNS CONFIGURATION CLASSES & LAUNCH MODES

In accordance with the study guidelines, consideration was limited to RNS propellant tanks of 33-foot diameter, launched by the INT-21. Three configuration classes were investigated with major emphasis given to the single tank and hybrid types. The single tank offers basic design simplicity and was evaluated over a wide range of geometric and other factors. A variation of this design -- the modified dual cell -- was assessed to see if it offered any advantages in terms of nuclear radiation shielding, and propellant management, using the same external tank geometry as the optimum single tank. The hybrid class has potential attractions in terms of engine disposal, ground checkout, development testing, propellant location control, and orbital assembly. Various geometric forms of the main and auxiliary tanks were considered to optimize the design with respect to both radiation shielding and engine disposal.

Although the RNS boost vehicle is ground-ruled as the INT-21, consideration was given to integral (engine-stage mated on ground) as well as non-integral launch. In the latter case, the main propellant tank is launched by the INT-21 (preferably inverted to minimize aerodynamic loads and heating as well as impact on KSC facilities) and the NERVA engine or engine-auxiliary tank module (in the case of the hybrid configuration) is launched by the space shuttle. Use of an inverted launch minimizes the need for an RNS aft skirt which improves both launch and mission performance. The non-integral launch requires in-orbit mating and checkout of the engine or auxiliary tank module with the main LH_2 tank as well as design of the avionics and propellant management subsystems for inverted launch.

PHASE III RNS CONFIGURATION CLASSES & LAUNCH MODES

CONFIGURATION CLASSES

LAUNCH MODES

MAJOR CONTRIBUTORS TO RNS PROGRAM

The basic goal of the Phase III study was to define attractive RNS concepts capable of providing low-cost manned/unmanned space transportation in the early 1980's. A high degree of reusability combined with maximum practical dependence on commonality is essential if RNS costs are to be kept low. The major space program elements which will contribute to the attainment of this goal are shown in the facing chart.

The space shuttle derived benefits are both operational and technological in nature. The vehicle serves as the low cost logistics system for delivery of propellant, NERVA engine, payload, replacement parts, and support equipment such as a shuttle-based maintenance element. It is also the source of programmed technology benefits in high performance insulation, reaction control subsystem, fuel cells for primary electrical power, and other items. Additionally, it is a potential source for an alternate fully or partially reusable booster for the RNS tank.

The Saturn V/Apollo will supply facilities, hardware designs, manufacturing techniques, as well as tooling, support equipment, and the RNS tank booster -- the INT-21. To obviate its strength limitations while minimizing booster development cost, launch of the RNS less NERVA can be accomplished in an inverted position. The slender cone of the baseline RNS tank (necessary for radiation attenuation to the payload) yields aerodynamic loads well within the baseline booster capability.

Space station also is a source of commonality benefits in such areas as astrionic equipment and software, the docking system, space handling equipment, and orbital operating procedures.

The manned spaceflight network will serve as mission ground control, support orbital operations, provide backup tracking/navigation aid, and perform trend data analysis for subsystem maintenance support.

MAJOR CONTRIBUTORS TO RNS PROGRAM

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MISSION OPERATIONS

The operations analysis for the RNS covered activities from prelaunch to end-of-life disposal, keeping in mind the basic objective of providing a reusable and economical transportation system. Although the detailed analysis considered launch of the RNS both in integral (stage and NERVA mated) and non-integral (stage and NERVA launched separately) configurations, emphasis is placed in this summary on the operations associated with the latter alternative.

Initial investigations show that vertical mating of the RNS tank and NERVA in the high bay aisle (for interface check) is the most attractive because of mating simplicity, as well as minimizing the GSE and facility modification requirements. Upon completion of leak and functional checks, the RNS will have demonstrated readiness for the mating operation with the INT-21 which will be accomplished after the RNS tank and NERVA are demated. While the RNS tank is being mated with the INT-21, the NERVA will be prepared for loading aboard the space shuttle orbiter for launch to earth orbit.

Orbital operations are initiated at the time the RNS tank is inserted into the operations orbit and terminated when the vehicle departs on its shuttle mission. Assembly operation for the 33-foot diameter RNS stage is primarily that of mating the NERVA to the RNS stage. Rendezvous and docking will be made with a variety of space program elements such as propellant depot, maintenance element, space shuttle and tug. The RNS will not make a hard dock to the space stations but will rendezvous to a stationkeeping position. Having made a rendezvous to a stationkeeping position or hard dock to an integrated space program plan (IPP) element, numerous operations such as payload transfer, propellant transfer, or orbital maintenance and repair can be accomplished. Analysis of foregoing operations in conjunction with mission design enable the generation of functional description from which RNS design requirements are derived. Taking into consideration all operations, mission design and relationship of RNS with other IPP elements, requirements derived will permit development of a RNS which is compatible with the overall space program.

MISSION OPERATIONS

PRELAUNCH OPERATIONS

The prelaunch operations considered activities of post-manufacturing acceptance, transportation of the RNS to KSC, and operations in the Vertical Assembly Building (VAB) in preparation for launch.

The post-manufacturing acceptance activities will primarily be oriented toward automatic checkout and vehicle acceptance test in conjunction with on-board checkout (OBCO). The major influence of the OBCO concept on post-manufacturing and prelaunch operations is in the area of ground equipment and test time. Under this concept, the manufacturing facility will conduct in-line checks and insulation breakdown (megger) and continuity tests at all interfaces as each subsystem is installed. Those interfaces that would require stage power to be applied in order to complete the subsystem installation test would be reserved for post-manufacturing checkout activity that would be conducted prior to final OBCO acceptance testing. The primary advantages to be achieved by this process is in the reduction of GSE requirements, test time, and potential reduction in schedule impact.

Following checkout the stage will be prepared for shipment in the same manner as that developed for transporting the Saturn S-II stage with the exception that the RNS stage will be shipped without an engine. Transportation to KSC will be accomplished in the same manner as the S-II with only minor modifications required to the transporter to accommodate the increased tank length between support rings. Water transport will utilize the AKD, presently used by S-II, to transport the RNS stage between Seal Beach and KSC.

Five major items will be received at KSC: the basic RNS tank, the NERVA engine, forward adapter, RCS pods, and the shroud. On arrival at the VAB, all items will be subjected to visual inspection and records checked for completion. During the RNS tank receiving operation, the NERVA engine, which has been received and checked out, will be delivered to the VAB and prepared for mating with the RNS tank. Initial study shows that vertical mating in the high bay aisle is most attractive because of mating simplicity as well as minimizing the GSE and facility modification requirement. Therefore, least recurring and non-recurring costs result, and major design constraints on the engine, tank and overall operation are eliminated.

Following tank/NERVA mating, the stage will be subjected to a leak and functional verification test. This will be followed by an overall system test performed by the onboard checkout (OBCO) equipment under control of the stage computer to verify total system operational capability. Upon completion of this activity, the RNS will have demonstrated readiness for the mating operation with the INT-21 which will be accomplished after the RNS tank and NERVA are demated. While the RNS tank is being mated with the INT-21, the NERVA will be prepared for loading aboard the space shuttle orbiter for launch to earth orbit. Major checkout operations associated with launch of both the RNS tank/INT-21 and NERVA/space shuttle have been identified for these activities in the study.

PRELAUNCH OPERATIONS

POST MANUFACTURE

- CHECKOUT
- PREPARE FOR SHIPMENT

TRANSPORTATION

- LAND
- WATER

KSC OPERATIONS

- NERVA-TANK INTERFACE MATING CHECK
- SHROUD INSTALLATION
- INVERTED TANK STACK ON INT-21
- NERVA LOAD IN SPACE SHUTTLE ORBITER

EARTH LAUNCH AND PAYLOAD HANDLING OPTIONS

Orbital operations are initiated at the time the RNS tank is inserted into the operations orbit and terminated when the vehicle departs on its shuttle mission. Post-boost operations, prior to mating NERVA to the RNS tank, include removal of protective shroud(s) not jettisoned during boost and removal or safing of range safety devices. Launch of the NERVA engine by the space shuttle is made with poison wires installed and control drums locked in the "scram" position, in keeping with safety considerations. The primary functions of the poison wires is to prevent the reactor from going critical if it should be immersed in water or liquid hydrogen. When the shuttle achieves orbit, the poison wires may be removed from the reactor by use of a retractable boom which is part of the engine delivery system. Following poison wire removal the engine is rotated 180 degrees and the engine is mated to the RNS stage. A check of the engine stage interface functions will be made following the assembly to verify that the operation has been successfully completed and the stage is now ready in an operational configuration for propellant transfer and payload mating.

A number of payload handling options are available through use of other IPP elements. Payloads can be delivered to the RNS in operations orbit by: (1) the space shuttle directly from ground launch, (2) the space tug following payload buildup at the space station, or (3) by the second stage of the INT-21. The last delivery option would be considered only if physical size and weight would prevent delivery of the payload by space shuttle for later assembly in earth orbit. Payload unloading could take place either in the 270 n mi return orbit or in the 260 n mi RNS operations orbit. The tug could be utilized to return either the payload alone or payload and RNS to the operations orbit for subsequent payload transfer to the space station or space shuttle.

In the case where the RNS returns to the 260 n mi operations orbit the payload may be retrieved by either the tug or space shuttle for final disposition. It can be seen that there is flexibility in the payload handling options which would permit contingency operational alternates.

EARTH LAUNCH & PAYLOAD HANDLING OPTIONS

POISON WIRE REMOVAL AND ENGINE MATING CONCEPT

The baseline RNS is configured for separate launch of the RNS tank assembly and NERVA engine from the **earth's** surface. The INT-21 launch vehicle will be utilized to launch the RNS tank assembly whereas the NERVA engine will be launched by the **space shuttle**.

Launch of the NERVA engine is made with poison wires installed and control drums locked in the "**scram**" position. No antricriticality device will be installed on the engine since suborbital starts of the NERVA will not be employed with separate launch of RNS tank and NERVA engine. Beside the structural interface of the NERVA pressure vessel and NERVA delivery support structure, an interface will be provided between the **shuttle and NERVA docking structure** so that status of the NERVA engine can be monitored during prelaunch, launch, and orbital coast up to the time of NERVA deployment from the **shuttle cargo bay**.

The primary function of the poison wires is to prevent the reactor from going critical if it should be immersed in water or liquid hydrogen. Once in orbit, the poison wire may be removed from the reactor through use of a retractable boom mounted on the aft bulkhead of the cargo bay as shown in the figure. Satisfactory removal of the poison wires can be verified through a **TV camera system** built into the NERVA delivery support system. Status of control drum position would of course be monitored prior to poison wire removal and NERVA engine deployment from the cargo bay.

Prior to deployment of the engine from the cargo bay, poison wires are removed from the NERVA reactor by retracting the poison wire removal boom. The neuter docking system between the engine delivery system and NERVA is uncoupled, the NERVA engine translated aft to clear the docking mechanism and the entire engine delivery system rotated 90 degrees to the longitudinal axis of shuttle. The NERVA is now clear of the cargo bay but the engine must be rotated 180 degrees to permit the NERVA to be docked to the RNS. Having completed the 180 degree rotation, the NERVA is now in position to be mated with the active neuter docking system of the RNS. The shuttle maneuvers the **NERVA** to dock it to the RNS. Following engagement of the docking system and verification of lockup of the NERVA engine to RNS, the delivery support structure releases its hold on the pressure vessel and is returned to the stowed position in the **cargo bay**.

Having verified structural engagement and lockup, the active docking ring on the RNS stage will be retracted to a locked position. Fluid and electrical connections **then will be made** through engagement of the retractable interface plates. Upon completion of this operation, a functional checkout will be performed on the total system employing on-board checkout (OBCO) to verify system operational readiness.

POISON WIRE REMOVAL & ENGINE MATING CONCEPT

PROPELLANT TRANSFER AND LOCATION CONTROL

Liquid acquisition and position location control during transfer is of pivotal importance, as it impacts both vehicle and depot design as well as operation. The study analysis considered methods shown on the chart. To date, positive displacement techniques have been used mainly for small tanks containing storable propellant. Metallic bellows are preferred over polymeric bladders for cryogenic applications. The largest metallic bellows being studied is six feet in diameter (Arde Corporation). It is highly questionable if a suitable positive expulsion device will be available for large tankers even in the late 70's. Capillary devices can be used for total propellant control (e.g., multiple screens in series), for partial control, or as a means of propellant acquisition and collection via capillary pumping (channels and ducts). Artificial gravity fields may be grouped into techniques as linear acceleration, centrifugal acceleration due to spin or pitch axis rotation, and dielectrophoresis. These require either supplemental propulsion or power, but are considered the most attractive method for RNS.

The methods of receiver tank thermodynamics evaluated include no vent filling, thermodynamic venting, direct venting, and pre-chilldown. As a result of the study, refueling is recommended using venting as required or using pump transfer from source tank to RNS with closed loop return of RNS hot gas to pressurize the source tank. In general, it is necessary to prevent excessive pressure buildup during filling if no vent filling is employed and to vent at the highest possible temperature if vent filling is employed.

Based on the analysis conducted, the methods of propellant position control during transfer can be compared on a semi-quantitative basis. Assuming equal weight for the evaluation criteria of: propellant losses, operational complexity, technology, system and design complexity, subsystem requirements and hydrodynamic control in receiver, two methods are attractive. The two best methods for transfer are (a) linear acceleration using source tank RCS sized specifically for propellant transfer, and (b) centrifugal acceleration using rotation about the pitch axis.

PROPELLANT TRANSFER & LOCATION CONTROL

LIQUID LOCATION CONTROL METHODS

- ACCELERATION
- BELLOWS
- BLADDER
- CAPILARY
- ELECTROHYDRODYNAMIC

RECEIVER TANK THERMODYNAMICS DURING FILLING

- NO VENT FILLING
- THERMODYNAMIC VENTING
- DIRECT VENTING
- PRE-CHILL DOWN

PROPELLANT POSITION CONTROL DURING TRANSFER

- LINEAR ACCELERATION
- CENTRIFUGAL ACCELERATION
- SURFACE TENSION

PROPELLANT TRANSFER MODES

Many factors must be considered in comparing propellant transfer to the RNS through the use of an intermediary propellant depot (single transfer to RNS) versus direct transfer from the space shuttle (multiple transfer to RNS). The more important factors are outlined in the table.

Multiple transfer avoids the need for a propellant depot and affords the simplicity of a single step transfer operation. However, it is believed that these advantages are not sufficient to outweigh the advantages of an intermediary propellant depot. Through the use of the depot, RNS turnaround is not constrained by space shuttle delivery, as approximately ten space shuttles or space shuttle flights are required to refuel the RNS. The RNS need only rendezvous and dock once for refueling reducing operational complexity and radiation hazard. The depot can be specifically designed for efficient transfer and storage of propellant.

It is recognized that the need for a propellant depot is dependent on the traffic model the RNS must satisfy as well as the availability of space shuttles providing the logistic support. Early in the lunar exploration program when the lunar shuttle traffic is low, the multiple transfer directly from the space shuttle could meet the program needs. However, as the tempo of space operational activities increases, the lunar shuttle traffic rates may not be able to tolerate the time constraints imposed on the RNS during "turnaround" by the space shuttle multiple transfer. In this case the propellant depot could best meet the overall space program needs since it would serve other IPP elements beside the RNS.

PROPELLANT TRANSFER MODES

*TRANSFER MODE	ADVANTAGES	DISADVANTAGES
MULTIPLE TRANSFER (SPACE SHUTTLE)	<ul style="list-style-type: none"> • PD NOT REQUIRED • SINGLE STEP TRANSFER 	<ul style="list-style-type: none"> • RNS TURNAROUND CONSTRAINED BY SPACE SHUTTLE LAUNCH RATE • MULTIPLE RENDEZVOUS & DOCKING WITH RNS • REQUIRE DEVELOPMENT OF BELLOWS OR DIELECTROPHORESIS TECHNIQUE
SINGLE TRANSFER	<ul style="list-style-type: none"> • RNS TURNAROUND NOT CONSTRAINED BY SPACE SHUTTLE LAUNCH RATE • SINGLE RENDEZVOUS WITH PD • DEPOT CAN BE DESIGNED FOR EFFICIENT STORAGE & TRANSFER 	<ul style="list-style-type: none"> • AN ORBITING PD IS REQUIRED • TRANSFER MUST BE ACCOMPLISHED IN TWO STEPS

SPACE SHUTTLE SUPPORT REQUIREMENTS

As indicated in the previous chart, selection of propellant transfer mode will be influenced by the RNS traffic rate and availability of logistic support vehicle. The facing chart shows parametrically the space shuttle support requirements which are influenced by quantity of consumables/material to be delivered to orbit and the time span in which delivery must be made. Factors such as launch pad operations and refurbishment (5 days), space shuttle orbiter cycle time (18-21 days), space shuttle booster cycle time (14 days), and launch pad requirements were derived from the contracted space shuttle study. A space shuttle orbiter payload capability of 33,000 pounds delivery to a 260 n mi circular orbit inclined at 31.5 degrees was assumed. In demonstrating the use of this chart, let's assume a requirement of delivering 130,000 pounds of payload to lunar orbit. 300,000 pounds of propellant would be required for the RNS and 35,000 pounds of propellant to account for boiloff and transfer losses. This total amount of payload (465,000 pounds) would require 15 shuttle launches assuming 31,000 pounds of useful space shuttle payload delivered to earth orbit with each flight (2,000 pound allowance for container, etc.). A 109 day lunar shuttle cycle was identified early in the program indicating a need to accomplish propellant delivery to the PD in less than 109 days to provide time for transfer from PD to RNS. The intersection of the 100 day line and 15 shuttle launches indicates that this can be accomplished with one launch pad, three orbiter vehicles, and two booster vehicles at a launch rate of one every six days. A 54.6 day lunar logistic cycle for the same propellant quantities would require delivery to the PD in approximately 50 days. This results in a requirement of two launch pads, six orbiter vehicles and 4 to 5 boosters with a launch rate of one every three days. These launch rates, pad and space shuttle vehicle requirements are compatible with the projected space shuttle usage. The OLS logistic delivery requirements during the lunar space base activation (four launches in 27 day intervals) exceed the projected space shuttle usage which may necessitate around the clock operation and/or additional pad and space shuttle vehicles.

SPACE SHUTTLE SUPPORT REQUIREMENTS

OPERATIONS RELATED TO ENGINE RUN RADIATION

Operation of the RNS near a space station or PD raises the question of personnel radiation hazard. In order to provide a safe separation between the RNS and the propellant depot during nuclear engine operation, the RNS is assumed to make a Hohmann transfer (with the RCS) to an orbit 10 n mi higher than the PD. A coast period of two hours places the RNS 10 n mi above and 160 n mi (approximately 2.5 degrees) behind the PD at engine startup. The resulting flight path is shown to scale in the figure.

The relative motion of the RNS and PD is indicated on the left in the figure and shown in detail on the right in an RNS coordinate system, for the first few minutes of powered flight. Closest approach to the PD (90 n mi) occurs after ten minutes and at no time during boost does the line-of-sight to the PD point more than 112 degrees from the nose of the RNS. Additional separation could be achieved easily by transferring to a higher orbit or by coasting longer before startup (starting farther from the PD).

In a similar manner safe separation is provided in departing from the space station in lunar orbit. The TEI powered phase begins at a position 130 n mi behind the OLS and ends before the RNS has quite caught up with the OLS. The closest approach to the OLS (55 n mi) occurs a few minutes after shutdown.

The table at the bottom of the chart presents the NERVA full power dose rates in the forward, radial and aft directions. It can be seen from the table and distances between the RNS and PD that the radiation hazard problem can be resolved through operational procedures, i.e., transfer to a higher orbit with RCS to provide the safe separation and attitude during thrusting.

OPERATIONS RELATED TO ENGINE RUN RADIATION

NERVA FULL POWER DOSE RATE (REM/HR)					
DISTANCE (MILES)	FORWARD	RADIAL AT CORE CENTER		AFT	
		γ	n	γ	n
1	NEGLIGIBLE	4.3×10^2	4.3×10^3	2.9×10^2	7.9×10^3
3	NEGLIGIBLE	48	480	31	870
5	NEGLIGIBLE	17	170	11	310
10	NEGLIGIBLE	4.3	43	2.9	79
15	NEGLIGIBLE	2	19	1.2	34

OPERATIONS RELATED TO AFTER SHUTDOWN RADIATION

A radiation hazard exists after NERVA shutdown due to the fission product inventory decay. The after shutdown radiation is primarily gamma radiation. After shutdown iso-dose contours 24 hours after shutdown are shown on the facing chart. The radiation in the forward direction is negligible; however, the radial and aft directions pose a radiation hazard.

Rendezvous and docking schemes were evaluated to address this radiation problem and to determine rendezvous detection range and accuracy requirements.

A simplified rendezvous and docking example is illustrated for the recommended Hohmann transfer. Circular orbits of the propellant depot (PD) and RNS differ in altitude by 10 n mi. At a range of 25.6 n mi, a 17 fps impulse opposite to the direction of orbital motion initiates the transfer. Another 17 fps impulse, also opposite to the direction of orbital motion, completes the transfer. In the maneuver shown, the RNS is assumed to hold an inertially fixed attitude during coast, the apparent pitch motion being caused by the 180° rotation of the coordinate system during the transfer. With use of +X and -X translations as shown, the nose of the RNS can point continuously in the general direction of the PD.

Trajectories determined by similar maneuvers, such as the 130 degree transfer with line-of-sight thrusting, give similar values for range at time of orbital transfer initiation. Therefore, variations of the Hohmann orbital transfer were used to determine parameter values such as detection range and transfer delta-V.

Although the radiation problem can be resolved for rendezvous, maintenance operations in the area of the engine remains a problem because of the high radiation level and will require use of robots (unmanned tug) and special manipulators (teleoperators).

OPERATIONS RELATED TO AFTER-SHUTDOWN RADIATION

AFTER-SHUTDOWN
ISO-DOSE CONTOURS

MOTION OF RNS RELATIVE TO PD DURING TRANSFER

MAINTENANCE ANALYSIS

Three levels of maintenance have been identified which will be required to support RNS orbital maintenance and repair (M&R). In-orbit M&R actions will include remove and replace (R&R) operations of modular components and repair in place of those items, such as insulation, which cannot be returned to earth for repair. The modular replacement maintenance concept is as follows:

Level 1 - In Orbit - This level will be limited to fault isolation, module replacement, straightforward servicing and adjustment, and maintenance action verification.

Level 2 - Bench Maintenance - This level will be in direct support of the RNS, and is the prime maintenance level for mission support. The function may be performed in the maintenance shops at the launch site, or within the maintenance element support area and will include Level 1 maintenance to the modules.

Level 3 - Depot Maintenance - This level includes the previous levels of maintenance coupled with complete overhaul, refurbishment, and modification capability. The locale of this effort will be the contractor facility or launch site, with turnaround hardware requirements satisfied from spares inventory. Those items that require repairable support or are to be scrapped in ground based facilities will be returned by an available space shuttle.

A turnaround operations time line has been developed which identifies scheduled and unscheduled maintenance operations. Limited life items such as batteries and fuel cells will set the pace for scheduled maintenance activities, whereas random failures will determine unscheduled maintenance activities.

The "hands off" maintenance philosophy will be followed during the M&R checkout cycle. This basic philosophy states that "if the system has just operated on the previous flight according to specification, then leave it alone." In this instance the checkout cycle would verify that secondary or backup systems are operational.

The prime M&R concept drivers include: state-of-the-art, radiation environment and traffic rates. Two maintenance elements were postulated based on different traffic rate requirements. One concept would be delivered to the RNS by the space shuttle in an "on-call" basis, whereas the second concept utilizes a maintenance element permanently located in orbit. The first concept would meet the low traffic rate projected for the early portion of the lunar exploration program and would make use of the space shuttle developments for servicing satellites. A larger maintenance element located permanently in orbit, however, could better serve the RNS when traffic rates are high and turnaround time is critical. It is accepted that a remote maneuvering unit with manipulators will be required to perform maintenance operations in the area of the engine where radiation levels exist which would be injurious to maintenance personnel.

The radiation environment unique to the RNS poses the problem of personnel approaching the engine for the purpose of inspection or maintenance. Development of non-destructive testing techniques coupled with the OBCO would permit statusing of critical components in the vicinity of the engine without exposing maintenance personnel to the radiation environment. By the same token, development of teleoperator would permit maintenance operations in the areas of high radiation levels.

MAINTENANCE ANALYSIS

M&R MAINTENANCE LEVELS

- IN-ORBIT
- BENCH
- DEPOT

TURNAROUND OPERATIONS

- SCHEDULED MAINTENANCE
- UNSCHEDULED MAINTENANCE

M&R CHECKOUT CYCLE

M&R CONCEPT DRIVERS

- STATE-OF-THE-ART
- RADIATION ENVIRONMENT
- TRAFFIC RATES

SR&T IMPLICATIONS

- NON-DESTRUCTIVE TESTING TECHNIQUES
- TELEOPERATOR DEVELOPMENT FOR SPACE APPLICATION

MAINTENANCE/REFURBISHMENT CHECKOUT CYCLE

Both scheduled and unscheduled maintenance operations will be performed on the RNS to meet the objective of reusability. Scheduled maintenance will be determined by limited life items such as batteries, fuel cells, and micron nav set. The projected life of a silver-zinc battery is one year for discharge/recharge cycles of approximately 200 with depth of discharge limited to 15 per cent. The fuel cell with a 5,000 hour life must be changed at least once during the 3 year life of the RNS. The micron nav set has a mean time between failures of 5,000 hours which poses no problem to mission success since four units are used in the design. However, a random failure in redundant systems must be repaired to maintain the probability of mission success goal. These random failures for long life components necessitate the unscheduled maintenance operations.

The Maintenance/Refurbishment Checkout Cycle in the facing figure combines a logic-functional flow approach to present the RNS maintenance and repair (M&R) and checkout philosophy. The intent of the chart is to display the "hands off" concept to be used during vehicle turnaround M&R operations. The "hands off" concept is one which will entail the performance of corrective-type (unscheduled) M&R functions complemented with minimum preventative-type (scheduled) M&R activities. If all vehicle primary systems operated properly and no scheduled maintenance is required, then the M&R subsystem checkout cycle would verify the secondary or back-up systems.

Vehicle status data, namely the onboard checkout (OBOCO) system outputs (e.g., data printouts), visual inspection reports, and planned M&R requirements, are the basis for establishing system readiness. With the system deemed ready, scheduled maintenance, as required, is performed in place or by a spare replacement, with the necessary testing performed to ascertain system readiness. When the system is not functioning properly, unscheduled maintenance is enacted by fault detection/troubleshooting methods to determine the problem. As the problem is defined, the action cycle (same as with scheduled maintenance) is activated with the necessary testing performed to ensure system readiness.

MAINTENANCE/REFURBISHMENT/CHECKOUT CYCLE

SPACE SHUTTLE DEPLOYED MAINTENANCE ELEMENT

The approach for accomplishing maintenance functions can range from extensive utilization of space shuttle delivered EVA crew to elaborate space hangars. These extremes were immediately ruled out as being too hazardous or too costly for performing maintenance operations on the RNS. It is accepted that a remote maneuvering unit with manipulators will be required to perform maintenance operations in the engine area where radiation levels exist which would be injurious to maintenance personnel. Use of a maintenance element, with the capability of providing a suitable environment for personnel performing maintenance operations not hampered by pressure suits and reducing or eliminating the need for EVA, is considered a desirable means for accomplishing the maintenance function. The concept shown here lends itself to delivery to the operations orbit by the space shuttle on an "on call" basis which is very attractive for low traffic rates.

The element is designed as an integrated module, deliverable and deployable from the 15-foot diameter x 60-foot long shuttle cargo bay, capable of all normal maintenance and replacement by remote controlled mechanical manipulation, and with a "cherry-picker" pod incorporated which can be manned to afford direct manual inspection and/or service of special case problems that might rise. It is equipped with a pair of special manipulators to service the RCS modules, plus a general manipulator capable of all of the other servicing functions including emplacement of the manned pod.

The pair of RCS manipulators are located nearest the RNS end of the service element together with provisions for storage of a pair of RCS modules. Each manipulator consists of a translating carriage railed to the service element chassis, a swing-out arm and flip-flop spindle with dual (opposed) RCS module pickups.

A general manipulator, together with a complete complement of spare equipment modules, is mounted behind that for the RCS system. The manipulator and spares are arranged so that any modular group of components in the RNS can be exchanged with a replacement from the maintenance element. The manipulator embodies a multiple articulate boom mounted off a ring around the maintenance element chassis and capable of all necessary general reaches and orientations, incorporating an anchor pad and translation/rotation head at the boom end to facilitate the local manipulations.

The manned pod is attached to the maintenance element chassis through an airlock for pressurized entry and exit; and is equipped with suitable connections so that it can be picked up by the general manipulator and transported to the vicinity of any of the equipment modules. By incorporating arm sockets and an external complement of tools, various special servicing and/or repair functions can be accomplished.

SPACE SHUTTLE DEPLOYED MAINTENANCE ELEMENT

NERVA REMOVAL CONCEPT

Prior discussion was oriented to maintenance operations in the forward skirt area where radiation levels were low enough to permit direct repair and servicing operations by man. Following NERVA operation, the radiation level in the immediate vicinity of the engine will require the use of a remotely controlled manipulator such as an unmanned space tug shown in the figure.

The procedure to be followed for NERVA removal begins with disengagement of fluid and electrical connections at the docking interface plate. This would be followed by extending and unlocking the docking rings and finally disengaging the engine and translating it aft with engine removal manipulator. The engine tilt actuator would rotate the engine 60 degrees for mating with the tug's acquisition probe. The tug would then maneuver to a position for engaging its extendible acquisition probe with the engine neuter docking ring. After a lock between the engine and acquisition probe is verified, the RNS engine manipulator would be released permitting engine removal and disposal as required.

It is recognized that the tug would have to have radiation hardened components and would more than likely be unmanned and remotely controlled.

NERVA REMOVAL CONCEPT

RNS DISPOSAL LOCATION AND CONFIGURATION ALTERNATES

Disposal of the RNS stage (or NERVA engine) are influenced by three major factors which are identified as: (1) disposal configuration alternate for disposal, (2) the place or location from which the spent stage is disposed of, and (3) the place or location to which the disposal is made. These three factors with their options are shown in the facing figure.

The spent nuclear stage or engine may be disposed of in a number of ways after it has reached its end-of-life operating time and the implications of the various discard modes will be touched upon in the following discussion. An option identified deep ocean disposal, however, this mode of disposal is not considered desirable as a normal disposal mode because of the potential hazard to the earth's population. This mode of operation would be considered only in emergency conditions where eventual earth impact would occur as a result of a failure and a controlled disposal to specific earth location would be least hazardous to the general populous.

Placing the spent stage in a long life earth orbit from the RNS operations orbit or synchronous orbit is a desirable location for spent stages; however, the disadvantage to this location is that it requires surveillance to insure noninterference with future missions. Lunar orbit or lunar surface impact is not desirable from the standpoint of creating lunar mission hazards or contaminating the lunar surface with radioactive material and thus jeopardize future surface exploration.

Heliocentric orbit disposal is considered desirable from the standpoint of energy ($\approx 11,000$ ft/sec from low earth orbit and $\approx 3,500$ ft/sec from lunar orbit) requirement and from the standpoint of safety. This approach, however, could pose a potential hazard of earth impact if injection is unsuccessful.

Various disposal configurations include disposal of the entire stage, the engine with an auxiliary propellant tank, and the engine alone by the tug.

RNS DISPOSAL LOCATION & CONFIGURATION ALTERNATES

GEOCENTRIC LONG LIFE
ORBIT 660 X 660 NMI

HELIOCENTRIC
ORBIT DISPOSAL

DISPOSAL CONFIGURATIONS

COMPLETE
RNS STAGE

RNS PROPULSIVE
MODULE

ENGINE ALONE
(VIA TUG)

RECOMMENDED END-OF-LIFE DISPOSAL MODES

Since there is no assurance that the engine life limit and the RNS tank life limit will be reached at the same time because of engine cycle limit and stage life limit, disposal of the engine in long life earth orbit by a space tug is recommended. Although other disposal modes may be less costly in terms of propellant required for disposal from an unmanned planetary mission or from lunar orbit, tug disposal offers program flexibility. The RNS tank and subsystems need not be discarded in event the NERVA engine reaches its life limit before the RNS tank and subsystems reach its life limit. **Also**, disposal of the engine independent of the RNS tank to a 660 x 660 n mi x 45° inclination orbit is within the capability of a tug with a 50,000 pound propellant tank capacity.

Use of the tug for disposal provides for NERVA malfunction contingency should the NERVA suffer irreparable damage (operational or accidental). An engine **disposed** of in earth orbit can be kept under surveillance more readily than a stage disposed of in heliocentric orbit whose trajectory periodically crosses the earth's orbit. The 660 n mi circular disposal orbit allows for retrieval of the engine at some future date either for examination and analysis or reclamation.

Alternate RNS stage disposal modes are possible through program scheduling such that the end of vehicle life coincides with an unmanned planetary injection mission or from lunar orbit provided the end-of-life coincides with a no payload return shuttle flight. These disposal modes are attractive from the standpoint of relatively small propellant requirement for disposal to a heliocentric orbit

The hybrid configured RNS stage offers the possibility of using the auxiliary tank and adaptor unit for disposing the engine to a long life earth orbit.

RECOMMENDED END-OF-LIFE DISPOSAL MODES

ENGINE DISPOSAL BY TUG RECOMMENDED (660 X 660 N MI, 32° TO 45° INCL ORBIT)

- TUG CAPABILITY
- PROVIDES FOR NERVA MALFUNCTION CONTINGENCY
- ENGINE LOCATION CAN BE READILY DETERMINED IN EARTH ORBIT
- ENGINES MAY BE RETRIEVED AT SOME FUTURE TIME

ALTERNATE DISPOSAL MODE BY RNS (HELIOCENTRIC ORBIT)

- CLASS II PLANETARY INJECTION WHEN END-OF-LIFE COINCIDES WITH IPP SCHEDULE
- FROM LUNAR ORBIT PENDING COINCIDENCE OF END-OF-LIFE & LOGISTIC SCHEDULE

INTERFACE WITH OTHER PROGRAM ELEMENTS

In the course of RNS operations from earth launch to end-of-life disposal, the RNS will have physical and functional interface with a number of other program elements. The interface elements shown include the planned and postulated interfaces conceived at this time. Some of these interfaces may not come into being whereas other new interfaces may be established as the integrated space program evolves. An example of a changing interface function may be that of propellant transfer directly from the space shuttle orbiter early in the program when RNS lunar shuttle rates are low. However, with higher frequency lunar logistic flights it may be necessary to utilize an orbiting propellant depot.

The physical interfaces identified here would include the structural interface the RNS would have with the INT-21 launch vehicle and the normal docking interface the RNS would have with program elements such as payloads, propellant depot, maintenance element, etc.

The functional interfaces relate to functions such as stationkeeping, communications, rendezvous, tracking, lighting, etc.

INTERFACE WITH OTHER PROGRAM ELEMENTS

INTERFACE COMPATIBILITY REQUIREMENTS

The facing chart shows that as many as 12 interfaces are possible between the RNS and other IPP elements. Interfaces have been divided into physical and functional categories as indicated. The functional category produces the largest number of possibilities.

Docking/structural interfaces are most numerous in the physical category. Interfacing elements in this case would include space shuttle, space tug, INT-21, manned and unmanned payloads, propellant depot, and maintenance element. Manned payloads will impose requirements on the RNS which are over and above those required by unmanned payloads. These requirements include electrical power, oxygen and water for the crew based on tug and orbiting lunar station studies.

The functions of rendezvous, command/control, telemetry (T/M), tracking, and radiation environment protection are shared between RNS and many program elements. Compatibility required among these many space program elements will be a forcing function in subsystem design. Function commonality among the many program elements points to maximizing usage of common subsystems and/or components in order to reduce development costs and provide functional interface compatibility.

The RNS will play both the active and passive roll in the rendezvous function depending upon which element it will rendezvous with. It will be the active vehicle when rendezvousing with the maintenance element and propellant depot. It will also be active in establishing the stationkeeping position with the lunar or geosynchronous space station. In the case of payload transfer, however, it will be passive taking up an attitude hold position and the element making the payload transfer will be active in rendezvous and docking.

Provisions are made in the RNS for override control which will take the form of direct connection and/or RF link between manned program elements physically interfacing with the RNS. RF link will be provided for override control with those elements not interfacing physically with the RNS.

Voice communications link will be provided for the manned payloads and the RNS voice link can serve as a backup for other program elements physically interfacing with the RNS. The communication function has a potential of interfacing with eight program elements including ground stations.

Protection from radiation environment created by the NERVA must be provided for by all program elements either through shielding or operating in the shield shadow cone when in the near-vicinity of the RNS.

INTERFACE COMPATIBILITY REQUIREMENTS

IMPORTANT MISSION/PERFORMANCE ISSUES

Traditional and important objectives of mission and performance analysis are to test design alternatives on a payload basis and to provide a reference mission framework for subsystems analysis. Results in these areas are seen in the recommended RNS design, wherein payload capability has been a major selection factor. A broader objective, of particular importance in the present climate of competing space transportation concepts, has been to demonstrate the RNS to be an efficient transportation system element. Maximum competency as an earth-lunar inter-orbit shuttle has been the primary concern. The RNS defined by lunar mission requirements provides very high performance for geosynchronous orbit and transplanetary injection missions and is adaptable to the advanced manned Mars mission propulsion role. Effort toward meeting the broader objective has yielded a better understanding of how missions should be configured for best use of the projected transportation system. A corollary result is a more realistic definition of mission requirements to be met by the RNS.

The following charts concentrate on four important issues which arise when attempting to prove the attractiveness of the RNS concept. As outlined on this chart, these issues relate to the efficient design of lunar shuttle missions, the feasibility of performing other missions in a reusable mode, the operational characteristics of NERVA as applied to potential missions, and the compatibility of the RNS with other elements of a space transportation system from the viewpoint of performance effectiveness. Study results have provided favorable resolution to these issues in most cases, and have otherwise delimited potential problem areas.

IMPORTANT MISSION/PERFORMANCE ISSUES

ARE EARTH-MOON SHUTTLE FLIGHTS GEOMETRICALLY FEASIBLE?

- REPEATABILITY
- PLANE CHANGE REQUIREMENTS
- MINIMUM ENERGY TRANSIT TIMES

CAN A REUSABLE STAGE DO OTHER MISSIONS?

- GEOSYNCHRONOUS ORBIT SHUTTLE
- TRANSPLANETARY INJECTION
- MANNED MARS MISSION PROPULSION

ARE NERVA PERFORMANCE PENALTIES SERIOUS?

- I_{sp} , COOLDOWN, BOILOFF VS EFFICIENT MISSION DESIGN
- LOW T/W

CAN RNS USE OTHER SPACE TRANSPORTATION ELEMENTS EFFECTIVELY?

- LAUNCH MODES
- TUG ASSISTANCE
- PROPELLANT DEPOT ORBITS
- SPACE SHUTTLE

CHARACTERISTICS OF LUNAR SHUTTLE TRAJECTORIES

Emphasis in defining requirements has been on the lunar shuttle mission because it is the primary generator of RNS design requirements and because it presents novel trajectory features to be explored. In general, the flight of an RNS (or any shuttle) from an earth orbit station to a lunar orbit station - and return - would appear to require non-minimum energy transit times and sizable plane changes. Investigation shows, however, that by properly choosing the initial alignment of the earth and lunar orbits, the earth orbit regression rate, and the lunar orbit stay time, plane changes can be practically eliminated. For an OLS delivery flight, where the (polar) lunar orbit plane is free to be established at will, optimization of mission parameters results in a plane change only at TEI. Moreover, by optimizing the initial earth orbit node (Ω_{REF}) and selecting the second TEI opportunity, near-minimum energy transit times are obtained together with near-zero plane change. The departures from an ideal "minimum-energy mission" are in fact quite negligible.

To facilitate return to a previously established OLS orbit it is convenient to choose an earth orbit whose regression rate is an exact small fraction ($1/N$, preferably $1/2$) of the moon's mean orbital rate. The earth-moon-orbit geometry favorable for initiating a mission will then recur after exactly N lunar sidereal months, or 54.64 days for $N = 2$. If the initial mission is "minimum energy" so also will be the subsequent revisit flights if the 54.64 day cycle is maintained. The demonstrated feasibility of minimum energy coplanar maneuvers and a repeatability cycle assures that there is no penalty inherent in the concept of repetitive shuttle flights between established terminal orbits at earth and moon.

CHARACTERISTICS OF LUNAR SHUTTLE TRAJECTORIES

OLS DELIVERY - UNCONSTRAINED LUNAR ORBIT

- COPLANAR TLI & EOI
- NEAR COPLANAR LOI - PLANE CHANGE AT TEI, IF ANY
- INITIAL EARTH ORBIT ALIGNMENT (Ω REF) AFFECTS PAYLOAD 10-15%
- OPTIMUM Ω REF, SECOND TEI, GIVE MIN ENERGY TRANSIT TIMES PLUS NO P/C

OLS REVISIT - CONSTRAINED LUNAR ORBIT

- ORIGINAL MISSION CHARACTERISTICS REPEAT EVERY 54.6 DAYS (TWO LUNAR MONTHS)
- OTHER DEPARTURE OPPORTUNITIES APPROX EVERY 9 DAYS
- PAYLOAD PENALTY UP TO 30% FOR OFF-NOMINAL FLIGHTS

LUNAR SHUTTLE MISSION CYCLES

Typifying the results of lunar shuttle mission analysis, this chart presents a pattern of flight opportunities following the assumed delivery of an OLS early in 1985 (payload values do not reflect maximum outbound delivery capability shown later). Translunar injection opportunities for revisit flights occur approximately every nine days with a nearly identical, high payload mission repeating every sixth opportunity. The slow regression of the moon's plane will produce a slight drop in payload over a period of many months. In practice, such effects would be easily accommodated by an occasional slight "tune up" of the earth orbit orientation.

Intermediate to each pair of nominal high payload flights are five other TLI opportunities which could be used should the need arise to break the 54.6 day cycle for an emergency mission. The payload loss in so doing varies from 6 to 35 per cent of maximum. For these flights, LOI and TEI plane changes, although freely optimized, dominate the performance picture.

The severity of the requirement to revisit a pre-established OLS may be judged from the dashed line data representing an unconstrained lunar orbit. In these cases LOI is near-coplanar and payloads are raised as much as 40,000 pounds. The remaining penalty, of 5,000 to 17,000 pounds compared to the minimum energy flights, is due to the requirement to return to the pre-established earth orbit. However, return to a free earth orbit offers no guarantee of desirable geometry for the next flight and the concept of repeatable high payload missions is not preserved. All things considered, constraining the earth and lunar orbits appears to be a significant scheduling and performance asset in that it assures high payload flights on a predictable basis.

LUNAR SHUTTLE MISSION CYCLES

BASELINE RNS CAPABILITY FOR SHUTTLE MISSIONS

The minimum energy repeatable lunar mission has been selected as a baseline for performance evaluation and reference mission definition. However, other mission models have been considered in this and related studies, as detailed on this chart. The major variable is the amount of plane change capability to allow in the lunar vicinity maneuvers to assure a "flyable" mission and to account for possible abort requirements.

Original study guidelines specified the 30/30 degree plane change case but initial results of the repeatable mission analysis showed that an optimized mission would require a plane change only at TEI if at all. A mission model constructed on this premise was referred to as "nominal" because the initial earth orbit alignment was midway between best and worst values. Further analysis identified the "minimum energy" mission model, as already explained. A fourth model, here termed "lunar abort", was introduced under the rationale of anytime-abort from lunar orbit. Studies of OLS and LSB missions, however, have not identified a need for RNS (or chemical shuttle) abort to earth orbit. In any case, rapid return should be the desired feature as opposed to the arbitrary 90 degree plane change capability. At this stage of definition of lunar logistics operations and related emergencies, there appears to be no justification for a non-minimum energy mission as a design requirement.

The baseline RNS flying the baseline minimum energy mission provides a lunar delivery capability of 200,000 pounds, far higher than originally contemplated for a 300,000 pound LH₂ capacity stage. The payload capability here noted reflects the inclusion of all performance related features such as realistic modeling of NERVA I_{sp} and cooldown, boiloff, RCS translational maneuvers, and velocity losses. Estimated requirements for OLS and LSB missions range up to 130,000 pounds delivered with 15,000 to 20,000 pounds return payload and the occasional delivery of modules up to 160,000 pounds in weight. These requirements are readily handled by the baseline RNS.

Manned missions to geosynchronous orbit, although not studied to the extent that lunar missions have been, promise ultimately to create additional demand for shuttle vehicle flights. Requirements imposed on the RNS do not differ substantially from those of the lunar mission except in regard to mission phase durations, which are much shorter. A space station module of over 160,000 pounds could be delivered by the baseline RNS on an unmanned flight (with no external shielding). The dashed line indicates payload capability for manned flights based on detailed performance modeling similar to the lunar shuttle cases.

BASELINE RNS CAPABILITY FOR SHUTTLE MISSIONS

LUNAR MISSION PERFORMANCE IMPROVEMENTS

The large uncertainty as to the nature and extent of lunar mission logistics support in the 1980's may delay for sometime assessment of the most desirable vehicle size. Alternates to the 300,000 pound LH₂ baseline RNS therefore continue to deserve consideration in terms of a desire for greater cost effectiveness and mission flexibility. Increasing RNS tank size is the most obvious route to a better payload to gross weight ratio but results of OLS and LSB studies are pointing to the need for, or at least the acceptability of, smaller lunar shuttle capability. The use of the space tug with the RNS in a two-vehicle lunar mission mode is another alternative with attractive possibilities.

A dramatic increase in payload capability follows when a space tug is used to assist the return of the RNS at the end of a lunar mission. By refueling the RNS in elliptical orbit, the tug permits an 80 percent increase in maximum lunar payload delivery. If the tug simply retrieves the RNS, using tug propulsion only, a 50 percent gain is still possible. This chart illustrates the gain in terms of propellant quantity (RNS and tug) per pound of delivered payload. More important than the increase in the already large payload capability is the indication that an RNS sized for moderate mission requirements could occasionally employ an alternate flight mode for handling much larger payloads or for emergency non-optimum flights. The tug performance parameters and practical tank size range are taken from a recent NR study of space tug missions and design. A relatively large tug is advantageous as a retriever, but size makes little difference for a refueler over the range studied.

A remaining question is whether the baseline RNS plus tug makes a more efficient system than would a larger RNS alone, since propellant-to-payload ratio will decrease also for larger stages. The dotted bars in the chart represent an RNS alone, scaled up to provide the same payload as the RNS/tug combination. On this basis tug assistance still offers better efficiency. Equivalent RNS sizes range from 335,000 to 388,000 pounds in the retriever cases and from 395,000 to 462,000 pounds in the refueler cases.

Refueling the RNS in lunar orbit represents another technique for the occasional delivery of larger than normal payloads. A propellant stockpile or a second RNS in service would be required, but a one way payload could exceed 330,000 pounds.

LUNAR MISSION PERFORMANCE IMPROVEMENTS

TRANSPLANETARY INJECTION CAPABILITY

As an injection stage for unmanned planetary missions, the RNS can function effectively in a variety of flight modes as shown in this chart. For low energy missions to Mars and Venus, the RNS offers 100-200,000 pound payload capability when used alone and returned to the earth operations orbit. Payload requirements identified for this type of mission range up to 150,000 pounds for an MSSR concept studied by Northrup.

Expending the RNS on an injection flight, at the end of its useful life, pushes its capability up to the energy level of outer planet missions and beyond. Greater importance, however, lies in extending the energy range as a reusable injection stage by coupling with a high performance chemical expendable stage. The simplest flight mode, a single continuous boost from circular orbit, provides capability beyond 30,000 fps. The large velocity loss on such a flight can be significantly reduced by the multi-orbit injection technique with a 1500 to 2000 fps gain in injection velocity as shown in the chart. Self-return to a 260 n mi circular orbit following payload injection seriously reduces the mission velocity level the RNS can reach. However, the addition of expendable upper stages, typified by Centaur and space tug, bring fast Grand Tour and outer planet direct flights well within reach in spite of the self-return velocity penalty.

Payload studies for medium to high energy missions have generally contemplated much lower launch vehicle capability than here presented. Nevertheless, there is no evidence that the overall effectiveness of a planetary exploration program could not be enhanced by the greater capability of the RNS/chemical stage system. Increased payload allows more experiments, greater equipment redundancy, and use of more favorable planetocentric orbits. In addition, the larger available launch windows ease mission and program planning. Payloads previously identified for Grand Tour and comet missions have ranged up to 50,000 pounds, predicated on Saturn V capability. Within the RNS operational time period, the fast Grand Tours and direct outer planet flights will offer a chance to continue outer planet exploration after the conclusion of the regular (Jupiter swingby) Grand Tour cycle.

Tug assistance is applicable to reusable transplanetary injection just as it is to the lunar mission return leg. The maximum possible gain in injection delta-V is approximately 4000 fps, a useful increment where it is desired to extend the capability of a specified RNS/chemical stage system.

TRANSPLANETARY INJECTION CAPABILITY

MANNED MARS MISSION CONFIGURATIONS AND SIZING

In the area of manned Mars missions, there is a continuing concern to correlate lunar shuttle configurations and sizing with planetary propulsion requirements, to assess advanced subsystems requirements, and to explore the interaction of planetary mission design with propulsion requirements. The Phase III study addressed these problems in a limited way by sizing 33-foot-diameter modules in three configurations and for two contrasting mission models (1986 opposition and 1990 conjunction). One of the chief subsystems alternatives - HPI/boiloff reliquefaction - was explored, as was the effect of tug assistance for RNS retrieval.

The trends observed in initial mass requirements are not unexpected. Reliquefaction appears to be moderately attractive, tandem departure stages excel over parallel stages due to the benefits of multi-orbit injection, and tug refueling of returning stages is desirable up to the limit of tug utilization (escape velocity). This chart illustrates sizing of propulsion and propellant modules for a 1986 OVS mission when HPI without reliquefaction is used for the interplanetary nuclear stage (INS).

Configuration 1 represents a standard approach to an all-reusable system. The first of the equal sized departure nuclear stages (DNS) supplies about 6000 fps of the required velocity increment in three burns before returning to circular orbit. The second DNS also makes three perigee burns to achieve trans-Mars injection and then must return to circular orbit from hyperbolic speed. The INS ultimately returns itself and the planetary mission module to an elliptical (24 hour) earth orbit. Application of 100 percent reliquefaction to the INS (with reoptimization of HPI thickness), reduces initial mass in earth orbit (IMEO) by seven percent in this case. For the 1990 conjunction mission, with its much longer duration, the gain is 12 percent, an improvement to be weighed against the extra complexity, development cost, and risk of failure with such systems.

The disadvantages of parallel staging are evidenced by the higher IMEO for Configuration 2. No weight is dropped until the earth departure maneuvers are completed and both stages must return from hyperbolic speed. The velocity losses are somewhat lower with the doubled T/W but with the multi-orbit injection technique the effect is not enough to offset the above handicaps.

The considerable reduction in IMEO (about 30 percent) when tanks are expended is shown by Configuration 3. One NERVA performs all propulsion tasks and returns ultimately to earth orbit. Outboard tanks are jettisoned prior to the last trans-Mars injection burn while the third tank is jettisoned in Mars orbit.

Modest tailoring of configuration and mission assumptions would result in modules readily compatible with the base-line RNS size. Considering the variety of possible planetary propulsion configurations, it appears likely that a large degree of overlap can always be obtained between lunar and planetary sizing requirements within the framework of the single-tank concept.

PLANNED MARS MISSION CONFIGURATION & SIZING

ADVANTAGE OF MULTI-BURN MANEUVERS

Each type of mission proposed for the RNS offers the possibility of reduced theoretical velocity requirements with multi-impulse maneuvers. However, the degradation in NERVA performance associated with shorter burns (reduced effective specific impulse and increased total cooldown loss) has caused concern that the RNS cannot realize its full theoretical potential. Results throughout this study have nevertheless shown that multi-impulse maneuvers can substantially increase payload; that is, NERVA performance degradation is generally outweighed by efficient mission design.

In the case of translunar injection (TLI) a velocity loss of 800 fps attends a single continuous burn. This can be reduced to less than 200 fps by splitting TLI into two perigee burns, and diminishing reductions follow for a split into three or more parts. When velocity requirements are combined with NERVA performance characteristics, the total required LH_2 (including cooldown) is reduced substantially by using two or three burns as shown on the left of this chart. The further reduction in velocity loss in going to four burns does not compensate for the lower NERVA performance. Based on this result, a two-burn TLI is recommended as a desirable flight mode for the RNS.

A similar approach applied to lunar orbit insertion (LOI) and transearth injection (TEI) maneuvers yields the results on the right. Sizable plane changes coupled with these maneuvers strongly favor a three burn profile because of the much lower delta-V's. Only below about five degrees plane change is a one burn maneuver superior. Again, cooldown LH_2 is included in the comparison as well as the variation of NERVA effective I_{sp} with burn time.

It is important to note that the relatively low T/W of a NERVA powered stage is not a significant factor so long as multi-burn departure maneuvers are utilized. This is the case for the reusable transplanetary injection modes and manned Mars mission stages as well as for the shuttle missions. Only in the case of an expendable transplanetary injection maneuver is there an irreducible velocity loss, occasioned by the necessarily continuous burn from the last elliptical orbit up to final injection energy.

ADVANTAGE OF MULTI-BURN MANEUVERS

TRANSLUNAR INJECTION

LUNAR ARRIVAL/DEPARTURE

TYPICAL RNS PROPELLANT BREAKDOWN

A breakdown of RNS propellant allocations for typical missions affords another view of the propulsion efficiency to be obtained with NERVA. This chart presents that information for three reference missions defined during this study. The interim lunar mission (first column) indicates a larger amount of boiloff and residual LH_2 than that obtained with the final propellant management system. Because of the relatively short duration of the geosynchronous mission, there is no boiloff with the baseline RNS design. Cooldown LH_2 represents about 5-1/2 percent of the total in each case, being slightly higher in proportion to impulse propellant in the first and third cases which involve more (therefore shorter) NERVA burns than does the baseline lunar mission.

In powering a typical lunar shuttle mission, NERVA achieves an overall effective specific impulse of 775 seconds when startup, shutdown, cooldown and cooldown delta-V are accounted for. This represents a six-percent loss from the nominal steady-stage value of 825 seconds, a loss comparable to that suffered by LOX/LH_2 propulsion systems when the effects of chardown, line loss, and inherently greater boiloff are included. Accurate modeling of NERVA performance has been included in all the quoted payload evaluations.

TYPICAL RNS PROPELLANT BREAKDOWN

PROPELLANT ALLOCATION	WEIGHT (POUNDS)		
	'NOMINAL' LUNAR MISSION	MIN ENERGY LUNAR MISSION	GEOSYNCH ORBIT MISSION
IMPULSE	275,625	280,311	281,051
COOLDOWN	15,554	15,439	16,059
BOILOFF	2,971	1,360	0
RESIDUAL	5,850	2,890	2,890

AVERAGE Isp
(SEC)

770

775

770

Space Division
North American Rockwell

TOTAL LOGISTICS BREAKDOWN FOR RNS LUNAR MISSIONS

Another type of performance penalty, of potentially serious proportions, involves the efficiency of space shuttle operations in support of RNS flights. Since the largest component of RNS earth orbit logistics requirements consists of low density LH_2 , the cargo volume of the shuttle (15-foot diameter x 60-foot length) is a limiting factor. With early shuttle designs this was not the case, but subsequent trends point to an ultimate shuttle payload capability considerably in excess of the 40,000 plus pounds of LH_2 which can be accommodated in the groundruled cargo bay. Several ways to ameliorate this problem await further investigation, including use of an alternate operating orbit and transport of mixed cargo with a higher average density than LH_2 .

The possibility of obtaining a higher effective cargo density stems primarily from the considerable amount of LOX which is required to be transported to orbit for space tug, RCS, and orbit adjust requirements. Based on data from OLS and LSB studies, this chart indicates a potential effective density of 6.6 pounds per cubic foot, a 50 percent increase over that of LH_2 . The packaging and transfer problems associated with this concept have not been investigated, so these figures must be considered only as an upper limit on what could be accomplished in this way. When allowance is made for containers and related equipment, the maximum payload weight in the 10,600 cubic foot shuttle cargo volume will vary from about 50,000 (LH_2 only) to 70,000 (mixed) pounds.

TOTAL LOGISTICS BREAKDOWN FOR RNS LUNAR MISSIONS

MISSION LOADING	%
• RNS LH ₂ (INC 10% LOSS)	55.8
• RCS & FUEL CELL LOX/LH ₂ (INCL 10% LOSS)	1.4
• CONTAINERS	5.7
• PAYLOAD	
• TUG LOX/LH ₂ (60% OF PL)	20.3
• CONTAINERS	2.0
• OTHER CARGO (DENSITY 10 LB/FT ³)	11.5
PRO-RATED ORBIT MAINTENANCE	
• ORBIT ADJUST LOX/LH ₂	3.0
• DORMANT FUEL CELL LOX/LH ₂	0.3
EFFECTIVE DENSITY OF ABOVE "AVERAGE" PAYLOAD	
• 6.6 LB/FT ³	
REPRESENTATIVE SPACE SHUTTLE LOADINGS (10,600 FT ³)	
• LH ₂ AT 4.4 LB/FT ³ PLUS CONTAINER & EQUIPMENT	50,000 LB
• "AVERAGE" PAYLOAD AT 6.6 LB/FT ³	70,000 LB

EFFECT OF PAYLOAD DENSITY ON SPACE SHUTTLE UTILIZATION

Although the space shuttle design is undergoing further change, the magnitude of the penalty which can result from a cargo volume limit may be judged from a recent space shuttle design point (47,000 pounds of payload delivered to 270 n mi, 55 degrees). The number of space shuttle flights required to support one RNS lunar mission is a good indicator of efficient utilization, and consequently of operational cost. This parameter is shown here as a function of effective payload density. It is assumed that the earth operations orbit to which the shuttle flies has the correct regression rate for the 54.64 day lunar mission cycle but can otherwise vary in altitude and inclination subject to $h \geq 100$ n mi and $i \geq 28.5$ degrees.

When payload density is low, it is best to operate from the highest possible earth orbit. This minimizes the RNS lunar mission delta-V's and the propellant for orbit decay correction. Under the stipulated constraints the proper orbit is 292 n mi, 28.5 degrees as on the left of the chart. As payload density increases, a point is reached where cargo bay weight capability exceeds shuttle performance capability and a lower altitude is indicated. This trend continues until (1) the 100 n mi limit is reached or (2) an optimum altitude is reached, below which the increase in orbit decay cost outweighs the increase in shuttle performance. These situations are met in the lower and upper curves. A further increase in payload density has no effect since the problem is now constrained by the orbit mechanics rather than by the cargo bay volume.

Relating these trends to the 50,000 pound cargo bay limit noted on the previous chart (equivalent to an effective density 4.7 pounds per cubic foot), it is seen that number of flights falls in the volume limited region. This point (11.4 flights) represents a 25 to 40 percent increase over the unconstrained cases. The range noted reflects uncertainty in the assumptions associated with orbit decay between missions. Whether such a penalty is viewed as serious depends partly on whether an alternative system can do any better. As comparative studies now stand, the inherent advantages of nuclear over chemical propulsion appear to hold up against a 25 to 40 percent shuttle efficiency penalty.

EFFECT OF PAYLOAD DENSITY ON SPACE SHUTTLE UTILIZATION

SUMMARY

Following the nominal OLS delivery flight, TLI opportunities will occur approximately every 9 days with a nearly identical, high payload recurring every sixth opportunity (54.6 days). This basic pattern will hold year by year with slowly varying flight characteristics caused by the changing inclination of the lunar orbit plane. Ample mission planning flexibility appears to exist within the constraints imposed by pre-established earth and lunar orbit orientations.

The RNS traffic rate will be a strong factor in determining propellant transfer modes and maintenance element configurations. Early in the program when traffic rates are low, the space shuttle could deliver propellant directly to the RNS without affecting turnaround time. By the same token the space shuttle delivered maintenance element could be delivered to orbit on an "on call" basis for maintenance operations without being a detriment to turnaround time. This maintenance element concept could be developed as an outgrowth of the orbiter's requirement to service orbiting satellites. High RNS traffic rates on the other hand could dictate the use of a propellant depot and maintenance element located permanently in orbit.

Although the induced radiation environment, unique to the RNS, poses a hazard to operations, realistic operating procedures can mitigate the radiation problem. The mission operations are designed such that departure and arrival orbits pose no radiation hazard to space program elements. The same holds true for rendezvous approaches.

High shuttle traffic rates will result in engine cycle limit being reached before the three year stage life limit. Engine disposal by tug would thus permit continued tank usage when engine operation cycle limit is reached before RNS tank life limit is reached. It also provides for disposal in the event the engine is inoperable and irreparable.

Due to the many common interface functions between RNS and other program elements, interface compatibility and commonality will be major sub-system design drivers.

Beyond the basic lunar application, the RNS possesses strong capabilities for other high payload missions. Shuttle flights to geosynchronous orbit pose no new requirements not subsumed by the lunar mission. Transplanetary injection capability with chemical upper stages exceeds that of any previously contemplated ground launched boost systems. The velocity requirements for RNS return to orbit are sizable, but nevertheless can be accommodated without unduly compromising the attractiveness of reusable injection mode.

For all classes of missions, NERVA performance characteristics are enhanced by the use of efficient multi-burn maneuvers. A two-burn TLI maneuver, for example, raises lunar delivery payload by 15,000 to 20,000 pounds. The possibility of refueling the RNS by space tug likewise offers significant performance improvements in many mission situations.

The apparent hindrance to best use of the space shuttle due to the low density of LH_2 requires further investigation. Easy mission/performance compatibility with other space transportation elements is evidenced by the variety of feasible launch modes for the RNS tank, by the utility of tug assistance, and by the wide choice of acceptable earth operations orbits for the RNS missions.

SUMMARY

- DESIGN OF LUNAR SHUTTLE MISSIONS FOR REPEATING FLIGHT CHARACTERISTICS IS PRACTICAL
- MAINTENANCE ELEMENT CONCEPT COMPATIBLE WITH RNS SUPPORT REQUIREMENTS & SERVICING PHILOSOPHY
- RNS RADIATION ENVIRONMENT CAN BE ACCOMODATED WITH REALISTIC OPERATIONS PROCEDURES
- ENGINE DISPOSAL BY TUG PROVIDES FOR PROGRAM FLEXIBILITY & INOPERABLE ENGINE CONTINGENCY
- INTERFACE COMPATIBILITY & COMMONALITY MAJOR SUBSYSTEMS DESIGN DRIVER
- RNS EXHIBITS SUBSTANTIAL CAPABILITY OVER WIDE RANGE OF MISSIONS
- TWO-BURN LOW EARTH DEPARTURE AND OTHER MULTIBURN MANEUVERS ENHANCE RNS PERFORMANCE
- LARGE PAYLOAD GAIN - ACHIEVABLE THROUGH TUG ASSIST IN RETURN TO RNS OPERATIONS ORBIT - INCREASES PERFORMANCE EFFECTIVENESS & OPERATIONAL FLEXIBILITY
- MOST EFFECTIVE USE OF SPACE SHUTTLE REQUIRES FURTHER INVESTIGATION

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BRIEFING OUTLINE

- INTRODUCTION - T. M. LITTMAN

- MISSION REQUIREMENTS & OPERATIONS - A. R. JUSAK

- SYSTEM CAPABILITY - A. R. JUSAK

- SYSTEM DEFINITION - D. GARCIA

- PROGRAM PLANNING - W. G. ANTYPAS

- COST ANALYSIS - T. M. LITTMAN

- CONCLUSIONS & RECOMMENDATIONS - T. M. LITTMAN

SYSTEM DEFINITION ACTIVITIES

The design objective during the Phase III study was to establish attractive RNS concepts. The attractiveness of the design arrangements was measured in terms of cost/performance effectiveness, which is directly relatable to components weight including external radiation shield, structure, thermal and meteoroid protection, as well as mechanical, fluid, and astrionic subsystems. Additional design drivers considered that also affect performance and cost include impact on the launch vehicle, facilities requirements, and complexity of flight operations.

The systems and subsystems analyses performed were subdivided into two separate but interrelated tasks, i. e. , integrated stage design development and subsystems design. The first concentrated on the development of stage geometrical arrangements taking into consideration radiation environment, performance, interface with NERVA and launch vehicle, as well as ground and space operations including engine removal and disposal. These required parametric tradeoffs where the dimensional characteristics of stage design concepts were varied to assess their influence on structure, thermal/meteoroid protection, radiation dose to the tank top, and external shield weight. Loading impact on the launch vehicle also was assessed trading stage geometry, boost mode, and assembly mode. These analyses together with the results of interface design definition, mission operations requirements, thermodynamics, propellant management, and astrionics investigations were integrated into total systems for stage design evaluation and selection.

The subsystems were evolved concurrent with the stage configurations. The emphasis in the analyses in this case was placed in the investigation of interactions with other subsystems/systems and commonality with current NASA programmed efforts. In addition, simplicity of manufacturing, inspection and repair, as well as maintainability and reliability requirements were leading drivers in the design and selection of each subsystem, and the components thereof. Weight and performance complete the list of variables in the tradeoffs performed for stage concept evaluation, as well as for the selection of subsystems and components as applicable.

The selections in each area were integrated into the reference RNS system, and critical supporting research and technology was defined for a timely development of the system.

SYSTEM DEFINITION ACTIVITIES

- CONFIGURATION EVALUATION & SELECTION
- SUBSYSTEMS DESIGN
- REFERENCE RNS DEFINITION
- SUPPORTING RESEARCH & TECHNOLOGY

CONFIGURATION EVALUATION AND SELECTION

The design issues depicted in the chart relate to performance and cost evaluation criteria and served as guides in the design investigations. For example, stage length is inversely proportional to the aft bulkhead cone angle and end cap radius. As stage length increases so does surface area and weight. On the other hand, the weight of the external shield required for radiation dose attenuation to the payload decreases with increase in fineness ratio of the cone. Consequently, the tradeoff yields a point of minimum total weight which denotes the stage design with the highest flight performance. Yet, to complete the tradeoff, the resulting stage geometry has to be evaluated for impact to the launch vehicle, orbital operations, and facilities in order to arrive at a realistic and attractive solution.

Other design issues that must be considered in the system evaluation include stage and engine interface and NERVA disposal requirements. Interface design must respond to a wide range of considerations when applied to hybrid and single tank designs. These include the inherent complexities associated with mating fluid lines and electrical receptacles in earth orbit. Engine disposal also presents a wide range of design considerations that must be considered in the alternate configurations evaluation and selection. Additionally, cost effectiveness must be one of the prime drivers in the vehicle selection, because in the final analysis operational efficiency is measured in terms of cost.

CONFIGURATION EVALUATION & SELECTION

- WHAT IS BEST GEOMETRY? →

- HOW DOES IT INTERACT WITH RADIATION SHIELDING? →

- IS IT COMPATIBLE WITH LAUNCH VEHICLE? →

- IS IT COMPATIBLE WITH FACILITIES? →

- IS IT COMPATIBLE WITH NERVA? →

- DOES IT FACILITATE ENGINE DISPOSAL? →

- IS IT COST EFFECTIVE?

PHASE III RNS CONFIGURATION CLASSES AND LAUNCH MODES

The three RNS configuration classes investigated are shown in the adjacent chart. One is a single tank design. A second is a modified dual cell which maximizes the column of propellant available for radiation attenuation during the critical last engine burn when the radiation dose rate is reaching its peak. The third is a hybrid or two-tank design which can be employed in end-of-life engine disposal while minimizing radiation dose to the payload.

The analytical approach taken was to tailor tank aft bulkhead geometry to minimize payload shielding requirements. This is done by (1) controlling the angles for radiation incident on the tank, (2) using the conical aft bulkhead to achieve dual cell benefits, and (3) taking advantage of the inverse square law to attenuate radiation beamed to the payload. These gains are achievable at the expense of reduced structural efficiency and increased operation complexity. However, radiation shielding for manned flights represents the major design driver and controlling factor in achieving best overall performance.

The analytical investigations were initiated by optimizing the performance of the single tank design for the lunar shuttle mission and then assessing the operational impact including launch to orbit. Also, the single tank optimum design point was employed to evolve attractive dual cell and hybrid concepts, aimed at improving an already acceptable overall performance.

The loading impact of selected configurations on the INT-21 was next investigated. The proposed INT-21 has been upgraded recently by NASA to accept the loads imposed by a standard payload envelope, 33 feet in diameter and 141 feet in length including a biconic nose cone. The in-flight winds were restricted to 50 meter/sec compared to the 75 meters/sec for the Saturn V, thus reducing launch availability. This was done to minimize the uprating requirements of the S-IC and S-II. Alternate launch modes as shown in the chart were investigated to mitigate the loading impact on the INT-21 and thereby minimize development cost of the booster.

The non-integral launch requires in-orbit mating and checkout of the engine or auxiliary tank module with the main LH₂ tank as well as design of the astrionics and propellant management subsystems for inverted launch.

PHASE III RNS CONFIGURATION CLASSES & LAUNCH MODES

CONFIGURATION CLASSES

LAUNCH MODES

DIMENSIONAL AND RELATIVE RADIATION DOSE DATA

A parametric analysis on a single tank conical aft bulkhead configuration was conducted to determine the "optimum" geometry with respect to flight performance consistent with the radiation dose constraint to the tank top of 10 REM and compatible with the INT-21 boost vehicle structural capability. Vehicle size was compatible with the ground ruled LH₂ capacity of 300,000 pounds. The tank configuration consisted of three basic segments: a forward elliptical bulkhead of 1.5 aspect ratio, a cylindrical section 33 feet in diameter and a conical aft bulkhead. The latter's geometry was varied by altering the half cone angle from 5 to 15 degrees and the cap radius from 25 to 125 inches. The results of the analysis are depicted in the adjacent chart. Stage length variation including engine and astrionics bay is given in the left-hand graph. Overall length difference between extreme configurations is approximately 90 feet. This length difference is, of course, accountable to both the change in half cone-angle and cap radius. However, the latter is the stronger parameter. For example, increasing the cap radius from 25 to 125 inches decreases the stage length for the 5-degree half cone-angle by 74 feet or nearly 85% of the maximum stage length variation. Surface area variation of the tanks is summarized in the center graph. Over the range of the parameters being considered it is less than 25%, whereas that of length is 190%. Since surface area can be related to both weight and heat leak, this result implies that a relatively small inert weight variation can be expected between the tank geometries under study.

With reference to a conical aft bulkhead geometry of 15 degrees half cone-angle and 100-inch cap radius, the last set of curves indicate the relative gamma kerma dose at the tank top based on a residual LH₂ of 5000 pounds at engine shutdown. Both Pressure Vessel and Reactor Assembly (PVARA) and external sources contributions are included. The curves reflect closely the stage height findings, i. e., half cone-angle is not as strong a parameter as cap radius. For instance, at 100-inch radius the change in radiation dose is 25 percent while that at 25-inch radius is 60 percent. At the 5-degree, 25-inch radius the relative radiation dose is only 4.5 percent of that of the reference point. The underlying causes for these results are: (1) greater source-to-tank top detector separation distance, (2) reduced effective energy deposition and scattering centers in the aft end of the propellant tank, and (3) greater depths of the LH₂ propellant column for radiation attenuation at any specified time during the tank draining cycle, for a fixed propellant capacity. For example, the single tank configuration employing an 8-degree half cone-angle, 25-inch end cap radius, exhibits a source to detector separation distance of 561 inches (46.75 feet) greater than a configuration employing a 15-degree, 100-inch cap radius for a fixed LH₂ tank capacity of 300,000 pounds. Additionally, the depth of the LH₂ column at the end of the engine burn (5000 pound LH₂ residual) is 198 inches greater for 8-degree half cone-angle, 25-inch end cap radius than a 15-degree half cone-angle, 100-inch end cap radius. This increased column depth represents approximately two photon relaxation lengths in LH₂.

DIMENSIONAL & RELATIVE RADIATION DOSE DATA

STAGE WEIGHT VARIATION

Following the initial dependent variables determination, i. e., stage height and surface area and relative gamma dose, the parametric analysis of the single tank conical aft bulkhead proceeded with the structural as well as environmental protection studies, the latter including meteoroid and thermal protection, and external radiation shielding sizing. The results are presented in the adjacent chart as a function of half cone-angle and end cap radius. In addition to the tank, the stage dry weight curves include forward and aft skirts; foam and high performance insulation for ground and space thermal protection; meteoroid protection for three years at 0.995 probability of no impact to the tank wall; and fixed weight components consisting of auxiliary propulsion, astrionics, thrust structure, etc. The resultant weights indicate a maximum difference between extremes of approximately 2,600 pounds, the majority of which is in the meteoroid and thermal protection system which responds to the surface area variation. Since the latter also affects the heat input to the tank the center graph depicts the related increase in effective weight due to boil-off. A maximum variation of approximately 1,300 pounds between geometry extremes is noted. The final curves in the chart present the external shield weight required for a tank top integral dose criterion of 10 rem. Shield weights were derived from PVARA as well as for external on-axis integral tank top dose contributions compatible with a residual LH₂ capacity of 5,000 pounds (for after-shutdown cooling requirements) at termination of last burn. The shield weights vary from approximately 2,000 pounds for the 5-degree, half cone-angle and 25-inch cap radius to 13,000 pounds for the 15-degree, 100-inch cap radius design point. As can be seen in the chart, shield weight is essentially insensitive to cone-angle variations at large end cap radii (≥ 100 inches). This is due to the small variation in tank length, height of propellant column towards the end of the last burn, and reactor core view angle at large cap radii discussed in the previous chart.

STAGE WEIGHT VARIATION

DESIGN TRADEOFF MAP

The results of the parametric analyses including structure, thermal and meteoroid protection, boiloff penalties, and external shield are expressed in the opposite graph in terms of payload weight variation with tank geometry. An 8-degree half cone-angle with a 25-inch cap radius was found to yield near-optimum performance. It is possible that performance may continue to increase somewhat beyond this point as shown by the dotted lines in the figure; however, only parametric radiation analysis has been performed beyond the noted design point and consequently, the results need further verification. The data as depicted in the chart show the 8-degree half cone-angle, 25-inch cap radius configuration to have a 21,000 pound payload gain over the reference 15-degree half cone-angle, 100-inch cap radius. This payload gain results from a 9,000 pound variation in external radiation shield weight between the two designs less a 1,300 pound increase in structure weight for the longer tank. The net weight difference of 7,700 pounds yields the payload gain previously quoted compatible with the lunar shuttle mission.

The loading impact of the selected configuration on the INT-21 was next investigated. The proposed INT-21 has been upgraded recently by NASA to accept the loads imposed by a standard payload envelope, 33 feet in diameter and 141 feet in length including a biconic nose cone. The in-flight winds were restricted to 50 meters/sec compared to the 75 meters/sec for the Saturn V, thus reducing launch availability. This was done to minimize the uprating requirements of the S-IC and S-II. Since the selected RNS single tank configuration length, including nose cone and NERVA, is approximately 220 feet, alternate launch modes were investigated to mitigate the loading impact on the INT-21. The selected approach corresponds to a non-integral launch mode, i.e., engine and stage delivered to earth orbit separately via space shuttle and INT-21, respectively, with subsequent mating and assembly in orbit. The stage atop the booster in an inverted position results in a 165-foot-long payload. A flexible body dynamic analysis at the max ($q \alpha$) condition at wind speeds of 75 meters/sec for a 100 percent launch availability resulted in loadings on the INT-21 below those experienced with the 141-foot NASA baseline payload. This is due to the low air loading induced by the slender 8-degree cone coupled to an aft movement of the center of pressure.

The maximum acceptable tank length for the configurations under study atop the INT-21 (inverted position) is 190 feet. This, as shown in the chart, is also the ground facilities (KSC) constraint. The limit height of a right-side-up RNS launch configuration with no engine but with a nose cone must be restricted to 141 feet and is shown in the lower right-hand side of the figure. The launch availability of this configuration is the same as that of the NASA baseline INT-21, that is, less than 100 percent. The data show that to preclude further modifications (and increased development cost) to the INT-21, this configuration must be limited to a 12-degree half cone-angle with approximately a 90-inch cap radius. Therefore, an inverted launch of the stage is necessary to maximize payload and cost effectiveness without increasing the development cost of the baseline booster. Furthermore, only with an inverted launch of the nuclear stage less NERVA, relaxation of the wind criteria, or an off-optimum performance design concept such as a larger half cone-angle and tank cap radius can the impact on the booster be avoided. It is also of importance to note at this point, that a non-integral launch requiring in-orbit mating and assembly neither creates new requirements nor violates any existing ones, because NERVA design criteria specify engine and stage assembly and disassembly capability in earth orbit.

DESIGN TRADE-OFF MAP

MODIFIED SINGLE TANK CONFIGURATION (DUAL CELL)

The adjacent chart depicts the configurational arrangements of the RNS using an inner cell with a capacity commensurate with the last cooldown plus residual propellant requirements identified as 5,950 pounds. The objective is dual in nature: (1) by trapping propellant within the inner cell, the requirement for the lower capillary barrier would be eliminated, and (2) the resultant column of propellant can be used effectively as a radiation attenuation shield in the critical latter seconds of burn when the radiation dose rate is reaching its peak.

Since the selected single tank baseline of 8-degree and 25-inch cap radius was shown to be optimum, the inner cell concept study was confined to that configuration only. A number of inner cells geometries were considered by varying the radius of the cylinder. The limiting upper radius was 60 inches when the bulk-head becomes tangent to the sidewalls of the cone.

The optimum cell geometry was based on the minimization of the algebraic sum of tank and inner cell structure as it increases with pressure and of shield weight as it reduces with increasing inner cell height. The increase in pressure is equal to the propellant transfer line height from outer to inner cell, multiplied by the density and acceleration and reaches a maximum when the outer cell is near depletion. Losses in and at the transfer line intake are relatively small. The results of the analyses indicated that the optimum configuration has an inner cell top radius of 50 inches.

As the drain history of this configuration parallels that of the single tank to the 5,950-pound LH₂ level, a time interval of only 10 seconds is available for secondary gamma scattering effects to occur in the hydrogen gas between the inner cell and the outer tank wall prior to engine shutdown at the end of the TEI burn. Additionally, there are fewer available scattering centers due to the drastically reduced volume between the inner cell and outer tank as can be seen in the configuration outline shown. However, the design shows a 1,450-pound structural weight penalty over the single tank design due to the weight of the inner cell, additional capillary devices, and an increase in tank wall thickness resulting from additional pressure line losses and hydrostatic head to the top of the inner cell.

MODIFIED

DUAL CELL

SINGLE TANK CONFIGURATION

HYBRID CONFIGURATIONS

The hybrid configuration exhibits an auxiliary tank mated to the NERVA. This approach is attractive from the standpoint of ground mating with the engine, amelioration of start-up pressurant when the main tank has a large ullage, and disposal of the engine.

In addition to the aforementioned factors, consideration also must be given to other significant design drivers such as nuclear radiation, system weight and shuttle cargo bay compatibility for the auxiliary tank. A spectrum of hybrid configurations shown on the opposite page were synthesized for an initial screening prior to a more detailed trade study to identify the most attractive candidate in this class. A wide range of auxiliary tank geometries and capacities were screened on the basis of shuttle cargo bay dimensional compatibility, producibility, radiation scattering, LH_2 capacity in relation to disposal capability, and interface compatibility with the main tank. Various main tank cap radii and half cone-angles were considered in the context of radiation attenuation and stage mold line symmetry. Auxiliary propellant tank capacity from 3,000 to 9,300 pounds of propellant maintaining compatibility with the space shuttle cargo bay geometry were considered. The 9,300 pounds upper bound stems from recommended safe orbit altitude and inclination propellant requirements for engine disposal identified in orbital operations requirements. The 3,000 pounds propellant capacity yields the most attractive geometry from the standpoint of radiation doses to the payload while maintaining excellent structural compatibility with the main tank. The single tank study results which indicated significant weight reductions for designs with small half cone-angles and cap radii were employed as guide in the hybrid design screening.

HYBRID CONFIGURATIONS

ALTERNATE AUXILIARY TANKS 260 IN.	AFT CAP R (IN.)	FRONT CAP R (IN.)	LH2 CAPACITY (LB)
25	50.7	3000
	25	90	8600
	50	90	9300
	68	82.5	9300
	77 CYLINDER

HYBRID CONFIGURATION EVALUATION

The evaluation of the most promising configurations is summarized in the adjacent chart. The evaluation criteria include vehicle length and weight, external shield weight, and disposal capability from low earth orbit. The minimum empty vehicle weight, integral tank top radiation dose and external shield weight are seen to occur for the hybrid configuration employing a 3,000-pound capacity LH₂ auxiliary tank (8-degree half cone-angle, 25-inch cap radius). Shielding weight advantages of about 2,800 to 3,400 pounds are derived from the 25-inch end cap radius in the lower auxiliary tank of this hybrid configuration as well as the overall vehicle length of 211 feet. However, the reduced LH₂ capacity of 3,000 pounds yields a high altitude disposal orbit at an inclination that could pose future safety problems with increased space traffic. The hybrid configuration employing an auxiliary tank with a 7.5-degree half cone-angle and 68-inch end cap radius aft bulkhead geometry; and a 7.5-degree half cone-angle, 112-inch end cap radius aft bulkhead geometry for the main tank, although heavier than the latter configuration has an improved end-of-life disposal capability and therefore should be retained as a potential hybrid configuration.

HYBRID CONFIGURATION EVALUATION

CONFIGURATION ⁽¹⁾	EVALUATION CRITERIA				
	DISPOSAL CAPABILITY FROM LOW EARTH ORBIT	VEHICLE LENGTH (FT)	EMPTY WEIGHT (LB)	TANK TOP RAD DOSE ⁽²⁾ (REM)	EXTERNAL SHIELD WEIGHT (LB)
MAIN TANK - 8 ⁰ , 40 IN. R CAP AUX TANK - 77 IN. R CYL 9300 LB-LH ₂	660 X 660 NMI 45 ⁰ INCLINATION	209	80400	88	6000
MAIN TANK - 8 ⁰ , 40 IN. R CAP AUX TANK - 8 ⁰ , 25 IN. R CAP - 3000 LB-LH ₂	660 X 660 NMI 32.5 ⁰ INCLINATION	211	78950	32	3800
MAIN TANK - 8 ⁰ , 77 IN. R CAP AUX TANK - 77 IN. R CYL - 9300 LB-LH ₂	660 X 660 NMI 45 ⁰ INCLINATION	193	79810	107	6600
MAIN TANK - 7.5 ⁰ , 112 IN. R CAP AUX TANK - 7.5 ⁰ , 68 IN. R CAP - 9300 LB-LH ₂	660 X 660 NMI 45 ⁰ INCLINATION	186	79230	109	6600

⁽¹⁾ AUXILIARY PROPULSIVE MODULE COMPATIBLE WITH EOS PAYLOAD CAPABILITY; GN&C MODULE
REQUIRED FOR ENGINE DISPOSAL

⁽²⁾ 5000 LB LH₂ RESIDUAL LEVEL

CANDIDATE SYSTEMS SUMMARY

The opposite chart presents a brief summary of the most attractive design concepts in each of the configuration classes considered. The burnout weight including shielding (directly relatable to payload performance and cost effectiveness) shows that the single tank design has an advantage of more than 3,000 pounds over the 8-degree hybrid, and more than 6,000 pounds over the 7.5-degree design. It also shows an advantage of 1,500 pounds over the modified dual cell. In the case of the latter, the weight difference is due to the higher tank design pressure discussed earlier plus the weight of the inner cell assembly.

Over 2,500 pounds of the penalty for the 7.5-degree hybrid are due to the higher external shield weight. The other 3,500 pounds are due to increases in stage weight empty over the single tank design. In the case of the 8-degree hybrid, the configuration shows a small external shield advantage over the single tank. However, the weight empty difference in favor of the latter is also close to 3,500 pounds. These are basically in structure, and meteoroid and thermal protection, and are due to increases in surface area of both pressurized and unpressurized shells as well as a slight increase in tank design pressure due to losses accrued by transferring propellant from the main to auxiliary tank. From the standpoint of delivery to earth orbit, all configurations are about equal. They all require an inverted launch on the INT-21 to preclude modifications to the S-II and S-IC. Engine disposal favors the hybrid employing the auxiliary tank. However, an inoperable NERVA makes it mandatory to rely on an alternate vehicle such as the space tug for engine removal and disposal.

On the basis of the data summarized in the chart, the single tank design employing INT-21 launch of the RNS tank in an inverted attitude is the most attractive configuration while meeting current standards for onboard personnel protection and therefore has been selected as the baseline vehicle.

The hybrid configuration employing an auxiliary propellant tank with a capacity of about 3,000 pounds offers an attractive structural arrangement from the standpoint of radiation attenuation to the payload and consequently, payload performance. However, further work is necessary in this configuration to establish safe orbit disposal requirements including stability and control of the vehicle.

CANDIDATE SYSTEMS SUMMARY

ITEM	SINGLE TANK	MODIFIED DUAL CELL	HYBRID	
MAIN TANK HALF CONE ANGLE (DEGREES)	8	8	7.5	8
MAIN TANK CAP RADIUS (IN.)	25	25	112	40
AUXILIARY TANK LH ₂ CAPACITY (LB)	-	-	9,300	3,000
AUXILIARY TANK CAP RADIUS (IN.)	-	-	68	25
TANK TOP RADIATION DOSE* (REM)	38	37	109	32
TANK DESIGN PRESSURE (PSIA)	27.5	28.5	28.2	28.2
EMPTY WEIGHT (LB)	75,540	77,000	79,230	78,950
EXTERNAL SHIELDING WEIGHT** (LB)	4,050	4,050	6,600	3,800
BURNOUT WEIGHT - INCL EXT SHIELDING (LB)	81,450	82,910	87,700	84,610
ENGINE DISPOSAL (HI EARTH ORBIT)				
NORMAL MODE***	TUG	TUG	AUX TANK	AUX TANK
INOPERABLE NERVA	TUG	TUG	TUG	TUG
RELIABILITY IMPLICATIONS	NO SIGNIFICANT DIFFERENCE BETWEEN CONCEPTS			
RECOMMENDED CONFIGURATION	✓			

*5000 LB LH₂ - LAST COOLDOWN PROPELLANT

**10 REM CRITERIA

***APPLICABLE FOR STAGE LIFE EXCEEDING ENGINE

SUBSYSTEMS DESIGN

The subsystems were evolved concurrent with the stage configurations. The emphasis in the analyses in this case was placed in the investigation of interactions with other subsystems/systems and commonality with current NASA programmed efforts. In addition, simplicity of manufacturing, inspection and repair, as well as maintainability, and safety and reliability requirements were leading drivers in the design and selection of each subsystem and the components thereof. Weight and performance complete the list of variables in the tradeoffs performed for the selection of each component.

SUBSYSTEMS DESIGN

SUBSYSTEMS

- STRUCTURE
- THERMAL PROTECTION
- PROPELLANT SYSTEM
- ASTRIONICS
- REACTION CONTROL SYSTEM
- ELECTRIC POWER SYSTEM

PRIMARY DESIGN CONSIDERATIONS

- COMMONALITY WITH CURRENT PROGRAMMED SYSTEMS
- INTERACTION WITH OTHER SUBSYSTEMS
- STATE-OF-THE-ART
- MANUFACTURABILITY
- SAFETY & RELIABILITY
- PERFORMANCE
- WEIGHT

STAGE STRUCTURAL SUMMARY

The accompanying chart summarizes the design criteria and conditions, and the construction and materials of the various elements of the stage. The ultimate and limit safety factors indicated under manned and unmanned criteria apply to both pressurized and unpressurized structures and are multipliers for the applied load. At limit, the resultant load must not exceed the yield stress usually identified as 0.2% offset point on the stress strain curve, and must be able to sustain the ultimate load without failure, albeit with permissible permanent deformation.

In the case of pressurized shells, the factors of safety are applied where pressure is the critical design condition, as in the hoop direction of the pressure vessel. When the pressure is relieving, as in the longitudinal direction of a cylindrical pressure vessel, a factor of 1.0 is applied to the pressure.

The lifetime is consistent with the duration of 10 round trips, i. e., three years and the meteoroid probability of no penetration has been identified as 0.995 for this duration.

The max ($q\alpha$) is the design condition of the tank sidewalls stiffening and forward and aft skirts. It is based on an attitude-attitude rate control system for the maximum wind condition compatible with a 95% probability of no occurrence combined to (0.85) 99% wind shear build-up with an embedded jet gust of (0.85) 9 meter/sec. The resultant angle of attack is 10.1 degrees at $q = 735$ psf and occurs at 73 seconds in the flight. The loads compatible with ground winds defined as 64.4 knots at 60 feet altitude considered the effects of a damper and are based on wind tunnel data. However, they are only applicable to the S-IC and S-II stages of the INT-21. The shear distribution pattern over the payload, in this case the inverted RNS stage, due to the damper are not well defined. Therefore it was assumed that the damper is located at the intersection of the nose cone and cylindrical section of the RNS and a maximum limit bending moment of 25×10^6 inch-pound results. The thrust structure is designed by the engine thrust loads, and the tank membrane by the hoop load due to maximum differential pressure which occurs during space operations.

The tank sidewalls when subjected to ground winds must resist, unpressurized, an ultimate loading of 254 lb/in. at room temperature and 365 lb/in. during max ($q\alpha$) at -423°F with an internal relieving pressure of 25.5 psig. The limit load for the latter condition is a tensile load of 493 pounds. The compressive load at ultimate affects approximately one-third of the cylindrical section of the tank, i. e., 12 feet. The tank wall stiffening is as shown in the chart. The forward and aft bulkheads are designed by pressure differential and therefore are monocoque construction. However, since the conical aft bulkhead is 100 feet long, handling requires that formers be included to retain the shape of the cross-section. The spacing will be determined by manufacturing considerations. The membranes are tapered to match the hoop loading. A minimum thickness of 0.030 inches is assumed, achieved by chem-milling. The design loading for the skirts is dictated by the max ($q\alpha$) condition and varies in ultimate loading from 1,600 lb/in. at the base of the aft skirt to 3,980 lb/in. at the forward end of the forward skirt. The construction identified for the skirt, i. e., integral stiffening at 5-inch pitch, at the end of Phase II study is retained based on simplicity of manufacturing. The fiber glass 4-foot-long heatblocks are 2-inch honeycomb sandwich with 0.040-inch facings. The thrust structure material, titanium alloy, was selected on the basis of low conductivity (relative to aluminum) and nuclear activation (relative to steel). The design condition is the engine thrust plus the shear resulting from 3 degrees of gimbal.

The meteoroid protection/thermal insulation composite is unchanged from Phase II and is designed for a three-year life at 0.995 probability of no penetration.

STAGE STRUCTURAL SUMMARY

- DESIGN CRITERIA

MANNED (1.05 PROOF, 1.1 YIELD, 1.4 ULTIMATE SAFETY FACTORS)
UNMANNED (1.05 PROOF, 1.1 YIELD, 1.25 ULTIMATE SAFETY FACTORS)
3 YEAR LIFETIME (99.5% METEOROID PROBABILITY OF NO IMPACT)

- MAJOR DESIGN CONDITIONS

GROUND WINDS - 64.4 KNOTS AT 60 FT
MAX (q_a) - ATTITUDE RATE CONTROL SYSTEM
95% WIND PROBABILITY OF NO OCCURRENCE, 99% WIND SHEAR & GUST
MAX AXIAL ACCELERATION AT S-IC CENTER ENGINE CUT-OFF
NERVA THRUST - 75,000 LB
PRESSURE DIFFERENTIAL - 27.5 PSIG AT -250°F

- TANK CONSTRUCTION & MATERIALS

SIDEWALLS - SKIN (0.104 IN.) + 0-90° WAFFLE (18 X 36 X 1.5 X 0.13) AI 2014-T6
BULKHEADS - TAPERED MONOCOQUE - RING FORMERS ON AFT BULKHEAD AI 2014-T6

- SKIRTS CONSTRUCTION & MATERIALS

COMPOSITE - INTEGRAL LONGITUDINAL STIFFENING + FRAME AI 7075-T83
FIBERGLASS HONEYCOMB SANDWICH HEAT BLOCK (4 FT LONG)

- THRUST STRUCTURE CONSTRUCTION & MATERIAL

SKIN (0.10 IN.) - RINGS - Ti-6Al-4V

- METEOROID PROTECTION

DOUBLE BUMPER ENCASING GAC-9 HPI (2.0 IN. SPACING)
FIBERGLASS SHEETS - 0.030 IN. OUTER, 0.010 IN. INNER

THERMAL PROTECTION DESIGN

The RNS operating lifetime is three years, although the thermal cycle is only 28 days. Based on Apollo data, the optical properties of the thermal control coatings have been degraded from an initial value of $\alpha_s/\epsilon = 0.18$ to $\alpha_s/\epsilon = 0.4$ to account for the anticipated degradation of long-term exposure to solar flux and particles. The application of the lower ratio may become feasible in the future, but no currently available coatings ($\alpha_s/\epsilon = 0.18$) have proven practical for long mission duration, large surface area application. The time averaged temperature of 332°R is based on detailed environmental analyses that take into consideration vehicle optical properties as well as orientation throughout the mission profile. It should be noted that no solar shield considerations have been included, but such influences would be effective in reducing heat transfer.

Both semi-rigid and spacerless insulations were considered in the tradeoff analyses to select an HPI system. The candidates considered include GAC-9, DAM-NM, NARSAM, and SUPERFLOC. GAC-9 was selected as the baseline high performance insulation with SUPERFLOC as an alternate. SUPERFLOC exhibits higher performance than GAC-9, but the latter offers a higher degree of development as well as better handling, installation, inspection and repair characteristics. However, it is of importance to note that this choice could change depending on the results of current NASA funded technology studies as well as the space shuttle development program.

THERMAL PROTECTION DESIGN

REQUIREMENTS

- 28 DAYS THERMAL CYCLE
- 3 YEAR MISSION LIFE

OPTICAL PROPERTIES

- $\alpha_s = 0.36$
- $\epsilon = 0.90$

SURFACE TEMPERATURE

- T (TIME AVG) = 332 R

SELECTION

- GAC-9 (BASELINE)
- SUPERFLOC (ALTERNATE)

THERMAL PROTECTION PERFORMANCE SUMMARY

A summary of the results of the thermal protection analysis performed for the lunar mission are presented in the adjacent chart employing the baseline single tank design. The vehicle payload performance variation is shown with respect to insulation type and thickness for two propellant management concepts, i.e., continuous boiloff venting, and destratification with zero boiloff. The initial vapor pressure employed for the comparison is 18 psia.

The continuous boiloff concept is an ideal system and was employed as a baseline to establish payload performance trends and thickness of insulation required for detailed propellant thermodynamic analyses. It assumes utilization of the hydrogen propellant heat of vaporization to consume the tank heat leak. The propellant tank is continuously vented at a rate equal to the hydrogen boiloff rate which is directly proportional to the heat leak through the tank structure, and insulation systems. Because of the low tank pressure associated with this approach, a substantial weight saving in tank pressure is possible. However, this weight saving must be compared to the amount of propellant lost by venting the boiloff.

The destratification-zero boiloff approach utilizes the liquid hydrogen as a heat sink to eliminate boiloff losses. All heat leaks through the propellant tank structure and insulation system are absorbed by the propellant bulk. The bulk temperature rise during the mission establishes the tank pressure design requirements. Although this method conserves valuable propellant by eliminating boiloff venting, it can be costly in terms of additional tank weight.

The results show that for the continuous boiloff system GAC-9 optimizes at a thickness of 1.5 inches. SUPERFLOC, on the other hand, optimizes at 1.3 inches and shows a payload improvement of 10,000 pounds over GAC-9. The destratification-zero boiloff concept shows a degradation in performance when applied with GAC-9 insulation. This is due to the increase in operating pressure to 38 psia compared to the 27 psia required for the continuous venting system. It also shows a shift in the optimum insulation thickness which is due to the tradeoff between tank and insulation weight. When employed with SUPERFLOC, the zero boiloff concept shows a slight improvement over the continuous boiloff. In this case, the savings in boiloff more than compensate for the increase in insulation thickness and tank pressure to 30.5 psia. This is due to the more effective, and lower density characteristics of SUPERFLOC compared to GAC-9. Also shown in the chart is the payload effect of 1-degree propellant subcooling with SUPERFLOC. The payload increase over the non-subcooled, zero boiloff concept is approximately 6,500 pounds and it is due to the possible reduction in tank pressure and insulation thickness.

As previously mentioned, SUPERFLOC exhibits higher performance than GAC-9, but the latter offers a higher degree of development as well as better handling, installation, inspection, and repair characteristics. For these reasons GAC-9 is currently favored. However, this choice could change depending on the results of current NASA funded technology studies as well as the space shuttle development program.

THERMAL PROTECTION PERFORMANCE SUMMARY

THERMODYNAMIC OPTIMIZATION

The two operational and propellant management design techniques studied are classified as destratification and passive propellant management. The former requires mixing and/or an ullage spray to assure a uniform temperature and thermodynamic equilibrium. In this case, the heat leak is taken up by sensible heating of propellant, and venting is precluded. Passive propellant management, on the other hand, exploits the inherent characteristics of the hydrogen, and the shape and size of the RNS. The characteristics utilized are the large stage volume, high fineness ratio, large ullage created by the TLI and LOI burns (2/3 of propellant utilized), ullage stratification during pressurization (for a diffuser-type pressurization inlet), and the natural tendency for hydrogen to stratify because of heat leak. Passive propellant management utilizes vertical ullage temperature gradients that occur during pressurization and the slow approach to thermodynamic equilibrium (long conduction path, small gradients) to reduce boiloff and burnout residuals.

For either of the two design techniques, one of the important objectives is development of a synergistic combination of propellant management with pressurization and thermal control. For example, for the passive propellant management technique, a capillary barrier is utilized for bulk propellant control to prevent liquid-vapor mixing. This capillary barrier also functions to augment feedout and minimize sloshing. Other objectives are operational simplicity and adaptability to interface elements. Elimination of the in-orbit vent requirement for the destratified technique is an example of the former. Ability to utilize depot propellant over a fairly wide range of temperatures and pressures is an example of the latter. Finally, as propellant and tankage represent the major portion of the stage weight, substantial savings in payload are potentially available through sound innovation and careful design tradeoffs.

THERMODYNAMIC OPTIMIZATION

- OBJECTIVES

- OPTIMIZE SUBSYSTEMS SATISFYING NERVA REQUIREMENTS

- OPERATIONAL SIMPLICITY

- COMPATIBILITY OF PROPELLANT MANAGEMENT WITH PRESSURIZATION & THERMAL CONTROL

- DESIGN APPROACHES

- PASSIVE PROPELLANT MANAGEMENT

- DESTRATIFICATION - ZERO BOILOFF

- INTERFACE ELEMENTS AFFECTING DESIGN

- PROPELLANT REFUELING IN ORBIT - 15.5 PSIA, 36.8°R

- PRESSURANT TAPPED FROM TURBINE EXHAUST

- PUMP PERFORMANCE

ULLAGE PRESSURE AND LIQUID VAPOR PRESSURE PROFILES

The approach taken in the development of a passive propellant management thermal control and pressurization methodology is to avoid the use of active mixing techniques and exploit the inherent characteristics of the RNS. Therefore, such devices as liquid mixers and ullage sprays, which attempt to maintain thermodynamic equilibrium, have been purposefully avoided. In addition, venting is limited to that required to prevent excessive ullage pressure and prevent locally heated regions within the liquid; venting to cool the bulk liquid to some lower temperature has also been avoided. Rather, operational simplicity and high performance have been sought by a technique called passive propellant management.

The approach requires only that energetic mixing of ullage with bulk propellant be avoided. This is achieved by a two-pronged approach: (a) RCS engine sizing and operational logic that minimizes vehicle disturbances and (b) a system of capillary devices to control propellant even if maximum vehicle perturbations do occur. All else follows as a natural consequence of meeting mission requirements utilizing autogenous pressurization. The TLI burn expends about 55% of the loaded propellant. At the end of this burn the ullage phase is highly stratified with temperature varying from 37° R in the liquid to about 230° R at the top of the tank with the average temperature about 160° R. Because of the large stage size and low g field (primary heat transfer mechanism is conduction), equilibrium is only slowly restored. Furthermore, heat leak into the ullage is of the same order as heat flow from the ullage to the walls and liquid. Inasmuch as this basic situation prevails throughout the mission, a hot stratified ullage is the expected thermodynamic condition. At first thought this situation may appear undesirable, but further consideration reveals that it can provide distinct advantages for the RNS, as discussed subsequently.

The thermodynamic analysis was conducted using the ELF computer program and a single node analysis. The ELF program¹ is a multinode analysis developed by NI Rocketdyne for NASA-MSFC and for NR's S-II program. The program determines ullage pressures, pressurant flow rates, gas and wall axial temperatures and gradients, heat transfer rates, and evaporation or condensation rates. It was utilized for preconditioning, bootstrap, steady burn, tailoff, and early cooldown analysis of the RNS. In principal, it is applicable to later cooldown and coast, but in practice computer cost for these long periods are prohibitive. Therefore, a single node method was developed to analyze these mission phases. This analysis assumes constant specific heat and ideal gas behavior, but does not assume perfect mixing and is entirely applicable for a stratified ullage. The single node analysis requires an independent source of information on heat and mass transfer rates. This information was obtained from several sources, particularly the work of Clark² and Blatt³. These data were cross checked and compared with ELF program predictions to assure accuracy. The ELF program and single node analysis were used in tandem to determine the thermodynamic condition of ullage and propellant for a nominal four-burn lunar mission.

One of the major results, the ullage pressure and liquid temperature history for the mission is presented in the adjacent figure. Nuclear heating of the liquid has been conservatively estimated from Phase II analytic results. Solar heating of the liquid was determined by apportioning the total heat leak between gas and liquid based on wetted wall area, inner wall temperatures, and Blatt's³ correlations of orbital flight data. Important features of the ullage pressure profile are the pressure histories during cooldown and coast following TLI and LOI. In the 3.3 days following TLI burn, pressure decreases monotonically to 21.8 psia at LOI burn initiation -- heat leak mitigating somewhat the rate of pressure collapse. During the 17.5 days in lunar orbit following the LOI burn, heat leakage rate eventually exceeds ullage cooling processes and pressure builds up again until at about 105 hours after LOI burn (186 hours into the mission) venting is required.

Venting has been delayed to this time by using the liquid as a heat sink to absorb part of the heat leak. Due to the advent of venting, bootstrapping is no longer required for the TEI and EOI burns. Maximum pressurization flow rate occurs during bootstrapping during LOI burn. Tank design pressure (27.5 psia) has been established for the critical case -- malfunction pump operation for the EOI burn.

1. M. Epstein and R. E. Anderson, "A Generalized Propellant Tank Pressurization Analysis," Advances in Cryogenic Engineering, Vol. 10, 1965
2. J. Clark, "A Review of Pressurization, Stratification and Interfacial Phenomena," Advances in Cryogenic Engineering, Vol. 10, 1965
3. M. Blatt, "Empirical Correlations for Pressure Rise in Closed Cryogenic Containers," J. Spacecraft and Rockets, Vol. 5, No. 6, 1968

ULLAGE PRESSURE & LIQUID VAPOR PRESSURE PROFILES (PASSIVE PROPELLANT MANAGEMENT)

THERMAL MANAGEMENT TECHNIQUES SUMMARY

Two techniques for the management of propellant and pressurant have been analyzed, passive propellant management and destratification.

The former approach avoids active systems such as liquid mixers. Capillary devices are employed for bulk propellant control. These devices provide effective feedout and slosh control for all mission phases. Passive propellant management exploits the ullage temperature gradients that occur during pressurization as well as the capillary devices provided for feedout and slosh control, and the large size and geometric shape of the RNS to decrease boiloff and residuals.

Tank design pressure, 27.5 psia, is determined by the malfunction (single pump) requirement for the last burn. Venting requirements are reduced by using LH_2 as a heat sink for a portion of the heat leak and a thermal control approach that allows venting at high temperatures. Heat leak is reduced by maintaining a dry wall in the ullage region. Total boiloff losses for the mission are 1,360 pounds.

Analyses conducted showed that heat transfer rates from the gas, and mass transfer at the interface is low. Calculated evaporation at the interface is less than 100 pounds for the entire mission. However, evaporation will increase if there is appreciable liquid sloshing. This is not anticipated, but considering the lack of definition of this problem area allowance was made for 1,000 pounds of evaporated propellant. Thus, the trapped residual is the initial ullage plus the pressurant added plus 1,000 lb totaling 2,890 lb.

Destratification requires mixing and/or an ullage spray to assure a uniform temperature and thermodynamic equilibrium. In this case all of the heat leak is taken up by sensible heating of the propellant and venting is precluded. Considering nuclear heating, solar heating, heat generated by the mixer, and the effect of pressurant, the tank pressure required to eliminate venting is 35.0 psia. Final residual H_2 was based on an average gas phase temperature of 67°R as the use of saturated vapor temperature is clearly overly conservative.

Payload performance was determined for the two methods of thermal management. The results show that passive propellant management yields a 30,000 pound payload gain over the system employing destratification.

THERMAL MANAGEMENT TECHNIQUES SUMMARY

PASSIVE PROPELLANT MANAGEMENT

REQUIRES AN IN-ORBIT VENT SYSTEM

REQUIRES CAPILLARY BARRIERS FOR BULK PROPELLANT CONTROL

- EFFECTIVE FEEDOUT
- SLOSH CONTROL

UTILIZES INHERENT SIZE & GEOMETRY OF STAGE

LIQUID HEAT SINK

DELAYED HIGH TEMPERATURE VENTING

INCREASED DRY WALL AREA

SIGNIFICANT PAYLOAD GAIN THROUGH REDUCTION OF DESIGN PRESSURE & LOSSES

● TANK PRESSURE (PSIA)	27.5
● BOILOFF (LB)	1,360
● RESIDUAL TEMPERATURE (R)	125
WEIGHT (LB)	2,890
● LUNAR PAYLOAD CAPABILITY	
(OUTBOUND/RETURN, 1,000 LB)	166/20

BASED ON 1.5 INCHES OF GAC-9 INSULATION

DESTRATIFICATION-ZERO BOILOFF

AVERAGES HEAT INPUT IN THE FLUID

REQUIRES MIXER AND/OR SPRAY TO MAINTAIN THERMAL EQUILIBRIUM

ELIMINATES VENTING

● TANK PRESSURE	35.0
● BOILOFF (LB)	0
● RESIDUAL TEMPERATURE (R)	67
WEIGHT (LB)	7,660
● LUNAR PAYLOAD CAPABILITY	
(OUTBOUND/RETURN, 1,000 LB)	136/20

ASTRIONICS, CONTROLS, AND POWER SUBSYSTEMS

Astrionic system definition effort encompasses a wide range of activities. These include definition of GN& C requirements to satisfy mission requirements, selection of GN& C equipment to satisfy the requirements, and selection of RCS and electrical power subsystems.

ASTRIONICS ,CONTROLS & POWER SUBSYSTEMS

- GUIDANCE, NAVIGATION & CONTROL
- REACTION CONTROL SYSTEM
- ELECTRICAL POWER

GUIDANCE AND NAVIGATION

An autonomous capability for orbit-to-orbit operations is recommended to provide maximum flexibility for carrying out RNS missions. The capability minimizes the need of earth based operations other than for monitoring, flight scheduling, and resupply. Another obvious advantage to providing the RNS with a completely autonomous capability is that redundant equipment is provided for back-up by the MSFN during orbit-to-orbit transfers, including trans-earth and translunar flights, and by both the MSFN and another autonomous vehicle during rendezvous and docking.

Automatic rendezvous and docking with manual override is consistent with the autonomous capability requirement. Also there are some obvious disadvantages to requiring a man-in-the-loop for rendezvous and docking. Visual cues would have to be provided either by having a man on board one of the vehicles or by having TV. If the man were on the earth or EOSS, for example, time lags introduced by data relay satellite systems could present a problem.

Geometrical and error sensitivity considerations were used to establish mission requirements for phasing orbit accuracies prior to a rendezvous transfer. These mission requirements translate into constraints which the GN&C must satisfy. For phasing orbits these constraints translate into establishing the orbit with sufficient accuracy to allow the RNS radar to detect a target vehicle with 99.9% probability at a desired detection range and within prescribed scan limits so that RSS of delta-V's to correct cross range altitude position errors and velocity errors in all three axes is less (within a 99.7% probability) than could be corrected by a delta-V of the same magnitude as that required for the rendezvous Hohmann transfer.

Using the phasing orbit as a baseline orbit, the GN&C is then required to accomplish translunar, transearth, and low earth orbit - geosynchronous orbit transfers so that injection into the new trajectory results in the RNS entering desired arrival corridor with a midcourse correction of 50 fps or less. Finally, the phasing orbit requirements must be met following insertion into a new orbit after these transfers.

The driving selection criterion has been commonality with other space elements, especially the space shuttle, and space station. In addition, inherent redundancy of the equipment has been emphasized to accrue the desired high reliability and safety necessary for the operation of the reusable nuclear stage. Also, ease of maintenance, performance accuracy, weight and state-of-the-art have been considered in the evaluation criteria employed in components evaluation and selection.

GUIDANCE & NAVIGATION

REQUIREMENTS

- AUTONOMOUS CAPABILITY FOR ORBIT-TO-ORBIT OPERATIONS
- AUTONOMOUS RENDEZVOUS & DOCKING WITH MANUAL OVERRIDE
- PHASING ORBIT ACCURACY FOR RADAR DETECTION RANGE (100-820 N MI) & RENDEZVOUS ΔV (34 FPS)
- ORBITAL INSERTION POSITION & VELOCITY END CONDITIONS MEETING PHASING ORBIT ACCURACIES

PHASING ORBIT ACCURACIES

- ± 3 N MI, ± 10 FPS CROSS RANGE
- ± 20 N MI, ± 15 FPS DOWN RANGE
- ± 1 N MI, ± 15 FPS ALTITUDE

SELECTION CRITERIA

- COMMONALITY WITH OTHER SPACE ELEMENTS
- INHERENT REDUNDANCY
- STATE-OF-THE-ART
- PERFORMANCE
- WEIGHT

GUIDANCE, NAVIGATION & CONTROLS COMPONENTS SUMMARY

A strapdown IMU is recommended for the RNS as a result of a detailed NR/SD trade study. The strapdown comparison performed resulted in the MICRON system being the preferred IMU candidate. MICRON, which is the acronym for "micro navigator", is a lightweight, low cost system that incorporates large scale integrated electronic technologies. The main constituents of the MICRON are two small electrostatically suspended gyros (ESG's), a miniature 4096-word/24-bit metal oxide semiconductor computer, servo electronics, power supply and battery, environmental control, and structural housing. The system weighs three pounds, occupies less than 90-cubic inches and consumes about 12 watts of power. The gyro has only one moving part, a solid ball, supported electrostatically. The micron-ESG can serve as an accelerometer as well as a gyro; therefore, two instruments per system provide complete information. Existing MICRON systems exhibit a 0.04 deg/hr rms random drift rate and a 2,000 MTBF. Projected capabilities of the spacequalified instrument are a 0.01 deg/hr and 16,000 hours, respectively. To provide a high reliability, use of four MICRON systems per vehicle is anticipated. Present accuracy of the MICRON acceleration sensing capability is in the order of 10^{-4} . Six additional accelerometers are included in the power, weight, and volume to cover any requirements for greater accuracy.

Vehicle inertial attitude can be determined by several methods. Determining line-of-sight to two stars was selected for the RNS because of its simplicity, accuracy, and applicability to all phases of the reference lunar mission. Mechanically gimbaled star trackers were selected for the attitude updating. These instruments have the capability of tracking stars within a field of view (FOV) determined by the gimbal freedom. Star acquisition is accomplished by scanning a portion of the celestial sphere in which a known star is expected. Since gimbaling can provide a large FOV, these instruments are, in general, less constrained operationally than other types evaluated.

A horizon sensor and an unknown landmark tracker (ULTRA) were selected for state vector position updating on the RNS. Horizon sensors used for the RNS must cover a large range of subtended angles. Discussions with Quantic Industries have indicated the feasibility of having a two optical assemblies in each sensor head creating a fully redundant 4-head system.

Autonetics has developed an ULTRA which can be used to provide good ephemeris data in earth or lunar orbit. However, it is not effective at altitudes above 1000 n.mi because of orbital resolution limitations. Thus, it complements the capability of the horizon sensor.

The Information Management System (IMS) includes digital computer(s) and communication and tracking subsystems. A multiprocessor computer system was selected for the RNS because of the maximum flexibility it allows, the expansion capability both for added processing and growth to planetary mission capability, and its lower size, weight, and power. Additionally, a multiprocessing system can be developed in parallel with the other subsystems because of its capability to adapt to changes in the subsystem requirements.

The Goddard Tracking and Data Relay Satellite (TDRS) system appears attractive for providing continuous communications. However, preliminary calculations indicate that a 3 to 13-foot antenna, depending on hardware (pre-amp, etc.), characteristics would be required to compensate for a TDRS nondirectional antenna link with the RNS. Direct links with earth would require much smaller antenna size and/or transmitter power, but provide only intermittent coverage in low earth orbit. Since no requirement for continuous communication has been identified, a system transmitting intermittently, while in contact with the MSFN, was selected. In the event that MSFN S-band channels are allocated, K-band frequencies at windows in the oxygen and water vapor absorption spectrums appear attractive. The UHF system provides omnidirectional vehicle-to-vehicle, EVA, and short range ground-RNS communications.

The X-band radar provides for rendezvous and may be time shared for vehicle-to-vehicle communications. Operating with a cooperative target, the radar, with a 2-watt transmitter and a 30-cm diameter parabolic antenna, has a detection range greater than the 820 n mi requirements for geosynchronous altitude autonomous rendezvous. The scanning laser radar accomplishes automatic docking and also provides for tracking a cooperative target at ranges up to 75 n mi.

GUIDANCE, NAVIGATION & CONTROLS COMPONENTS SUMMARY

REACTION CONTROL SYSTEM SUMMARY

The RCS provides the vehicle with delta-V capability in cases where the nuclear engine cannot be used; i.e., three axis attitude control with the nuclear engine shut down and roll control with the nuclear engine operational. Since the NERVA is gimbaled, the RCS is not required to counteract thrust misalignment. The most critical control maneuver demanded is docking with a payload in lunar orbit since a mission abort due to a single point failure at this time would be unacceptable. Also, although the docking maneuver is to be automatic, manual docking backup is required. Nevertheless, the maneuver requirements established to date for the lunar shuttle mission permit a relatively leisurely manner of accomplishment. That is, no maneuver needs to be accomplished in a few seconds; several minutes or longer can be taken. Additionally, continuous burn time of as much as 1,000 seconds is considered acceptable for thrusting. This design approach reduces the accelerations required for any given maneuver and permits employing relatively low level thrusters.

Of the concepts considered, the supercritical storage gaseous O_2/H_2 system is recommended with four quads located on the forward skirt. The supercritical system exhibits a minimum amount of development problems.

The storage concept for both hydrogen and oxygen has been used in previous spacecraft so valuable development experience is already available. Additionally, this type of system eliminates the problem of propellant acquisition since supercritical storage assures single phase delivery regardless of the gravitational conditions.

The major area requiring development is propellant conditioning. This is basically accomplished by heating the propellants in a heat exchanger which is provided energy from the flow of hot gases from an O_2/H_2 gas generator. Because of the heavy storage tanks and number of components, this system weighs more than the others. However, due to the high total impulse requirements for the RNS, the system hardware weight penalty has minimal impact on the selection.

REACTION CONTROL SYSTEM SUMMARY

SELECTION CRITERIA

- COMMONALITY WITH OTHER SPACE ELEMENTS
- DEVELOPMENT STATUS
- SIMPLICITY OF OPERATION

ELECTRICAL POWER SUBSYSTEM SUMMARY

The RNS self-sufficing concept requires that all electrical energy requirements be provided from on-board equipment during each mission cycle. Power cycles of 28 and 55 days have been investigated with a minimum stage lifetime of three years. Electrical power during dormant periods other than those in the mission cycle is assumed to be supplied from the propellant depot and/or other power modules. The subsystem concept and component selection is based on maximizing commonality with other space program elements (i. e., shuttle, EOSS, etc.) and minimizing special logistic requirements, where practical.

The redundancy requirements are dictated by the mission including considerations for manned payload missions. Three separate and independent busses are to be provided to satisfy redundancy requirements. Automatic circuitry is to be provided to permit paralleling of energy sources, inverters and/or converters with minimum transient effects and for load protection. Both back-up and emergency power requirements are considered in the final concept selection.

The electrical energy requirements identified in the chart are based on the "NR nominal" lunar mission. The stay time in lunar orbit is approximately 17 days, 13 of which the RNS is assumed to be in a dormant mode. Since the minimum energy transfer opportunities occur every 54.6 days, it has been assumed the RNS will be in a dormant condition for 24 days in earth orbit, allowing time for stage checkout, rendezvous with the propellant depot, and transfer of propellants.

Energy requirement of NR's representative lunar mission from time of earth departure to return to earth operations orbit is shown to be 800 kwh. The total energy requirement for a 54.6 day turn-around time is 1200 kwh. It should be noted that the long cooldown time and the long dormant times are the major contributors to the overall electrical energy requirement even though the power levels are modest.

ELECTRICAL POWER SUBSYSTEM SUMMARY

REQUIREMENTS

POWER CYCLE: 28-55 DAYS
0.5 KW (DORMANT)
POWER LEVEL: 1.75 KW (SUSTAINED)
5.75 KW (PEAK)
TOTAL ENERGY: 800 - 1200 KWH

CONCEPTS

FUEL CELL/BATTERY

SOLAR ARRAY/BATTERY

SOLAR ARRAY/REGENERATIVE
FUEL CELL

SELECTION CONSIDERATIONS

- LOW WEIGHT FOR 30 TO 60 DAY MISSION
- REACTANTS AVAILABILITY FROM RCS TANKS
- FLEXIBILITY IN PROVIDING PEAK LOADS
- LOW COST-DEVELOPMENT BY SPACE SHUTTLE
- ORIENTATION PENALTY
- OPERATIONAL COMPLEXITY
- HEAVIER BATTERY DUE TO LUNAR OPERATIONS
- SIMILAR TO SOLAR ARRAY/BATTERY
- REQUIRES LONG TIME FOR REGENERATION

SUBSYSTEM DESIGN

FUEL CELLS (3) 2 KW EACH
LIFETIME: 2000 HRS (CURRENT)
10,000 HRS (DEVELOPMENT)
INVERTERS (3) THREE-PHASE, 400 HERTZ
BATTERIES (2) SILVER-ZINC, ONE-YEAR LIFETIME

RELIABILITY/SAFETY ANALYSIS

Reliability and safety investigations during Phase III were oriented to analyses of the major RNS subsystems in the alternate stage configurations with emphasis on the selected baseline design, i.e., a single tank conical configuration with an aft bulkhead geometry employing an 8-degree half cone-angle and a 25-inch end cap radius. Reliability and safety analysis techniques were utilized throughout the study to ensure both a high probability of mission success for candidate RNS missions and to eliminate potentially hazardous mission operations that could endanger the safety of the earth's population and/or personnel in space.

Reliability studies of the major subsystems and components resulted in the elimination of all single point failures with the exception of the main propellant tank. Reliability logic diagrams, fault tree analysis (FTA) and failure mode, effects and criticality analyses (FMECA) were the principal tools employed in the reliability analyses of the subsystems based on current NASA-accepted fail operational/fail safe (FO/FS) and fail operational/fail operational/fail safe (FO/FO/FS) criteria for mechanical and electrical parts and components, respectively. The incorporation of high redundancy within each subsystem and a maintainability concept of RNS mission applications that was designed to include provisions for periodic replacement of parts and components, represented the dual approach employed during Phase III to ensure both reliable and safe system design and operation.

Both nuclear and non-nuclear safety implications of RNS launch operations at KSC were investigated. The well-established safety procedures and exclusion areas for Saturn V launch vehicles were found to be adequate for both the space shuttle and INT-21 launch operations of the RNS. The current nuclear safeguards employed by the NERVA engine for the ground test program appear adequate to eliminate potentially hazardous occurrences during launch operations.

Normal flight operations were investigated and shown to be within established nuclear safety criteria. NR concepts for RNS mission trajectories were tailored to avoid NERVA engine firings in the vicinity of other space program elements. Ancillary operations such as rendezvous and docking, crew transfer, orbital refueling, and maintenance have been shown feasible for the RNS baseline design if conducted at the forward end of the vehicle. However, operations near the aft end of the main tank and in the area of the engine/stage interface were shown to be limited due to the NERVA post-shutdown radiation environment. However, additional shielding can be minimized if sufficient decay time is allowed after the last engine firing prior to scheduled operations. Additionally, an unmanned space tug affords a potentially attractive method for performing certain space operations at the aft end of the vehicle (e.g., engine disposal).

Performance and trajectory analysis of lunar shuttle missions for abnormal or emergency operations could conclude with the RNS in a highly elliptical earth or lunar orbit. It was determined that safe return of the RNS crew to the earth operations orbit or to an OLS by means of a space tug is feasible and would adequately accommodate the majority of emergency situations. An emergency operations matrix for the lunar shuttle mission was developed to implement the contingency planning process. This process, in turn, was utilized to identify key or critical operations procedures to be developed for various mission segments during the evolution of the RNS program.

RELIABILITY/SAFETY ANALYSIS

RELIABILITY CRITERIA

- FO/FS FOR MECHANICAL
- FO/FO/FS FOR ELECTRICAL
- LONG LIFE WITH MAINTENANCE
- MAN RATING
- HIGH MISSION SUCCESS PROBABILITY

SAFETY CRITERIA

- MINIMIZE HAZARDS TO
 - EARTH'S INHABITANTS
 - SPACE PERSONNEL
 - RNS CREW/PASSENGERS

FAILURE MODES

- IDENTIFY THE FAILURE MODES
- DETERMINE THE CAUSES OF THE FAILURE MODES
- DETERMINE THE EFFECTS OF THE FAILURE MODES

EFFECTS AND CRITICALITY

- DETERMINE THE EFFECTS OF THE FAILURE MODES
- DETERMINE THE CRITICALITY OF THE FAILURE MODES
- DETERMINE THE PRIORITY OF THE FAILURE MODES
- DETERMINE THE MITIGATION OF THE FAILURE MODES

FAILURE AVOIDANCE

CONTINGENCY ACTION

TYPICAL FAILURE MODES, EFFECTS AND CRITICALITY RELATIONSHIPS

Three principal analysis techniques were employed during Phase III to investigate the reliability and safety implications of the RNS for candidate mission applications.

Reliability logic diagrams were developed and fault tree analyses were performed for each of four major RNS subsystems, i.e., main propulsion, RCS, EPS and GN&C to establish potential failure modes and effects as well as to assess compliance with previously established FO/FS and FO/FO/FS criteria.

Concomitantly, a failure modes, effects and criticality analysis (FMECA) was performed for each subsystem to determine the possible consequences that might arise from specific parts and component failures with respect to mission success probability and safety of both personnel in space and the inhabitants of the earth. Consequently, three general criticality categories were established. Criticality I - That hardware which, if failed, could cause injury to personnel or loss of mission. Criticality II - Hardware failures which could cause scrub of a mission, the adoption of an alternate mission mode, or which would require an alternate mode of operation. Criticality III - Hardware failures which downgrade the mission but would not seriously impair the mission as originally planned.

FMECA relationships of typical failure modes and effects are shown in the accompanying chart for representative components within a particular subsystem. It must be emphasized that although a large number of failure effects are possible they can be traced to two primary ones, i.e., loss of propulsive capability and/or loss of guidance, navigation, or control capability.

The Phase III study indicated that the main propellant tank was the only single point of failure classified as Criticality I. Certain other structural elements, e.g., thrust structure, also fall within this category. Since provisions for redundancy are obviously not feasible in these instances, adequate safety margins must be included in the design and verified during the test program. In addition, a comprehensive quality control program must be implemented during the fabrication phase to assure the quality of the manufacturing process and, by non-destructive testing techniques, to detect structural flaws or other areas of potential failure in order that they can be eliminated prior to vehicle use.

A highly redundant capability was incorporated into subsystem design and is reflected in the accompanying chart.

TYPICAL FAILURE MODES, EFFECTS, & CRITICALITY RELATIONSHIPS

SUBSYSTEM	COMPONENT	FAILURE MODE	POSSIBLE FAILURE EFFECT					CRITICALITY CATEGORY
			THRUST LOSS	ENGINE DISASSEMBLY	ELEC POWER LOSS	CONTROL LOSS	PROPELLANT LOSS	
MAIN PROPULSION	MAIN LH2 TANK	RUPTURE	X	*			X	I
	PROPELLANT SHUTOFF VALVES	OPEN/CLOSED	X				X	II
	COOLDOWN CONTROL & SHUTOFF VALVES	OPEN CLOSED	X	X			X X	II
	PRESSURIZATION CONTROL VALVE	OPEN/CLOSED	X				X	II
RCS	PROPELLANT TANKS	RUPTURE				X	X	II
	GAS GENERATOR PRESSURE REGULATOR	CLOSED				X	X	II
	RELIEF & ISOLATION VALVES	OPEN				X	X	II
EPS	FUEL CELLS	INOPERATIVE	X		X	X		III
	BATTERIES	INTERNAL SHORT, STRUCTURAL	X		X	X		III
	AC BUS	COMPLETE FUNCTION LOSS	X		X	X		III
G,N,&C	IMU STAR TRACKERS HORIZON SENSOR MULTIPROCESSOR COMPUTER	GYRO FAILURE INOPERATIVE HEAD FAILURE PARTS OR CIRCUIT FAILURE	X	*		X X X X		III

*EFFECT FOLLOWS LOSS DURING MAIN BURN OR COOLDOWN

FAILURE AVOIDANCE (GN&C RELIABILITY LOGIC)

The GN&C subsystem is a typical example of mission failure avoidance or minimization through design modifications which provide compliance with FO/FO/FS criteria. The system as illustrated consists of four MICRON's, six accelerometers, three star trackers, a four-head horizon sensor, unknown landmark tracker, various radars, communications and a multi-process computer. The accelerometers and MICRON's incorporate internal redundancies as well as being redundant to each other. Triple star tracker redundancy is provided for precise attitude determination. The landmark tracker provides a degraded backup mode, and is also redundant to the horizon sensors. The x-band radar is the primary means for rendezvous, with backup by ground tracking or the target vehicle. Similarly, the laser radar and TV are redundant for docking and are backed up by target vehicle equipment. The computer has no backup but has extensive internal redundancy.

Full compliance of this system with FO/FO/FS criteria is essential because of potential accidents involving RNS earth impact with a large inventory of fission products. These potentialities emphasize the requirement for a highly reliable engine control system (NDIC) as well as the GN&C.

High mission reliability is provided by basic design enhanced by maintenance between missions. The developed approach to maintainability provides for ensuring the high mission success probability desired for extended RNS mission applications. The maintainability concept generally consists of replenishment, as in the case of the main propellant tank, or removal and replacement types of operations. The latter type includes both scheduled and unscheduled replacement.

The effect of maintenance on cumulative mission reliability is also illustrated in the accompanying chart. Redundancy has been assumed for the example because virtually all components (excepting structures) operate in some form of redundancy. Curve A represents the cumulative mission reliability of redundant star trackers and indicates an 0.891 probability of at least one tracker functioning after 6,400 hours (ten missions) of use without any maintenance. A mean time before failure of 16,000 hours (manufacturer's data) was assumed. It can be seen that the probability of successfully completing one mission cycle of 640 hours is 0.9985; this probability is also valid for the second mission if no failures have occurred to that point. A probability of 0.891 is indicated for ten missions. Curve B indicates the increase in mission reliability which can be attained over a span of ten missions by effecting the necessary maintenance between missions. This principle is applicable to most RNS equipment even though the redundant elements may be dissimilar (e.g., TV camera and laser radar).

FAILURE AVOIDANCE (GN&C RELIABILITY LOGIC)

CONTINGENCY ACTION

(LOSS OF THRUST/TUG RESCUE)

Failure effects involving degraded propulsion capability will generally cause termination of RNS flight in an elliptical orbit at earth or moon. The return of the RNS and/or its payload to the earth operations orbit or to an OLS is envisioned as a task for an unmanned space tug. Assessment of such a rescue action has limited consideration to single stage tugs of up to 80,000 pounds propellant capacity. In the event of complete propulsion failure, the RNS terminal orbit may be hyperbolic. Rescue in this situation is beyond tug capabilities and appears doubtful by any means.

The payload weight which a tug can retrieve is shown in this figure as a function of delta-V which the tug must generate to reach the payload. Assumptions are that the tug is based in circular orbit (260 n mi at earth or 60 n mi at moon), that it transfers to and docks with the payload using the indicated delta-V, and that it returns with the payload to the original orbit using the same delta-V. Retrieval from highly elliptic earth orbit coupled with possible plane change requirements can escalate the tug delta-V well above 10,000 fps. The curves show that a single stage tug is not well suited to that velocity regime.

Superimposed on the tug capability curves is a weight history of a typical lunar mission presented in terms of delta-V above circular orbit velocity during the main maneuvers. RNS propulsion failure during or prior to a maneuver poses a rescue task for the tug which can be roughly measured by the weight and delta-V level existing at the time of failure. Without detailed consideration of rendezvous and plane change requirements, it is nevertheless apparent that a tug can in many situations effect return of the RNS with payload and without propellant dumping. At other times, for example between TLI_1 and TLI_2 , an 80,000 pound capacity tug would not be able to return the RNS, even after propellant dumping. Alternatives in this case include separate return of RNS and payload or use of a two-stage tug.

CONTINGENCY ACTION

(LOSS OF THRUST/TUG RESCUE)

SAFETY/RELIABILITY CONCLUSIONS

Safety is of paramount importance for all space systems. However, the NERVA engine imposes unique requirements that must be met to ensure a high degree of mission success probability and minimum potential operational safety hazards that might endanger either personnel in space and/or the earth's population. This is achievable through a reliability/design approach that emphasizes practical components redundancy with inherent multifunction and compensating capabilities. A good example in this case is the guidance, navigation, and control system. The design selected not only offers a practical degree of redundancy, but the components can satisfy more than one function. Also, as in the case of the multiprocessor computer, the component has self-check capability and can compensate for an internal malfunction.

In-space checkout and trend data analysis permit incipient failure detection thereby permitting corrective maintenance action. These combined with redundancy assures minimum probability of catastrophic failures.

The inherent capability of NERVA to operate at reduced thrust for a long period of time enhances the operational safety capability of the system. Performance and trajectory analysis for lunar shuttle missions for abnormal or emergency operations involving degraded propulsion capability will generally conclude with the RNS in a high elliptical earth or lunar orbit. In this case, safe return of the RNS crew can be effected to the earth operations orbit or to an OLS by means of a space tug (80,000 pounds LH_2 propellant capacity). Another approach to the rescue operation, is the institution of a life-boat integrated with the payload. In this manner, the crew can separate from an inoperable RNS and return to an operational orbit or to an orbit within reach of a space tug.

SAFETY/RELIABILITY CONCLUSIONS

SAFETY IS OF PARAMOUNT IMPORTANCE FOR ALL SPACE SYSTEMS

- EARTH POPULATION
- SPACE PERSONNEL

NERVA IMPOSES UNIQUE DESIGN & OPERATIONAL CONSIDERATIONS

- NUCLEAR RADIATION ENVIRONMENT
- FISSION PRODUCT INVENTORY

REQUIRED RELIABILITY/DESIGN GUIDELINES

- COMPONENT REDUNDANCY, & MULTIFUNCTION CAPABILITY
- COMPENSATING SYSTEMS
- ELIMINATION OF SINGLE POINT FAILURES
- IN-SPACE MAINTENANCE

REQUIRED OPERATIONAL PROCEDURES

- INHERENT NERVA SAFEGUARDS
- COUNTERMEASURES
- WELL CONCEIVED & REHEARSED NORMAL & EMERGENCY OPERATIONS
- INCIPIENT FAILURE DIAGNOSTICS & MANUAL OVERRIDE CAPABILITY

REFERENCE STAGE DEFINITION

This section of the report covers the definition of the reference RNS stage incorporating the selected single tank design and subsystems into a fully integrated vehicle system. Detailed designs of subsystems as well as interfaces are presented.

REFERENCE STAGE DEFINITION

- DIMENSIONAL DATA
- SUBSYSTEMS INTEGRATION

THERMAL/METEOROID PROTECTION

PROPELLANT SYSTEM

ASTRIONICS

- INTERFACES

REFERENCE STAGE DESIGN

The layout of the selected baseline configuration is shown in the adjacent chart depicting the general arrangement of the vehicle. The single tank of 33-foot cylindrical diameter and 8-degree half cone angle with 25-inch cap radius aft bulkhead is 1827 inches long and accommodates 300,000 pounds of propellant with 5 percent ullage. Extension skirts are added to the forward and aft ends of the cylindrical portion for attachments of INT-21 booster and payload, and aerodynamic shroud, respectively. A thrust structure, also in the form of a skirt is provided at the end of the aft bulkhead to accommodate the engine installation. An active neuter docking system and supporting cone structure are incorporated in the forward skirt for stage docking to a propellant depot, maintenance element, or similar facility and for payload module connection. An annular astrionics bay for installation of the stage auxiliary subsystems is also integrated into the forward skirt structure. Another active neuter docking system is built into the thrust structure to facilitate orbital installation of the engine, as well as for orbital handling of the stage and NERVA during removal and disposal operations. The passive assembly of the neuter dock is attached to the engine forward thrust structure and the active assembly is attached to the stage thrust structure. The tank is compartmentalized by three perforated capillary barriers and a bottom screen to facilitate the passive propellant management during space operations.

A typical cross-section of the thermal/meteoroid protection system is also shown. It indicates the foam sublayer sprayed on to the outer surface of the tank wall and fluted longitudinally to permit venting and outgassing of the meteoroid/thermal protection composite. The latter consists of two shells of fiber glass, 0.030 and 0.010 inch for the outer and inner skins encasing three one half inch panels of GAC-9.

Both bumpers are continuous cylinders with the outer sheet beaded to impart enough stiffness to prevent local flutter. The two sheets are kept apart by rows of radial fiber glass posts spaced approximately 8.6 inches circumferentially and 8 feet longitudinally. The insulation panels "hang" from these posts. To avoid imposing excessive bearing loads on the "flimsy" aluminized mylar layers of the HPI, dacron straps are used on either side of the panel at each post location. These straps are used to discretely pick up the weight of the insulation at the density control pins. The latter consists of 30 mil nylon pins which are spaced longitudinally in line with each post and to which is applied a thin adhesive coating. By special techniques developed at NR/SD, precision perforations in the HPI panels are formed without incurring piercing or tearing of the insulation. The pins are then inserted in the perforations and bonded to the individual HPI layers of the panel in a single operation. The pin ends are bonded to the dacron straps.

The inner bumper is perforated to allow the purge gasses to escape into vertical channels precut in the tank sidewall closed cell sprayed foam.

REFERENCE STAGE DESIGN

SECTION A-A
ROTATED 90°

PROPELLANT ACQUISITION AND LOCATION CONTROL CONFIGURATION

The adjacent chart details the design proposed for RNS propellant management. Capillary barriers divide the tank into four major compartments named to denote their major function, beginning at the top of the tank they are the ullage compartment, the bulk propellant compartment, the cooldown compartment, and the restart compartment

The bulk propellant capillary barrier is mounted and supported at the intersection of the cylindrical and aft bulkheads just above the propellant level after the TLI. The barrier consists of two thin perforated plates with 0.060 inch chemically milled holes. The fraction of open area to total plate area for each plate is 0.10. The plates are supported by radial and circumferential frames and are conically shaped for structural purposes, with apex located two feet above base. The barrier will prevent gas-liquid interchange between compartments during rotation rates and lateral, negative, and centrifugal accelerations due to vehicle maneuvers and perturbations. Design features are based upon latest available technology. For example, fraction openness is based upon experimental data from work by Gluck* and by Paynter and Barksdale**. The small openness ratio effectively segregates ullage gas from propellant, yet results in negligible flow pressure drop during the TLI burn.

The restart and cooldown compartments assure adequate propellant flow at each engine restart. Hole size and fraction openness, while different for each compartment's capillary barrier, were established in much the same manner as discussed above. The gap between the plates provides a wicking and resealing capability. These compartments are thermally cooled by a thermodynamic vent system attached to the outside of the tank, as part of the in-orbit venting system. The restart compartment, which contains 500 lbs of propellant, supplies vapor free propellant at each engine restart. This compartment is also designed to assure adequate feedout and small trapped residuals for the last phase of the last engine cooldown. During cooldown propellant is fed from the restart compartment to the collector compartment (Sec A-A).

The cooldown compartment supplies the bulk of the cooldown propellant during the mission. This compartment is sized for 'worst case' hydrodynamics for the lunar mission. Cooldown following the TLI burn presents no problems as the bulk propellant compartment is essentially full and liquid flow to the cooldown compartment and on to the cooldown line - is assured. Likewise gas free liquid flow occurs at LOI burn initiation. The situation can be different after the LOI burn as the bulk propellant compartment is then partially depleted. This compartment will be settled by the early cooldown flow (0.4 - 1.7 lb/sec). However, during later cooldown flow (0.0 - 0.7 lb/sec pulsed flow) propellant could be dislocated to the upper end of the bulk propellant compartment. This would lead to vapor passage from the bulk compartment to the cooldown compartment as cooldown flow proceeds. As vapor accumulates within the cooldown compartment, vapor passage to the restart compartment is precluded by the wicking capability of the V-shaped wicking clusters. These clusters are sized to provide cooldown flow to the restart compartment under zero to minus 10^{-5} g's. At TEI burn initiation bulk propellant can still be dislocated in the bulk and cooldown compartments. The wicking clusters cannot provide adequate flow at engine burn initiation and, therefore, the cooldown compartment must be settled by RCS propellant prior to the burn, 40 lbs of RCS propellant is sufficient to settle the restart compartment. As this is not sufficient to settle the bulk propellant, a small amount of vapor could enter the cooldown compartment while thrust build-up occurs. During TEI steady burn, it is expected that vapor will be 'burped' out of the cooldown compartment at the barrier vertex. This is the basis for the small cone angle and large open area specified for the cooldown barrier.

This same sequence of events will occur during TEI cooldown and EOI restart. An estimate of the maximum amount of vapor that could accumulate in the cooldown compartment has been made. No allowance was made for mitigation of this problem by vapor burping during steady burn. A total of 5,350 lbs of liquid could be displaced by vapor due to adverse location during cooldown flow, withdrawal of vent fluid from the cooldown line, and restart flow. Therefore, the cooldown plus restart compartments have been sized for a capacity of 5,350 lbs plus 5,950 lbs (11,300 lbs total), the 5,950 lbs is the maximum propellant needed for the TEI cooldown.

Feedout during this last cooldown requires wicking clusters in the restart compartment. Four V shaped clusters are required to lead fluid into the collector compartment. V-shaped channels are specified as they are self emptying, thereby, reducing trapped residual in the compartments. The collector compartment capillary barrier is contoured to avoid the two pump outlet lines. Fraction openness increases from 0 to 0.55 with increasing distance from the centerline. This is done to prevent vapor pull through (interface dip) over the outlet line.

* D. F. Gluck, 'Propellant Position Control by Capillary Barriers During Spacecraft Rotational Maneuvers', Journal of Spacecraft and Rockets, Vol. 7, No. 3, 1970.

** H. L. Paynter and F. R. Barksdale, 'Criteria for Passive Propellant Control Schemes', Journal of Spacecraft and Rockets, Vol. 7, No. 6, 1970.

PROPELLANT ACQUISITION & LOCATION CONTROL CONFIGURATION

PROPELLANT SYSTEM

The propellant system depicted in the adjacent chart provides an overview of the propellant fill, drain, feed, vent and pressurization system. Also included are the details of the ground and orbital fill system. Propellant is fed to the NERVA engine as shown in the enlarged view via two propellant feed lines 9.5 inches diameter each through the propellant shutoff valves. Cooldown propellant is supplied through the 3.0 inch-diameter cooldown line and is controlled by the cooldown flow valve (17).

In the upper left-hand corner is shown the ground fill and drain schematic. As ground fill and drain is accomplished with the tank in an inverted position, ground fill and drain is accomplished via line (1) and pumped out (using a different pump) should ground draining be required. A 3.5-inch-diameter line is adequate for this purpose. Just prior to launch, this line is capped off. To assure that inadvertent drainage or venting does not occur during the flight, the valves (2) are normally maintained in the closed position. The ground fill and drain system utilizes, in common, some of the lines of the refueling or emergency vent system (2).

An artificial gravity field for propellant positioning using linear or rotational acceleration is employed to facilitate refueling in orbit in the no-vent mode or can also be used with venting for conditioning of the tank and its contents.

Filling in the no-vent mode utilizes a divider valve (3) which proportions the entering propellant between the fill line (5) and the ullage spray (4). If tank pressure should rise excessively, flow is diverted to the ullage spray (4) to reduce tank pressure, the divider valve operating off tank pressure. The design utilizes an inlet diffuser (6) in the lowermost capillary compartment to reduce and dissipate inlet momentum. The diffuser uses a plug at the end of the refill line, with many holes machined along the length of the tube. The holes are uniformly distributed along the length of the tube within the lowermost compartment. Hole size is not too important as long as the size is greater than about 0.25 inch and less than two inches. Total hole area is at least ten times the cross-sectional area, that is, 1.96 ft^2 , thus reducing the inlet momentum by a factor of about 100. In addition to the effect of the diffuser, the capillary barriers which subdivide the tank into compartments dissipate part of the energy. Using a refueling rate of 30,000 lbs/hr (10 hrs to accomplish refueling) the pressure loss through the valves and the fill line (5) is about 2.2 psi for the 6-inch-diameter line recommended. During ground fill the orbital fill standpipe doubles as a ground vent system since launch of the stage is inverted and the diffuser is in the ullage. The associated vent line and valves for this operation are denoted by (7) in the chart. Two valves in series are normally closed; they open if tank pressure reaches an incipient critical condition. Using two valves in series prevents inadvertent venting if one valve fails open. If one valve fails closed (normal position) venting can be accomplished through the orbital fill system (8) and (3). In this case, both shutoff valve (8) and the divider valve (3) are opened to permit full flow through the fill line (5). Both valves open if they sense a pressure of 1 psia greater than the pressure at which valve (7) is programmed to open.

Preventing pressure excursions and maintaining subcooled propellant within the capillary compartments is controlled by the thermodynamic vent. The system shown in the chart originates at the aft end of the conical aft bulkhead where the thermodynamic vent tubes are manifolded and in turn connected to the cooldown line by way of a throttling valve. The tubes emanating from this aft manifold have a one-half-inch hemispherical cross-section and run in parallel the length of the vehicle to intermediate manifolds until they reach the most forward manifold which discharges overboard through two symmetrically placed vent heads. This design which is in effect an open loop refrigeration system operates by withdrawing liquid through the cooldown line and expanding the liquid to a lower temperature and pressure.

Under nominal conditions the vent side fluid flows to heat exchanger tubes (19) at the bottom of the RNS tank to assure that this important region and the feedout lines receive adequate cooling. The tubes cool the liquid propellant and tank walls, and intercept heat leaks through the high performance insulation. The flow rate is regulated so that the cooling fluid becomes completely vaporized upon reaching the ullage compartment. This is accomplished by flow rate control valves which sense the temperature of the vented fluid, assuring that the vent fluid is substantially superheated.

This system is subject to the override that if tank pressure begins to rise rapidly, cold throttled fluid is by-passed directly to the cooling tubes along the ullage compartment (10). If necessary, all the throttled fluid can be by-passed in this way resulting in rapid cooldown, including possible ullage vapor condensation, of the ullage compartment. The proportioning valves (11) controlling this by-pass operate off tank pressure.

A schematic of the autogenous pressurization subsystem is also shown in the chart. Autogenous pressurization utilizes engine heat to vaporize some of the LH_2 pumped from the tank and thus the hydrogen gas is returned to the tank maintaining the tank's pressure as the expulsion continues. A turbopump system is used to feed propellant to the engine - with turbine exhaust working fluid, supplied via a by-pass from the engine, being used as pressurant. The start-up phase when engine chamber pressure, pump spin rate, and turbine exhaust supply are building up is the critical operational period. Pressurant supply rate requirements are appreciably higher during this period than during steady engine burn rate requirements surging to stay ahead of pump requirements. A bang-bang control system is used operating off tank pressure. Orifice (12) allows less flow than that required for steady engine burn; thus, this flow is supplemented as required with flow through the larger orifice (13) by activation of the on-off valves (14) as required. Orifice (12) plus (13) are oversized for the maximum flow requirement during pressurization. As required during pressurization, the flow through (13) is shut off by valves (14) to give the proper pressure buildup.

Pressurant enters the tank through the diffuser (15). This causes flow to enter horizontally and at low velocities to minimize ullage mixing. However, during pressure buildup and steady burn the pressurant flow rates are 180 and 27 ft^3/sec . These high volumetric flow rates will require a baffle system as well as a diffuser to reduce the incoming momentum of the pressurant.

PROPELLANT SYSTEM

ENGINE-TANK INTERFACE

STANDARD ENGINE

The design illustrates the feasibility of incorporating a standard neuter docking system in the engine/tank interface for automatic orbital mating of the engine and tank. The latest engine configuration is used, with the following modifications: the main feed lines are jogged inboard to accommodate the docking system geometry and to facilitate connection to the 25-inch radius tank cap; and six large electrical connectors of the same total pin area are employed in place of the 46 smaller connectors as specified in the latest ANSC reports. The active docking assembly is incorporated in the stage thrust structure, and a passive docking ring is attached to the engine forward thrust plate. An actuated coupling plate inside of the active docking section on the stage provides for holdback of the line and electrical connections during structural acquisition and lockup, with subsequent positive alignment and coupling of the subsystems.

The stage thrust structure is a titanium skirt sized to attach to the tank conical aft bulkhead at the same station as the lower propellant retention screen and to integrate with the docking installation. The arrangement permits commonality in the tank features for the thrust structure and screen attachments; affords good path length for minimization of thermal leakage; and fully encloses the system elements of the interface.

If the turns in the main propellant lines at the TPA inlets are objectionable, the TPA's can be rotated to align with the propellant lines without significant effect on the overall configuration. Jogging of the lines has the advantage of bringing them closer to the engine gimbal center, thus, reducing the amount of flexibility required to accommodate the engine gimbal motions.

If access for inspection, etc. of the forward side of the interface is desired, the thrust structure can be constructed as an open truss rather than a monocoque tube.

ENGINE - TANK INTERFACE (STANDARD ENGINE)

ENGINE-TANK INTERFACE MODIFIED ENGINE

In order to maximally exploit the radiation shielding advantages of the small (25 inch) aft cap radius of the baseline tank, it would be desirable to contract the engine accessory systems ahead of the external shield - particularly the TPA's which may be significant secondary scatterers. In the current engine configuration, the TPA's are located 35 inches out from the center-line and the valves/plumbing project beyond the nominal 50-inch external shield radius in many places. The preliminary study conducted has indicated that, with a nominal lengthening of the thrust structure ahead of the shield, the TPA's can be pulled in to 22 inches from center and the valves/plumbing contracted to a 38.5 inch radius envelope without otherwise significantly altering the present arrangement and component concepts. The depicted interface design employs this contracted engine configuration.

The stage thrust structure, docking and coupling provisions are the same as for the standard engine configuration with joggled feed lines. The principal difference is that the feed lines do not require joggling. If the engine design can in fact be so modified, additional shield weight savings might be accrued and a simpler interface design accomplished.

ENGINE - TANK INTERFACE (MODIFIED ENGINE)

ASTRIONICS BAY

The astrionics bay contains all the electronic equipment including guidance, navigation, controls, communication, and the NERVA NDIC. In addition RCS, electrical power system, and environmental control are housed in this area.

The distribution of the equipment both within and without the astrionics bay is displayed in the opposite chart. All instrumentation and components identified during the course of the study are shown on the developed views of the inner and outer surfaces of the astrionic module structure.

The majority of the inside surface is covered by the two RCS propellant tankage quadrants with a third quadrant occupied mainly by three fuel cells with their associated inverters and controls. Also shown are the two batteries required during peak NERVA operations. The NDIC is located next to the electrical power system. The last internal quadrant is used for the instrumentation connected with horizon sensors, communication, station keeping and docking, IMS computer and RACU's.

The external surface of the astrionics bay is mostly covered with the environmental control system radiators and by the four RCS pods. In addition, the star trackers, the communication antenna, the ULTRA tracker and horizon sensors and the rendezvous and docking equipment consisting of TV camera, laser radar chaser and target and the X-band radar antenna occupy the remaining available space.

As can be seen, the surface and volume occupied by the electronics and equipment, other than the RCS and its propellant requirement, is relatively small. Their relocation to suit additional or new refurbishment requirements can be accommodated.

ASTRONICS BAY

SUPPORTING RESEARCH & TECHNOLOGY

In keeping with its Phase A status, the NR investigation has defined in considerable detail an attractive RNS concept. The benefits of reusability, commonality, design and operational simplicity, and autonomy were exploited to the maximum practical extent to achieve space flight economy safely and reliably as a key element of NASA's integrated program plan (IPP). With a capability of satisfying a wide range of proposed manned and unmanned shuttle missions, combined with an inherent growth potential for manned planetary exploration, the RNS represents a timely versatile space transportation system for the 1980s and beyond. However, in spite of being judged technically feasible and economically attractive, considerable room remains to resolve outstanding technical uncertainties, establish practical degree of reusability, and maximize cost benefits through commonality with other space elements. The key activities to be accomplished to resolve these critical problems are listed in the accompanying chart. These cover both analytical and testing investigations and are intended to serve as necessary decision-making points prior to Phase D initiation.

Some of the more significant activities include the development of a flutter-free thermal/meteoroid protection design integrating an efficient and easy to produce, maintain, and repair high performance insulation. Additionally, passive propellant management employing a stratified ullage to minimize residuals has shown potential payoffs that are worthy of concentrated investigation. Another area needing immediate attention is the definition of an economical in-space maintenance operations concept that benefits all the currently programmed space elements.

SUPPORTING RESEARCH & TECHNOLOGY

OVERALL OBJECTIVES

- RESOLVE OUTSTANDING UNCERTAINTIES
- ESTABLISH PRACTICAL DEGREE OF REUSABILITY
- ACHIEVE COST BENEFITS THROUGH COMMONALITY

KEY ACTIVITIES

- METEOROID/THERMAL PROTECTION
- OUTER SHELL DYNAMICS
- RADIATION EFFECTS & SAFETY
- PASSIVE PROPELLANT MANAGEMENT
- O₂/H₂ RCS DESIGN
- NAVIGATION ACCURACY
- IMS DEVELOPMENT
- IN-SPACE MAINTENANCE OPERATIONS
- INCIPIENT FAILURE DETECTION

CONCLUSIONS

The preliminary design analyses performed to date have demonstrated that the single tank design (8-degree half cone-angle, 25-inch cap radius) employing INT-21 launch of the RNS tank is an inverted attitude is clearly the most attractive configuration while meeting current standards for onboard personnel protection and therefore is recommended as the baseline design. The hybrid configuration employing a 3,000-pound capacity auxiliary tank offers an attractive structural arrangement from the standpoint of radiation attenuation to the payload and consequently, payload performance. However, further work is necessary in this configuration to verify engine safe disposal capability, and to establish stability and control requirements.

One other aspect examined in the comparison of the single tank with the hybrid consists of possible hot test facility cost savings with the latter configuration. That is, in the hybrid concept, savings might be possible by use of the propulsion module (auxiliary tank plus NERVA) with a facility LH₂ tank for hot testing. Although this approach would permit early initiation of hot testing, the result of the reliability/test cost analysis of four alternate development programs suggest that cost benefits are unlikely.

Of the propellant management concepts considered, passive propellant management is recommended based on the possible residuals reduction -- hence significant payload gains over a destratified-zero boiloff system. Furthermore, when combined with a judicious capillary device design for propellant feed and location control, the integrated system results in improved thermal control and overall operational simplicity.

GAC-9 was selected as the baseline high performance insulation with SUPERFLOC as an alternate. SUPERFLOC exhibits higher performance than GAC-9, but the latter offers a higher degree of development, as well as better handling, installation, inspection, and repair characteristics. However, it is of importance to note that this choice could change depending on the results of current NASA funded technology studies, as well as the space shuttle development program.

Commonality with other space elements was given prime consideration during the study. However, the effort was only partly successful due to the limited programmatic definition existing in some areas. For example, further work is necessary on HPI before commonality with the space shuttle can be attained. In astrionics, a large degree of commonality was established with the space station study. Nevertheless, further equipment definition is necessary before recurring and non-recurring cost benefits can be accrued.

CONCLUSIONS

- SINGLE TANK DESIGN OFFERS HIGHEST PERFORMANCE OF CONCEPTS CONSIDERED
- HYBRID CONFIGURATION EMPLOYING 3000 LB CAPACITY AUX TANK OFFERS ATTRACTIVE STRUCTURAL ARRANGEMENT
- ORBITAL MATING & ASSY OF STAGE/NERVA IS FEASIBLE & NECESSARY TO MAX. PERFORMANCE
- INVERTED TANK LAUNCH FEASIBLE & NECESSARY TO MINIMIZE BOOSTER DEVELOPMENT COST
- SLENDER CONE-ANGLE ARRANGEMENTS YIELD DYNAMICALLY STABLE VEHICLE DESIGNS
- PASSIVE PROPELLANT MANAGEMENT OFFERS DESIGN SIMPLICITY & PERFORMANCE BENEFIT
- GAC-9 OFFERS COMPARATIVE EASE OF INSTALLATION & REPAIR - SUPERFLOC OFFERS POTENTIAL PERFORMANCE GAINS
- COMMONALITY EXPLOITED IN LIMITED WAY -- FURTHER GAINS APPEAR POSSIBLE

BRIEFING OUTLINE

- INTRODUCTION - T. M. LITTMAN
- MISSION REQUIREMENTS & OPERATIONS - A. R. JUSAK
- SYSTEM CAPABILITY - A. R. JUSAK
- SYSTEM DEFINITION - D. GARCIA
- PROGRAM PLANNING - W. G. ANTYPAS
- COST ANALYSIS - T. M. LITTMAN
- CONCLUSIONS & RECOMMENDATIONS - T. M. LITTMAN

PROGRAM PLANNING ACTIVITIES

Phase III program planning activities included four major areas (1) manufacturing requirements, (2) development test requirements, (3) facilities definition, and (4) program scheduling. In manufacturing, the study was oriented towards extending the Phase II results to consider stage configurations exhibiting a high fineness ratio conical aft bulkhead, and alternate launch modes (inverted RNS tank attitude on INT-21 with shuttle delivery of NERVA to orbit). The major areas evaluated included manufacturing feasibility utilizing production techniques applicable to MAF and Seal Beach as well as utilization of existing S-IC and/or S-II tooling, facilities, and GSE. Three alternate manufacturing and assembly techniques were evaluated. Also, a limited analysis of HPI installation was conducted, and a preliminary concept developed.

The objective of the development test program was to establish an approach to RNS testing including operational requirements, test facilities, test articles, and preliminary test schedules compatible with the study guidelines. Test criteria were defined to attain the required operational confidence at each test level commensurate with program goals of low cost transportation, flexibility of operation, reliability, and safety. To accomplish this goal, four alternate ground test programs were evaluated.

Facilities definition included the evaluation of four areas (1) establishing facilities requirements, (2) determining utilization of existing resources, i. e., MAF, Seal Beach, KSC, MSFC, etc., (3) analysis of concurrent utilization of MAF and Seal Beach for RNS/S-IC and RNS/S-II, respectively, and (4) preparation of a preliminary integrated facilities plan.

To assure consideration of the most promising program alternatives, schedules were prepared for the four alternate development test programs. For the selected baseline program, an integrated schedule was prepared covering all major development activities from Phase A-IV through the completion of flight testing and a ten year operational program.

PROGRAM PLANNING ACTIVITIES

MANUFACTURING AND ASSEMBLY TECHNIQUES

An analysis of the single tank conical RNS structure shows that the tank has a basic similarity to the S-IC and S-II structures except for the aft bulkhead structure which has an eight degree half cone angle and 25 inch end cap radius, and the additional requirement of HPI installation.

The manufacturing and assembly techniques considered are based on a concept which is common to the three alternates; i. e., the tank will be assembled from two major assemblies, the forward section referred to as the cylinder section (33-foot diameter), and the aft bulkhead referred to as the conical section. This is necessary because of the length of the stage, and to make maximum practical use of existing resources; i. e., facilities, tooling, GSE, etc. The alternates considered are presented on the opposite page. The main approach is to pattern the major structural assembly breakdown after the concepts verified by the S-IC and S-II techniques.

In alternate I, cylindrical assemblies are utilized for the cylindrical section of the tank as well as the conical aft bulkhead and are weld joined circumferentially to build up the completed tank assemblies.

The prime objective in the development of the second alternate concept was to reduce the number of major structural components. In this case, the tank configuration is assembled by longitudinally weld joining full length, formed, rectangular panel assemblies. The conical aft bulkhead is assembled in three sections longitudinally weld joined from long, formed, tapering panel assemblies. The three aft bulkhead assemblies then are mated and joined to the cylindrical sections of the tank by circumferential welds.

Alternate III makes maximum utilization of the benefits provided by the two aforementioned concepts. Here the tank configuration is weld joined circumferentially from cylinder assemblies as in the first concept, (thus utilizing existing tooling and techniques), but the aft bulkhead is weld joined longitudinally as in the second concept. The closeout of both sections is conducted while the tank is in a horizontal attitude.

MANUFACTURING & ASSEMBLY TECHNIQUES

APPROACH

- IDENTIFY TANK FABRICATION REQUIREMENTS
- EVALUATE S-IC/S-II FABRICATION CONCEPTS
- IDENTIFY CONCURRENT USAGE OF TOOLING/TECHNIQUES

ALTERNATIVES FOR TANK WELD JOINING

	I	II	III
TANK CYLINDERS	CIRCUMFERENTIAL	LONGITUDINAL	CIRCUMFERENTIAL
CONICAL AFT BULKHEAD	CIRCUMFERENTIAL	LONGITUDINAL	LONGITUDINAL
CLOSE-OUT WELDING	CIRCUMFERENTIAL	CIRCUMFERENTIAL	CIRCUMFERENTIAL
TANK ATTITUDE	HORIZONTAL	VERTICAL	HORIZONTAL

BASELINE
CONFIGURATION

MANUFACTURING AND ASSEMBLY TECHNIQUES COMPARISON

The table presents the evaluation of the three manufacturing and assembly techniques considered, and represents the salient characteristics. The check marks represent the advantages of the major areas considered for each of the alternates.

Alternate I provides the advantages of the joint utilization of existing modified S-IC and/or S-II tooling, facilities, and MPHE, and the maximum utilization of key skills. The concept has the disadvantage of requiring a large number of components, a large number of weld joints, a long schedule flow, and a design restricted to the S-IC and/or S-II manufacturing concept.

Alternate II results in a fewer number of major components and a reduced number of weld joints with an increased weld joint reliability. Additionally, it does not impose design constraints, reduces the number of control documents, and effects an improved schedule flow. However, the concept introduces the disadvantages of all new tooling requirements for the RNS. This eliminates the advantage of the joint utilization of existing S-IC and/or S-II tooling and the maximum utilization of key skills.

Alternate III provides the maximum joint utilization of S-IC and/or S-II tooling, facilities, and MPHE, the maximum utilization of key skills, reduces the degree of design constraint, and provides a reduced number of major structural components.

The most attractive and cost effective technique was found to be weld joining the tank cylinder assemblies circumferentially, the conical aft bulkhead longitudinally, and closeout to be weld-joined circumferentially while the stage is in a horizontal attitude. With this fabrication and assembly technique, maximum utilization of S-IC and/or S-II tooling, facilities, GSE, and technical skills is provided. This approach allows for an increase in major assembly size, and consequently this will reduce the number of structural components which contributes to cost avoidance. The analysis showed that additional cost avoidance can be implemented by utilizing existing Apollo tooling and facilities to fabricate and assemble the lower 10-foot length section of the conical aft bulkhead. However, new tooling is required for the aft bulkhead. This requirement is common to the three manufacturing and assembly alternates.

MANUFACTURING & ASSEMBLY TECHNIQUES COMPARISON

ALTERNATES SALIENT CHARACTERISTICS	I CIRCUMFERENTIAL WELD	II* LONGITUDINAL WELD	III COMBINED
JOINT UTILIZATION S-IC/S-II TOOLING	✓		✓
MAX UTILIZATION OF MPHE	✓		✓
MAX UTILIZATION OF KEY SKILLS	✓		✓
CONCURRENT UTILIZATION OF RNS/S-II OR RNS/S-IC	✓		✓
REDUCED NO. OF MAJOR STRUCT COMP	✓	✓	✓
REDUCED NO. OF WELD JOINTS		✓	✓
REDUCED CONTROL DOCUMENTS		✓	✓
INCREASED WELD JOINT RELIABILITY		✓	✓
MINOR FACILITY MODS	✓		✓
IMPROVED SCHEDULE FLOW		✓	

NOTE: AFT BULKHEAD NEW TOOLING COMMON TO THREE ALTERNATES

* NEW TOOLING REQUIRED PECULIAR TO RNS TANK

TANK MANUFACTURING AND ASSEMBLY FLOW SEQUENCE

The LH₂ tank will be assembled by circumferentially weld joining four cylinder assemblies. A forward LH₂ bulkhead, formed by meridian welding 12 gore sections subsequently covered with spray foam insulation, will be circumferentially welded to the upper end of the cylinder section to close out the tank. The forward skirt and docking adaptor, assembled by utilizing mechanical fasteners, then will be attached to the upper cylinder section and the entire assembly placed in an inverted flight position.

The transition section, assembled by vertical weld joining quarter panel sections, will be circumferentially weld joined to the inverted cylinder/forward bulkhead assembly. The aft skirt assembly then will be attached with mechanical fasteners. Aft and intermediate conical bulkhead assemblies, made by vertical weld joining tapered assemblies, will be circumferentially weld joined in a vertical position.

The LH₂ tank assembly and the conical bulkhead assembly will be located in a horizontal position and the circumferential closeout weld will be made. The stage structure then will be moved to a remote test site for pneumostatic testing. After the pneumostatic testing is completed, a post-pneumostatic x-ray and fluorescent penetrant inspection will be performed on all LH₂ tank and conical bulkhead structural welds. The final cleaning operations then will be performed on the interior of the stage. Subsequently, the stage is positioned so that the forward LH₂ bulkhead access opening is attached to a clean room.

Spray foam insulation and HPI will be applied with the stage structure in a horizontal position. The final systems installation, followed by final post-manufacturing checkout, is to be performed with the stage in a horizontal position.

TANK MFG & ASSEMBLY FLOW SEQUENCE

TANK INSULATION

The application and processing of spray foam insulation on the tank will be accomplished by utilizing the equipment and techniques developed for S-II. This contributes to cost avoidance which is one of the main objectives of the manufacturing and assembly technique developed. Although maximum use of existing resources, (i.e., tooling, facilities, and GSE) is considered, new tooling will be required for scalloping (machining) of the foam. Limited analysis was conducted with reference to the installation of the HPI/meteoroid protection shield. To minimize HPI installation costs and handling operations, the maximum size panel subassemblies should be employed.

TANK INSULATION

APPROACH

- MAXIMIZE USE OF AVAILABLE TOOLING, FAC, & TECHNIQUES FOR FOAM
- MINIMIZE HPI HANDLING OPERATIONS

SPRAY FOAM APPLICATION

- USE TECHNIQUE DEVELOPED FOR S-II
- NO NEW APPLICATION EQUIPMENT REQUIRED
- NEW TOOLING REQUIRED FOR MACHINING (SCALLOPING)

HPI & METEOROID INSTALLATION (PRELIMINARY CONCEPT)

- USE SUBASSEMBLY PANELS ~ 5 X 10 FEET
- THREE GAC-9 BLANKETS & OUTER BARRIER
- SUBASSEMBLY PANELS ARE STEPPED FOR NEXT ASSEMBLY OPERATION
- JOINT OVERLAP OF PANELS
- OPTIMIZE PANEL SIZE ASSEMBLIES & JOIN PRIOR TO INSTALLATION ON TANK
- HPI INSTALLATION WITH TANK IN HORIZONTAL ATTITUDE

HPI AND METEOROID SHIELD INSTALLATION

The polyurethane foam layer between the tank outer wall and the inner meteoroid bumper will be generally applied to the completed tank after the pneumostatic test and dye penetrant inspection of the welded joints. In localized areas such as the aft skirt, the foam will be applied prior to the pneumostatic test due to inaccessibility after the aft skirt installation. The foam spacer will be sprayed by rotating the tank under fixed spray heads. Foam machining is accomplished with special tooling developed for the longitudinal groove machining operation.

The analysis conducted with reference to the installation of the HPI, including the meteoroid protection shield, shows that subassembly sections will have edges stepped to provide necessary joint overlap when attached to the next subassembly. Specific joint overlap area and configuration will be determined after detail insulation system design. Size of individual blankets will be fixed by available material widths. Blanket length will be approximately ten feet. Subassemblies consisting of the outer meteoroid bumper, insulation blanket (three layers) and inner meteoroid bumper, will be joined to provide maximum sized sections for installation onto the tank and skirt structures. The quarter section (ten foot only) insulation subassemblies will be installed starting at the forward end where attachment to the structure is accomplished. Major subassemblies will be installed progressively by joining four quarter sections then progressively working the tank length by joining sections of approximately ten feet in length to complete each 360 degree section before progressing on toward the aft end of insulation assembly. The manufacturing plan of providing maximum sized insulation panel sections will reduce time for installation onto the stage structure and permit maximum utilization of manpower at the fabrication and subassembly level.

HPI & METEOROID SHIELD INSTALLATION

DEVELOPMENT TESTING

The test philosophy in conjunction with the methodology for identifying and analyzing test issues, is the framework upon which the test program is built. This program covers all test activities from materials and components through orbital activation and flight tests of the first RNS vehicle and its subsequent operational support.

The primary objective of the test study effort was to establish a plan for determining the suitability of the RNS design in meeting operational requirements and to specify test articles, and a preliminary test schedule compatible with the study guideline IOC date of CY 1981. Test criteria were defined to attain the required operational confidence at each test level commensurate with program goals of low cost transportation, flexibility of operation, reliability, and safety.

As a result of the development test program analysis, test requirements were identified and classified as shown on the opposite page.

Development testing is concerned with the engineering evaluation of prototype hardware in selected environments and configurations for the purpose of acquiring data for reliability evaluation, identifying sensitive parameters, and evaluating the development configuration functional performance. The test activities will encompass materials, design feasibility, breadboard tests, and component tests through integrated systems testing.

Structural testing will be conducted at the manufacturing site; cold flow, and hot tests are to be conducted at NRDS (E/STS-2); and the dynamic tests will be accomplished at MSFC (in conjunction with an INT-21). Flight testing includes those flight tests required to determine the compatibility, functional, and operational suitability of the RNS systems in the actual space and nuclear radiation environments.

DEVELOPMENT TESTING

OBJECTIVES

- DETERMINE SUITABILITY OF DESIGN TO MEET ANTICIPATED OPERATIONAL REQUIREMENT
- QUALIFY RNS TO MEET DESIGNATED MANRATING, RELIABILITY, & CONFIDENCE STANDARDS
- IDENTIFY PROBLEM AREAS EARLY IN DEV PROG TO AID DESIGN EFFORT & MINIMIZE REDUNDANT TESTING
- MAXIMIZE USE OF MULTIPLE TEST ARTICLES TO MINIMIZE COST

MAJOR DEVELOPMENT TEST ACTIVITIES

- STRUCTURAL
- COLD FLOW
- HOT TEST
- DYNAMIC
- FLIGHT

ALTERNATE DEVELOPMENT TEST PROGRAMS

An essential part of the RNS vehicle development effort is the ground test program. The ultimate goal is to demonstrate the RNS readiness for flight tests to finalize man-rating of the vehicle. To achieve this goal, in the most effective manner, four alternate ground test programs were evaluated. The alternates employed differently configured test articles as shown on the opposite page.

ALTERNATE DEVELOPMENT TEST PROGRAMS

BASELINE PROGRAM

- EMPLOYS FOUR TANKS OF ALL FLIGHT HARDWARE
- ONE ARTICLE USED FOR BOTH COLD FLOW & HOT TESTING

BOILERPLATE CONCEPT

- USE FULL SIZE TANK
- FABRICATED OF HEAVY GAUGE MATERIAL (NON-FLIGHT WEIGHT)
- USED FOR BOTH COLD FLOW & HOT TESTING
- OTHER TEST ARTICLES ARE COMMON WITH BASELINE PROGRAM

MINI-STAGE APPROACH

- EMPLOYS S-II FORWARD BULKHEAD & ONE CYLINDER SECTION
- NEW FLIGHT CONFIGURED CONICAL AFT BULKHEAD
- USED FOR STRUCTURAL, COLD FLOW, & HOT TESTING
- ENGINEERING MOCKUP CONSTRUCTED OF WOOD
- REQUIRES NEW DYNAMIC TEST ARTICLE

GROUND TEST MODULE (GTM)

- FABRICATED OF EXISTING S-II HARDWARE
- NEW FLIGHT CONFIGURED CONICAL AFT BULKHEAD
- USED FOR BOTH COLD FLOW & HOT TESTING
- OTHER HARDWARE IS COMMON WITH BASELINE

PROGRAM EVALUATION METHODOLOGY

The objective of this program is to relate the reliability aspects of the RNS to the integrated test program, to evaluate development test program alternatives, and to determine the value of orbital flight testing prior to mission applications. The logic diagram depicts the major requirements and constraints fed into the optimization process, and the output derived as a result of the computer analysis. In the optimization process, testing is incrementally allocated to available test articles (an available test article is an article whose testing is below maximum test level) for which confidence improvement per unit of cost is maximized. The allocation process continues until either all the overall requirements are met or test limits (resources) are expended. This approach was used to evaluate different program options to determine the most attractive development test program that meets the mission requirements.

The major output of the program are: (1) the test cycle allocation, (2) lowest cost program for a given reliability/confidence level, and (3) the distribution of costs. The specific data outputs obviously depend on the inputs and constraints employed. Preliminary variable and fixed costs of facilities, test articles, GSE, support personnel, test operations, and durations were incorporated in the program optimization process.

PROGRAM EVALUATION METHODOLOGY

BASELINE TEST PROGRAM CYCLE ALLOCATION

This chart depicts the input factors such as test items considered, and minimum and maximum cycles. The output of the program shows the actual testing allocated by the computer optimization program to each test item, whereas the equivalent cycle reflects the relative worth of the test cycles performed in terms of an actual lunar shuttle mission. The minimum cycles were based on specific test analyses, whereas the maximum cycles were defined by the schedule constraints and statistical contribution limits of the test items. The output indicates that approximately 45 mission equivalent cycles are required to demonstrate the 90 percent confidence goal (reliability goal of 0.95 is held constant). To achieve this level of demonstration, 430 actual cycles of testing must be performed. Thus, the average mission equivalency weighting factor is approximately 0.10 (45 divided by 430). The allocation sequence indicates that 9 of a total of 15 test items were selected by the computer program to have additional testing performed above the minimum level initially established in the input. The subsystems testing was selected as the most cost efficient within the available test item alternatives. The cold flow, flight, and structural testing, and most component testing were test activities not selected. This result should not be misunderstood as an indication that the latter test sources are poor contributors to the overall test program demonstration. It simply means that the computer program determined that it would be too costly to select these test items for additional testing above the minimum required testing imposed by the input data.

BASELINE TEST PROGRAM CYCLE ALLOCATION

RELIABILITY INPUT FACTORS

- RELIABILITY = 0.95
- CONFIDENCE = 0.90

TEST ITEM	CYCLES			
	MIN	MAX	ACTUAL	EQUIVALENT
RNS-TA-1 (COLD FLOW)	5.00	25.00	5.0	0.80
RNS-TA-1 (HOT FLOW)	10.00	30.00	25.0	14.00
RNS-TA-2 (DYNAMIC VEHICLE)	10.00	25.00	25.0	4.50
RNS-TA-3 (FLIGHT)	2.00	10.00	2.0	2.00
STRUCTURAL TEST ARTICLE (STA)	3.00	10.00	3.0	0.27
ENGINEERING MOCKUP	30.00	40.00	30.0	0.60
PROPULSION SUBSYSTEM	20.00	45.00	45.0	6.75
REACTION CONTROL SUBSYSTEM	25.00	40.00	40.0	4.40
ELECTRICAL POWER SUBSYSTEM	25.00	40.00	40.0	2.00
GUIDANCE & CONTROL SUBSYSTEM	30.00	40.00	40.0	4.00
INFORMATION MGMT SUBSYSTEM	30.00	40.00	40.0	2.00
DOCKING SUBSYSTEM	20.00	40.00	40.0	0.80
COMPONENTS (<5 LB)	30.00	40.00	30.0	0.30
COMPONENTS (5 LB TO 25 LB)	25.00	40.00	25.0	0.50
COMPONENTS (>25 LB)	20.00	40.00	40.0	2.00
TOTALS			430.0	44.92

TEST PROGRAMS COMPARISON

The summary table indicates that the "baseline" test plan - utilizing all major articles fabricated of flight hardware - is the most cost effective. In addition, it was determined to have the most realistic time span and testing sequence. However, it did not permit meeting the guideline IOC date of CY 1981. At the present time, this is not believed to be a serious constraint and therefore, the baseline plan is recommended.

TEST PROGRAMS COMPARISON

REQUIREMENT

- 95 PERCENT RELIABILITY AT 90 PERCENT CONFIDENCE

COMPARISON

TEST PROGRAM ALTERNATIVE	TOTAL PROGRAM COST	
	% ABOVE BASELINE	RANKING
BASELINE	-	1
GROUND TEST MODULE	8	2
BOILERPLATE	33	3
MINI-STAGE	46	4

BASELINE TEST ARTICLE UTILIZATION CYCLE

The flow sequence of the baseline development test articles shows the four major articles, functions, and utilization cycle. The development test program approach requires independent hydrostatic testing of the forward and aft bulkhead with several combinations of cylindrical sections for the structural test article. These assemblies then will be mated with the necessary tank cylinders to form the basic flight tank and will be pneumostatically tested to permit early identification of weld assembly deficiencies and provide assurance that the completed assembly will meet design requirements.

The structural test article (RNS-S) then will be reallocated for ultimate use as an engineering dimensional simulator at the manufacturing site to provide a configuration mockup. This approach will permit early identification of possible configuration interferences and other potential engineering design deficiencies. This, in turn, will minimize flight article engineering design changes and their associated costs.

The cold flow/hot test article (RNS-TA-1) first will be used for facility activation and checkout at NRDS. Upon completion of the facility activation, the article will be used for the cold flow testing program at NRDS. Upon completion of cold flow testing, RNS-TA-1 will be hot tested on the same test stand (E/STS-2). The RNS-TA-2 vehicle will provide backup for RNS-TA-1 in case of damage during cold flow or hot testing. While acting as backup, the RNS-TA-2 will be used to activate Kennedy Space Center (KSC) and then be shipped to Marshall Space Flight Center (MSFC) for dynamic and acoustic testing in conjunction with INT-21. If RNS-TA-2 is required for completing the tests at NRDS, the flight stage (RNS-TA-3) will be used for KSC activation and checkout and the structural test article, which was previously scheduled to become the engineering dimensional simulator, will be cycled to MSFC for dynamic testing. In this case, a dimensional simulator could be fabricated of wood at the manufacturing facility.

The RNS development test program includes exposure of the flight test article (RNS-TA-3) to cryogenic environment at launch pad 39 while the RNS is stacked on top of the INT-21 (propellant tanking during countdown demonstration test (CDDT)). This approach eliminates the necessity for a special off-pad cold flow acceptance testing facility. This recommendation is based on the confidence and experience derived from the Saturn program, and predicated on the following: (1) early laboratory cryogenic tests, (2) extensive cold flow and hot test programs at NRDS, and (3) the manufacturing technique which includes structural testing of individual tank sections in process testing and post-manufacturing checkout.

BASELINE TEST ARTICLE UTILIZATION CYCLE

FACILITIES DEFINITION

The facilities definition study compliments the cost effectiveness of the manufacturing and assembly technique, and the cost avoidance approach of the development test program.

The main objective of the facilities definition analysis is maximum use of existing resources, minimizing modifications, and reducing cost by concurrent usage of RNS/S-IC and RNS/S-II.

Four manufacturing site alternatives were evaluated as shown on the opposite page.

The facilities definition study was based on consolidating RNS production operations with either S-II or S-IC operations. This will provide significant facility economies without deterring from other program costs. Savings will accrue from common use of facilities and equipment, as well as a major reduction in facility area requirements when compared to using separate sites.

Due to vehicle similarities, except for the conical aft bulkhead section and HPI installation, assembly of the RNS at Michoud or Seal Beach can be accomplished with minimum manufacturing changes. Existing equipment and/or facilities will be used in a common or dual role, enabling NASA to effect a smooth, economical, orderly buildup of the RNS program. Common use of equipment, material and personnel handling equipment (MPHE), tooling, etc., is practical and, in many instances, they can be utilized on either stage.

"Maximum utilization of existing resources" was the primary consideration in using this approach in the development of RNS facility plans for Seal Beach and Michoud.

FACILITIES DEFINITION

APPROACH

- MAX USE OF EXISTING FACILITIES
- MINIMUM MODIFICATIONS POSSIBLE
- CONCURRENT USAGE: RNS/S-IC, RNS/S-II

MANUFACTURING SITE ALTERNATIVES

- MICHoud ASSEMBLY FAC (MAF)
- SEAL BEACH (SB)
- JOINT UTILIZATION: MAF & SEAL BEACH
- UTILIZATION OF APOLLO FACILITIES/TOOLING

FACILITIES COMPARISON

The table depicts the major manufacturing operations and the level of facilities modifications at MAF and Seal Beach.

Preliminary results of facilities requirements and utilization analyses showed that the RNS tank can be manufactured at either MAF or Seal Beach. Cost estimates of necessary new and modified facilities indicated a negligible total cost difference. However, a saving may be possible by employing both facilities for RNS manufacturing and assembly. This alternate takes maximum advantage of applicable existing resources. The facilities cost savings could amount to approximately \$1.3 million. However, a number of factors that will affect these savings were not taken into consideration, such as (1) whether either or both sites are operational when the RNS manufacturing program starts, (2) cost of transporting components and large sections of the RNS tank from one site to the other, (3) number of stages required, (4) yearly production rates, and (5) results of S-IC/S-II consolidation studies conducted by NASA-MSFC. Irrespective of the location for manufacturing of the RNS tank (MAF, Seal Beach, or both), a requirement exists for new tooling for the manufacture and assembly of the conical aft bulkhead section of the tank. Apollo tooling is a good candidate for the manufacture and assembly of the 8-degree half cone-angle and 25 inch cap radius section, up to a length of ten feet. This cost savings was not credited to the RNS during the facility and tooling utilization analysis.

FACILITIES COMPARISON

MAJOR MFG OPERATIONS	FACILITY MODS LEVEL	
	MAF	SB
● BUILD-UP AREA & AUTOCLAVE INSTALLATION	MINOR	
● LH ₂ BULKHEAD FAB & ASSEMBLY	MINOR	
● BULKHEAD HYDROSTATIC TEST	MINOR	
● CLEAN ROOM ~ 2000 SQ FT	MINOR	
● TANK CLEANING AREA	MAJOR	MAJOR
● HORIZONTAL SPRAYFOAM APPLICATION & MACHINING	MAJOR	MINOR
● FINAL SYSTEMS INSTALLATION & CHECKOUT	MINOR	MAJOR

PRELIMINARY RESULTS

- FAC MODS COST SAME FOR MAF & SEAL BEACH (~ \$2.0M)
- JOINT UTILIZATION SAVINGS ~ \$1.3M

OUTSTANDING QUESTIONS

- FACILITY AVAILABILITY AT START OF RNS PRODUCTION PERIOD
- TRANSPORTATION OF LARGE SECTIONS BETWEEN SITES
- YEARLY PRODUCTION RATE & NUMBER OF STAGES REQUIRED
- RESULTS OF S-IC/S-II CONSOLIDATION PROGRAM

MAJOR KSC FACILITY MODIFICATIONS

The use of the Saturn INT-21 for the launch of an RNS tank in an inverted configuration offers the significant advantage of nearly complete utilization of existing assembly and launch facilities at Launch Complex 39. Effectively, no facility modifications will be required below the RNS tank/INT-21 interface. Modification requirements are concentrated in the area of RNS tank support, including vehicle access and servicing.

The total weight of the Saturn V/Apollo at ground ignition is approximately 6.5 million pounds while the maximum weight of the Saturn INT-21/RNS tank at ground ignition is 6.3 million pounds. This is approximately 200,000 pounds less than Saturn V/Apollo weight. Therefore, no modifications are required to the mobile launcher, crawler transporter, crawler way or launch pad structure because of loading.

The existing S-IVB umbilical is located at approximately level 220. The new fill line of the RNS tank is located at approximately RNS tank station 1842. This is approximately 6.6 feet above the S-II/RNS tank interface point or level 219 on the umbilical tower. Therefore, the position of S-IVB umbilical arm has to be modified to accommodate the RNS tank LH₂ ground service port/orbital vent. Functions of the new arm include LH₂ fill and drain, pressurization, pneumatic, instrumentation and electric system support. In addition, various mobile launcher lines and plumbing will require modifications at that level.

The Saturn V/Apollo requires the 335,860 gallons of LH₂ per flight. This quantity does not include chilling, losses, topping, etc. The current LH₂ storage capacity at LC 39 is 850,000 gallons. The difference reflects a factor of 2.54 gallons of LH₂ storage capacity per gallon of LH₂ loaded on the Saturn V vehicle at ground ignition. Based on RNS tank propellant requirements, an additional storage capacity of approximately 400,000 gallons is required.

The modification costs were estimated at approximately 12.0 million dollars.

MAJOR KSC FACILITY MODIFICATIONS

MODIFICATIONS

- REPOSITION PLATFORM LEVELS AS REQUIRED FOR RNS SERVICING
- EXTEND AND/OR NEW C/O REQUIREMENTS FROM S-II TEST CELL TO HIGH BAY AISLE
- HIGH BAY AISLE NO MODIFICATION REQUIRED

- RELOCATE AND/OR MODIFY REQ PLATFORM(S) ABOVE LEVEL 212
- RELOCATE AND/OR MODIFY UMBILICAL SWING ARM AT ~ LEVEL 219
- RELOCATE OR MODIFY HIGH PRESSURE SYSTEM, ELECTRICAL SYSTEM, AND PNEUMATIC SYSTEM IN SUPPORT OF RELOCATED UMBILICAL SWING ARM
- INCREASE LH_2 STORAGE CAPACITY ~ 400,000 GALS

PROGRAM SCHEDULING

The program development schedule is a graphical display or timeline of the principal activities and milestones which comprise the development program. It represents one of the resource requirements, and has a significant effect on time, cost, expenditure rate, manpower, facilities, etc. Although the objective of schedule development is a realistic appraisal of time requirements and activity phasing, there is necessarily considerable uncertainty in both requirements and optimal phasing at the present Phase A state of program development.

To assure consideration of all the most promising test program alternatives, schedules have been prepared for four alternate programs. This was done to assure determination of the optimal schedule.

The major guidelines considered in the preparation of the development schedule are shown in the facing page.

PROGRAM SCHEDULING

APPROACH

- EVALUATE ALTERNATES TO DETERMINE OPTIMUM SCHEDULE
- SYNTHESIZE INTEGRATED ACTIVITIES & MAJOR MILESTONES TO SATISFY GUIDELINES

GUIDELINES

- 10 YEAR OPERATIONAL PROGRAM
- TIME REQUIRED FOR MAJOR TESTING OPERATIONS
 - COLD FLOW 9 MONTHS
 - HOT TESTS 24 MONTHS
 - DYNAMIC 12 MONTHS
 - FACILITIES CHECKOUT 6 MONTHS
 - FLIGHT 18 MONTHS

RNS PROGRAM SCHEDULE

This schedule depicts the integrated set of activities and events necessary to accomplish RNS development and operational phases. All major development activities from Phase A-IV through the completion of flight testing and a ten-year operational program are included. To assure consideration of the most promising program alternatives, schedules were prepared for four alternate development/test programs. However, only the baseline schedule is summarized here.

The following major ground rules and assumptions were used in the preparation of the integrated program schedule. They were derived from NASA study guidelines and the results of Phase III technical analyses: (1) NASA dates for first flight and IOC are treated as guidelines but not as constraints; (2) a 12-month spacing between the starts of Phases A-IV, B, and C was assumed; and (3) a nine-month period was assumed for Phase C, since it was assumed that Phase D would commence immediately at the conclusion of Phase C with continuous NASA review throughout the program rather than a three-month gap between phases. A maximum of three RNS/S-II stages per year can be produced with a single set of tooling. The production of two articles per year is considered the minimum economical rate, due to utilization of manpower and maintenance of technical skills. To minimize the cost and time during DDT&E, a production rate of three stages per year was employed. Six RNS stages (and six S-IC/S-II) are required for operations with each RNS flying ten missions, evenly spaced in time at an annual rate of six per year. As a compromise between storage time and production rate economy, it was assumed that one RNS and one S-II stage would be produced annually.

The minimum times required for major system tests as derived in the test planning study are as follows:

Cold flow	9 months
Hot tests	24 months
Dynamic tests	12 months
Facilities checkout	6 months
Flight tests	18 months

The baseline program development schedule reflects the desired manufacturing time spans and testing sequences within the aforementioned constraints but does not meet the guideline IOC date (CY 1981). An earlier IOC date could be realized by shortening the length of the hot test or flight test period, or shortening Phases A-IV and/or B. However, the time spans used are the best current estimates of realistic requirements.

RNS PROGRAM SCHEDULE

PROGRAM PLANNING SUMMARY

Results of the program planning study show that an RNS tank with 8-degree half cone-angle & 25-inch cap radius, 160 feet in length can be fabricated and assembled at either MAF or Seal Beach. This can be accomplished without imposing design constraints on the tank or subsystems. Additionally, the recommended technique permits concurrent usage of existing resources for RNS/S-IC and RNS/S-II with minimum modification to either facility.

Based on the development test program requirements and the computer optimization process, the baseline development test program was found to be the most attractive alternate. It will achieve the objectives of minimum cost, (or maximum reliability and confidence level for a given amount of testing), and minimize the number of test articles/operations.

The baseline program schedule reflects the best manufacturing time spans and testing sequences. The schedule will permit early resolution of structural changes, as well as allow the majority of these changes to be incorporated on an in-line basis, thus keeping modification and change costs to a minimum.

The schedule will not meet the study guideline IOC date of CY 1981, but this appears unlikely to be a critical factor. However, the schedule does contain a certain amount of flexibility with a reasonable amount of time slack for unforeseeable program delays.

PROGRAM PLANNING SUMMARY

COMBINED TANK MANUFACTURING & ASSEMBLY ALTERNATE DEFINED

- TANK MANUFACTURING & ASSEMBLY FEASIBLE W/O IMPOSING DESIGN CONSTRAINTS
- MANUFACTURING TECHNIQUE APPLICABLE TO MAF & SEAL BEACH
- COMMON UTILIZATION OF S-IC & S-II TOOLING, FAC & GSE
- FAC MODS SAME FOR MAF & SEAL BEACH (\$2.0M)

BASELINE DEVELOPMENT TEST PROGRAM DEFINED

- MINIMUM TEST ARTICLES
- ALL STAGES BUILT OF FLIGHT HARDWARE MOST ATTRACTIVE
- LOWER COST FOR GIVEN RELIABILITY & CONFIDENCE

BASELINE INTEGRATED SCHEDULE DEFINED

- REFLECTS EXPECTED MFG & TEST REQTS

ADDITIONAL STUDY NEEDED

- LOW PRODUCTION RATES
- LONG TIME TO IOC
- EXTENDED STORAGE OF OPER STAGES

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BRIEFING OUTLINE

- INTRODUCTION - T. M. LITTMAN
- MISSION REQUIREMENTS & OPERATIONS - A. R. JUSAK
- SYSTEM CAPABILITY - A. R. JUSAK
- SYSTEM DEFINITION - D. GARCIA
- PROGRAM PLANNING - W. G. ANTYPAS
- COST ANALYSIS - T. M. LITTMAN
- CONCLUSIONS & RECOMMENDATIONS - T. M. LITTMAN

COSTING GUIDELINES AND ASSUMPTIONS

The principal groundrules employed in performing the cost estimation for the RNS baseline program are described here. There are a number of significant departures from the guidelines employed during Phase II with regard to the categorization of items as well as the cost base employed.

COST BASE

All costs were normalized to government fiscal year 1971 dollars. The Phase II study employed a GFY 1970 base. The MSFC baseline INT-21 vehicle used to boost the RNS tank to orbit has a procurement cost of \$95M and a launch operations cost of \$12M. These numbers were obtained from NASA-MSFC for the Phase III RNS study. No specific charge was made for INT-21 DDT&E, nor was a production learning curve employed. Orbital delivery of the NERVA engine and all logistics (expendables including LH₂ resupply, stage replacement parts) is via the space shuttle at a cost of \$5.0M per trip. The baseline program employed a "25K" shuttle (delivery of 33,000 pounds to a circular 260 nmi x 31.5° orbit at a cost of \$162/lb of useful payload). NERVA unit procurement was taken at \$13.0M including the engine and all accessories (external shielding, electronic controls). No DDT&E costs were considered nor was a production learning curve employed. One engine was assumed to be required for each stage -- that is, no replacement engines or parts were considered. Contractor fee was excluded from the cost analysis.

DDT&E

One flight test stage, plus associated booster and launch costs, was charged to development. In Phase II, only 20 percent was included in DDT&E with the remainder charged to the operational program. Mission support requirements for two equivalent lunar flights (including purchase and delivery of 600,000 pounds of LH₂) also were included.

Cost of transporting one RNS (RNS-TA-1) (excluding engine) to NRDS for cold flow/hot testing includes road modifications as well as operations and is contained under mission operations. Development costs associated with supporting space program elements (maintenance facility, propellant depot, space tug, etc.), special equipment (teleoperators, engine "hanger" shields), and RNS payloads were excluded. The docking system and fuel cells were assumed to be developed by prior programs (space station and shuttle). Therefore, only requirements to adapt the technology/hardware for RNS usage were considered.

PRODUCTION AND OPERATIONS

Procurement costs for the aforementioned space program elements also have been excluded as well as the production and associated mission operations costs of RNS payloads. A ten year operational program was assumed with RNS production units of 2, 4, 6, 8, and 10. Corresponding annual flight rates of 2, 4, 6, 8, and 10 respectively, were used. Ten flights per RNS were assumed for all operational programs except for a parametric study in which lifetime was varied. For the low flight rate programs (2 and 4), stage lifetime requirements exceed the baseline value of three years. However, this was not accounted for in the design of meteoroid protection or other hardware. For costing purposes, the "optimum" production rate of stages was assumed. RNS storage costs were estimated from a recent S-II study and are included under mission operations. A maintenance element was used to support RNS orbital operations. It was assumed to make one round trip to orbit via the space shuttle (at a cost of \$5M) for each 10 RNS flights. No LH₂ transfer or boiloff losses prior to mission departure were accounted for in the cost analysis. Disposal of the RNS was assumed to take place on the last operational flight. Therefore, no supporting system (space shuttle or tug) was required.

COSTING GUIDELINES & ASSUMPTIONS

COST BASE

- GFY 1971 DOLLARS
- INT-21 PROCUREMENT = \$95M + LAUNCH OPERATIONS = \$12M
- SPACE SHUTTLE OPERATIONS = \$5M / FLIGHT
- NERVA PROCUREMENT = \$13M
- NO DEVELOPMENT COSTS CHARGED FROM OTHER SPACE PROGRAMS

DDT&E

- ONE RNS FLIGHT TEST STAGE & TWO EQUIVALENT LUNAR MISSIONS

PRODUCTION & OPERATIONS

- ANNUAL PRODUCTION RATE OF ONE RNS & ONE S-II
- 10-YEAR OPERATING PROGRAM
- 6 FLIGHTS PER YEAR
- 10-FLIGHT RNS LIFETIME WITH MAINTENANCE
- RNS SELF-DISPOSAL ON LAST MISSION
- SPACE SHUTTLE DERIVATIVE MAINTENANCE ELEMENT
- NO LH₂ TRANSFER OR BOILOFF LOSSES PRIOR TO MISSION START

RNS BASELINE PROGRAM COSTS

The top level costs for the baseline single tank configuration (six production units, delivery of NERVA and logistic supplies to orbit by the space shuttle at a cost of \$162/lb, boost of RNS tank in inverted attitude on INT-21) are shown in the chart along with a plot of annual funding requirements. These data are expressed in terms of 1971 dollars and exclude NERVA and INT-21 DDT&E. RNS production cost averages \$45.4 M, including \$13.0 M for NERVA. Although the operational program covers a period of only ten years, the earlier funding requirements result from procurement of the INT-21 boosters. Reference is made to the program schedule which assumed concurrent RNS/S-II production of one stage each per year. Thus fabrication of the INT-21 for the first operational RNS is initiated six years prior to IOC.

The major cost item (mission operation) results from the delivery of RNS propellant to orbit by the space shuttle. Additional substantial costs are due to procurement of the INT-21 launch vehicle and supporting launch operations.

RNS BASELINE PROGRAM COSTS

WBS LEVEL	WBS IDENTIFICATION	DDT&E \$M	PROD \$M	OPER \$M	TOTAL \$M
2	RNS PROGRAM	809.2	272.3	3749.3	4830.8
3	LAUNCH VEHICLE	119.4	0.0	582.1	701.5
3	LAUNCH OPERATIONS	47.6	0.0	285.9	333.5
3	RNS	585.6	272.3	29.4	887.3
3	MISSION OPERATION	56.6	0.0	2851.9	2908.5

• AVERAGE RNS PRODUCTION COST - \$45.4M

COST COMPARISON OF ALTERNATE CONFIGURATIONS

A comparison of the baseline RNS with the two more attractive hybrid designs is shown in the chart. Although there are only modest differences in DDT&E and Prod/Oper costs, there are sufficient differences in structural and external shielding weight to show payload benefits for the baseline which in turn translates into better cost effectiveness. For the hybrid designs to deliver the same maximum outbound lunar payload (200,000 pounds) as the single tank, there is an additional cost per flight of 3.0 and 6.5 million dollars for the 8-degree and 7.5-degree hybrids, respectively. Furthermore, the 8-degree hybrid (in comparison to the 7.5-degree design) has a more limited capability for self-disposal using the auxiliary propellant tank.

One other aspect has been examined in the comparison of alternatives. In the hybrid concept, savings might be possible by the use of the propulsion module (auxiliary tank plus NERVA) with a facility LH₂ tank for hot testing. Although this approach would permit early initiation of hot testing, the results of the reliability/test cost analysis of four alternate development programs suggest that cost benefits are unlikely. In this study, the least costly test program (taking into consideration hardware, facilities, GSE, operations, etc.) was found to be the one using all full-size ground test articles fabricated from flight-type hardware.

Even the alternate program, employing a mini-stage for cold flow/hot testing -- which is somewhat analogous to the use of the hybrid propulsion module -- was more expensive. This resulted from the lower worth of the mini-stage in reliability confidence development and the requirement to perform additional flight tests which are the most costly.

COST COMPARISON OF ALTERNATE CONFIGURATIONS

CONFIGURATION	DESIGN FEATURES	DDT&E \$M	PROD/OPER (\$M/FLT)	RELATIVE PAYLOAD DELIVERY COST ⁽¹⁾
SINGLE TANK	8° x 25-IN.	809.2	67.03	1.00
HYBRID	8° x 40-IN. MAIN TANK, 3000 LB LH ₂ AUX TANK	839.8	67.29	1.06
HYBRID	7.5° x 112-IN. MAIN TANK, 9300 LB LH ₂ AUX TANK	832.0	67.23	1.11

(1) BASED ON MAXIMUM OUTBOUND PAYLOAD TO
LUNAR ORBIT WITH ZERO RETURN PAYLOAD

RNS PROGRAM COST CENTERS & REDUCTION POSSIBILITIES

An assessment of cost-influencing parameters or drivers is essential to increase visibility and understanding of each item's impact on the overall cost picture. Displayed here are the relative contributions of the top level cost centers for the baseline RNS program based on the data presented in an earlier chart.

In a reusable space transportation system, propellant delivery to orbit is a dominant cost. For the baseline RNS, 300,000 pounds of LH_2 must be delivered prior to each mission (excluding transfer and other pre-flight losses). This makes up the major share of total program operations cost. Thus, the importance of minimizing space shuttle delivery costs cannot be emphasized too strongly.

Booster procurement and ground operations associated with use of an INT-21 (ground-ruled booster for the NR study) to launch the RNS tank to orbit also represents an important share of total operating cost. Significant reductions in this cost item are likely only through change to a reusable or partially reusable booster.

Although the items of propellant and RNS tank delivery represent the two major cost drivers, there are numerous other areas in which important cost avoidance or reduction is possible. Thus, further benefits should be possible through greater hardware/software IPP commonalty- particularly on HPI, astrionics and operations. Increased component lifetime (without imposing excessive requirements on maintenance) would permit reduction in number of stage buys and hence reduce total production cost.

Numerous savings in operating cost or cost effectiveness are possible through more efficient use of propellant. These include higher I_{sp} (even at the expense of shorter NERVA life), increased stage size, and use of a tug-assist flight mode.

RNS PROGRAM COST CENTERS & REDUCTION POSSIBILITIES

RNS PROGRAM COST SENSITIVITY

The transport of logistic supplies -- particularly RNS propellant-- represents a major fraction of program cost. If the space shuttle delivery cost can be reduced to \$100/lb (50,000 pound payload capability to 260 n mi x 31.5 degrees at \$5M/flight) from the baseline study value of \$162/lb, RNS cost effectiveness would be improved by 26.7 percent. On the other hand, if shuttle orbital delivery cost is \$200/lb, it would increase the cost of moving payload to lunar orbit by 16.4 percent. The parametric data are presented in the left hand graph on the facing page.

Another way to reduce RNS cost may be possible if a vehicle other than the expendable INT-21 is employed for launch of the RNS tank to orbit. One promising approach is to use the space shuttle booster with an expendable second stage (ESS). Another alternative is to temporarily configure the RNS as an ESS (designated here as CPS-chemical propulsion module) by adding an LO₂ tank and a space shuttle engine. Since the engine/LO₂ tank module would be recovered for reuse by a later regular shuttle flight after it delivers the NERVA, this concept is equivalent to a fully reusable system. The bar graph opposite compares the baseline launch mode with the ESS and CPS. As can be seen, the potential reductions in lunar delivery cost are 8.1 and 14.3 percent, respectively. Furthermore, DDT&E cost savings of 5.8 and 10.8 percent, respectively also were estimated.

RNS PROGRAM COST SENSITIVITY

RNS COST SENSITIVITY (CONTINUED)

Performance studies have indicated that cost effectiveness is most sensitive to NERVA specific impulse. Thus, a ten-percent increase in I_{sp} would reduce lunar payload delivery cost by nearly 25 percent as shown in the left hand graph opposite. Since the Los Alamos Scientific Laboratory is currently developing carbide-type fuel elements which have a potential I_{sp} well over 900 seconds, the economic benefit is apparent. Higher specific impulse can be even more significant than NERVA lifetime. Thus, an eight-percent increase in I_{sp} is equivalent to the \$13M production cost of a complete engine. Stage inert weight, while not as strong a cost driver as specific impulse, nevertheless offers important benefits (or penalties). The sensitivity of cost to weight is nearly a 1:1 ratio. On the other hand, with the use of the multi-perigee thrusting technique for minimizing RNS gravity losses, the low thrust of the 75K NERVA imposes only a small performance penalty. Thus, thrust level will have a negligible effect on cost effectiveness.

RNS service life was assumed to be ten equivalent lunar round trips during a three-year period. The effect of varying mission cycles (without changing the maximum time in space, which influences meteoroid protection and thermal coating requirements) was determined and the results are plotted in the right hand figure. The sensitivity to service life reduction below the baseline value is quite evident, particularly below eight flights/RNS. Some cost benefit may be possible by extending lifetime beyond ten flights if it does not impose undue maintenance requirements or degrade safety/mission success probability.

The RNS production rate is an important cost factor since it affects operating efficiency as well as requirements for facilities, tooling, and special test equipment. During Phase II, it was estimated that a rate of two stages per year (two RNS's or one RNS plus one S-II) would increase production costs by about 13 percent compared to the optimum rate of three at the Seal Beach facility. A reduction to one/year would nearly double the unit production cost. On the other hand, the number of production units was determined to have no appreciable effect on recurring cost over the range considered in this study (2-10) if lifetime is considered to be constant. However, if DDT&E is amortized over the production run, there is a substantial impact when only a few operational stages are built.

RNS PROGRAM COST SENSITIVITY (CONT)

RNS COST SENSITIVITY (CONTINUED)

An important factor in design optimization is RNS size. Due to the current absence of firm mission requirements, the major study effort was concerned with a nominal LH₂ capacity of 300,000 pounds -- based on previous mission studies. However, the effect on performance and cost of changing the size was evaluated as shown in the left hand plot opposite. The cost data excludes DDT&E. Significant improvement in cost effectiveness with size is quite evident and would justify development of a larger stage if heavier payloads are needed. Overall program economies should dictate sizing the RNS based on the anticipated traffic density and payload weight range. If there is need to carry a heavier payload only occasionally, alternate flight modes are available (such as tug-assist on the return to earth orbit). Future studies need to more fully explore this area, including the economics of flying the RNS with off-loaded propellant.

A study of four alternate test plans was made with the goal of achieving minimum cost for a target reliability of 0.95 at a confidence level of 0.90. The pertinent cost data, presented in normalized form in the program planning section indicates that use of all flight-type hardware is optimum. Follow-on studies are necessary to evaluate other test mixes and program risk associated with use of a minimum number of test articles. The effect on the baseline test program cost of varying confidence level for a fixed reliability goal is shown in the right hand graph here. The apparent insensitivity of test cost to reduction in confidence below 0.90 is due to the method employed in the study. Fixed costs (test hardware, facilities, GSE, etc.) were kept constant while variable costs (personnel, consumables, M&R, etc.) were allowed to change in order to meet changing test cycle requirements corresponding to alternative confidence levels. For the baseline program, fixed costs represent 67.5 percent of the total. Further studies would be valuable to determine the effect on cost and confidence of allowing fixed cost items to vary.

RNS PROGRAM COST SENSITIVITY (CONT)

CONCLUSIONS

NR's Phase III study has clearly met its intended objective of defining an attractive RNS program as summarized in this briefing and described in more detail in the final report (SD 71-466). The benefits of reusability, commonality, design and operational simplicity, and autonomy were exploited to the maximum practical extent to achieve the study goal. Repetitive, realistic lunar shuttle cycles were established -- employing precision flight mechanics techniques -- with payload delivery capabilities equal to or exceeding the requirements identified in other studies. Similarly, a substantial capability was defined for other shuttle missions as well as injection flights (particularly in conjunction with an expendable tug) covering both unmanned explorers and manned Mars payloads. A stage concept (high fineness ratio single tank) and launch mode (inverted tank on INT-21 with NERVA delivered by space shuttle) have been defined which appears about optimum within the study constraints.

As a result of the investigation to date, a considerable amount is known about anticipated ground, orbital, and flight operations. Furthermore, although the RNS must operate in the unique and hazardous nuclear radiation environment associated with this superior power source; routine, safe operations -- including maintenance -- appear possible by proper design and planning of operating procedures. Furthermore, although numerous uncertainties exist and will persist until the RNS program reaches a greater state of maturity, no significant problem areas have been uncovered which might threaten its technical feasibility or substantially affect its economic attractiveness.

The study also identified a number of potentialities for further cost avoidance or reduction. Some of these are largely beyond the control of the RNS program (such as space shuttle logistics delivery cost). Others probably can be implemented by changes in design, operations, or program activities. Typically these include higher NERVA I_{sp} (even at the expense of reduced life), use of a fully or partially reusable booster for the RNS tank, alternate mission flight modes, and the other items listed on the chart.

CONCLUSIONS

RNS TECHNICALLY FEASIBLE & ECONOMICALLY ATTRACTIVE

- WIDE SPECTRUM OF POSSIBLE APPLICATIONS FOR BASELINE
- LATITUDE IN MISSION MODELING
- LOW COST THROUGH REUSABILITY, DESIGN EFFICIENCY, & COMMONALITY
- HIGH RELIABILITY & SAFETY ACHIEVABLE

RADIATION ENVIRONMENT MAJOR DESIGN & OPERATIONS INFLUENCE

- CREW PROTECTION READILY ACCOMMODATED
- MANNED OPERATIONS POSSIBLE AT FORWARD END OF RNS
- ACCESS TO AFT END USING ROBOT/TELEOPERATOR EQUIP.
- APPROACH PATTERN & ATTITUDE CONTROL REQUIREMENTS CAN BE MET

FURTHER COST SAVINGS POSSIBLE

- REDUCED LOGISTICS DELIVERY COST
- ALTERNATE BOOSTER
- HIGHER Isp
- LOWER INERT WEIGHT & RESIDUALS
- LONGER LIFE
- ALTERNATE FLIGHT MODE - TUG ASSIST

RECOMMENDATIONS

With a capability of satisfying a wide range of proposed manned and unmanned shuttle missions, combined with an inherent growth potential for manned planetary exploration, the RNS represents a versatile space transportation system for the 1980's and beyond. However, in spite of being judged technically feasible and economically attractive, considerable room remains for additional pre-Phase B studies to (1) expand the base for RNS applications and better definitize its role in the IPP; (2) further reduce nonrecurring and recurring costs through design and operating improvements with emphasis on hardware/software commonality; (3) resolve outstanding technical questions by specific analytical and experimental efforts; and (4) increase confidence in the design/program recommendations through greater depth of analysis as well as concurrent SR&T.

Further work in the first area is essential to establish (or substantiate) a realistic RNS size coupled to existing demand but with sufficient flexibility to permit economic delivery of smaller as well as larger size payloads than nominal. Alternate flight modes (particularly tug assist) are of special interest.

As indicated in this briefing, dramatic benefits in cost and cost effectiveness should be possible by further exploring design and operational approaches outside the constraints or scope of the Phase III study. As more detail definition becomes available from other IPP elements, notably space shuttle and space station, the advantages of commonalty can be more fully exploited. Particular attention needs to be given to the problem of shuttle cargo bay volume constraint in conjunction with the delivery of LH_2 . Current space shuttle program concern with this area should be encouraged and alternate solutions investigated.

The benefits of long hardware life and unique subsystems/operating procedures (such as stratified ullage yielding reduced residuals) need to be verified by further analysis and probably concurrent SR&T. Wherever possible, early space experiments should be undertaken to mitigate against costly changes in later phases of the program.

The aforementioned suggestions have briefly touched on the more significant areas where additional studies should be beneficial. Obviously others can be and will be considered as the RNS program evolves and more substantive information becomes available from other interfacing programs. However, it should be noted in concluding that subsequent studies are expected to lead us in the direction of further improvements in an already attractive concept for space transportation in the 1980 period and beyond.

RECOMMENDATIONS

BETTER DEFINITIZE RNS ROLE IN INTEGRATED SPACE PROGRAM

- APPLICATIONS
- FLIGHT MODES
- PAYLOAD/RNS INTERFACE REQUIREMENTS
- MISSION/TRAFFIC SCHEDULING

FURTHER REDUCE COSTS EMPHASIZING REUSABILITY & COMMONALITY

- ALTERNATE LAUNCH VEHICLES
- SPACE SHUTTLE CARGO BAY VOLUME CONSTRAINT
- ADAPTATION OF EXPECTED DEVELOPMENTS IN HPI, ASTRIONICS, OPERATIONS
- INTEGRATED NERVA/RNS HOT TESTING

RESOLVE OUTSTANDING TECHNICAL QUESTIONS

- HARDWARE LIFETIME & RELIABILITY
- HPI DESIGN FOR PRODUCIBILITY & BOOST ENVIRON COMPAT

INCREASE CONFIDENCE IN DESIGN/PROGRAM BASELINE

- GREATER DEPTH OF ANALYSIS
 - PROPELLANT MGMT - STRATIFIED ULLAGE & THERMO VENTING
 - OPERATIONS - LOGISTICS/MAINTENANCE
 - SAFETY - FAILURE DETECTION, ABORT, & RESCUE
- SR&T