Final Report - Phase I, SENSE Study

STANDARDIZATION AND ECONOMICS OF NUCLEAR SPACECRAFT

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ABSTRACT

Feasibility and cost benefits of nuclear-powered standardized spacecraft are investigated. The study indicates that two shuttle-launched nuclear-powered spacecraft should be able to serve the majority of unmanned NASA missions anticipated for the 1980's. The standard spacecraft include structure, thermal control, power, attitude control, some propulsion capability and tracking, telemetry, and command subsystems. One spacecraft design, powered by the radio-isotope thermoelectric generator, can serve missions requiring up to 450 watts. The other spacecraft design, powered by similar nuclear heat sources in a Brayton-cycle generator, can serve missions requiring up to 2000 watts. Design concepts and trade-offs are discussed. The conceptual designs selected are presented and successfully tested against a variety of missions. The thermal design is such that both spacecraft are capable of operating in any earth orbit and any orientation without modification. Three-axis stabilization is included. Several spacecraft can be stacked in the shuttle payload compartment for multi-mission launches. A reactor-powered thermoelectric generator system, operating at an electric power level of 5000 watts, is briefly studied for applicability to two test missions of diverse requirements. A cost analysis indicates that use of the two standardized spacecraft offers sizable savings in comparison with specially designed solar-powered spacecraft.

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1.0 INTRODUCTION

This report presents the results of a three-month study and preliminary design effort on standardized nuclear-powered spacecraft. The study was performed for the AEC/NASA Space Nuclear Systems Division in support of the current NASA program aimed at reducing the cost of space missions.

1.1 MOTIVATION

In its role as the future space booster, the Space Shuttle, through standardization, vehicle re-use, and large payload capacity, will significantly reduce the cost of delivering payload to orbit. This has a strong effect on overall cost structure and provides an incentive to extend the savings to payload systems. Much of the NASA effort is concentrated on the definition of standard components and subsystems which can be incorporated into future spacecraft without redevelopment and requalification.

However, even with standard components, a major portion of development, test, and evaluation (DTE) costs will remain if the spacecraft itself, its structure, electrical power system, thermal control and vehicle dynamics continue to be highly mission and orbit dependent, as at present. This study investigates whether the use of nuclear power in the era of the Space Shuttle would permit standardization of the spacecraft as a whole. The motivation behind such standardization is the effort to eliminate, or sharply reduce, the costs associated with developing a new, specially designed, spacecraft for each mission. The DTE costs have historically accounted for approximately 70% of the overall cost of space programs.

These factors suggest the need for a thoroughly new approach to spacecraft design, and a reevaluation of the commonly accepted basis for selecting the spacecraft electric power systems. Hitherto, weight and volume considerations have been paramount and have usually dictated that spacecraft be tailor-made to fit each mission. In these circumstances the power system with the lowest weight-per-watt or overall cost-per-watt is the logical choice. In most cases this has, quite properly, resulted in the selection of solar-array power systems for auxiliary power. Since the spacecraft is being specially designed to begin with, the special requirements of the solar-array system for the particular orbit conditions of the mission can usually be satisfied in the design.

The unique properties of the Space Shuttle, however, imply that changes in the groundrules are necessary. The large lifting capacity and the substantially reduced cost of placing a pound of payload in orbit will obviate much of the incentive for specially-designed payload equipment to perform each mission. In addition, the mandated use of the Shuttle as a space ferry for all missions will strongly suggest the development of "standard spacecraft" with standard integration modes and procedures for use with this standard booster. These standard spacecraft should be capable of performing a variety of different missions without redesign or requalification. They would simply be outfitted with the specific equipment required for each mission. In concept, this equipment would only need to be integrated with the standard "housekeeping" systems provided by the standard spacecraft, such as telemetry, attitude control, and electrical power generation.

Under these groundrules, a nuclear system may be the most cost-effective choice for auxiliary power for most missions, even if the recurring cost of the generator itself were to remain relatively higher than that of a solar-array system of comparable capability. The fundamental advantage of the nuclear system in this context is that it provides power (and heat) which is not dependent on the sun-angle or occultation period, and is not degraded by trapped Van Allen or solar radiation. These highly orbit-dependent factors generally require solar-array systems to be specially designed for each mission. The use of solar arrays, in turn, tends to require the development of a custom-designed spacecraft for each mission, despite the relaxation of the weight and volume limitations permitted by the shuttle.

Nuclear systems, by contrast, are largely insensitive to orbital and environmental factors. For example, the same basic nuclear generator (MHW-RTG) is planned to be used in earth orbit (LES 8/9) and for deep-space missions (Mariner/Jupiter-Saturn). Moreover, the availability of solar-independent waste heat from the nuclear systems should permit the design of a thermal-control system which is largely orbit-and mission-independent, and which permits the adaptation of various types of mission equipment without redesign of the standard spacecraft. This waste heat can be used to compensate for variations in solar input, earth reflection, and load power consumption, thus maintaining equipment temperature within acceptable limits for different orbits and mission-power profiles.

It is not suggested here that solar-array power could not continue to be used for Shuttle-delivered missions; only that nuclear power more readily lends itself to the design of multi-purpose spacecraft, which would thereby result in significant cost benefits to such missions.

This study was focused on determining the feasibility of designing such a multiple-purpose spacecraft or family of spacecraft, examining its applicability to a wide variety of missions, and estimating the potential cost benefits.

1.2 OBJECTIVES

The primary objectives of the phase I study were defined as:

- Study of the projected NASA mission profile for the post-1979 period to determine the most probable spectrum of future mission requirements.
- Definition of hardware development requirements necessary in order to confirm the feasibility of solar-powered vehicles.
- Evolution of standardized spacecraft configurations utilizing three different nuclear power systems, which would be capable of satisfying the requirements of a wide variety of missions. This includes estimation of the size, weight, and cost of each configuration.
- Estimation of the overall potential savings to be achieved through use of standardized space vehicles, based on the government-supplied mission data.
- Definition of operational characteristics of the spacecraft, especially those relating to the nuclear power system and those which differ from characteristics of solar-powered vehicles.
- Definition of hardware development requirements necessary in order to confirm the feasibility of the proposed spacecraft concepts.

1.3 SCOPE AND DIRECTIVES

The study originally considered the entire unmanned portion of the NASA-von Braun - Fleming Mission Model, applicable in the post-1979 time period.

By direction, the following assumptions were made:

- The nuclear power systems considered are: radioisotope thermoelectric generator (RTG), radioisotope Brayton-cycle system, and the uranium/zirconium-hydride reactor thermoelectric system. The respective power ranges of the three systems are: up to several

hundred watts, from several hundred to about two thousand watts, and several thousand watts.

- The launch vehicle is the Space Shuttle, in certain cases aided by the Agena or Centaur (prior to 1983) or by the chemical tug (1983 and later).
- Of the total NASA Mission Model of July 1972, manned missions interplanetary missions, and Van Allen belt instability experiments are excluded because of their special requirements.
- Effort would be concentrated on power system designs applicable to a large number of missions and/or launches, such as in the 100-2000 watts power range. In particular, relatively little effort would be devoted to the high-power level reactor spacecraft.

In addition to those directed assumptions, the decision was made to postpone consideration of DoD missions to later phases of the study. The lack of an unclassified mission model has prevented their inclusion except by inference. Further, the indefiniteness of program information for pertinent NASA missions in the 1980's required that the Mission Equipment/Standard Spacecraft physical interface areas be extrapolated from presently defined earlier missions in the same program categories.

1.4 STUDY PROCEDURE

Information on the nuclear electric power systems was furnished by the Atomic Energy Commission.

The NASA Mission information was derived from the following two documents:

- Ref. 1-1 Aerospace Report ATR-72-(7231)-1 Vol. II
(August 1971) "Integrated Operations/Payloads/
Fleet Analysis Final Report"
- 1-2 Aerospace Report ATR-72(7312)-1 Vol. II
(August 31, 1972) "NASA Payload Data Book, Payload
Analysis for Space Shuttle"

Information dealing with the Space Shuttle was derived from the following three documents:

- Ref. 1-3 "Space Tug Economic Analysis, Final Presentation", Lockheed
Missiles and Space Co., Sunnyvale Calif. April 12, 1972.
- 1-4 Space Shuttle Baseline Accommodations for Payloads, May 1972,
NASA - Marshall Space Flight Center.
- 1-5 Space Shuttle Proposal - North American Rockwell Corp.

The cost model used in the economic analysis and some information on economic comparisons were drawn from the following:

- Ref. 1-6 "USAF Space Planners Guide", Air Force Systems Command
1-7 "Payload Effects Analysis Study", Lockheed Missiles & Space Co., Final Report Contract NAS W-2156.

In addition, information was obtained through meetings and discussions with scientists and engineers working on various space programs, in particular, on the following:

Earth Observation Satellite (EOS)
Synchronous Earth Observation Satellite (SEOS)
Tracking and Data Relay Satellite System (TDRSS)
Advanced Technology Satellites F and G, ATS-F/G

The steps involved in carrying out the study are shown sequentially in flow diagram form in Figure 1-1, with sequence indicated by arrows. Arrows on lines running upward represent the "feedback" of later stages upon preceding decisions, typical of any engineering effort.

1.5 CONCLUSIONS

The following are the fundamental conclusions derived in the course of the study:

- Analysis of the most recent NASA mission model for the post-1979 time period revealed that many different missions can be divided into just a few groupings of common power, weight, and size.
- Two standard radioisotope-powered spacecraft, using present-day technology, can satisfy 80-90 percent of the unmanned, earth-orbit missions listed in the NASA mission model for the 1979-1990 period. The spacecraft provide structure, thermal control, electrical power, attitude and velocity control (AVCS), and tracking, telemetry, and command (TTC) subsystems. The only missions excepted are those few which exceed either the weight-carrying capacity or the attitude control accuracy of the spacecraft as currently designed.
- Two nuclear power system designs, one for each spacecraft, can satisfy all the missions served. The systems are the Radioisotope Thermoelectric Generator (RTG) and the Radioisotope Brayton (RIB) system. For each spacecraft mission, power is provided as required in modular steps, either by addition of complete RTG's to provide power up to 450 watts(e) or by adding fuel capsules to the Brayton system to cover the range from 500 to 2000 watts(e). Both systems can use the same standardized heat source.

- Waste heat from the nuclear generators can be used to maintain spacecraft equipment temperatures within acceptable limits for all altitudes, sun angles, and equipment duty cycles. This is the key factor which permits standardization of the entire spacecraft despite the wide variety of mission conditions encountered.

- Although DoD missions were not included in the study, a separate review indicates that approximately 80 percent of these missions can be served by one or the other of the standard spacecraft designs.

- Savings of 64 percent for four missions using the standardized RTG spacecraft and 43 percent for six missions using the Brayton-spacecraft, in comparison with mission-specialized solar-powered spacecraft, were estimated on the basis of the cost model in the USAF Space Planners Guide. The savings over these missions, over and above those to be derived from the component and subsystem standardization program, are estimated to be 1.7 and 2.6 times the respective development cost of the standardized spacecraft. Thus the development cost can be amortized over relatively few missions and savings thereafter are appreciable. The calculations included both recurring and non-recurring costs, but excluded specialized payload equipment, launch and operations costs.

- Basic problems arising from the use of radioisotope fuel have been successfully met in previous missions. The routine use of nuclear systems in standard spacecraft should reduce the cost of the specialized ground handling and testing equipment required. A clearer definition of ground operations and safety requirements must await further specification of the shuttle ground time line and abort modes.

- For missions requiring approximately 5 kilowatts(e), the study demonstrated that a standard Reactor Thermoelectric System (RTS) can fit within the shuttle vehicle and be mated to two such diverse missions as a low altitude radar mapping experiment and a synchronous orbit communications system. In this power range, reactor electric power is less costly than solar-array power on a dollar per watt(e) basis. The mission flexibility of the RTS may be expected to provide a further cost advantage in DTE when multiple missions are met.

FIGURE 1-1. STUDY TASK FLOW DIAGRAM

2. BACKGROUND INFORMATION

2.1 GENERAL

In this section characteristics of the nuclear power system and of the launch vehicle (space shuttle) are discussed in general terms only, with particular reference to the interaction of these systems with other space-bus systems. Section 2.2 treats the nuclear power systems and section 2.3 is devoted to the launch vehicle.

2.2 POWER SYSTEM CHARACTERISTICS

This section describes some of the principal features of the three nuclear power generators studied. These are: the RTG (low power), the radioisotope Mini-Brayton generator (intermediate power), and the reactor thermoelectric system (high power).

2.2.1 RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG)

RTG's have been selected as the most applicable nuclear power systems in the low power range; 40-500 watts(e). These devices are characterized by long lifetime, power output stability, and completely static operation. RTG's that have been produced and flight qualified to date typically provide 40 watts(e) at the beginning of life (BOL). For the purposes of this study, the RTG systems are represented by the SNAP-19/Pioneer generator and Multi-Hundred-Watt (MHW) generator. The latter is expected to be operational by the end of 1974 and will provide approximately 150 watts BOL (Ref. 2-1). The pertinent characteristics of each of these generators are described below, with emphasis on those interface properties of most concern to the spacecraft.

2.2.1.1 Configuration

A cutaway view of the SNAP-19/Pioneer generator is shown in Figure 2-1, and of the MHW in Figure 2-2. Externally, both generators are ribbed, or finned, cylinders. Internally, the SNAP-19 unit consists of six flat panels containing thermoelectric couples deployed in a hexagonal arrangement around matching hexagonal faces of the heat source exterior. The MHW thermoelectric couples are mounted in a 12-sided configuration internally. The top and bottom faces of both units are electrically inert and thermally insulated.

FIGURE 2-1

40-WATT SNAP-19 RTG

150-WATT RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG)

FIGURE 2-2

The SNAP-19 thermoelectric elements are spring-loaded against the heat source, resulting in conductive heat transfer between fuel and converter. After fueling, the unit is closed with a calibrated mixture of Helium and Argon entrapped and is seal-welded around the top flange. This gas mixture, used to suppress sublimation of the thermoelectric materials, will leak through the Viton O-ring (which seals the electrical connector to the case) at a small, well-known rate. The released gas is replaced by Helium vented from the fuel capsule into the converter space at approximately the same rate. In advanced units, (e.g. SNAP-19/Viking, 1975 launch), the Helium produced by the fuel as a result of radioactive decay will be vented into a separate reservoir and the gas entrapped in the converter will be permanently sealed in. Because of the complexity of fuel insertion and gas-charging, the fueling and sealing operation of the SNAP-19 unit is generally done at the factory.

The MHW thermoelectric elements do not contact the heat source, but are separated from it by a small radiation gap. This simplifies the fueling procedure, since no special tooling is required to spring-load the elements during fuel insertion (as for SNAP-19). The end domes are fastened in place by 24 bolts and sealed to the outer can by metallic O-rings. The thermoelectric couples are also sealed to the outer can by metallic O-rings. As a result, an inert gas fill is required after fueling and while the unit is still on the ground, in order to suppress oxygen infusion into the hot interior of the generator. Although the seals are not perfect, they are expected to retain a significant gas charge for at least 30 days after fueling and before launch. Once the MHW system is launched, the gas must be permitted to vent to space by positive action, requiring an opening of a valve or vent tube.

The design of the MHW system makes in-flight fueling a possible option; for example, by using the shuttle-vehicle manipulators. This would eliminate the need for gas overpressure during ground operation.

Both RTG systems are designed for cantilever mounting to the spacecraft via mounting lugs at one end of the case as shown in Figures 2-1 and 2-2.

Table 2.1 lists the major parameters for each of these generators.

TABLE 2-1
REFERENCE RTG PARAMETERS *

Parameters	SNAP-19/ Pioneer	MHW
Nominal Max. Output Power, watts (e);		
BOL	40	150
3 years	32 design life	135
5 years	--	130
12 years	--	115 design life
Design Voltage at Max. Power, volts	4.2	30
Initial Heat Source Loading, watts	650	2400
Active Radioisotope (PU-238) half life, years	86.4	86.4
On-Pad Power Available, watts (e)	40	50-90
Generator; Length, inch	11	21
Envelope diameter, inch	20	16
Fin Height, inch	6.75	2
Case Material	Mg-Th	Be
Heat Source; Length, inch	6.5	16.60
OD, inch	3.5	7.25
Weight, lb.	30	85
Thermoelectric Couples, no.	90	312
series x parallel array	45 x 2	156 x 2
Nominal BOL Temperatures, °F		
radiator fin-root	330	550
TE hot junction	975	1830
heat source exterior	1050	1940
fuel centerline	1815	2600
Nuclear Radiation at 1 meter; R/hr neuts/cm ² sec.	0.3 3 x 10 ³	
Magnetic Field at 1 meter, gammas	40-70	(No data available)

*Data furnished by United States Atomic Energy Commission

2.2.1.2 Nuclear Safety Interface

The fuel capsules have been designed and qualified to withstand launch aborts and intact re-entry from earth orbit without additional protection. However, the ability of the fuel capsules to withstand launch-pad explosion and fire of the space-shuttle solid boosters and liquid propellant is not fully confirmed at this time. This capability is currently being studied under a NASA contract. If it is concluded that the fuel capsules cannot withstand the shuttle-peculiar abort environment, then the power systems will have to be designed for in-flight fueling using the shuttle's manipulators. The fuel would then have to be transported in blast shields or caskets, removed following orbit attainment, and inserted into the converters by the manipulators. It should be noted that this is essentially the procedure used in the Apollo program, where the SNAP-27 generator is fueled by an astronaut on the lunar surface using a specially designed tool.

For the purpose of this study, however, it has been assumed that the fuel capsules can withstand the shuttle's abort environment without further protection.

2.2.1.3 Startup and Control

In general, startup of an RTG will occur spontaneously once fuel is inserted and a heat sink provided to maintain the cold side temperature of the thermoelectrics at or near proper operating temperature. Because of the gas fill, discussed in Section 2.1.1, the MHW will only be capable of partial power on the ground (~ 50 We). The gas reduces the effectiveness of the thermal insulation, thereby increasing system heat loss and reducing the converter hot junction temperature. Therefore, the gas valve or vent tube must be opened at launch or in orbit in order to start up the system for full-power operation.

In-orbit control requires maintaining the proper DC load across the RTG terminals at all times. Methods for achieving this are discussed in Section 5.

2.2.2 RADIOISOTOPE MINI-BRAYTON

The Radioisotope Mini-Brayton (RIB) system has been selected as the most applicable nuclear power system in the intermediate power range (0.5-2 kWe).

The RIB is a dynamic conversion system using an inert gas as the working fluid in a single-phase closed loop. Although the system is expected to have a long lifetime without significant degradation, component and system tests to date have

generally been limited to the order of 10,000 hours. The completely sealed system using hydrodynamic gas bearings is expected to operate for much longer periods; providing, however, that hermeticity of the gas loop is not compromised. The only rotating component is a solid metal brushless rotor completely supported on a film of inert gas. Fully redundant electronic controls will further serve to enhance system reliability.

The system is designed for use with the same heat source being developed for the MHW-RTG. This heat source, designed for service at 2000°F, will be used at a maximum surface temperature of 1800°F in the Mini-Brayton system.

Recent studies under NASA contract have shown that the same Brayton system can be used with high efficiency over a large range of output power by varying only the heat input and system gas pressure. This is a particularly useful characteristic where system weight is not of primary importance. An attractive option under these circumstances is to design the system for minimum weight at a relatively high power level; e. g., using three MHW heat sources. This same system can then be used with either one or two heat sources with little or no loss in efficiency. In fact, with two heat sources, system efficiency is predicted to be somewhat higher because the oversized (for this case) radiator will reduce the compressor inlet temperature nearer to the optimum efficiency point. The projected operating conditions are listed in Table 2.2.

2.2.2.1 Configuration

The system consists of six major components, arranged schematically as shown in Figure 2-3. The turbine, alternator, and compressor are mounted together on a single shaft and enclosed as a unit; the Brayton Rotating Unit (BRU). The other major components can be located about the BRU in various ways, consistent with the desire to minimize pressure drops and heat losses in the connecting ducts. Figure 2-4 shows a photograph of a model with a single heat source heat exchanger in one possible configuration. Not shown on the photograph is the thermal insulation required on the exterior surfaces of all the components (except the radiator and its connecting ducts) to reduce parasitic heat loss due to radiation in space. For minimum size and weight, the insulation should be multifoil super-insulation. However, in order to permit ground tests of the system prior to flight, a high-temperature fibrous insulation such as Min-K, would be

TABLE 2- 2

REFERENCE BRAYTON PARAMETERS * *

Parameters	No. of Heat Sources		
	1	2	3
Net Conditioned Output Power, watts (e); BOL	689	1588	2273
Conversion Efficiency, %; BOL	28.7	33.1	31.6
Design Output Voltage, volts DC	120*		
Gas Flow Rate (approximate), lb/sec.	0.1	0.2	0.3
BRU Rotation Rate, RPM	50,000		
Gas Temperatures (nominal), °F			
radiator inlet	300		
radiator outlet	70 *		
compressor outlet	250		
recup. outlet	1250		
turbine inlet	1600		
Heat Source Skin Temp. (max.), °F	1800		
Weights, lb. ; BRU	25.3		
recuperator	68.4		
radiator	161.6*		
ducting	11.7		
HSHX	171.0		
superinsulation	78.0*		
structure	55.0*		
electronics	15.0		
parasitic load	20.0		
heat sources	45	90	135
TOTAL	651	696	741

* These values may not apply to the specific designs shown elsewhere in this report.

** Data furnished by NASA Lewis Research Center

SCHEMATIC OF RADIOISOTOPE BRAYTON SYSTEM

FIGURE 2-3

FIGURE 2-4

MODEL OF 550 WATT BRAYTON CYCLE GENERATOR

preferable. Therefore, applying guidelines which encourage the exchange of increased weight for reduced system cost and complexity, Min -K was selected as the primary insulation material for the RIB system in this study.

The system is designed for constant BRU rotating speed and power output. Therefore, a parasitic load must be provided to dissipate excess power. The control module automatically adjusts power to the parasitic load as a function of BRU rotating speed. The output of the alternator is 3-phase AC, which provides the frequency signal for the controller. The controller itself, however, operates on the DC side of the 3-phase rectifier, and consumes 40 - 90 watts of the output power.

The alternator is cooled by ducting the compressor discharge gas over the stator back iron. This appears to be a feasible cooling approach up to at least 2500 watts(e) output power, and permits the simplest gas-flow arrangement with very little loss of loop efficiency.

Table 2.2 lists the parameters of a RIB system assuming fixed hardware, but varying number of MHW heat sources, and gas fill to optimum pressure in each case. These parameters are based upon the latest available contractor data, and differ somewhat from data used elsewhere in this report (the latter reflect earlier NASA data predicting lower conversion efficiencies). If the higher conversion efficiencies are, in fact, achievable, then the Brayton spacecraft designed in this study will still be valid and will be conservative in its ability to dissipate the required heat load.

It should be noted that the RIB system is capable of producing power levels intermediate to those listed if heat source power is provided at intermediate levels. This opens the possibility of using partially decayed heat sources for missions requiring less power than the maximum available from fresh fuel capsules. This can greatly reduce the cost of fuel by substantially lengthening the useful lifetime of a recovered fuel capsule without reprocessing the fuel.

2.2.2.2 Nuclear Safety Interface

Since the RIB system uses the same heat source as the MHW-RTG, the nuclear safety considerations are quite similar to those described in section 2.2.1.2. If necessary, in-flight fueling can be done here as well; moreover, the RIB system offers a greater possibility for designing the heat source heat exchanger

(HSHX) as a self-contained blast and fire shield, if necessary, by adding protective members to the exterior of the unit. These members would also serve to attenuate the nuclear radiation emanating from the heat source.

If ground fueling is the accepted procedure, then means must be provided for cooling the heat source after fueling and prior to launch. This can be achieved by an auxiliary heat exchanger surrounding the working fluid heat exchanger, through which a cooling gas is circulated in an open loop (as described in Reference 2-2). Auxiliary cooling could possibly be removed prior to launch, because of the slow heating rate of the heat source without any cooling, resulting in at least a two hour period between removal of auxiliary cooling and achievement of start-up temperature (Reference 2-2).

2.2.2.3 System Start-Up

The system is started by allowing the HSHX to warm to operating temperature ($\sim 1700^{\circ}\text{F}$). At this point motoring power is applied to the alternator (450 Hz, 40V, 10A) for a short period (~ 1 minute). This accelerates the rotating unit to self-sustaining speed, from which point it bootstraps to rated operating speed. If the system is started while still attached to the orbiter vehicle, motoring power could be provided by the on-board power system. For remote starts, or to provide a restart capability, a battery and motor-start inverter would be added to the system (as shown in Figure 2-3).

2.2.3 REACTOR THERMOELECTRIC SYSTEM

This system has been chosen to cover the power range of approximately 3 to 10 kWe. It consists of a nuclear reactor cooled by a circulating liquid metal (primary coolant) which heats a group of compact thermoelectric converters. These converters are cooled by a secondary liquid metal loop which circulates around the converters and through the space radiator. Each loop is driven by a thermoelectric pump and contains a bellows-type expansion compensator to compensate for changes in coolant density as the reactor is throttled up or down in power.

A major feature of the reactor, as compared to radioisotope-fueled systems, is its ability to be shut down during ground handling and launch.

This essentially eliminates the need for auxiliary cooling. However, sufficient nuclear radiation may be present, as a result of pre-flight testing, to restrict ground handling procedures to some extent.

2.2.3.1 Configuration

The basic system schematic is shown in Figure 2-5, and the general configuration in Figure 2-6. The various components of the system are described functionally below. The system parameters are listed in Table 2-3. Most parameter data is available only for the 5-kWe system. The 10-kWe system is simply a scaled up version of 5-kWe system. It is estimated that the system could be scaled up to approximately 20 kWe with the same basic technology; limited only by the amount of burnable gadolinium poison which can be safely introduced into the reactor at BOL.

Reactor--The reactor consists of a cylindrical assembly of cylindrical fuel rods, radially surrounded by a hollow conical reflector (Figure 2-7). The reflector has four connected sliding segments which are moved for start-up and control. The fuel rods consist of an alloy of uranium-hydride and zirconium-hydride.

Shield--The shield consists of a disc of high density, high melting point ($\sim 1300^{\circ}\text{F}$) material; e.g., tungsten, for gamma shielding, followed by lithium-hydride for neutron shielding in the shape of a truncated cone with domed ends.

Converters--The converters are a group of tubular "compact-converter" modules (Figure 2-8), each module consisting of a solid stack of lead-telluride disc-like elements separated by solid wafer insulators. Each module has a power output of approximately 320 watts at nominal operating temperatures.

Coolant Systems--Both primary and secondary coolant systems use the eutectic mixture of sodium and potassium (NaK) which has a freezing point of 12°F . Each loop is pumped through one segment of a dual-throat electro-magnetic pump powered by additional converter modules, and is directed to flow through the center (primary) and around the outer surface (secondary) of the converter tubes. Each active heat transfer surface, including the fuel elements, contains helical vanes or swirlers to improve heat transfer to the coolant fluid.

REACTOR-THERMOELECTRIC SYSTEM SCHEMATIC

FIGURE 2-5

ZrH REACTOR THERMOELECTRIC SYSTEM

FIGURE 2-6

FIGURE 2-7 REFLECTOR-CONTROLLED U/ZrH REACTOR

Figure 2-7

● INNER DIAMETER (IN.)	0.75
● OUTER DIAMETER (IN.)	1.52
● ACTIVE LENGTH (IN.)	15.07

THERMOELECTRIC MODULE FOR REACTOR SYSTEM

Figure 2-8

TABLE 2-3

REFERENCE REACTOR/TE PARAMETERS * *

Parameters	5 kWe System		10 kWe System	
	BOL	EOM#	BOL	EOM#
Electrical Power at 30 VDC, KW				
Gross	5.13	5.13	10.3	10.3
Net at Mating Plane		5.0		10.0
Reactor Thermal Power, KWT	86.2	93.0	198	213
Reactor Outlet Temperature, °F	1140	1200	1142	1200
Flow Rate, Lbs/Sec				
Primary Loop	4.71	4.67	10.8	10.7
Secondary Loop	2.85	2.84	6.7	6.6
Pressure Drop, PSI				
Primary Loop	0.84	0.84	1.07	1.04
Secondary Loop	1.25	1.25	1.7	1.6
Converter Efficiency, %	6.83	6.29	6.1	5.6
Radiator Area, Ft ²		246		522
Base Diameter, Ft*		6		9
System Overall Height, Ft*		22.3		31.5
Radiator Cone Half-Angle, °*		8		8.5
Number of TE Modules				
Power		16		36
Pump		3		6
Total Weight, Lbs		1785		2940
Distance from Base Diam. to CG, Ft*		15.1		20.6
Temperatures, °F, Primary Loop				
Reactor Outlet	1140	1200		
Power Module ^A T	82	28		

Based on five-year lifetime.

* These values may not apply to the specific designs shown elsewhere in this document.

** Data furnished by United States Atomic Energy Commission

TABLE 2-3 (Continued)
REFERENCE REACTOR/TE PARAMETERS

Parameters	5 kWe System		10 kWe System	
	BOL	EOM#	BOL	EOM#
Temperatures, °F; Primary Loop (Continued)				
Pump Module ΔT	98	102		
Reactor Inlet	1060	1110		
Reactor ΔT	80	90		
Temperatures, °F; Secondary Loop				
Radiator Outlet	459	471		
Power Module ΔT	124	134		
Pump Module ΔT	145	156		
Radiator Inlet ΔT	584	607		
Radiator ΔT	125	136		
Temperatures, °F; Radiator Fin Root				
Inlet	582	605	(Data not furnished)	
Outlet	457	469		
Total Radiation Dose at Base Plane (20 ft), NVT RAD		10^{12}		
		10^6		
Weight; Lbs:				
Reactor		482		
Shield, Gamma		138		
Shield, Neutron and Case		223		
Converters		192		
EM Pumps		77		
Volume Accumulators (3)		38		
Piping		127		
Radiator/Structure		452		
Electric Wiring		46		
Miscellaneous		<u>10</u>		
Total		1785		<u>2940</u>

2.2.3.2 Nuclear Safety Interface

The major safety precaution which must be observed in a reactor system is to delay system startup until the desired orbit is achieved and verified. Although some ground testing may be anticipated, this will be limited in power and time so that the fission-product build-up does not result in a strontium-90 inventory which would represent an unacceptable hazard in the event of a launch pad fire or uncontrolled reentry. Because the fuel elements are not designed to withstand such launch aborts intact, the permitted level of fission products during launch would be small. This will simplify ground handling from the viewpoint of personnel access. Accessibility restrictions, less severe than for the radioisotope systems, may be expected to be minimal prior to start-up in space.

The availability of the shuttle will make reactor recovery feasible following completion of the mission. A recent study (Ref 2-3) has concluded that a reactor can be retrieved by the shuttle vehicle for return to earth without excessive irradiation of the crew, provided that retrieval occurs at least 10 days after shutdown, and that the system is positioned with the reactor at the aft end of the cargo bay. Whether or not such retrieval will be permitted, in view of the hazards which would be associated with a re-entry or landing accident, is not now known. If it is permitted, however, the use of reactors in lower earth orbits than are now permitted will be possible. If such retrieval is not permitted, however, then orbit restrictions may be expected to apply, in which case orbit lifetimes of the order of several hundred years will be required prior to reactor startup.

2.2.3.3 Start-Up and Control

Following orbit verification, the reactor will be started by ground command. The movable reflector segments will be stepped toward the closed position while neutron levels and system temperatures are monitored on the ground. Assuming that the proper precautions have been taken to prevent NaK freeze-up during the launch sequence, primary coolant pumping under battery power will be initiated when reactor fuel temperature begins to approach operating conditions. The primary system will then start coolant circulation in the secondary system. The entire start-up sequence may take several hours, in which case some added precautions may be necessary to prevent freezing of the coolant in the radiator prior to circulation of the secondary loop.

Following start-up, the system is self-adjusting to maintain constant thermal output. Short term changes in load requirements must be accommodated by control of the thermoelectric output. However, relatively long term changes in power requirements would probably be accompanied by a change of reactor thermal output via ground command. A reduced load requirement would then result in a reduction of all system temperatures and fuel burn-up, thereby prolonging the system lifetime.

2.3 LAUNCH VEHICLE

The Space Shuttle system consists of an Orbiter, a manned, reusable delta-winged vehicle with an expandable external propellant tank, and two solid rocket motors. Figure 2-9 depicts the space shuttle system and shows some basic dimensions. The concern here is with the Orbiter's delivery of payloads to earth orbits. The Orbiter's fuselage contains a payload bay which can accommodate single or multiple payloads within a cylindrical envelope 15.0 feet in diameter and 60.0 feet long. The payload is attached to the Orbiter outside this envelope. Figure 2-10 depicts the Orbiter and its payload volume. Reference 2-4 is the primary information source.

The payload capabilities of the Orbiter are : 65,000 pounds to a 100 nautical mile circular orbit with 28° inclination, 25,000 pounds to a 270 nautical mile circular orbit at 55° inclination, and 40,000 pounds to a 100 nautical mile circular polar orbit. The requirements of the payload center of gravity location relative to the Orbiter's forward payload bay wall are shown in Figure 2-11.

Delivering payloads to synchronous orbit requires an orbit transfer vehicle such as the proposed Tug. The payload capability depends on whether the Tug is reused or expended. Present data indicate that an expendable Tug can deliver a payload of about 10,000 pounds and a reusable Tug can deliver a payload of approximately 3,000 pounds to synchronous orbit.

The payloads can be deployed and retrieved by the orbiter via a remote manipulator arm system. In an orbital mission the manipulator arms can deploy a 65,000 pound payload in ten minutes at approximate rates of .05 feet per second and 0.1 degrees per second. Figure 2-12 depicts the use of the remote manipulator system.

FIGURE 2-9 SPACE SHUTTLE SYSTEM

FIGURE 2-10 ORBITER AND PAYLOAD VOLUME

FIGURE 2-11 PAYLOAD LONGITUDINAL CENTER-OF-GRAVITY LIMITS

FIGURE 2-12 REMOTE MANIPULATOR

Payloads mounted in the Orbiter will be subjected to acceleration loads, vibration, shock and acoustic environments. The preliminary limit load factors for the payload, as quoted in the literature, ~~may~~ may be summarized by the following quasi-steady factors for preliminary payload structural investigations:

<u>Condition</u>	<u>Nx</u>	<u>Ny</u>	<u>Nz</u>
Orbit Boost	3.25	<u>+0.75</u>	<u>+1.00</u>
Re-entry & Ldg.	-1.00	<u>+0.75</u>	-3.50
Flyback	-0.75	<u>+1.25</u>	<u>+2.75</u>

At the present time, there are no specific vibration criteria although available information indicates that the vibration environment should not exceed current launch vehicle payload environments. Shock spectra are also undefined at this time. The acoustic environment has been defined and indicates that the Orbiter's payload bay interior will not experience more than a 145 dB sound pressure level. The thermal environment of the payload during the launch and on-orbit sequence is also relatively undefined at this time. NASA document MSC-06900 lists some baseline thermal conditions for the space shuttle; however, as stated previously, a detailed knowledge of the payload design, payload mission, and shuttle design is necessary before the thermal interface can be defined. Dynamic environments for the orbiter payload will probably be more definitive as the shuttle program progresses.

The shuttle Orbiter system, with its large payload volume and weight capabilities, lends itself to the use of standardized vehicles (or "buses") with integrated nuclear-electric power sources for most anticipated future space-craft missions. The standardization will evolve around the three nuclear electric power generating systems discussed previously, and will be based upon analysis of the requirements of the different missions which are discussed in the following section (Section 3) of this report.

- Ref. 2-1 Multi-Hundred Watt Radioisotope Thermoelectric Generator Program, Annual Report, July 1970 - June 1971, General Electric Co. Nuclear Systems Programs, Report GESP-7071, GEMS-406.
- Ref. 2-2 NASA TM X-68072: Isotope Brayton Electric Power System for the 500 to 2500 Watt Range, R. P. Macosko, G. J. Bama, H. B. Block, B. D. Ingle, Lewis Research Center.
- Ref. 2-3 Manned Space Flight Nuclear System Safety, Doc. No. 72SD4201-1-2, January 1972, General Electric Co., Space Div, Philadelphia, Pa. Final Report, Contract No. NAS 8-26283.
- Ref. 2-4 Payload Design Requirements for Shuttle/Payload Interface, May 4, 1972, MSFC. Extracts re. Payload Interface, Proposal of North American Rockwell Corporation.

3. MISSION MODELS

3.1 INTRODUCTION AND SUMMARY

Forecasts of missions to be performed as part of the planned space activities during the 1978-1990 time period were studied in order to define the requirements for the diverse application and thereby arrive at realistic criteria for conceptual spacecraft designs. The primary forecasting references used in the study were those made by the Aerospace Corporation in 1971 and 1972 (References 1-1 and 1-2). Reference 1-1 is a compendium of all non-military space operations planned for the subject period and Reference 1-2 is a revised, and more detailed, description of NASA planned operations. It should be noted that Reference ~~3~~-2 indicates a reduction in total scope of space traffic from that planned in Reference ~~3~~-1.

Analyses of the forecasted missions indicates that considerable commonality exists for a majority of the missions, both in the 100-500 watt range and in the 500-2000 watt range. These missions generally are in the categories of Communications or Earth Observations and typically are performed in 700 n. mi circular orbits or in geosynchronous (19,323 n. mi) orbit.

3.2 APPROACH

References 1-1 and 1-2 served as guides for the survey of planned space activities for the 1978-1990 time period. The survey goal was to identify common basic characteristics for a majority of mission objectives and requirements in order to permit establishment of design criteria for standardized, modularized, multipurpose spacecraft using nuclear power subsystems. Upon further investigation, it became clear that many of the forecasted missions are not yet firmly defined and that specific mission requirements will undoubtedly depend upon such factors as availability of launch vehicles (both conventional and shuttle launched),

availability of "standard buses", and availability of ample power supplies. Thus the forecasts were viewed primarily as being indicative of generalized mission requirements enabling identification of basic design criteria.

The many mission types and resultant spacecraft characteristics were initially classified as follows:

(1) Purpose

- a. Scientific
- b. Application
- c. Operational

(2) Power Level

- a. Low power ($< 500W$)
- b. Intermediate power (500 to 2000W)
- c. High power (2kW or more)

(3) Size

- a. Low weight
- b. Intermediate weight
- c. High weight

Analysis of this initial classification identified those missions whose characteristics logically fall beyond a "normal" set of broad boundary conditions. For example, manned and interplanetary (Earth escape) missions were eliminated from further consideration for this phase of the study. Manned mission requirements will continue for some time to be too unique for the concept of "all purpose design" as well as being, generally, of too short a duration to permit efficient use of nuclear power. Earth escape missions are generally quite suitable for nuclear power. However, these were left as a separate category for future consideration, mainly because nuclear power is frequently selected for such missions on the basis of its unique deep-space capability. Also eliminated from further consideration in this first-phase study were certain experimental missions requiring highly specialized spacecraft, such as the radio astronomy observatory (consisting of three connected subsatellites forming a 10km Antenna) and two relativity experiments (requiring an angular

stability of 0.0005 arc sec/year--clearly a unique requirement).

The remaining missions were then examined in detail and categorized according to power level, payload size and complexity, and type of mission (see Tables 3.1 through 3.11). The results, grouped according to power levels, are summarized as follows:

a) Low Power Requirements: (100 to 400 watts)

Table 3.1 Astronomical Experiments

Table 3.2 Physics Experiments

Table 3.3 Communication/Navigation Missions

Table 3.4 Earth Observation Missions

b) Low-to-Intermediate Power Requirements (500 to 700 watts)

Table 3.5 Physics Experiments

Table 3.6 Communication/Navigation Missions

Table 3.7 Earth Observation Missions

c) Intermediate Power Range (900 to 2000 watts)

Table 3.8 Communication/Navigation Mission

Table 3.9 Earth Observation Missions

Table 3.10 Large Astronomical Observatories

d) High Power Range (4 to 10 kilowatts)

Table 3.11 Communication and Radar Mapping

These power level groupings are readily associated with the types of nuclear power sources being considered. The low power range (Tables 3.1 through 3.4) falls into the RTG category. The intermediate power range (900 - 2000 watts) is applicable to the RIB System and the high power range is suitable for the reactor thermoelectric system. The "low-to-intermediate power" range covered by Tables 3.5 through 3.7 includes those missions which could conceivably be performed using either RTG or RIB Systems; depending on the detailed mission requirements.

The conclusion drawn from these mission analyses, as reflected by an examination of Tables 3.1 through 3.11, is that a great deal of commonality is evident on both the mission and spacecraft levels. In fact, the vast

Table 3.1 RTG - ASTRONOMY MISSIONS

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FILMS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. ML. PERIG } APOG } INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	PURPOSE/SENSORS
EXPLORER (LEO)		NA2-1 4-1	8	40	171 373 373	300 300 28	BIAS MOM 10 SEC.	ASTRONOMY ----- OPTIC, X-RAY, γ-RAY
ASTRON. EXPLORER	NAS14A A4-14		15	100	250 860 860	270 270 28	3 AXIS 10 SEC.	SAME AS ABOVE ----- TV CAM, UHF, γ-RAY, LASER
EXPLORER (SYNCH)		NA2-2 4-7	8	40	171 373 ≈ 2800	SYNCH 28	BIAS MOM 10 SEC.	SAME AS NA2-1
ASTRON. EXPLORER	NAS14B A4-15		9	100	250 860 6000	SYNCH 0	3 AXIS 10 SEC.	SAME AS NAS-14A
SOLAR ORBIT PAIR A	NAS-7 A4-9		2	200	750 1820 ≈ 12,000	SYNCH 30	SPIN 4 DEG.	MONITOR SOLAR FLARES, SUNSPOTS, SOL. WIND ----- TV, X-RAY, COUNTERS & DETECT.
OSO	NAS-15 A4-16		1	300	500 1900 1900	350 350 30	3 AXIS 5 SEC.	MONITOR SUN IN X-RAY & γ-RAY SPECTR. ----- SPECTROMET; UV SENS X-RAY SENS VIDICON
OPT INTER- FEROMETER A & B	NA5-9 -10 A4-16		1	230	750 3010 ≈ 20,000	SYNCH ≈ 30	3 AX. 1 SEC.	PAIR TO MEASURE STELLAR DIAMETERS ----- LASER IR BOLOMETER

Table 3.2 RTG - PHYSICS MISSIONS

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. MI. PERIG } APOG } INCLIN. DEG	ATT. CONTR. TYPE ACCUR.	SENSORS
EXPLORER (UPPER ATMOSP. LOWER MAGNETOSPHERE)	NSP-1 A4-17	NP2-13 5-1	12	100	100 1160 ≈ 1600	180 1800 90	SPIN* +2 DEG.	ELECTRON MULTIPL.; VLF; CURR. COLL.; UV DETECT MASS SPECTROM; DRAG; ELECT. FLD; MAGNETOM.
EXPLORER (MIDDLE MAGNETOSPHERE)	NSP-2 A4-18	NP2-14 5-7	12	100	100 **** 570 2300	1000 20,000 0 → 90	SPIN +2 DEG.	SOLID ST. DET; MAGNETOM; ION & ELECTR. CURRENT; PHOTPMULT; SCINTILL CTR; E-FIELD; CHARGE COLL.
EARTH PHYS. SAT.	NEO-5 A4-25		7	200	150 580	400 400 90	3 AXIS 1 SEC.	MICROWAVE ** LASER IMAGE DISSECT.
GEOPAUSE		NE2-45 7-45	2	100 to 400	125 710 ≈ 3000	≈ 12000 ≈ 12000 *** 0	3 AXIS	PROOF MASS PRECIS RADAR RANGING LASER
	* DUAL SPIN VARIABLE RATE, ALSO DESPIN MODE ** GEODESY, CONTINENTAL DRIFT, SURFACE STRAIN *** DRAG-FREE ORBIT, ZERO ECCENTRICITY **** SLIGHT WEIGHT DIFFERENCES, NSP-2 AND NP2-14							

Table 3.3 RTG - COMM/NAV

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. MI. PERIG } APOG INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
SATS		NC2-47 8-7	24	150	40 300 2100	SYNCH 0	3 AX. 0.1 DEG.	TRANSPOND; ANTENN COMMUN. EXP.
SATS (POLAR)		NC2-48 8-13		100	40 300	300 300 → 3000 90	3 AX. 0.1 DEG.	SAME
COOPER APPL.	NCA3A A4-36		2	420	250 820 ≈ 6000	SYNCH 0	3 AX. 10 SEC.	HF; VHF; SOL. ST. DET.
COOPER APPL. (POLAR)	NCN3B A4-37		2	420	250 820	300 3000 90	3 AX. 10 SEC.	SAME
FOREIGN DOMEST. COMM.	NCN-9 A4-41		26	230	285 1000 7500	SYNCH 0	SPIN 0.2 DEG.	ANT; TRANSM; UHF; VHF; HF.
NAV./ TRAFFIC	NCN10A A4-42		6	200	135 700 5000	SYNCH 0 → 5°	3 AX. 0.5 DEG.	HF; VHF; UHF
NAV./ TRAFFIC	NCN10B A4-43		10	200	135 700	16,000 30,000 29	3 AX. 0.5 DEG.	SAME

Table 3.4 RTG - EARTH OBSERV.

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLTS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. ML. PERIG } APOG } INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
TOS METEOROL	NEO-7 A4-27		12	200	250 1000 ≈ 1500	700 700 100	3 AX.	BOLDMET; TV CAM; VIDICON
TIROS-0		NE2-40 7-15	12	250	580 1380 ≈ 2000	900 900 * 103	3 AX. 0.15 DEG.	RADIOMET; VERT. TEMP. SOUND; DATA BUOY COMMUN.
TIROS 4th GENER.	NEO-6 A4-26		3	200	245 1000	700 700 100	3 AX.	RADIOM; SCAN RADIOM; TEMP. RADIOM.
SYNCH. MET. SAT.	NEO-8 A4-28		2	300	250 1000 ≈ 7500	SYNCH 0	3 AX. 10 SEC.	VIS SCAN; IR SCAN; ENVR. MON; DATA COLL; DATA RELAY.
SYNCH. MET.	NEO-15 A4-30		12	300	250 1000 ≈ 7500	SYNCH 0	3 AX. 10 SEC.	SAME
SYNCH EARTH RESOURCES	NEO-4 A4-24		7	400	350 1000 ≈ 7500	SYNCH 0	3 AX. 10 SEC.	IR SCAN RADIOMET; μ WAVE; IMAGE SPECT.
SYNCH EARTH RESOURCE	NEO-11 A4-29		8	400	350 1000 ≈ 7500	SYNCH 0	3 AX. 10 SEC.	T.V. CAM; IR SCAN; μ WAVE; RADAR

* SUN SYNCHRONOUS ORBIT

Table 3.4 RTG - EARTH OBSERV. (CONT'D)

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. ML. PERIG } APOG } INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
SYNCH EARTH OBSERV.	NEO3 A4-23		6	400	350 1000 ≈ 7500	SYNCH 0	3 AX. 10 SEC.	EAR SENS; μ WAVE; METEOR SENS; IR SCAN; SPECTROM
SYNCH EARTH OBSERV.		NE2-43 7-33	6	400	1400 2640 ≈ 20K	SYNCH 0	3 AX. 0.1 DEG.	IR; VISIB RADIAT; μ WAVE SCAN; IMAGE SPECTROM.

Table 3.5 RTG OR BRAYTON - PHYSICS MISSIONS

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. ML. PERIG } APOG INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
GENERAL RELATIVITY	NSP6 A4-20		2	700	350 1450	300 300 90	3 AX 1 SEC	GYROSCOPE; CHG. PARTICLE; IR; MAGNETOMETER

Table 3.6 RTG OR BRAYTON - COMM/NAV

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. MI. PERIG } APOG INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
TRACK & DATA RELAY		NC2-49 8-19	10	430	219 1760 ≈12K	SYNCH 0	DUAL SPIN 0.1 DEG	HF; UHF; VHF; LASER; SOLID STAT DE LINKS TO OTHER SATS & GROUND.
TRACK & DATA RELAY	NCN-5 A4-38		10	680	600 2300 ≈16K	SYNCH 0	SPIN 0.1 DEG	COMMUN; CMD; TRACK; VHF; LASER.
COMMUN SAT	NCN-7 A4-39		11	585	355 1420	SYNCH 0	SPIN 0.2 DEG	MULT. ANT; TRANSPONDER; UHF; LASER; VHF.
SMALL ATS	NCN-2A A4-34		12	500	150 600	300 3000 0 → 90	3 AX 10 SEC.	TV; IR; LASER; CHG. COLLECT NANO-G.
SMALL ATS	NCN2B A4-35		12	500	150 600 ≈4500	SYNCH 0	3 AX. 10 SEC.	SAME
SYSTEM TEST SAT.		NC2-51 8-33	8	600	315 2860 ≈20K	SYNCH 0	3 AX. 0.1 DEG.	VHF; UHF; SHF; EHF.

Table 3.7 RTG OR BRAYTON - EARTH OBSERV.

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. MI. PERIG } APOG INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
POLAR EARTH OBS.	NEO2 A4-22		12	600	850 2500	500 500 100	3 AX. .07 DEG	μ WAVE; IR RADIOM; RADAR SCATT; IR SCAN; OPTICS
POLAR ERS	NEO16 A4-31		22	600	850 2500	500 500 100	3 AX. .04 DEG.	IMAG RADIOM; IR RADIOM; μ WAVE; RAD. SCAN; IR SCAN.
POLAR ERS.	NEO-17 A4-32		6	600	850 2500	500 500 100	3 AX. .04 DEG	μ WAVE; VISSCAN; IR RADIOM; IR SCAN; RADAR

Table 3.8 BRAYTON - COMM/NAV

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. ML. PERIG } APOG } INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
US DOMEST COMMUN	NCN8 A4-40		21	900	1000 3425	SYNCH 0	SPIN 0.2 DEG.	MULT. ANT; UHF; TRANSPONDER; VHF LASER.
DISASTER WARNING		NC2-50 8-27		900	500 1760 ≈ 12K	SYNCH 0	3 AX. 0.1 DEG.	VHF UHF SHF
FOLLOW ON SYST DEMONST.	NCN13 A4-46		20	1000	315 2000	SYNCH 0	3 AX. 0.2 DEG.	COMM; NAVIG; DATA RELAY; EARTH RESOUR.
ATS		NC2-46 8-1	6	1000	488 3000 ≈ 20K	SYNCH 0	3 AX. 0.03 DEG.	SPEC. ANTEN.; IKW 12GHZ TRASM.
MEDICAL NETW	NCN-11 A4-44		2	1000	550 2000	SYNCH 0	SPIN -	UHF VHF HF LASER
EDUC BROAD- CAST	NCN-12 A4-45		2	2000	1000 3400	SYNCH 0	SPIN 0.2 DEG.	UHF; LASER VHF; HF

Table 3.9 BRAYTON - EARTH OBSERV.

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLITS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N. MI. PERIG } APOG } INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
EARTH OBSERV.		NE2-38 7-1	8	1200	910 2400 ≈ 3000	500 * 500 98	3 AX. 0.15 DEG.	RADIOMETER; SPECTROPHOTOM; THEMATIC MAP; ATMOSPH & POLLUT. SENS.
SEOS (SYNCH E. OBS)		NE2-39 7-7	5	1500	900 2500 ≈ 18K	SYNCH 0 AND 30	3 AX. 0.1 DEG.	IMAGE SPECTR & IR SENS; OR: μ WAVE SENSOR**; OR FILM CAMERA
EARTH RESOURCES		NE2-42 7-27		1200	310 1800 ≈ 2000	500 500 98	3 AX. 0.15 DEG	SAME AS NE2-38
	* SUN-SYNCHRONOUS ** HIGH DATA RATE							

Table 3.10 BRAYTON - LARGE ASTRON. OBSERV.

NAME	REF. (1971) NO/ PAGE	REF. (1972) NO/ PAGE	NO. OF FLTS	POWER WATTS	WEIGHT SENS. IN ORBIT IN SHUTTLE	ORBIT N.MI. PERIG APOG INCLIN. DEG.	ATT. CONTR. TYPE ACCUR.	SENSORS
LARGE SPACE TELESCOPE	NAS-1 A4-5	NA2-5 4-27	3 (5)	1462 (1500)	12K (8K) 18K (21K)	330 330 28.5	3 AX. CMG 1 SEC.	3 METER OPTICS; TV; PHOTO; SPECTRO- PHOTOMET; POLARIMETER.
LARGE SOLAR OBS.	NAS-2B A4-6	NA2-7 4-41	1 (4)	2000 (1500)	16K (7K) 37K (27K)	300 300 28° → 90°	3 AX CMG 1 SEC. (0.1 SEC) *	PHOTOHELIOGRAPH AND TELESC; SPECTRO- HELIOGRAPH; UV SPECTROHELIOGR; CORONA; X-RAY; ETC.
LARGE RADIO OBSERV.	NAS-3 A4-7		3	2000	10K 19K	350 350 30	3 AX. 1 SEC.	BOLOMETER; ELF DETECT; VHF DET; UHF; μ WAVE.
HIGH ENERGY ASTRON	NAS-4 A4-8	NA2-3 4-13	4 (6)	750 (820)	10K (12K) 18K (21K)	250 250 28.5	3 AX. CMG 1 MIN.	TELESC; SPECTROM; (X-RAY)
HIGH ENERGY TELESC. (X-RAY)		NA2-9 4-55	1	1500	9K 16K	400 400 28to90	3 AX CMG	X-RAY SOURCES (QUASARS PULSARS) HI-RESOL. X-RAY TELESC; COUNTERS; ETC.
	* SUN-DIRECTED							

Table 3.11 REACTOR MISSIONS

[illegible]

majority of missions may be satisfactorily performed with a rather small series of spacecraft designs. The results of the analysis are amplified in the following paragraphs of this section.

3.3 LOW POWER MISSIONS

Tables 3.1 through 3.4 briefly describe the characteristics of missions in the low power range (single or multiple RTG's). These missions are:

- Astronomy Experiments - require spacecraft whose function is to make astronomical observations in various regions of the electromagnetic spectrum.
- Physics Experiments - generally associated with the measurements of environmental factors near Earth.
- Communication/Navigation Missions - satellite communication with Earth stations and/or with other satellites.
- Earth Observation Missions - Earth viewing applications for meteorology and Earth resources observation.

The lists include both the Reference (1-1) and Reference (1-2) descriptions of often identical missions, and were used with discretion, in the knowledge that specific mission details are, in many cases, still conceptual.

Tables 3.1 through 3.4 indicate that an RTG-powered spacecraft may accomplish the majority of the listed missions provided that it is capable of:

1. supplying 100 to 400 watts of power, as required,
2. supporting an in orbit weight range of 1000 to 1500 lb.,
3. Earth pointing control of 0.1 degree accuracy, with a similar accuracy pointing at the sun (with some scanning capability).

The majority of missions require Earth-oriented operation from either a synchronous equatorial orbit or from a near polar, circular orbit of 700 to 1000 n. mi. altitude.

The spacecraft should be designed for three-axis stabilization during normal orbital operation to satisfy sun-, star-, or Earth-pointing requirements, and to provide for angular offset pointing or scanning as necessary. (A few missions are presently planned as "spinners", according to Tables 3.1, 3.2, and 3.3; however, the requirements of these missions could be met by fully stabilized spacecraft.) Provisions must be made for the installation of a variety of Earth-viewing sensors, sunviewing sensors, and for the deployment of a variety of antennas. Thermal control for all spacecraft and payload equipment must also be provided.

Specific capabilities beyond the above must be provided as part of the payload. For example, astronomical payloads with pointing requirements of 10 arc-seconds will probably require a vernier control system (e.g., highly accurate sensors and fine control wheels) in addition to the "coarse" system (of the order of 0.1° accuracy). Similarly, telemetry capacity beyond the normal capacity of the USB system should be provided by the payload.

Analysis of Tables 3.1 through 3.4 indicates that most missions will require spacecraft in the 700 to 1500 lb. range. A single spacecraft could readily meet this requirement. By its inherent nature, such a vehicle will probably be overdesigned, and, therefore, too heavy for some of the smaller listed missions (400 - 500 lbs). However, the use of small payloads in a standard spacecraft, even though not weight effective, may well be cost-effective when the relatively low transportation costs of the shuttle are considered, along with the cost of developing a new custom spacecraft for such missions.

3.4 INTERMEDIATE POWER MISSIONS

Tables 3.8, 3.9, and 3.10 reveal two distinct sub-groups within the intermediate power class. Tables 3.8 and 3.9 list spacecraft requirements

and configurations similar to the previously examined low-power range, but with higher power levels and with weights in the two-to-four-thousand pound class. Table 3.10 lists the large astronomical observatories, with weights of the order of 10,000 pounds, some of which have greatly increased equipment and operations complexity. The large observatories are thus fundamentally different from the applications-oriented Earth-viewing or communications vehicles listed in Tables 3.8 and 3.9. The differences are not only a matter of size; the large observatories require more precise pointing control and, therefore, superior thermal control of large optical structures.

It was concluded that, if all mission requirements in this class are to be met, at least two different spacecraft designs would be needed. Due to the limited scope of this study it was decided to concentrate the effort toward meeting the requirements of the Earth-oriented missions and to leave for future consideration the possible implementation of a common observatory design. On this basis, then, all remaining missions in this class are Earth-oriented, in either equatorial, synchronous or near polar orbit, at 500 - 700 n. mi. altitude. (The latter may be sun-synchronous orbits, maintaining constant local time at the nodes.) Power requirements range from 900 watts to 1.5 kW (one exception at 2 kW), with in-orbit weight in the 1800 to 3500 lbs. range. Three-axis attitude control (one axis being vertical) is required with an accuracy of the order of 0.03 to 0.1 degree.

Provisions must be made for sufficient Earth viewing area to accommodate all the sensors, as well as for the deployment of various antennas. Provisions should also be made for offset pointing to any location on the visible hemisphere, and for thermal control of all payload equipment. Such a spacecraft will be capable of meeting all the requirements of the listed missions.

3.5 LOW-TO-INTERMEDIATE POWER REGIME

Sections 3.3 and 3.4 above discussed anticipated mission requirements in the 100 to 400 watts range, and in the 900 to 1500 watts range. There also exist planned missions in the range of 400 to 700 watts (as listed in Tables 3.5, 3.6, and 3.7) which are categorized by payload as follows:

Table 3.5	Physics Experiment
Table 3.6	Communications/Navigation
Table 3.7	Earth Observation

Detailed examination of each category indicates that the requirements for missions in Table 3.6 and 3.7 do not differ substantially from those which would form the basis of an RTG or RIB powered spacecraft design, except for the power level.

Further study and economic analysis indicates that the Tracking and Data Relay and Systems Test Satellites in this series would be more economically mated to the Brayton spacecraft. It may be supposed that this will be true for the other missions in this series as well, but these were not studied in detail.

The physics relativity experiment of Table 3.5 was not considered further in this study, because of the stringent attitude control requirements and special precautions needed to minimize thermal distortion and local mass shifts. These requirements are not compatible with the philosophy of a universal, standardized and relatively inexpensive design.

All of the missions in the low and medium power ranges, except for the physics experiment of Table 3.5 and the large observatories of Table 3.10, have been summarized in Tables 3.12 and 3.13, covering the power ranges below and above 500 watts, respectively. These tables indicate that classification by power level in general provides a similar classification by weight. The two tables also indicate a high degree of

TABLE 3.12

MISSION CHARACTERISTICS - LESS THAN 500 WATTS REQUIRED

(1971 and 1972 Mission Models)

PURPOSE	LOW ORBITS					SYNCHRONOUS ORBITS			
	No. of missions	No. of flights	Elec. power watts	Nominal s/c Wt. lbs.	Orbit N. Mi.	No. of missions	No. of flights	Elec. power W	Nominal s/c Wt. lbs.
Earth Observ.	2	15	200	1000	700	5	35	300/400	1000
	1	12	250	1400	900	1	6	400	3000
Comm. Nav.	3	12	420	1000	Ellipt.	1	24	150	500
						3	34	200/400	1000
						1	10	430	1000
Physics	1	7	200	600	400 P				
	3	26	400	1000	Ellipt.				
Astronomy	1	8	40	400	300	1	8	40	400
						1	9	100	1000
	1	15	100	1000	300	1	2	200	2000
	1	1	300	2000	300	1	1	230	3000
TOTALS	13	96				15	129		

Table 3.13

MISSION CHARACTERISTICS -OVER 500 WATTS REQUIRED

(1971 & 1972 MISSION MODELS)

	SYNCHRONOUS ORBITS				LOW ORBITS				
	NO. OF S/C	NO. OF FLTS.	ELECT. POWER WATTS	NOMINAL S/C WT. LBS.	NO. OF S/C	NO. OF FLTS.	ELECT. POWER WATTS	NOMINAL S/C WT. LBS.	ORBITS N. M.
EARTH OBSERV.	1	5	1500	2500	3	40	600	2500	500P
					2	9	1200	2500	500P
COMM./ NAV	1	12	500	600	1	12	500	600	ELLIP.
	1	11	580	1500					
	4	33	700/1000	2000					
	2	14	600/1000	3000					
	2	23	900/2000	3500					
TOTALS	11	98			6	61			

commonality in orbit requirements and direction-maintaining or orientation requirements.

3.6 HIGH POWER APPLICATIONS

Table 3.11 lists the anticipated missions requiring power in the multikilowatt range. Present indications are that only the direct broadcast mission from synchronous altitude and a polar orbit low altitude radar mapping mission qualify in the power range compatible with the reactor thermoelectric system (4 to 10 kW). The side-look radar mission is not listed in either Reference (1-1) or (1-2), but initial planning and consideration are being given to this mission. Neither of these missions are specified in sufficient detail to lead to specific design criteria. However, the principal design constraints and the bulk of the spacecraft volume will be dictated by the reactor-thermoelectric and heat rejection systems. Thus, for these missions, the additional requirements are:

- a) Direct broadcast: installation of high power (4 to 8 kW) transmitters and associated antennas; provision for Earth-Oriented control in synchronous orbit.
- b) Side-look-radar mapping: a typical requirement consists of mounting a 24-ft.-long, 30-inch-high slotted waveguide antenna longitudinally (with a "look angle" depressed approximately 48 degrees from the horizontal) to map a swath of 45 n. mi. width from an altitude of 300 n. mi. Power requirements are of the order of 5 kW. For further details, see section 5.

3.7 SELECTED TEST MISSIONS

The design goal established for the study is the generation of spacecraft configurations and subsystem designs that enable the performance of any of the forecasted missions. In order to arrive at an assessment of the spacecraft's ability to attain this desired goal, a "test" of candidate design concepts must be made. For purpose of this study, the "test" was defined as the ability of a potential design concept to satisfy

the requirements of a set of selected missions which are representative of all the forecasted missions. Specifically, a potential design should be evaluated for both Communications and Earth Observations missions, for both Low-Earth polar orbits and geosynchronous orbits, and for missions requiring the overall range of electrical power. The set of representative missions selected for design evaluation is shown in Table 3.14.

TABLE 3.14
SELECTED MISSIONS FOR DESIGN EVALUATION

Spacecraft	Mission/Description
1. RTG	a. TIROS-O (Table 3.4) - Intermediate Weight and Power, circular near-polar orbit, Earth Observations b. Track/Data Relay Sat (Table 3.6) Maximum weight and power for RTG series, geosynchronous orbit, Communications
2. Brayton	a. ATS (Table 3.8) - 1kW, near maximum of weight range, geosynchronous orbit, Communications experiments. b. EOS (Table 3.9) - 1.2 kW, middle weight range, near-polar orbit, Earth Observation
3. Reactor-Thermo-electric	a. Side-Look Radar Mapping (Table 3.11) - polar orbit, lower end of reactor power range.

4.0 STANDARDIZED SPACECRAFT DESIGN APPROACH

4.1 REQUIREMENTS

Analysis of the mission model establishes mission system functional requirements which in turn define subsystem requirements. Table 4.1 lists a number of spacecraft functions necessary to many missions and the subsystems which perform these functions. The subsystem requirements, together with conceptual design approaches to their solutions, provide design criteria for spacecraft and subsystems, which for standardized spacecraft must match the requirements of a wide variety of missions.

System requirements for the standardized spacecraft include the following:

1. Meet launch vehicle (shuttle and tug) space, weight, interface, and environmental constraints.
2. Provide the payload with a satisfactory space-weight envelope, mechanical and thermal interfaces, and environmental conditions (temperature, linear and angular motion, radiation) during pre-launch, injection and on-orbit operation. This includes means of quick access to all spacecraft/payload components for replacement or repair.
3. Function satisfactorily with adequate system life in any orbit contemplated in the selected mission model.
4. Provide orbit maintenance, i. e., stationkeeping maneuvers and minor orbit adjustment, for specified mission duration.
5. Provide electrical power to the payload as required for different missions without significant change to the spacecraft mechanical and thermal design or to equipment placement.
6. Provide attitude control on orbit and provide an unobstructed view for directional payloads.
7. Provide tracking, telemetry, and command (TT & C) functions for mission accomplishment and for payload requirements up to some specified data rate.
8. Provide for isotope fuel loading of the nuclear power system at the later stages of prelaunch.

TABLE 4.1 SPACECRAFT FUNCTIONS AND SUBSYSTEMS

<u>FUNCTION</u>	<u>SUBSYSTEM</u>
Physical support of payload and service elements, tie to booster	Structure
Provision of satisfactory environment, disposal of heat	Thermal control, shielding
Provision of electrical power	Power
Propulsion for Stationkeeping, orbit adjustment	Propulsion or AVCS
Orienting spacecraft as required	AVCS (+ TTC if ground command)
Provision of unobstructed view for directional elements	Structure + AVCS
Provision of communications for payload and service elements (limited rate)	TTC
Provide command and control of spacecraft	TTC

These requirements lead to the design objectives set forth in the next section.

4.2 DESIGN CRITERIA

Based on the mission objectives and requirements discussed in Sec. 3, coupled with the power system characteristics outlined in Sec. 2 and the subsystem requirements of the preceding section 4.1, the following design goals and criteria were established for the standardized spacecraft.

(i) Electrical Power Source

Spacecraft with electrical power requirements in the 100 to 450 watt range can be served by RTG units, with power supplied at modular levels of 150, 300, and 450 watts by the use of one, two, or three MHW-RTG units. Most of these spacecraft are in the low weight range.

Spacecraft with 500 to 2200 watt electrical power requirements can be served by the Brayton system, also working at modular power levels fueled by one, two or three of the same radioisotope heat sources used in the MHW-RTG (see Sec. 2.2.2.).

The upper range covered by the RTG unit can also be served by the Brayton unit; in this power range it is necessary to trade off the higher weight and volume of the Brayton unit and its associated spacecraft against the use of three RTG's and the resultant higher recurring cost of the power system.

(ii) Power System Voltage

The voltage supplied by the spacecraft should be consistent with the NASA subsystem standardization program. For the purposes of this study, it has been assumed that the spacecraft will furnish a regulated supply, $28V \pm 2\%$.

(iii) Weight

An in-orbit weight of up to 2000 pounds, providing for up to 750 pounds of mission equipment, appears to satisfy almost all of the missions at power levels of less than 500 watts, and is an appropriate design goal for this spacecraft.

For the spacecraft to be powered by the Brayton unit, in-orbit weight of 4000 pounds, with payload of up to 1900 pounds, covers all of the anticipated missions except for the large observatories. Accordingly these values may be set as design goals.

No weight goals were set for the reactor spacecraft because of the scarcity of definite requirement data.

(iv) Propulsion-orbit adjustment.

Many missions have no requirements for propulsion for orbit adjustment. A reasonable goal is the requirement for stationkeeping of synchronous satellites for six years (typical for communication satellites). The same level of propulsion would then be able to perform a certain amount of orbit adjustment. For example, a ΔV capability of 350 m/s (1170 ft/sec) could provide either the synchronous orbit stationkeeping, or could circularize an orbit with a 700 to 1000 mile apogee and a 100 to 270 mile perigee altitude, thus easing the tug requirements for placing payloads in orbits beyond the shuttle parking orbits.

(v) Attitude Control

Three-axis stabilization is required for many of the missions and is therefore a design requirement. Control within accuracy of 0.1 degrees satisfies most of the missions and can be set as the objective; attitude control is to be achieved by automatic on-board means. Present experience suggests firing of reaction jets should not be required more than twice per week. Since large external appendages are common, the system should be conservatively designed to stabilize against suitably large unbalanced solar pressure torques. Control accuracy of greater than 0.1 degree, achievable by more precise sensors, actuators and control systems, should be considered as an add-on feature for those missions requiring it.

The spacecraft shall be capable of being pointed in any direction.

Moments of inertia should be suitable for stable spinning of the spacecraft, either where this is the operational orbital mode, or if required by the propulsion system.

(vi) Telemetry

A bit rate of 10^6 bit/second is consistent with current frequency assignments and with system needs of both standard spacecraft plus payload needs for a large number of missions. This value can be set as the design goal; when payload requirement exceed this value the additional equipment can be considered as part of the mission equipment.

(vii) Temperature Control

A temperature range of between -10°C and $+40^{\circ}\text{C}$ for all equipment compartments, to be maintained in all orbits and all conditions, is a reasonable design goal, and quite conservative for most electronics and sensor packages.

The design goals are briefly summarized in Table 4.2.

TABLE 4.2 STANDARDIZED SPACECRAFT DESIGN GOALS

	<u>RTG S/C</u>	<u>BRAYTON S/C</u>
1. Power:	Up to 450 W(e) in modular steps	500 to 2200 W(e) in modular steps
2. Weight:	1000 to 2000 lbs	up to 4000 lbs
<u>RTG and BRAYTON S/C</u>		
3. Ferrying:	Stacking capability within shuttle, Agena/Centaur or chemical tug interface as necessary.	
4. Propulsion:	Small orbit change or stationkeeping capability.	
5. Power:	28 Volt DC, $\pm 2\%$ regulation	
6. Attitude Control:	Three-axis stabilization, 0.1 degree accuracy	
7. TT & C:	Standardized, 1 MBS data rate	
8. Orbit:	Low orbit, any inclination, up to synchronous orbit	
9. On-orbit life:	3 year minimum	
10. Thermal environment:	Equipment thermal environment between -10°C and $+40^{\circ}\text{C}$ for all orbits, spacecraft orientations, sun occultations, and spacecraft power levels.	
11. Pointing:	Any direction, including earth-oriented	

4.3 RTG SPACECRAFT CONFIGURATION SELECTION

The chief thermal-mechanical problem in a spacecraft powered by the RTG system is disposing of the waste heat emanating from the isotope source, which is significantly higher than the total input from any other source. A thermal management system should provide temperature regulation for the spacecraft and contained equipment and dispose of the excess heat to space by radiation. A good design, in the process of disposing of a large heat flow from the nuclear source, should be able to smooth out the highly variable but smaller magnitude heat inputs from other sources (primarily the sun, earth, and on-board equipment), and provide a relatively constant thermal environment for the equipment.

Three approaches to thermal management are possible. One is to isolate the RTG from the spacecraft and to merely radiate its heat to space directly. This does not make use of the waste heat from the nuclear source to permit thermal regulation of the spacecraft.

Another approach is to integrate the RTG with the spacecraft and use the spacecraft walls to radiate the heat to space. In order to produce a heat rejection temperature low enough to provide benign environment to the payload, a large area spacecraft wall is required.

The third approach is to dispose of most of the heat from a high temperature radiator external to the spacecraft, and allow enough of the heat to flow into the spacecraft, in order to offset variations from other sources. Temperature differences between different locations within the spacecraft can be minimized by appropriate means such as heat pipes.

Preliminary consideration of the three options led to adoption of the third approach for further detailed design study.

It is necessary to achieve this thermal control in a spacecraft configuration with reasonable dimensions, symmetry, and moments of inertia. An initial approach was to provide mass balance and uniform thermal radiation to the spacecraft equipment from the RTG radiator by mounting the nuclear system and the spacecraft equipment on opposite ends of the vehicle. This arrangement was incompatible with a reasonable location of the thruster, which must act along an axis of symmetry which is the major axis of inertia to provide for stable spin during thrust. The thruster had to be located either between the two faces or coincident with one of them.

Therefore, RTG mounting locations were placed symmetrically around the spacecraft spin axis or center line, giving the vehicle the form of a short cylindrical shell from which the RTG's are cantilevered. Figure 4-1 shows the overall configuration that was adopted. With this shape, mounting the payload equipment on the two faces of the cylinder provides both symmetry and a clear view for directional apparatus, together with relatively uniform thermal radiation from the nuclear heat source. The thruster may then be mounted along the axis of the cylinder. For the range of missions anticipated, with different propulsion requirements and with multiple firings needed for stationkeeping, standardization implied a liquid propellant engine. Mounting the fuel tank on the central axis rather than in separated peripheral tanks avoided problems of balancing propellant utilization.

Figure 4-2 illustrates some of the design choices and considerations which were involved in determining the final configuration.

4.4 RADIOISOTOPE BRAYTON (RIB) SPACECRAFT CONFIGURATION SELECTION

Efficient operation of the Brayton electric system requires a relatively low compressor-inlet temperature and therefore a low radiator outlet temperature. This desired temperature is in the same range which is desirable for spacecraft equipment (10-40C). Consequently the radiator was designed in the form of a uniform low-temperature shell forming the outer skin of the spacecraft and isolating it from varying solar and earth-originating thermal inputs. The dimensions of this shell are limited by the 15 foot diameter shuttle of the cargo bay. The radiator was eventually sized at 13 feet in diameter because of structural considerations, and because of the need to provide space for mounting to a supporting pallet for multiple payload launches or for connection to a tug vehicle. To maintain a uniform thermal environment within the spacecraft shell an approximately isothermal radiator design was selected in preference to a more complex design, which would result from the use of a lighter-weight, non-isothermal radiator.

To support the structure and included payloads, a four-foot diameter structural inner cylinder and interface ring was established. Figure 4-3 shows the spacecraft configuration as it eventually evolved. The outer shell was made non-load-bearing for primary loads and consequently removable from the spacecraft for separate testing of the power plant prior to spacecraft assembly. The entire gas flow loop of the Brayton system was therefore designed to be integrally mounted to the outer shell. This permits testing of the power system separately from the spacecraft, and subsequent mating of the power system to the spacecraft late in the system assembly sequence.

FIGURE 4-1. BASIC RTG SPACECRAFT-EXTERNAL VIEW

FIGURE 4-2
SPACECRAFT DESIGN APPROACH
RTG SPACECRAFT CONFIGURATION SELECTION

POWER CONFIGURATION SELECTION

SPACECRAFT FORM AND SIZE

SPACECRAFT PACKAGING ARRANGEMENT

FIGURE 4-3 BASIC BRAYTON SPACECRAFT EXTERNAL VIEW

Spacecraft and payload equipment can then be mounted on the two faces of the cylinder, readily accessible without penetrating the radiator shell, and with a clear view outward. Individual panels surrounding the inner structural ring provide mounting surfaces for the equipment, and are hinged to provide quick access. The radiator design temperature was selected to provide reasonable temperature regulation of the equipment in the face of thermal input perturbations (40C). With the completely enclosing radiator it was determined that no active temperature control is necessary if the radiator temperature is properly selected. In order to maintain this temperature with a different number of heat sources, a portion of the radiator must be thermally shielded from space when less than three heat sources are used.

Figure 4-4 illustrates some of the design tradeoffs involved in reaching this configuration.

4.5 ADAPTABILITY TO OTHER LAUNCH VEHICLES

The configurations which were ultimately adopted were designed to use the Space Shuttle as primary launch vehicle, and either expendable Agena or Centaur tugs or the reusable chemical tug for missions requiring such propulsion.

The RTG spacecraft diameter of 114 inches fits within the shroud of the launch vehicles listed below (120 inch diameter).

<u>LAUNCH VEHICLE</u>	<u>LAUNCH COST (\$millions)</u>	<u>PAYLOAD (lbs) FROM KSC</u>	
		<u>100 n. mi. orbit</u>	<u>Synch. Orbit</u>
Atlas-Agena	14	7000	1400 *
Atlas-Centaur	18	10000	1800 *
Titan III C-Transtage	25	15000	3800 *
Titan III D-Centaur	30	n/a	5500 *
		n/a	8000

* with solid apogee kick motor.

The anticipated weight of the RTG spacecraft payloads fit generally within the Atlas booster capabilities and should not require the TITAN III for most missions. To maximize payload for synchronous orbit launch with an Atlas booster, the liquid propellant engine of the RTG spacecraft would probably be replaced by a higher specific impulse solid motor. Stationkeeping and attitude control propellant could be supplied from two 15-inch diameter spherical tanks with 100 pound total fuel capacity.

FIGURE 4-4
RADIO ISOTOPE BRAYTON SPACECRAFT
CONFIGURATION SELECTION

POWER CONFIGURATION SELECTION

SPACECRAFT FORM AND SIZE

SPACECRAFT PACKAGING ARRANGEMENT

The Brayton spacecraft diameter of 163 inches is significantly larger than any existing upper stage shroud, so that mounting it to present vehicles would require design and qualification of a special bulbous shroud, with extensive work on structures and dynamics. To fit this spacecraft within a 120-inch envelope would require a significant change in spacecraft design. Because use of the shuttle vehicle was considered paramount in the present study, and designing the Brayton spacecraft to fit within existing boosters would have compromised its applicability to the shuttle, the latter was not taken as a constraint in the design. It is believed that a similarly useful standard Brayton-powered spacecraft can be designed to fit within existing launch vehicles. However, this was beyond the scope of this phase of the study.

For the two missions for which the reactor-thermoelectric vehicle was considered, requirements are specialized enough so that considerations of launch vehicle adaptability are not pertinent.

5.0 SPACECRAFT DESIGN

5.1 GENERAL

This section of the report presents detailed descriptions of the two standardized spacecraft design concepts, and of their subsystems. Because both the RTG and the Brayton vehicles are meant to be multi-purpose "buses", there is a great deal of commonality among some of their subsystems.

Section 5.2 below discusses this common ground with respect to Power Distribution, Attitude and Orbit Control, and Telemetry, Tracking and Command subsystems. This is followed by Sections 5.3 and 5.4 which describe the RTG and Brayton specific spacecraft designs, respectively, with emphasis on structures and thermal control, which differ strongly from one to the other. Section 5.5 discusses reactor applications, but no standardized spacecraft design is presented.

5.2 COMMON RTG & BRAYTON SYSTEMS CONSIDERATIONS

5.2.1 ELECTRICAL POWER SUBSYSTEM

The functional elements of a spacecraft electrical power subsystem are:

1. The power source and converters.
2. The power conditioner.
3. The power distribution system.

The power source and converters have been discussed previously in Section 2.2. The power conditioning elements are constrained both by the character of the power source and converter and by the selection of parameter values for the power distribution and control functions. Standardization must be based upon qualitative and quantitative studies of the payload interface; the systems chosen here are not conceived as fixing standards. Rather, it is expected that this approach will be placed by one consistent with the results of the NASA subsystem standardization program. Table 5.1 summarizes the characteristics chosen for the power system. These selections are discussed below:

5.2.1.1 AC vs DC

Because the Brayton system generates AC, it is natural to investigate the desirability of providing an AC supply as part of the

TABLE 5.1 SELECTED POWER SYSTEM CHARACTERISTICS

CHARACTERISTIC	OPTIONS	NOMINAL SELECTED CHARACTERISTIC
NUMBER OF BUS VOLTAGES	SINGLE OR MULTIPLE POWER BUS	SINGLE
BUS VOLTAGE LEVELS	+5 VDC; +28 VDC; +50 VDC; +120 VDC; 120 V, 400 Hz	+28 VDC
REGULATION	REGULATED (2%); UNREGULATED (20 to 30%)	+ 2%
OUTPUT IMPEDANCE	≤ 0.1 ohm FOR ≤ 10 KHz; 0.5 to 1.0 ohm ≤ 10 KHz	0.1 ohm
FAULT DETECTION AND ISOLATION	UNDER VOLTAGE; OVER VOLTAGE	BOTH
LOAD PROTECTION	FUSE; CIRCUIT BREAKER (RELAY OR WITH A MAGNETIC AMPLIFIER CURRENT SENSOR); CURRENT LIMITERS	CIRCUIT BREAKERS WITH SOLID-STATE SWITCHES; CURRENT LIMITERS
PEAK LOAD CAPABILITY	μ SEC PEAKS; WATT-HR PEAKS	μ SEC PEAKS
POWER SWITCHING	CENTRALIZED; DE-CENTRALIZED	CENTRALIZED FOR STANDARD SUBSYS.
POWER MANAGEMENT	RELAY CONTROL; SOLID-STATE ELECTRIC LOGIC (SOSTEL)	SOSTEL

general power subsystem, to avoid unnecessary conversions. However, partially as a result of the trends in past power systems, there are very few demands for such a supply in most unmanned spacecraft (in contrast to aircraft, where loads such as lighting and rotating machinery constitute a sizable portion of the total). Because of this, use of square-wave designs is common in AC applications, and efficient conversion systems have been designed to produce square-wave AC from DC. Hence it appears that relatively little would be saved by providing an AC supply, in terms of cost, weight, or reliability. Therefore, the decision was made to provide DC only, on both the RTG and Brayton spacecraft designs. This retains the DC control mode of the Brayton system as currently designed.

5.2.1.2 Bus Voltage Levels

In general, a number of different voltage levels are required for the various spacecraft subsystems and devices. Normally, for custom-designed spacecraft, after the spacecraft subsystems are defined and their voltage levels are specified, trade-offs are made to determine whether to provide a centralized or multiple voltage bus system. Subsystem data on voltage levels reviewed during this study do not fall into neat groupings as to level and the amount of total power taken at each level. Therefore, since the object of this study is to develop a series of standardized spacecraft to satisfy a number of missions, one bus voltage level will be selected for distribution in each spacecraft configuration. The 28 VDC level has been chosen for each of the standardized spacecraft configuration because it corresponds to present transistor circuitry voltage requirements and in most cases may be used directly by payload subsystems and by existing auxiliary power conditioning equipment.

5.2.1.3 Regulations:

The 28 VDC bus is regulated to $\pm 2\%$. Rather than provide an unregulated bus for the primary loads to save the weight of the low-voltage regulator (since some loads require particular power conversion to higher voltage), it was decided to specify a voltage tolerance consistent with most scientific and solid-state RF-type experiment packages and one which will normally present no problem.

5.2.1.4 Output Impedance:

The main power bus is designed to have an output impedance of 0.1 ohm at frequencies up to 10 KHz. This is consistent with the selected voltage regulation tolerance.

5.2.1.5 Fault Detection and Isolation

Load fault detection is achieved by providing current sensors on each power line with provision to automatically disconnect a line when its current exceeds a predetermined value.

Overvoltage detection is more closely related to failure of power conditioning circuits. If a failure occurs, standby-redundant circuits are switched in to replace the main unit. When several nuclear generators are paralleled, the use of bus contactors can remove an inoperative unit. The reconfiguration and protection of the power subsystem circuits is performed automatically with provision to override, or otherwise alter the system, by ground command.

5.2.1.6 Load Protection

As indicated above, current sensors are used in each line, with additional circuitry provided to supply a trip indication signal. The switch may be a relay or a solid-state switch. The latter is preferred because of its "soft" turn on characteristics. However, each power subsystem configuration will provide a small complement of lines protected by current limiting circuits instead of the on-off protection switches for certain critical functions (such as the command receiver). It is assumed that under normal circumstances other load protection devices, such as fuses, will not be incorporated in the design units, since this function is provided by the standard spacecraft.

5.2.1.7 Peak Load Capability

The question here is whether to provide for relatively long-term peak loads, such as would generally require a battery supply, or to limit the standard subsystem to a short-term (typically microseconds) peak-load capability. Because the latter would be readily available in the Brayton system, and can be made available in any event by capacitive storage, it was decided that only short-term peaks would be provided as a standard capability. Longer term peak loads which may be required by specific missions will require the installation of a battery supply as part of the payload package.

5.2.1.8 Power Switching

Power switching can be done on either a centralized or decentralized basis. For the standard spacecraft, a division appears likely to be made between payload and housekeeping. A large portion of the command system is

concerned with delivering signals for the on-off operation of spacecraft loads. Use of a standardized command system would logically suggest use of a centralized power switching system, at least for the housekeeping functions. Considerable effort (Ref. 5-1) has been directed toward the development of solid state power-switching techniques for high-power large-scale avionic power systems. Such systems employ computer control of power management functions and the operation of remote load controllers using signal multiplexing techniques developed originally for weight-optimized spacecraft. These devices make a straightforward design of a centralized power-management subsystem possible.

5.2.1.9 Summary

The characteristics summarized above and in Table 5.1 have been applied to the design of electrical power subsystems for the two nuclear-powered spacecraft configurations to be discussed in subsequent sections. These characteristics do not necessarily lead to a minimum-weight electrical power system design for every set of mission specifications, but will permit the design of efficient, reliable and flexible electrical power subsystems capable of supporting a wide range of load power levels and characteristics within each spacecraft configuration. Further details of the individual power systems are presented in Section 5.3 and 5.4.

5.2.2 ATTITUDE AND ORBIT ATTAINMENT AND CONTROL

In order to establish the design approach to the attitude and orbit control function a set of criteria has been defined (Section 3). These included:

- Capability for operation in an earth-facing mode, three-axis stabilized, in an arbitrary low earth orbit or in a synchronous circular equatorial orbit.
- Provisions for attitude maneuvers such as earth scan, offset pointing, or inertially fixed attitude stabilization.
- Guidance, control and propulsion to adjust and maintain the desired orbits.

The approach which evolved to meet these requirements for standardized vehicles performing a variety of missions is described in this section.

5.2.2.1 Orbit Achievement and Maintenance

Large propulsion requirements are provided by the shuttle or shuttle-tug combination, but initial orbit adjustments, stationkeeping, and

minor orbit adjustment are still required for certain missions. To achieve these missions, a propulsion system capable of repeated firings is necessary. A hydrazine system has been chosen for this purpose: it may impose a small weight penalty for certain missions where a solid propellant motor could be used for a known velocity requirement (example: circularizing a 1000 n. mi. orbit after boost to apogee by the tug), but it has the advantage of providing the desired propulsion without having to design and integrate a solid propellant motor for a specific mission. The additional advantages are the possibility of repeated firings and the common use of the hydrazine system for attitude control, reflecting the adaptability of the design to many purposes and missions.

The design goals proposed for different missions are:

- (a) For orbits with only minor propulsion requirements (e. g. up to 100 ft/sec), the hydrazine system shall provide these completely.
- (b) For larger propulsion requirements, e. g. insertion into approximately circular near polar orbits with 700 to 900 n. mi. altitudes, the shuttle or shuttle-tug combination places the spacecraft in a transfer orbit with approximately correct apogee, perigee of 100 to 300 n. mi., and inclination angle leaving no more than 1.5 degree of plane change to be accomplished. Orbit circularization requires approximately 1000 ft/sec., and the inclination change increases the requirement to 1170 ft/sec. If the latter value is set as the design goal for the propulsion system, the tug can be relieved of the duty of circularizing the spacecraft orbit before returning to the shuttle. In order to keep the propulsion system simple, the need for vectored thrust should be avoided. Thus, the spacecraft should be designed to be spun about the nominal thrust axis during sustained firings of the engine. The integration of the hydrazine thruster with the reaction control system then provides a simple and effective subsystem.
- (c) For geosynchronous orbits, it is assumed that the space tug supplies propulsion to place the spacecraft on the desired nominal orbit, so that the spacecraft propulsion and control systems are not required through injection. The hydrazine tank need then be only partially filled, or a complete fuel charge can provide six years of North-South and East-West stationkeeping. With the spacecraft designed for stable spinning, in certain cases the

demands made on the tug can be reduced if the central tank and thruster of the spacecraft are replaced with a more efficient solid rocket motor with enough impulse to circularize the orbit at apogee. Smaller hydrazine tanks would then have to be added outboard to provide fuel for stationkeeping and attitude control. When feasible and appropriate, such an arrangement can relieve the tug of some requirements and may give it additional versatility.

5.2.2.2 Attitude Control

(a) Sustained Propulsion By Spacecraft System.

During these periods, the spacecraft is spun about the axis of symmetry, which is the nominal thrust direction, to average out the effect of thruster vector misalignments from the vehicle center of mass. The original spin is imparted by hydrazine reaction motors which also realign the angular momentum vector or remove the spin as necessary. Earth and sun sensors operating in a spin scan mode provide attitude information.

(b) Normal Orbital Operation

The preponderance of missions require operation in an earth-facing, 3-axis stabilized mode, with provisions for offset pointing (pointing at any point on earth, rather than solely at the nadir) and, in some cases, for scanning an angular field. Other missions require arbitrary orientation to heavenly bodies, again including a scan requirement (e. g., solar observatories). Accuracy will be essentially determined by sensors; for earth facing operation, sensor resolution and pointing requirement are fortuitously compatible (0.1 degree pointing). In general, three schemes have been used for controlling such vehicles:

(1) Biased Momentum Wheel

A flywheel carrying substantial angular momentum provides stability to minimize the effects of disturbance torques. Such a control system is very simple; the resulting vehicle dynamics are similar to those of a spinner configuration. It is very suitable to an earth-facing mission, but only if there are no offset pointing requirements. The angular momentum must be aligned normal to the orbit plane, providing only a single degree of freedom for rotation about this axis. Since the standard spacecraft must be more versatile this scheme is not recommended.

(2) Three Orthogonal Reaction Wheels

By controlling the angular velocities of three relatively small, orthogonal wheels, momentum may be exchanged between wheels and spacecraft, permitting pointing in any space direction. The "stiffness" to

external disturbances of the "biased momentum" system (which carries a very large momentum normal to the orbit plane) is eliminated; the attitude control sensor must continuously monitor the motion and regulate the wheel speeds. Very fine control is easily achievable, limited only by sensor resolution. The integrated effect of unidirectional external torques requires gradually increasing absorption of this angular momentum in the wheel system. When rated wheel speed is reached this momentum must be unloaded by external means, for example, by means of reaction jets (thrusters) or by torquing against the Earth's magnetic field.

(3) Control Moment Gyros

Control moment gyros are gimballed rotating wheels, permitting exchange of momentum between wheels and vehicle by changing the relative orientations of the wheel momentum vectors, rather than by changes in wheel speed as described in (2). Configurations employing either two or three double-gimballed wheels are possible. Such a system operates with a net bias momentum (vector sum of the several wheel momenta) which is torqued to control and position the vehicle. The control system is more complex than the control of reaction wheels, but larger torques resulting in more rapid orientation maneuvers are possible. The penalty is increased weight and the complexity and attendant lower reliability due to addition of the gimbal system.

For the smaller spacecraft (RTG and Brayton) the reaction wheel control mode is recommended and has been incorporated in the design. For the larger (Reactor) spacecraft either reaction wheels or control moment gyro could be employed efficiently. Although a reaction wheel system has been incorporated in the present design, this should not imply that this selection is essential. Substantial additional study of the vehicle and its requirements are needed, and the possibilities of control moment gyros should be further explored.

Wheels will be selected to meet reasonable maximum momentum absorption demands of a variety of spacecraft configurations. The basic concept of a standard spacecraft implies that external components (e.g., antennae) may not be in optimum aerodynamic balance, and therefore larger than usual solar pressure torques are to be expected in some configurations.

5.2.2.3 Sensors and Computations

The basic sensor system consists of redundant earth-horizon sensors which will provide the pitch and roll signals. These earth sensors will be mounted to view the 17° earth from synchronous orbit, or to view four segments of the horizon for craft in low orbit. The pitch and roll signals control the yaw pointing axis. Motion about this axis (yaw) will be controlled by means of a series of 8 sun sensors. The entire system will be monitored and controlled by (redundant) digital computers which will program both normal stabilization as well as offset and scanning modes. Provision will be made to incorporate one of several star trackers, as required, to enable finer control of specific missions; such additional sensors being nominally defined as "payload equipment".

5.2.2.4 Summary Specifications

The RTG and Brayton vehicle orbit and control systems will be sized to provide:

- (a) Spinning orbit insertion, with adequate spin propellant. The spin rate is to be adequate to reduce asymmetry and induced nutation to less than one degree.
- (b) Sufficient propellant capacity for a velocity change of 1170 ft/sec. This is sufficient for an apogee circularization boost of 1,000 ft/sec. and a simultaneous 1.5 degree inclination change. For synchronous orbit missions, this propellant may be allocated to initial orbit adjustment and correction of the luni-solar inclination disturbance with North-South stationkeeping, estimated to require approximately 175 ft/sec. per year.
- (c) Attitude control propellant to provide for "dumping" of momentum induced by secular torques.
- (d) Reaction control wheels of sufficient size so as to require unloading no more than once a week.
- (e) Redundant hydrazine jet control torquing about all three axes, as well as redundant E-W and N-S thrusters. (The propulsion functions among these several duties may be combined).

(f) Earth and sun sensors of sufficient resolution to provide 0.1 degree attitude orientation in earth-facing mode. Provision to control in sun- or celestial-facing modes through additional (optional) sensors where required.

(g) Magnetic torquers in low earth orbit, sized to overcome expected internal moments.

(h) Dual (redundant) digital controllers.

5.2.2.5 Control Operations Sequences

(a) Synchronous Equatorial Orbit

Several options exist, depending on the mode of utilization of the tug, particularly on when tug-spacecraft separation takes place. Presumably the spacecraft power system is turned on before separation, and the tug provides initial attitude information to the spacecraft. The principal control sequences from low orbit include the following:

- (1) Boost into transfer orbit to synchronous altitude apogee.
- (2) Attitude control during transfer orbit and alignment for orbit circularization impulse at apogee.
- (3) Orbit circularization boost at apogee.
- (4) Reorientation to proper on-orbit alignment.
- (5) Orbit-trimming propulsion and associated realignments.

Step (5) is normally performed by the spacecraft, and steps (1) through (3) by the tug, after which separation can take place. However, other tradeoffs are possible. For example, step (3) may be performed by a solid rocket motor in the spacecraft, leaving the tug free to return immediately to lower orbit, or to proceed to another mission from the transfer orbit. This, however, implies a weight penalty for the spacecraft, i.e. the inclusion of a full stage of propulsion.

(b) Low Orbits Above Shuttle Capability (up to 1000 mile circular).

Again, the alternatives are complete insertion by the tug, or use of the spacecraft propulsion system for orbit circularization

at apogee. In the former case, the spacecraft needs no propellant for orbit change, unless later orbit changes are required. In the latter case, the spacecraft must be spun about the thrust axis prior to firing, despun afterward, and then reoriented to the orbital configuration. The spinning up may take place after separation near perigee, and may be used for attitude control during transfer orbit, in which case reorientation is required before the apogee boost. Attitude may be determined during spin by sun and horizon ~~sensors~~ sensors operating in a spin scanning mode.

(c) Orbits With Shuttle Capability

No spacecraft propulsion is required, unless a future orbit change is needed. A small amount of propellant is needed for attitude control only; even this may be obviated by the magnetic torquing system in some cases.

(d) Normal Operation In Orbit (earth oriented case)

The spacecraft is maintained in earth viewing attitude by accelerating and decelerating momentum wheels as required. Angular momentum due to secular torques is absorbed in the appropriate wheels until one of the wheels approaches its design limit momentum. At this point angular momentum is removed by firing the appropriate combination of hydrazine jets. Such maneuvers occur in general once per week, or less frequently, and may be executed automatically or under ground supervision. The synchronous orbit spacecraft may also control its orbit with occasional east-west corrections (7 ft/sec per year) and north-south thrusting (170 ft/sec per year) to counter the inclination disturbance due to sun and moon when this is required. The latter correction must be properly phased with respect to the ecliptic plane; the correction must be made at a specific time of day. Since the disturbances are small, infrequent (once per week) corrections, properly timed, are normally sufficient.

(e) Pointing Accuracy Considerations

For the 1980-1990 period pointing accuracy requirements for directional equipment, such as telescopes or antennas, do not differ significantly from those which prevail currently. Study of the missions described in References 1-1 and 1-2 and discussed in Section 3 of this report shows that an accuracy of 0.1 degree satisfies a majority of the missions. To set the specification for the standard spacecraft tighter than this introduces

more expensive manufacturing, alignment and calibration procedures than are necessary, thus increasing the cost of the standard vehicle unnecessarily. Consequently, 0.1 degree is set as the specification; when accuracies greater than this are required, the modifications required are considered as mission equipment. These modifications depend on the nature of the mission and upon whether the requirements involve only more precise attitude indication or more precise attitude control. Some of the mission equipment, e. g. telescopes or antennas, may be capable of detecting its target and acting as its own sensor, either full-time or intermittently. In other cases the mission equipment may have to depend on other sensors, which may have to be mounted close to the critical component. In general, where accuracies of better than one minute of arc are involved, in-orbit calibration procedures are necessary.

Among the various options available for precise pointing of experiments are the following:

- (1) Adding vernier momentum wheels and high accuracy sensors to the main spacecraft to increase its control accuracy.
- (2) Gimbaling the experiment with respect to the spacecraft and providing it with its own control system, either with or without its own sensors.
- (3) Use of the payload equipment itself for sensing. Examples are optical telescopes, or communications antennas that can be used intermittently for this purpose when not devoted to other needs.
- (4) Use of sustained or repeated observations and application of appropriate filtering techniques to improve the accuracy of observations.

Some computations and details of parameter choice for the attitude and orbit control subsystems for the RTG and Brayton spacecraft are presented in Appendix B.

5.2.3 TRACKING, TELEMETRY AND COMMAND

5.2.3.1. Approach

The approach to Tracking, Telemetry and Command (TT & C) subsystem has been to modularize to the extent that the "standard" TT & C subsystem can handle low to medium data rates and can be made compatible with the Tracking and Data Relay Satellite System (TDRSS) and/or the existing earth networks. The TT & C subsystem will operate at VHF for low data rate

($10^3 - 10^4$ bps) payloads and at S-band for medium data rate ($10^4 - 10^6$ bps) payloads.

Also included in the standard package are tape recorders for data storage as well as a command memory and programmer. A totally redundant system including all the above equipment has been budgeted for the standard spacecraft.

There are essentially three types of missions to be considered in terms of T & C equipment as a function of orbit and the TDRSS interface. These are synchronous-orbit missions, and low-earth-orbit mission with and without TDRSS interface. These missions can further be divided into three types in terms of data rate; these are:

- Low data rate (LDR); 10^3 bits/sec.
- Medium data rate (MDR); 10^3 to 10^6 bits/sec.
- High data rate (HDR); 10^6 to 5×10^7 bits/sec.

The equipment provided in the RTG and Brayton spacecraft will handle low and medium data rates. High data rate telemetry equipment is considered to be an option since it will require frequencies higher than the standard VHF and S band telemetry frequencies. The HDR option will consist of the modulation and RF equipment for operation at X band or Ka band. TDRSS users will operate in the Ka band. In a number of cases, (such as some LEO missions) the HDR is a requirement only for playback of stored information. For these missions, interface with the TDRSS could change them from HDR to MDR missions with only real time information transfer.

Table 5.2 shows the complement of standard TT & C equipment for typical missions of the various types for both RTG and Brayton missions. Thus, it is seen that low data rate missions have completely redundant VHF T & C subsystems, whereas medium data rate missions have completely redundant S-band T & C subsystems. Alternatively, the MDR mission can have VHF backing up the S-band system with a loss of information bandwidth. The HDR missions are provided with a VHF system primarily as a backup to allow command capability to the spacecraft during those periods when the high gain antennas inherent in the HDR systems are not pointing at the ground station or the Tracking and Data Relay Satellite.

5.2.3.2 Memory and Programmer Considerations

With respect to the memory and programmer portion of the command decoders, and the tape recorders, these will generally be required for low orbit missions which do not interface with TDRSS, since there will be periods of time when the spacecraft is not in view of the ground station during which commands should be executed and data should be collected. This may still be true with missions which interface with TDRSS since the spacecraft will not always be in view of a TDR satellite or may be in view of a TDR satellite which is "busy." The memory and programmer is an optional part of the command decoder. The tape recorder will be of video quality for HDR mission and will have a capability of 1 Mbps for LDR & MDR missions.

The multiplexers are essentially the same for LDR and MDR missions, the only difference being the clock rate. The number of data points which can be multiplexed will be varied on a modular basis.

5.2.3.3 Transmitter and Antenna Requirements

The transmitters have enough power to interface with the TDRSS as a baseline. This should be the highest power requirement and the power varies down from this on a modular basis. It is estimated that the peak RF power requirements are on the order of 12 watts for S band and 5 watts for VHF.

With respect to antennas, the only high gain antenna requirement in the standard system is for the S band link with the TDR satellite. In the satellite ground links, earth coverage or omni gain will generally be sufficient with the transmitter power discussed above as the upper limit.

Table 5.3 presents an estimate of the weight, size, and power requirements for the standard TT & C components.

COMPONENT	MISSION TYPE								
	SYNCH. ORBIT			LOW ORBIT (WITH TDRSS)			LOW ORBIT (WITH GROUND STATION)		
	LDR	MDR	HDR	LDR	MDR	HDR	LDR	MDR	HDR
VHF RCVR	2		1	2		1	2		1
S-BAND RCVR		2			2			2	
CMD DECODER (REDUNDANT) • MEM & PROGRAMMER (REDUNDANT)	1 NO	1 NO	1 NO	1 POSSIBLE	1 POSSIBLE	1 POSSIBLE	1 YES	1 YES	1 YES
MDR MUX (REDUNDANT)		1			1			1	
TAPE RECORDER				POSSIBLE	POSSIBLE	POSSIBLE	2	2	2
VXF XMTR	2		1	2		1	2		1
S-BAND XMTR		2			2			2	
ANTENNAS • S-BAND HI GAIN • S-BAND EARTH COVERAGE • VHF OMNIS	X	X	X	X	X X	X	X	X	X

TABLE 5.2. TT&C EQUIPMENT COMPLEMENT
TABLE 5.3

TELEMETRY & COMMAND SUBSYSTEM WEIGHT, SIZE, AND POWER ESTIMATES

ITEM	SIZE (INCHES)	WEIGHT (POUNDS)	POWER (WATTS)
S Band Receiver	9 x 9 x 2.5	8	12
S Band Xmtr	11 x 11 x 2	4	33
VHF Receiver	9 x 9 x 2.5	6	12
VHF Xmtr	11 x 11 x 2	4	16
LDR Mux (redundant)	10 x 6 x 6	6	4 [†]
MDR Mux (redundant)	10 x 6 x 6	6	4 [†]
Recorders (redundant)	15 x 15 x 4.5*	40	20 [†]
CMD Decoder (redundant)	10 x 6 x 4	5	2 [†]
Note: S band requires 2.1 ft antenna for interface w/ TDRSS VHF requires omni whips			
	TOTAL	79	103

* volume per recorder

[†] power per unit

5.3 RTG SPACECRAFT

5.3.1 MECHANICAL CONFIGURATION

The structure and dimensions of the RTG spacecraft evolved from the following design criteria:

- Total vehicle weight in the 1000 to 2000 pound range.
(See Table 3.12 for source of this range in the mission analysis)
- Propellant weight of up to 300 pounds and corresponding tank volume (see Sec. 5.2.2 for source of requirement)
- Equipment mounting surface heat dissipation of 20 watts per square foot, corresponding to a temperature of 33°C in sunlight, with an α_s of 0.2.
- Adaptability to interface with the orbiter, pallet, or tug.
- Ability to mount from one to three RTG units producing 150 electrical watts each (BOL), with provision for varying the amount of waste heat which can flow from the RTG to the spacecraft so as to regulate spacecraft temperature.

The final configuration selected took the form of a hollow hexagonal prism with a seven-foot outer diameter, a hollow core of 3.5 feet inside diameter, and a three-foot depth. The structure is divided into six compartments. The RTG units are mounted symmetrically on the outer faces of the prism; if only one is used, it is counterbalanced. A sketch of the spacecraft is shown in Figure 4-1, and a cutaway showing the equipment compartment in Figure 5-1.

Preliminary structural weight estimate is approximately 260 pounds. With approximately 240 pounds for 3 RTG's the reference vehicle dry weight is 500 pounds. This would allow payload of 500 to 1500 pounds in the total vehicle weight range which was set in the design criteria above.

The hexagonal core allows the installation of spherical propellant tank having a diameter of 2.67 feet with a volume up to ten cubic

feet, which can satisfy the propellant requirements for the maximum weight RTG configuration.

At 20 watts per square foot, approximately 450 watts output would require 22.5 square feet of heat dissipating surface. The mounting provided with the 7.0 foot outside diameter yields about 56 square feet for both ends of the structure.

The vehicle volume is about 86 cubic feet or 14.3 cubic feet per compartment. Since each end is an equipment mounting face, the access to the equipment will be accomplished by hinged doors around the periphery of the structure.

Adapting the RTG spacecraft directly to the orbiter or to a pallet will require an adapter truss since the 7-foot diameter has to reach 15-foot span support points. Figures 5-2 and 5-3 depict installations on a pallet and directly into the orbiter respectively. In addition, figure 5-4 shows a possible scheme of stacking seven spacecraft on a pallet which is in turn mounted to the orbiter. The 3.5-foot inner diameter can be used to adapt the vehicle to a 5-foot diameter tug interface and the 7-foot outside vehicle diameter can be adapted to a 10-foot diameter tug interface.

Figure 5-5 shows the structural system. Three RTG's are shown but the structure can accommodate 2 units directly opposed or one unit with a directly opposed ballast unit if vehicle spin-up is required by the mission. The two end mounting surfaces are one-half-inch honeycomb panels with integrated circular heat pipes. The access to the mounting surface is by split hinged doors on the three flat surfaces of the periphery. These doors are one-quarter inch honeycomb panels for handling rigidity and strength requirements.

The core is a stiffened shell structure, to which the propellant tank is tied, with ribs forming the compartments and in turn supporting the outer shell. The ribs are truss structures which allow access to adjacent compartments, and, most importantly, allow thermal interchange between the compartments. As an overall structure, the system is a 3-foot cylindrical beam supported at four points.

FIGURE 5-1. RTG SPACECRAFT, CUTAWAY VIEW

FIGURE 5-2 RTG SPACECRAFT INSTALLATION ON PALLET IN ORBITER

DIRECT MOUNTING ON ORBITER OF STANDARD RTG SPACECRAFT

FIGURE 5-3

FIGURE 5-4. RTG SPACECRAFT STACKED ON PALLET

MISSION POWER REQUIREMENT	NUMBER RTG REQUIRED	LOCATION OF RTG
150 WATTS	1	LOCATION "A"*
300 WATTS	2	LOCATION "A" & "C"
450 WATTS	3	LOCATION "A," "B," & "D"

* FOR 150 WATT MISSION A BALLAST IS REQ'D AT LOCATION "C" FOR DYNAMIC BALANCE.

PITCH & ROLL THRUSTERS
8 PLACES

EARTH

4 25# HYDRAZINE THRUSTERS

SPACECRAFT PICK-UP POINT
4 PLACES

YAW CONTROL THRUSTER
8 PLACES

LOCATION "D"

RTG (FOR QUANTITY SEE TABLE)

FIGURE 5-5 BASIC RTG SPACECRAFT, PLAN VIEW

Access to all of the contained equipment is achieved by removing those outer panels on which RTG's are not mounted. These panels contain no mounted or attached equipment but serve only as thermal insulation structures.

5.3.2 ELECTRICAL POWER CONDITIONING SUBSYSTEM

5.3.2.1 Block Diagram and Description

A block diagram of the RTG Electrical Power Subsystem is shown in Figure 5-6. The subsystem includes up to three 150 watt RTG units connected in parallel. It will deliver load power over a range of 150 to 450 watts, depending upon the number of RTG units. Each RTG unit is connected to the main bus through a current sensor and bus contactor. The load-controller circuit of the shunt regulator senses and regulates the voltage on the bus by dissipating the power not required by the spacecraft load. In the case of a three- RTG configuration with no load on the power system, at least 450 watts will be dissipated in the shunt regulator. Power from the main bus is distributed to the loads through load controllers. The controllers on each load provide the function of applying and removing power to the load. Each load controller has its own current sensor which is set to open the line when a current limit is exceeded. When this event occurs, a trip indication signal will be supplied to the telemetry system.

5.3.2.2 RTG Source Operating Point

The operating voltage point is determined by two considerations; one is the setting of the voltage point to coincide with the maximum power voltage point at end of mission, and the second is the prevention of the voltage operating point from being at or near open-circuit voltage. The latter consideration is necessary since operation at or near the open-circuit point will produce rapid degradation due to heating effects on the RTG thermocouple converter. The shunt regulator achieves both objectives. Figures 5-7 shows the output current voltage characteristic for an RTG with the end points labeled I_{sc} (short circuit current) and V_{oc} (open circuit voltage).

FIGURE 5-6. BLOCK DIAGRAM OF RTG ELECTRICAL POWER SUBSYSTEM

FIGURE 5-7. RTG OUTPUT CHARACTERISTIC AND OPERATING POINTS

This characteristic can represent one RTG or the composite of several RTG's in parallel. The degraded end-of-mission characteristic is shown as a dashed line. The shunt regulator is set to the maximum power point of the degraded RTG characteristic. This is to the left of the beginning-of-mission maximum power point, thus moving operation further away from the open-circuit voltage.

The effect of paralleling RTG's and thereby not operating at each maximum power point is not critical. The power equation in terms of Figure 5-7 is

$$P = - \left(\frac{I_{sc}}{V_{oc}} \right) V^2 + I_{sc} V$$

the maximum power point is at $V = (1/2) V_{oc}$, and therefore, $P_{max} = (1/4) I_{sc} V_{oc}$

The ratio

$$P/P_{max} = \frac{4V}{V_{oc}} \left(1 - \frac{V}{V_{oc}} \right)$$

defines the ratio of output power to maximum power.

At $V = (1/2) V_{oc}$ the ratio of P/P_{max} is, of course, unity. Even an operating voltage varying as much as 10% from the maximum power point would give a power ratio of 0.99, or only a 1% loss.

5.3.2.3 Fault Detection

The fault detection circuitry senses under- and over-voltage and any deviation from normal current being delivered by each RTG device. The action taken on the basis of any of these faults depends on a failure mode and effects analysis which is performed in the detailed design phase. The results of such an analysis determines which remedial actions are automatically taken and which actions are ground commanded.

In general, an under-voltage condition results from the loss of power in one of the RTG's or from faulty operation of the load

regulator. For the former event the payloads could be automatically disconnected and a reduced mission power profile could be configured. For the latter situation a standby regulator would be the solution. Overvoltage would probably result from a load controller/shunt regulator failure and would be corrected by switching to a standby unit.

5.3.2.4 Priority Bus

Failure of the main bus or the removal of all the RTG's from the main bus would cause the battery to supply power on the priority bus for the spacecraft housekeeping system. This system would be designed to deliver power for approximately 12 hours of operation, providing time for ground control to attempt to restore the main bus.

5.3.3 THERMAL CONTROL SUBSYSTEM

5.3.3.1 Design Criteria

The thermal design criteria employed for this study were dictated principally by electronic component reliability for an extended mission. Requirements for temperature control become quite stringent and narrow temperature limits must be specified and achieved in order to assure dependable operation for long life missions. The availability of waste heat from the nuclear power subsystems provide possible design alternatives which are not ordinarily available. These alternatives were explored for their ability to enhance the flexibility of the thermal control subsystem.

(1) Temperature Limits

It was assumed that all electronic components, except for the battery and the hydrazine attitude control subsystem, must be maintained between -10°C and $+40^{\circ}\text{C}$ during normal spacecraft operation. The battery requires a temperature range of -5°C to 25°C while the hydrazine must be maintained above 5°C .

(2) Environmental Considerations

The mission model for proposed spacecraft missions in the

100-to-500 watt range requires that the RTG spacecraft function satisfactorily in any orientation with respect to the earth and sun. Thus, the spacecraft could have a surface oriented toward the earth at an altitude of 200 n.mi. or could be inertially oriented with no solar input on the equipment panel in a synchronous orbit. The worst-case hot orbit is a 200 n.mi. noon orbit (sun in orbit plane) with one of the equipment panels oriented toward the earth. The worst-case cold orbit for the RTG spacecraft is a synchronous orbit with no solar input incident on either equipment panel.

(3) Heat Dissipation

It was assumed that all the electrical power produced by the RTG's will be dissipated on the equipment panels. Any generated power not required by the payload will be dissipated by shunts attached to the equipment panels. Thus, the total heat dissipation on the equipment panels will be fairly constant, except for the slow degradation of the power source. After about five years, the electrical power generated will have dropped to approximately eighty percent of its original level.

5.3.3.2 Thermal Control Subsystem Design

The thermal control subsystem for the RTG spacecraft is required to maintain the equipment within the acceptable temperature limits described above. The subsystem selected to accomplish this objective consists of (a) two equipment platforms which are isothermalized by five heat pipes and (b) an active thermal control device which senses the average equipment panel temperature and regulates the quantity of heat to be transmitted from the RTG to the equipment shelf. This regulated heat flow minimizes the effect of variations in heat dissipation of equipment on the shelf and variations in environmental heat sources. The internal surface of the equipment panels is painted black to maximize thermal coupling. All other internal surfaces are bare aluminum to specularly reflect any incident radiation (see Figure 5-8). This thermal design concept will enable the RTG spacecraft to operate within an acceptable temperature range for all altitudes and orientations.

Figure 5-8 illustrates the features of the selected thermal control concept. The thermal control coating on the equipment panel radiators is silver on FEP teflon. It was selected because of its stable thermal properties (solar absorptivity does not degrade much) and its ease of application. All other exterior surfaces are covered with multilayer insulation. The radiating area on the equipment panels will be adjusted according to the number of RTG's used (150, 300, or 450 watts).

This configuration was selected after rejection of two other approaches described below. All of the concepts were based on use of the waste heat from the RTG's in the thermal control process.

Concept I

Radiator Wall Design: In this concept it is assumed that the RTG is thermally connected to the spacecraft shell using heat pipes to distribute the thermal load evenly over the spacecraft wall. The heat pipes also reduce thermal gradients induced by solar heating. Figure 5-9 illustrates the concept. The waste heat from the outer case of the RTG is distributed over the entire side wall of the cylindrical spacecraft by a system of heat pipes. Since the spacecraft sidewall acts as a large radiator, it will run rather cool, yielding improved RTG efficiency. In this design, the mission payload is mounted to either the earth-viewing or top surface. Temperature control of the equipment shelf is accomplished by use of passive radiators. (If required for some special equipment items, active devices such as heaters, louvers, or heat pipes can be used). Heat from the radiator wall "saturates" both the top and earth-viewing shelf, and completely over-shadows the temperature effect of power variation and environment heat variations. Table 5.4 shows the radiator and equipment shelf temperatures for a spacecraft with 10-ft. diameter and 5-ft. height in a worst-case orbit, with this thermal control concept. Figure 5-10 shows the effect of the diameter-to-length ratio on temperature.

FIGURE 5-8 RTG SPACECRAFT THERMAL DESIGN CONCEPT

FIGURE 5-9 RADIATOR WALL SPACECRAFT CONCEPT

FIGURE 5-10 RADIATOR WALL TEMPERATURES FOR VARIOUS D/H RATIOS

Table 5.4

Temperature Range - Radiator Wall RTG Spacecraft

(10 ft. diameter, 5 ft. height, 200 nm Orbit)

<u>Power</u>	<u>Wall Temp. ($^{\circ}\text{C}$)</u>	<u>Shelf Temp. ($^{\circ}\text{C}$)</u>	<u>Radiator Area (ft^2)</u>	<u>Insulation Area (ft^2)</u>
150	12--17	20-25	None	157
300	35--42	24-30	36	121
600	73--80	29-30	126	31

Concept advantages:

- Modules can be stacked for shuttle multiple payloads
- RTG's operate at low cold junction temperatures
- Spin axis inertias are favorable

Concept disadvantages:

- Complex heat pipe arrangement
- Poor accessibility to equipment
- Zero "g" tests may be difficult
- Large spacecraft

This last disadvantage was ultimately decisive. The large radiator area which was required to make the radiator-wall concept work, resulted in a spacecraft with excessive weight and volume for the missions at these power levels. Although a certain amount of over-capability is desirable in a multi-purpose spacecraft, an excess of weight and volume capability would prevent optimum use of the Shuttle cargo bay.

Concept II

Hot Spot Design: The hot spot concept, investigated briefly, was based on mounting the RTG's in a stack on the axis of the spacecraft and external to it. Part of the waste heat of the inboard RTG would be allowed to enter the spacecraft body through a variable aperture. When the aperture control senses a cool equipment temperature, the opening is enlarged to permit additional RTG heat to enter the spacecraft. However, the design was complex,

resulted in unfavorable moments of inertia, and was soon refined to the present configuration, wherein the RTG's are mounted on the sides of the spacecraft.

5.3.3.3 Thermal Analysis of Selected RTG Spacecraft Configuration

A thermal analysis of the selected RTG spacecraft configuration was performed to demonstrate the capability of the thermal control subsystem. The analysis was performed on an RTG spacecraft which generates 450 watts of electrical power. However, the results of the analysis can be applied to other RTG power levels (150 or 300 watts) by scaling the effective radiating area of the equipment panels to the power level. This may be achieved by insulating part of the spacecraft end faces.

The active temperature controller (ATC), which regulates the amount of waste RTG heat entering the equipment compartment, is a movable vane device (see figure 5-11). The ATC is actuated by the expansion of a liquid-filled bellows which sense the average temperature of the two equipment panels. As shown in figure 5-11, the RTG is mounted to a radiator plate which is assumed to be at an average temperature of 350°F. The effective emittance of the ATC varies between 0.06 (closed) and 0.45 (full open). Figure 5-12 shows the variation of the net heat transmitted through the vane assembly to the equipment panels as a function of the average equipment panel temperature.

The hot orbit for the spacecraft is the 200 n.m. noon orbit with one equipment panel earth oriented. The orbit average equivalent space sink is defined as follows:

$$\sigma T_s^4 = \frac{\alpha}{\epsilon} \left[\bar{S} + \bar{\rho} \right] + \bar{\mu}$$

where:

σ	is Stefan-Boltzmann constant
T_s	is equivalent sink temperature
α	is solar absorptivity
ϵ	is IR emissivity
$\bar{S}$	is orbit average incident solar energy
$\bar{\rho}$	is orbit average incident earth reflected solar energy
$\bar{\mu}$	orbit average incident earth emitted energy

FIGURE 5-11 THERMAL SHUTTER ASSEMBLY

NET HEAT INPUT
TO EQUIPMENT
SHELF PER RTG (WATTS)

FIGURE 5-12 HEAT FLOW IN RTG SPACECRAFT

For the hot orbit, the equivalent sink temperature for the earth oriented panel, with silver on FEP teflon as the thermal coating ($\alpha/\epsilon = .1/.8$), is -21°C . The anti-earth face has a sink temperature -97°C . For the cold orbit, it is assumed that there is no external input on the equipment panels.

A simplified thermal model was developed to examine the feasibility of the thermal design. Figure 5-13 shows the results of this analysis. The Earth-viewing face can handle a power variation of 0-to-450 watts in all orbits within a temperature range of -3° to 29°C . At the end of life, with 20% less internal power (360 watts), the minimum temperature would be -4.5°C with zero watts dissipated on the surface.

Thus on a preliminary basis, this concept can maintain the spacecraft equipment within acceptable temperature limits. Since it is unlikely that the full range of heat dissipation (0-450 watts) on one surface will ever be realized, the temperature control subsystem can be said to maintain the equipment between 0 and 20°C . The battery will be mounted to the anti-earth face since it will have a narrower temperature variation. Thermostatically controlled heaters will be used to maintain the hydrazine above 5°C . This design is very flexible in that equipment can be mounted to both equipment panels regardless of dissipation.

5.3.3.4 Weight of Thermal Control Subsystem

The following is a weight breakdown of the thermal control subsystem designed for the RTG bus:

Table 5.5 Thermal Control Subsystem Weight-RTG Spacecraft

Heat Pipes	31.
Multilayer Insulation	18.2
Thermal Control Vanes (3)	6.9
RTG Radiator (3)	14.4
Coatings	
Ag on FEP Teflon	1.3
Black Paint	1.2
Misc. Isolators	2.0
Heaters	3.0
	<hr/>
	78.0 lbs

FIGURE 5-13 MOUNTING PANEL TEMPERATURE RANGE, RTG SPACECRAFT

5.3.4 STRUCTURAL ANALYSIS

The final RTG structural configuration that was discussed and depicted in Section 5.3.1 was analytically investigated for weights, mass properties, structural load continuity and integrity, and the capability of integrating it into the shuttle orbiter payload compartment.

The initial structural sizing for weight yielded a structural weight of about 190 pounds. With the electrical power system weight of 320 pounds, a basic vehicle weight of 510 pounds was used as a reference. These weights, along with the weights of the other spacecraft subsystems, are shown in Table 5.5. To this reference weight was added arbitrary payloads of 490, 990, and 1490 pounds evolving into total vehicle weights of 1000, 1500, and 2000 pounds. These payloads weights were uniformly distributed throughout the usable volume in determining the mass moments of inertia. The vehicle weight of 2000 pounds was used for the structural analysis. The mass moments of inertia variation as a function of gross weights is shown in figure 5-14.

With the preliminary limit load factors given in section 4.0 and using a safety factor of 1.4, the maximum payload (2000 pound spacecraft) was uniformly distributed to the mounting surface of one end and traced through the structure for load continuity and internal structural loads, as illustrated in figure 5-15.

In simplifying the solution of a redundant structure, the applied ultimate loads as shown in the sketch were reacted by the ribs, in turn reacted by the inner cylinder and then transferred by the two horizontal ribs to the support fittings. The panels were also investigated for the 145-dB acoustic environment.

This simplified analysis is conservative and allows for the unknown detailed weights which can not be investigated at this level of design. The highlights and basic assumptions of the analysis are presented in Appendix A.

TABLE 5.5

RTG S/C . PRELIMINARY WEIGHTS

PRIMARY STRUCTURE		ELECTRICAL POWER SYSTEM (CONTINUED)	
• SHELLS	84.0	• POWER CONDIT. UNIT	17.6
• RIBS	27.0	• LOAD CONTROLLER	3.0
• RINGS	24.5	• BATTERY CONT.	5.0
• SUPPORT FTG (4)	12.5	• MISC.	5.0 (320.0)
• STIFFENERS	20.0		
• RTG SUPT. (3)	6.0	S/C PROPULSION AND ATTITUDE	
• CONTIG (10%)	16.0 (190.0)	CONTROL SYSTEM	
EQUIPMENT SUPPORT STRUCTURE		• TANKS	25.0
• ELECT. POWER	3.0	• PROPELLANT	372.0
• SPS	7.0	• SENSORS	26.0
• TELE. & COMMAND	4.0	• WHEELS (3)	58.0
• COMM.	4.0	• ELECTRONICS	20.0
• EXPERIMENTS	7.0 (25.0)	• COMPUTERS	45.0 (546.0)
THERMAL CONTROL		TELE. & COMMAND SYSTEM	
• INSULATION	18.2	• S BAND REC.	8.0
• COATINGS	2.5	• S BAND TRANS.	4.0
• HEAT PIPES	31.0	• VHF REC.	6.0
• VANES (3)	6.9	• VHF TRANS.	4.0
• RTG RADIATORS (3)	14.4	• LDR MUX	6.0
• ISOLATORS & HEATERS	5.0 (78.0)	• MDR MUX	6.0
ELECTRICAL POWER SYSTEM		• RECORDERS (2)	40.0
• RTG (3)	240.0	• CMD DECODER	5.0 (79.0)
• BATTERY	37.0	BASIC S/C WEIGHT	
• SHUNT REG.	3.0	(1236.0)	
• FAULT DETECTOR	6.0	MISSION EQUIPMENT	
• SHUNT DISSIPATORS	3.4	(764.0)	
		TOTAL S/C WEIGHT	
		(2000.0)

FIGURE 5-14 RTG SPACECRAFT MASS PROPERTIES

FIGURE 5-15 MAX. LOAD DISTRIBUTION RTG SPACECRAFT

5.3.5 ATTITUDE AND ORBIT CONTROL SYSTEM DESIGN

The common attitude and orbit control system described in Sec. 5.2.5 has been applied to this vehicle. Appendix B lists the detailed design calculations and parameters. The component inventory and propellant requirement are summed up in the inventory listed below. (Note that low earth vehicles will include a magnetic torque compensation and unloading system not required in synchronous orbit.)

5.3.5.1 Propellant Requirements

Table 5.6 lists the required hydrazine for a conservatively designed spacecraft in a 5-year near-polar mission. This propellant will also be adequate for a craft placed in synchronous equatorial orbit utilizing this fuel for control and orbit maintenance.

Table 5.6

RTG Spacecraft Propellant Allowances (pounds)

Initial S/C weight	1000	1500	2000	2500
Dry weight	832.7	1251.3	1669.9	2088.5
Propellant allowances:				
1170 ft/sec ΔV (orbit insertion or 5 years N-S station keeping)	152	228	304	380
Spin up, Spin down, Spin Vector				
Precession	2	2	2	2
Orbit maintenance/correction (E-W)	10.4	15.6	20.8	26
Attitude Control	2.9	3.1	3.3	3.5
Total Propellant Weight	167.3	248.7	330.1	411.5

5.3.5.2 Attitude Control Components

The recommended system will contain the following components:

a) <u>Transfer mode:</u>	<u>Estimated weight, pounds</u>
2 Earth Sensors	4
2 Sun Sensors	2
Rate Gyro Assembly	4

b)	<u>Operational mode:</u>	<u>Estimated weight, pounds</u>
	3 Momentum Wheels	58 (total)
	2 Digital Computers	45 "
	2 Earth Sensors and Electronics	15
	Sun Sensors	2
	Actuator Electronics	<u>20</u>
	Total	150 lbs

In addition, the low orbit vehicles will require:

c)	Magnetic moment torquers	6 lbs
	Magnetometer	3 lbs

5.3.6 TT&C SYSTEM

This system, common to both the RTG and Brayton vehicles, is described in Section 5.2.6, Table 5.2.

5.3.7 BASIC SPACECRAFT EQUIPMENT

Table 5.7 lists the standard equipment of the basic RTG spacecraft. Figure 5-16 shows one possible layout of the equipment in the vehicle.

5.4 BRAYTON SPACECRAFT DESIGN

5.4.1 MECHANICAL CONFIGURATION

The structure and dimensions of the Brayton spacecraft evolved from the following ground rules:

- Total vehicle weights of 2000 to 4000 pounds (See Table 3.13)
- Capability of 660 pounds of liquid propellant (See Appendix B.2)
- Equipment mounting surface heat dissipation of 20 watts per square foot.
- Adaptability to tie to the orbiter, pallet, or tug
- Ability to accept from one to three isotope heat sources to power the Brayton cycle electrical generating system.

The configuration selected, shown in Fig 4-3, is 13 feet in diameter and 4 feet in length, with an inner diameter of 4 feet. The center hole on the ring is occupied by a spherical hydrazine tank and the nozzles of the

TABLE 5.7 EQUIPMENT LIST, BASIC RTG SPACECRAFT

ITEM	NAME	QTY.	SIZE	WT. (EA.)
1	REACTION WHEEL	3	13.7 DIA. X 4.7	19.7
2	EARTH SENSOR ELECT.	1	8.5 X 8.0 X 8.0	13.0
3	EARTH SENSOR	2	5.5 X 5.5 X 5.0	1.0
4	COMPUTER	2	12.0 X 11.0 X 8.0	22.0
5	SUN SENSOR ELECT.	1	4.0 X 5.75 X 5.25	3.0
6	RATE GYRO	1	4.0 X 6.0 X 4.0	4.0
7	SUN SENSOR	2	1.0 X 2.0 X 3.0	1.0
8	ACTUATOR ELECT.	1	9.0 X 12.5 X 7.4	20.0
9	SHUNT	6	3.5 X 10.0 X 2.5	2.3
10	LOAD CONTROLLER	1	5.0 X 5.0 X 5.0	3.0
11	POWER CONDITION.	1	6.0 X 8.0 X 12.0	17.6
12	VHF XMITTER	1	11.0 X 11.0 X 2.0	4.0
13	LDR MULTIPLEXER	1	6.0 X 10.0 X 6.0	6.0
14	BATTERY	1	8.5 X 6.0 X 11.5	37.0
15	RECORDER	2	15.0 X 4.5 X 15.0	20.0
16	MDR MULTIPLEXER	1	6.0 X 10.0 X 6.0	6.0
17	S-BAND XMITTER	1	11.0 X 11.0 X 2.0	4.0
18	COMMAND DECODER	1	6.0 X 10.0 X 4.0	5.0
19	VHF RECEIVER	1	9.0 X 9.0 X 2.5	6.0
20	S-BAND RECEIVER	1	9.0 X 9.0 X 2.5	8.0

FIGURE 5-16 EQUIPMENT PLACEMENT IN RTG SPACECRAFT

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propulsion system just as in the RTG spacecraft. The complete Brayton electric power system is installed within the structure. From one to three 2400-thermal-watt isotope heat sources may be installed without changing any of the power system hardware. The ring is divided by ribs into eight compartments or bays, with two occupied by the Brayton power system, two by the other standard subsystems, and four available for payload. Preliminary structural weight is about 490 pounds; the Brayton generator powered by three heat sources weighs 705 pounds, giving a total of about 1200 pounds for structure and electric power generating system.

At 20 watts per square foot, 1500 watt output would require 75 square feet of heat dissipating surface. Two bays will be used for the Brayton system, leaving six compartments with 90.5 square feet of surface on each side. The overall vehicle dimensions will comfortably satisfy the established requirements for the mounting surface. Half of the total vehicle volume of 534 cubic feet is available for mission equipment, which is generally to be mounted only on one of the end sections. This surface is the one which will be pointed away from earth; only earth viewing items are to be mounted on the opposite end.

Figure 5-17 shows the structural system in more detail. The outer shell structure constitutes the radiator of the Brayton power plant. This shell structure, in conjunction with a skeletal compartment frame (2 bays), is intended to be built up separately as an independent system which can be checked out and then installed over the main hub structure and secured. The equipment mounting surfaces will be approximately one-half-inch honeycomb panels that will rotate beyond 90 degrees through the panel center line providing access to the equipment. The earth viewing surfaces will be 1/4 inch honeycomb panels to support any earth viewing items. A stiffened cylindrical shell structure will constitute the central core to which the truss ribs will be tied. It will also constitute the inner cap of the end bulkhead.

The 13 foot outside diameter can be tied directly to a pallet or to the orbiter. Figure 5-18 depicts the installation of the Brayton spacecraft to

FIGURE 5-17. BASIC BRAYTON SPACECRAFT (1 OF 3)

FIGURE 5-17 BASIC BRAYTON SPACECRAFT (2 of 3)

EQUIPMENT LIST, "BASIC" BRAYTON BUS			
ITEM	DESCRIPTION	SIZE (W. X L. X DP.)	WEIGHT, LBS.
1	CONTROL BOX (MINI-BRAYTON)	13.0 X 13.0 X 3.5	15#
2	MOMENTUM WHEEL (3)	15.0 DIA. X 5.0	28# EACH
3	SUN SENSOR	2.5 X 2.5 X 2.0	1#
4	SUN SENSOR ELECTRONICS	4.0 X 5.75 X 5.25	3#
5	COMPUTER (2)	10.6 X 12.8 X 8.0	23# EACH
6	ACTUATOR ELECTRONICS	9.3 X 12.8 X 7.4	20#
7	S-BAND XMITTER	11.0 X 11.0 X 2.0	4#
8	VHF XMITTER	11.0 X 11.0 X 2.0	4#
9	S-BAND RECEIVER	9.0 X 9.0 X 2.5	8#
10	VHF RECEIVER	9.0 X 9.0 X 2.5	6#
11	LDR MUX	6.0 X 10.0 X 6.0	6#
12	MDR MUX	6.0 X 10.0 X 6.0	6#
13	COMMAND DECODER	6.0 X 10.0 X 4.0	5#
14	POWER CONDITIONING UNIT	8.0 X 20.0 X 7.0	35#
15	BATTERY (2)	8.5 X 11.5 X 5.5	37# EACH
16	LOAD CONTROLLER	5.0 X 5.0 X 5.0	3#

FIGURE 5-17 BASIC BRAYTON SPACECRAFT (3 of 3)

FIGURE 5-18 BRAYTON SPACECRAFT ON PALLET IN ORBITER

a pallet or directly to the orbiter. Figure 5-19 shows a possible scheme of stacking 6 Brayton spacecraft on a pallet. The pallet can also support a mixture of Brayton and RTG spacecraft, as shown in Figure 5-20.

To tie the Brayton spacecraft to a tug, either the four foot inside diameter ring or the 13 foot outside diameter ring can be used as a starting point to mate to a tug support ring of arbitrary diameter.

5.4.2 ELECTRICAL POWER SUBSYSTEM

5.4.2.1 Block Diagram and Description

The block diagram of the electrical power subsystem is shown in Figure 5-21. The isotope-Brayton assembly contains a heat source and a closed-cycle gas system which transforms the isotope heat to mechanical shaft work and then to electrical power by means of a solid-rotor alternator. Shaft speed is controlled by adjusting the electrical load on the alternator. Rated AC power output from the alternator is rectified and filtered to produce DC power, which is then shared between a parasitic load resistor and the useful load. Voltage control is maintained by compound connections to the shunt and series field windings on the alternator stator. A single assembly will be mounted in the spacecraft with output electrical power of 500 to 2200 watt, depending on mission requirements. The desired power levels will be obtained by varying the number of fuel capsules and by an appropriate adjustment of the gas pressure in the system.

The electrical output of the isotope-Brayton assembly available at the main bus will be 30 VDC. The remaining connections to the main bus are identical functionally to the RTG electrical power system. Payloads are supplied through an assembly of load controllers. A priority bus connected to the main bus is provided. Fault detection of over and under voltage at the main bus is also provided. When a payload requires its own battery for peak loads beyond the capability of the Brayton system, the battery system can be connected to the main bus through appropriate interface circuitry.

FIGURE 5-19. BRAYTON SPACECRAFT
STACKED ON PALLET

FIGURE 5-20. RTG AND BRAYTON SPACECRAFT
STACKED ON PALLET

FIGURE 5-21 BLOCK DIAGRAM, BRAYTON SPACECRAFT ELECTRICAL POWER SUBSYSTEM

4
5.3.2.2

Isotope-Brayton Start-Up

An additional requirement for the Brayton engine is a start-up phase with all equipment emplaced. The start-up is accomplished by shorting the field windings on the alternator and supplying three-phase, 400 Hertz power to the alternator stators. This will cause the alternator to act as a motor, driving the common shaft on which the compressor and turbine are also mounted to about one half of rated shaft speed. At this point the system will bootstrap itself up to rated speed.

The start-up procedure may be implemented by using the shuttle power supply. However, since a battery is required in the spacecraft for the priority bus, it can also be used to initiate the start-up through a suitably designed inverter. Approximately 15-20 watt-hours of battery energy is required, which is well within the rating of the battery required for the priority bus.

5.4.2.3

Three-Machine Concept

The selected power system concept is based upon the use of a single engine. During the study, consideration was given to a load sharing concept if three engines were used. The system would operate by sensing current from each alternator and supplying a correction voltage to the shunt fields of those alternators which are delivering less output current. The increased current in the shunt field would raise the output voltage from the alternator supplying less output current, thereby increasing the current output of these alternators. The net result is to have each alternator share the load current equally. Since the alternators are sharing load current on a DC power basis, many of the problems of paralleling alternators on a AC power basis are eliminated. Nevertheless, the system is not as simple as one with a single generator, as in the reference design.

5.4.2.4

Transient Performance

One of the most difficult areas to assess for the isotope-Brayton system is the matter of transient performance. This is one

of the more troublesome problems generally encountered in a spacecraft electrical integration phase. Some measurements have been made (Ref 5-2) on the Brayton 'B' 10 kW engine which indicated for a rather severe transient of a full-load to no-load step that the bus voltage would rise by at least 20% with the first crossover at 90 milliseconds. The settling time to within 1% of rated voltage is about 370 milliseconds, as shown in Figure 5-22. These results would of course, have to be examined for the smaller Brayton engine considered in this study, in which another control scheme is employed. Another problem will be to determine a standard of transient performance acceptability for the spacecraft--one which minimizes EMI and transmission problems. It is apparent that further effort will be required to define the Brayton electrical performance before the criteria can be stated.

5.4.3 THERMAL CONTROL

5.4.3.1 Design Criteria

The thermal design criteria employed for this study were dictated primarily by the need for electronic component functional reliability over a five-year mission. Several of the criteria, including use of the waste heat from the nuclear powered generator for thermal control, are identical to those established for the RTG spacecraft.

(1) Temperature Limits

It was assumed that all electronic components must be maintained between -10°C and 40°C during normal spacecraft operation. The battery requires a narrower temperature range of -5°C to 25°C , while the hydrazine must be held above 5°C .

(2) Environmental Considerations

Examination of the mission model for proposed spacecraft missions in the 500 to 2500 watt range indicated that the proposed Brayton spacecraft would have one surface earth-oriented (see Table 3.14). The worst case hot orbit is a 200 n. mi. noon orbit (sun in the orbit plane)

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FIGURE 5-22 TRANSIENT RESPONSE OF THE 10 kW(e) 1200-HERTZ BRAYTON-CYCLE POWER SYSTEM TO A 10 kW(e), 0.85 POWER FACTOR LOAD CHANGE

with one face earth-oriented. The worst case cold orbit is an earth-oriented synchronous orbit.

(3) Heat Dissipation

Using the same generator system and varying only the number of heat sources and the gas pressure, the efficiency of the Brayton cycle generator was assumed to increase monotonically with the number of heat sources used. For the purposes of this study, the following efficiencies were assumed:

<u>No. of Isotope Heat Sources</u>	<u>Thermal Watts</u>	<u>Efficiency</u>	<u>Electrical Watts</u>
1	2400	.23	550
2	4800	.26	1250
3	7200	.30	2160

It is assumed that all electrical energy is dissipated in the equipment compartment. When the equipment cannot use the total output power from the Brayton generator, the unused power is diverted to the parasitic load resistor mounted above the Brayton equipment compartment. Alternatively, the parasitic load can be distributed throughout the equipment compartments to maintain nearly constant dissipation with varying equipment duty cycles.

5.4.3.2 Design Description

The objective of the thermal control subsystem is to maintain the equipment within acceptable temperature limits regardless of the spacecraft altitude, orientation, or mission. As was described for the RTG spacecraft, the use of nuclear waste heat to minimize the temperature excursions of equipment due to variations in environmental inputs and internal dissipation was investigated.

The selected thermal control concept is the so-called radiator wall. It uses the Brayton system radiator, made approximately isothermal by use of heat pipes and counter-flow gas duct arrangement, surrounding the equipment compartments in a cylindrical wall. The radiator is sized for

40°C maximum temperature, and provides a radiant heat to the equipment compartment (see Figure 5-23). In contrast to the RTG thermal control system, no active control is required. The mission model for the Brayton spacecraft (see Table 3.13) indicates that all projected missions in the 500 to 2500 electrical watt range will be Earth-oriented. It has been assumed in developing the thermal control concept that heat-dissipating equipment will be mounted to the face pointed away from earth. This surface is desirable for such purposes since it is exposed only to solar energy whose effect can be minimized by employing a stable low " α/ϵ " thermal coating. Earth-viewing sensors, of course, must be mounted to the opposite face. To accommodate the requirements of any associated dissipative equipment, the radiative coupling from the equipment radiator to space can be adjusted.

Selection of this thermal design concept was based upon dimensional constraints, Brayton generator and equipment temperature requirements, and weight considerations. To allow the economic and the test-case aspects of the study to proceed, this configuration was frozen prior to the completion of the investigation of several other concepts. The merits of other concepts, which appear to provide greater flexibility at the expense of weight, cost, and maintainability considerations, are discussed briefly in Table 3.8.

Figure 5-23 illustrates the features of the selected thermal control concept. The radiating area on the gas radiator and equipment compartment must be adjusted according to the number of heat sources used in the spacecraft for a particular mission. Silver-on-teflon is used as the thermal coating on the gas radiator and on the equipment compartment radiator. This is a space-stable coating which exhibits a low ratio of solar absorptivity (α) to infrared emissivity (ϵ). Figure 5-24 shows the variation of the equipment panel temperatures with the dissipation of the equipment mounted on the panel. This is explained in greater detail in section 5.4.3.3.2.

The thermal control subsystem weighs 314 pounds, including the Brayton system gas radiator and thermal insulation.

EQUIPMENT COMPARTMENTS
EACH WITH ITS OWN RADIATOR;
SILVER-ON-TEFLON THERMAL COATING)

ISOTHERMAL GAS RADIATOR
(40°C IN LOW EARTH ORBIT;
SILVER-ON-TEFLON THERMAL COATING)

GAS RADIATOR
PAINTED BLACK
SIDE - Inside

BRAYTON
COMPONENTS

PARASITIC LOAD
RESISTOR LOCATED
ABOVE COMPONENTS

EARTH

EQUIPMENT
PAINTED
BLACK

MULTILAYER INSULATION
(ALUMINIZED SURFACE INSIDE)

FIGURE 5-23 BRAYTON SPACECRAFT - THERMAL DESIGN CONCEPT

FIGURE 5-24 EQUIPMENT PANEL TEMPERATURE - BRAYTON SPACECRAFT

As described above, the thermal design concept for the Brayton spacecraft consists of 40°C isothermal gas radiator radiatively coupled to the equipment compartments (see Figure 5-24). Heat-dissipating equipment is ordinarily mounted to the face pointed away from earth. This is not strictly necessary since experiment compartments can be designed for a particular mission to accommodate earth-viewing equipment. The effect on equipment temperature of variations in environmental heat sources and internal dissipation is minimized by the thermal coupling between the warm gas radiator and the equipment compartments. Several other thermal design concepts were investigated; the following section provides a brief description of the alternative concepts and their advantages and disadvantages.

5.4.3.3.1 Other Thermal Control Concepts

All of the thermal control concepts utilized a gas radiator surrounding the equipment, but the gas radiators considered were not all isothermal. Table 5.8 describes the features of several of the more promising concepts along with their advantages and disadvantages. The table is self-explanatory and the various concepts will not be discussed here. The configuration selected, (Concept 3 of Table 5.8), was chosen fairly early in the study in order to proceed with the economic and test case aspects of the study. Concepts 5 and 6 are two concepts which were investigated after Concept 3 was selected. These two concepts are more flexible than the selected configuration in that they can tolerate greater variations in equipment heat dissipation by utilizing a feedback louver system. Figures 5-25 and 5-26 show the variation of internal dissipation with temperature for Concepts 5 and 6, respectively. Both concepts involve greater complexity than the selected configuration and each has its own drawbacks. Further study is warranted to determine the optimum configuration with respect to mechanical complexity, cost, weight and thermal considerations.

TABLE 5.8
BRAYTON SPACECRAFT THERMAL DESIGN CONCEPTS

CONCEPT NO.	CONCEPT	DESCRIPTION	ADVANTAGES	DISADVANTAGES
1		<ul style="list-style-type: none"> • ALL HEAT GENERATED BY BRAYTON IS RADIATED FROM GAS RADIATOR. • LOUVERS COULD BE USED TO VARY RADIATIVE THERMAL COUPLINGS BETWEEN EQUIPMENT AND GAS RADIATOR. 	<ul style="list-style-type: none"> • EQUIPMENT SURROUNDED BY BENIGN ENVIRONMENT (0°C). • THERMAL DESIGN INDEPENDENT OF ALTITUDE. 	<ul style="list-style-type: none"> • REQUIRES LARGE SPACECRAFT 10' LONG x 15' DIA. FOR 7200 WATTS. • EQUIPMENT WATT DENSITY LIMITED (4.8 WATTS/FT.²). • GAS RADIATOR AREA TO SPACE MUST BE CHANGED WITH POWER LEVEL.
2		<ul style="list-style-type: none"> • GAS RADIATOR SIZED TO RADIATE BRAYTON WASTE HEAT AND SERVE AS 0°C SINKS FOR EQUIPMENT. • EQUIPMENT RADIATOR SIZED TO RADIATE MINIMUM EQUIPMENT DISSIPATION. 	<ul style="list-style-type: none"> • 0°C GAS RADIATOR TEMPERATURE COMPATIBLE WITH BRAYTON CYCLE REQUIREMENTS. • THERMAL DESIGN INDEPENDENT OF ALTITUDE. 	<ul style="list-style-type: none"> • REQUIRES LARGE SPACECRAFT: 7 FT. LONG x 15 FT. DIA. FOR 7200 WATTS. • RADIATING AREA MUST BE ADJUSTED WITH POWER LEVEL.
3		<ul style="list-style-type: none"> • ANTI-EARTH FACE IS PRIMARY RADIATOR FOR EQUIPMENT. • GAS RADIATOR SERVES AS CONSTANT HEAT INPUT TO EQUIPMENT TO MINIMIZE EFFECT OF VARIATIONS IN HEAT DISSIPATION & ENVIRONMENTAL SOURCES. 	<ul style="list-style-type: none"> • GAS RADIATOR PROVIDES UNIFORM HEAT INPUT TO EQUIPMENT AND A CLEAN INTERFACE EXISTS BETWEEN EQUIPMENT & GAS RADIATOR. • THERMAL DESIGN INDEPENDENT OF ALTITUDE. 	<ul style="list-style-type: none"> • HIGHER GAS RADIATOR TEMPERATURE WILL AFFECT BRAYTON EFFICIENCY. • RADIATING AREA MUST BE ADJUSTED WITH POWER LEVEL.
4		<ul style="list-style-type: none"> • ANTI-EARTH FACE IS PRIMARY RADIATOR FOR EQUIPMENT. • GAS RADIATOR PROVIDES A HEAT INPUT TO EQUIPMENT. 	<ul style="list-style-type: none"> • NON-ISOTHERMAL GAS RADIATOR IS SMALLER THAN ISOTHERMAL ONE. • AT REDUCED POWER LEVELS EXIT GAS FROM GAS RADIATOR WILL BE COLD THEREBY INCREASING BRAYTON EFFICIENCY. 	<ul style="list-style-type: none"> • TEMPERATURE GRADIENT WILL EXIST AROUND THE PERIPHERY OF GAS RADIATOR. • THERMAL COUPLING BETWEEN EQUIPMENT AND GAS RADIATOR MUST BE VARIED WITH POWER LEVEL. • ANALYTICALLY MORE DIFFICULT.
5		<ul style="list-style-type: none"> • EQUIPMENT MOUNTED ON BOTH FACES OF EQUIPMENT COMPARTMENT. • LOUVERS MODULATE HEAT FLOW BETWEEN EQUIPMENT AND ISOTHERMAL GAS RADIATOR. 	<ul style="list-style-type: none"> • THERMAL DESIGN IS MORE FLEXIBLE THAN CONCEPT 3 DUE TO FEED BACK ATC SYSTEM. 	<ul style="list-style-type: none"> • HIGHER GAS RADIATOR TEMPERATURE WILL AFFECT BRAYTON EFFICIENCY. • RADIATING AREA MUST BE ADJUSTED WITH POWER LEVEL.
6		<ul style="list-style-type: none"> • EQUIPMENT MOUNTED ON BOTH FACES OF EQUIPMENT COMPARTMENT WHICH IS ISOTHERMALIZED WITH HEAT PIPES. • LOUVERS REGULATE HEAT FLOW BETWEEN EQUIPMENT AND NON-ISOTHERMAL GAS RADIATOR. 	<ul style="list-style-type: none"> • THERMAL DESIGN IS FLEXIBLE AND HEAT PIPES ISOTHERMALIZED EQUIPMENT PANEL. 	<ul style="list-style-type: none"> • NON-ISOTHERMAL GAS RADIATOR MAKES STRUCTURAL DESIGN MORE DIFFICULT. • COMPLEX DESIGN. • PERIPHERAL TEMPERATURE GRADIENTS EXIST IN GAS RADIATOR.
7		<ul style="list-style-type: none"> • TWO EQUIPMENT COMPARTMENTS EXIST SEPARATED BY HOT PORTIONS OF GAS RADIATOR. • LOUVERS REGULATE HEAT FLOW BETWEEN EQUIPMENT COMPARTMENTS AND HOT PORTIONS OF GAS RADIATOR. 	<ul style="list-style-type: none"> • THERMAL DESIGN INDEPENDENT OF ALTITUDE AND POSSIBLE POWER LEVEL. • SMALLER GAS RADIATOR WITH COLD EXIT GAS TEMPERATURE TO ENHANCE BRAYTON EFFICIENCY. 	<ul style="list-style-type: none"> • COMPLEX PLUMBING ARRANGEMENT FOR GAS RADIATOR. • EARTH VIEWING EQUIPMENT COMPARTMENT IS ISOLATED LIMITED IN HEAT REJECTION CAPABILITY.

FIGURE 5-25 THERMAL CONTROL CONCEPT #5 - TEMPERATURE RANGE

FIGURE 5-26. THERMAL CONTROL CONCEPT #6, TEMPERATURE RANGE

5.4.3.3.2 Thermal Analysis and Design of the Selected Concept.

A thermal analysis of the selected configuration was performed to demonstrate the capability of the thermal control subsystem. The analysis was performed for the three-heat-source system which has 7200 thermal watts input and generates 2160 electrical watts. Lower power levels will require that both the effective radiating area of the gas radiator and the equipment compartment radiator be reduced. This can be seen from the following equations:

$$Q_w = R_{gR-S1} (\sigma T_{gR}^4 - \sigma T_{S1}^4) + R_{gR-E} (\sigma T_{gR}^4 - \sigma T_E^4) \quad (5-1)$$

$$Q_E = R_{E-S2} (\sigma T_E^4 - \sigma T_{S2}^4) + R_{E-gR} (\sigma T_E^4 - \sigma T_{gR}^4) \quad (5-2)$$

where:

Q_w	=	Brayton waste heat
Q_E	=	Equipment dissipation
R	=	Radiation coupling between spacecraft elements
T	=	Temperature
σ	=	Stefan-Boltzmann constant

subscripts:

gR	=	gas radiator
$S1$	=	equivalent radiating sink for gas radiator
$S2$	=	equivalent radiating sink for equipment
E	=	equipment

As can be seen, if the radiator couplings are reduced in proportion to the power level, the results of the analysis apply to all power levels.

As an aid in sizing the spacecraft, Equations (5-1) and (5-2) were plotted parametrically using the following parameters: (1) a 40°C isothermal gas radiator, (2) a 20°C equipment panel radiator, (3) 7200 thermal watts Brayton input, and (4) a 200 n. mi. Noon orbit environment.

Figure 5-27 is a cross plot of the required radiation coupling from the gas radiator and equipment radiator to space. Figure 5-28 shows the variation in required equipment radiator coupling to space as a function of equipment dissipation and radiation coupling between the gas radiator and equipment radiator. Utilizing dimensional and mechanical constraints, the Brayton spacecraft was sized to have a 13-foot outer diameter, be four feet high, and have an inner diameter of four feet. These dimensions are based upon 2160 watts of generated power.

The gas radiator temperature of 40°C resulted from a compromise between the desire to keep the exit gas temperature as low as possible to increase the Brayton cycle efficiency and the need to maintain a positive temperature difference between the gas radiator and the equipment.

Isothermal Gas Radiator

The isothermal gas radiator shown in Figure 5-29 consists of counterflow gas ducts and axial heat pipes mounted on a thin cylindrical radiator shell. The counterflow arrangement not only minimizes temperature gradients produced by the hot gas but also reduces gradients produced by other heat sources.

Preliminary feasibility analyses of the counterflow radiator indicated that the heat transfer film coefficient, pressure drops, and temperature gradients are reasonable. However, a more detailed analysis is required to verify the concept. The counterflow ducts were sized for 0.3 lb/sec. total flow rate and 0.3 psi pressure drop. The resulting inner diameter of the counter-flow tubes is 1.25 inches.

A study was performed to determine the lightest construction technique for the radiator. A fin efficiency of 90% was selected as a minimum value in order to keep the size of the gas radiator reasonable. Figure 5-30 illustrates the variation of fin efficiency as a function of skin thickness and fin length (this plot is the result of the linearized solution of the fin equation). One concept for the isothermal gas radiator was to have three sets

FIGURE 5-27 DISTRIBUTION OF RADIATION BETWEEN GAS RADIATOR AND EQUIPMENT PANEL RADIATOR

EQUIPMENT
DISSIPATION
(WATTS)

FIGURE 5-28 PANEL AREA VERSUS HEAT LOAD AT FIXED TEMPERATURE

HEAT PIPES
(APPROX. EVERY 5.5")

COUNTERFLOW GAS DUCTS

RADIATOR SKIN

COLD OUTLET
TO COMPRESSOR

HOT INLET FROM
RECUPATOR

GAS FLOW PATHS

FIGURE 5-29 ISOTHERMAL RADIATOR CONFIGURATION

FIGURE 5-30. FIN EFFICIENCY VS SKIN THICKNESS

of parallel counterflow pipes approximately 8 inches apart. Not including the increased weight of the steel pipes, the gas radiator would weigh 332 pounds. The concept utilizing axial heat pipes and a thin aluminum skin was finally adopted. Figure 5-31 is a plot of radiator weight versus skin thickness. As can be seen, no minimum exists in the curve to indicate an optimum skin thickness to number of heat pipes. A skin thickness of .02 inch and 8³/₄ heat pipes was selected for the Brayton spacecraft.

The analytic development is given in Appendix C.

Equipment Compartments

The spacecraft electronic components are distributed into six equipment compartments. The housekeeping components (TT & C, power conditioning, and attitude control) are mounted into two compartments leaving the remaining four compartments for mission equipment. The dissipating components are mounted to a honeycomb panel whose skin thickness and effective radiating area will be determined by the distribution of heat in the components. Each equipment panel will serve as the thermal radiator for the components mounted on it. The effective radiating area to space of each equipment panel will be adjusted to handle the attendant equipment dissipation (the packaging density is assumed to be between 20 and 30 watts/ft²). If the generated electric power is not used, the parasitic load resistor dissipates the excess power. This resistor is simply a high-temperature heater physically located on the panel cover of the power-plant bay. If necessary, some of the excess power could be used for heaters in a compartment requiring narrower temperature control.

Figure 5-32 illustrates the maximum temperature variation of the equipment as a function of the radiation coupling between the gas radiator and equipment. This figure is the result of steady-state computations and therefore represents a worst case variation. As can be seen, the greater the coupling, the smaller the temperature variation. A coupling of 50 ft.² was chosen based upon dimensional considerations. Figure 5-24 shows

FIGURE 5-31 GAS RADIATOR WEIGHT

FIGURE 5-32 MAXIMUM EQUIPMENT TEMPERATURE VARIATION VS. THERMAL COUPLING BETWEEN GAS RADIATOR & EQUIPMENT

the variation in equipment dissipation as a function of temperature and environment for a fixed equipment panel radiating area. The broken line assumes the thermal input to the Brayton is reduced by 20% from 7200 to 5760 watts. Note, however, that over five years the thermal output is reduced by only four percent. These curves could be shifted to the right (toward the 40°C temperature) by reducing the equipment radiating area to space. The important thing to note about Figure 5-24 is the range of tolerable heat dissipation. A variation in heat dissipation by a factor of 2-to-3 is tolerable without excessive temperature variation.

The effective radiating area of the equipment panels will be adjusted according to the dissipation of the equipment attached to that panel to produce a nominal 20-30 W/ft² for each equipment panel.

Brayton Generator Compartments

The Brayton generator components are located in a 90° sector of the spacecraft. The entire compartment is thermally isolated from other compartments on the bus by reflective multilayer insulation lining the compartment walls. The individual components of the Brayton engine are also insulated to preclude heat losses which will affect cycle efficiency.

Two types of insulation are used on the Brayton components:

1. Heat Sources Recuperator, and BRU - The isotope heat sources, recuperator, Brayton rotating unit, and connecting ductwork are covered with 5-inches of Min-K-2000 and 20 layers of standard multi-layer insulation. Min-K is used to permit ground-testing the generator, since the multi-foil insulation is relatively ineffective in the atmosphere, and may be degraded at high temperature in air.
2. Low-Temperature Ductwork - All lower-temperature ductwork is covered with multi-layer aluminized mylar separated by nylon net.
3. Gas Radiator - Inactive portions of the gas radiator which must be insulated when fewer than three heat sources are

used, are covered with standard multi-layer insulation consisting of 30 layers of 1/4 mil aluminized mylar separated by nylon net.

5.4.3.4. Weight of Thermal Control Subsystem

The weight breakdown of the thermal control subsystem is as follows:

<u>Item</u>	<u>Weight (lb.)</u>
Heat Pipes	54
Insulation	144.2
Thermal Coatings	
Silver on Teflon	17.0
Black Paint	5.7
Gas Radiator (skin and counter-flow heat exchanger)	92
Miscellaneous Isolators	5
Heaters	5
TOTAL	<u>313.7 pounds</u>

It should be noted that this weight includes the counter-flow gas radiator of the Brayton generator and the thermal insulation associated with the Brayton components, for a system capable of delivering 2200 watts (e).

5.4.4 STRUCTURAL ANALYSIS

The structural configuration that was discussed and depicted in the mechanical design section was analyzed for weights, mass properties, structural load continuity, structural integrity, and the capability of integrating it into the shuttle orbiter payload compartment.

The initial structural sizing resulted in a weight of 490 pounds. A more detailed analysis brought this weight down to 400 pounds. Table 5.9 gives the structural weights as well as the weights of other subsystems that would constitute a spacecraft mission weight. The mass moments of inertia as a function of gross weight are shown in Figure 5-33.

TABLE 5.9

BRAYTON S/C - PRELIMINARY WEIGHTS

PRIMARY STRUCTURE		ELECTRICAL PWR. SYS. (BRAYTON UNIT) (CONTINUED)	
● SHELL	113.0	● PLUMBING	12.0
● RIBS	55.5	● ELECTRONICS	15.0
● RINGS	83.5	● BATT. & INVERTER	59.0
● STIFFENERS	7.3	● PLR	20.0
● BRAYTON SPT. STR.	18.0	● INSULATION	<u>109.2 (705.2)</u>
● BUS SPT. STR.	90.0		
● CONTIG (10%)	<u>37.0 (404.3)</u>		
EQUIPMENT SUPPORT STRUCTURE		S/C PROPULSION AND ATTITUDE CONTROL SYSTEM	
● ELECT. POWER	5.	● TANKS	32.0
● SPS	10.	● PROPELLANT	560.0
● TELE. & COMMAND	5.	● SENSORS	17.0
● COMM.	5.	● WHEELS (3)	84.0
● EXPERIMENTS	<u>10. (35.0)</u>	● COMPUTERS (2)	45.0
		● ELECTRONICS	<u>20.0 (758.0)</u>
THERMAL CONTROL		TELE. & COMMAND SYSTEM	
● HEAT PIPES	54.0	● S BAND REC.	8.0
● INSULATION	35.0	● S BAND TRANS.	4.0
● COATINGS	17.0	● VHF REC.	6.0
● PAINT	5.7	● VHF TRANS.	4.0
● HEATERS	5.0	● LDR MUX	6.0
● MISC. ISOLATORS	<u>5.0 (121.7)</u>	● MDR MUX	6.0
		● CMD DECODER	5.0
ELECTRICAL PWR. SYS. (BRAYTON UNIT)		● RECORDERS (2)	<u>40.0 (79.0)</u>
● GAS RADIATOR	92.0		
● HEAT SOURCES (3)	305.0	BASIC S/C WEIGHT (2103.2)	
● RECUPERATOR	68.0	MISSION EQUIPMENT (1896.8)	
● TURBINE	25.0	TOTAL SPACECRAFT (4000.0)	

MOMENT
OF
INERTIA -
SLUG - FT²

FIGURE 5-33 BRAYTON SPACECRAFT MASS PROPERTIES

The structural analysis was done for 4000 pound vehicle. With the preliminary load factors estimated from Section 3 (Table 3.14, e.g.) and a safety factor of 1.4 for ultimate loads, the 4000 pounds was distributed as follows:

- 705 pounds to the 2 Brayton system compartments
- 560 pounds for the propellant system
- 525 pounds to one wall of each of the 6 compartments

Figure 5-34 illustrates the loads used in the analysis.

The mounting panels were investigated for the uniformly distributed inertia loads as well as the acoustic loading.

As discussed in the RTG structural analysis section the same simplifying assumptions were used in deriving the internal loads, structural continuity and basic structural sizes. The inertia loads were transferred by the mounting panels to the ribs, in turn to the 4 foot diameter cylinder and then by the horizontal ribs to the external support points.

5.4.5 BRAYTON CYCLE CONFIGURATION-ATTITUDE AND ORBIT CONTROL

Except for scaling in size, the Brayton system control concepts are identical to the concepts used for attitude and orbit control of the RTG spacecraft. The same apriori design assumptions have been made, namely that

- a) Spacecraft in high energy orbits (such as the synchronous equatorial orbit) will be inserted by means of a tug, and that therefore no auxiliary propulsion is required for orbit insertion. A 5-year supply of orbit maintenance propellant is provided.
- b) Spacecraft in low energy orbits (such as the 900 n.mi. near polar orbit) will receive the impulse to the transfer orbit from either the shuttle itself or from the tug. (Several spacecraft may be simultaneously boosted in this fashion). Each spacecraft may then circularize its orbit and make moderate inclination corrections

MAX. N_x LOADING

MAX. N_z LOADING

FIGURE 5-34 BRAYTON SPACECRAFT STRUCTURAL ANALYSIS LOADS

(1.5 degree). Such an insertion maneuver is made in the spinning mode, after which the spacecraft is despun for normal operation.

The angular momentum of the Brayton rotating unit (turbine-alternator-compressor) is relatively constant and too small to make a significant change in the spacecraft control characteristics. The moment of inertia (0.925×10^{-4} slug-ft²) at rated speed (800 rev/sec) results in a spin momentum of 0.465 lb-ft-sec, which is relatively small compared to the control wheel momenta considered for this vehicle (10-to-15 lb-ft-sec). Furthermore, the Brayton speed-control loop regulates speed rather tightly, within a 2% band. (The loop is designed so that a drop in spacecraft load slightly raises the speed of the BRU, causing parasitic load to be added at the rate of full load per 2% speed variation). Therefore, the greatest change - from full spacecraft load (zero parasitic load) to zero spacecraft load (full parasitic load) is accompanied by a momentum change of

$$0.02H_o = 0.93 \times 10^{-2} \text{ lb-ft-sec.}$$

which is well within the control range of the momentum wheels.

Therefore:

- a) The bias momentum of the Brayton machine is easily compensated with spin of the corresponding momentum wheel.
- b) The speed regulation of the system is well within the regulating capability of the wheel system
- c) There is no restriction on the orientation of the Brayton machine axis. Any orientation is acceptable, and none is preferred from an attitude-control viewpoint.

For details on the design of the attitude and orbit control subsystem for the Brayton spacecraft see Appendix B .

5.4.6 TT&C SUBSYSTEMS

The TT&C Subsystem for the Brayton spacecraft is the same as that of the RTG spacecraft described in section 5.2.3.

5.5 REACTOR CONFIGURATION

5.5.1 MISSION PROFILES

As reviewed in Section 3, the only unmanned NASA mission currently projected to require power in the multi-kilowatt range is a synchronous-orbit high-power ATS-type communications satellite. Designs of such a reactor-powered system have been studied in a previous program (Ref 5-3). In order to examine the versatility of the reactor system in a low-orbit mission of practical utility, a Side-Look-Radar (SLR) mapping mission, which appears practical at 5 kilowatts, was added.

It was not believed to be possible, given the priorities of the present study and the paucity of project mission requirements, to design and demonstrate cost-effectiveness of a multi-purpose "standard" spacecraft around the reactor. It was decided, rather, that a standard reactor should be investigated for applicability to the rather diverse requirements of the high power ATS and SLR missions.

The following parameters were established for a typical SLR mapper:⁺⁺

Table 5.10

PARAMETER	VALUE		
System Altitude, n.mi.	300 *		
Radar frequency	X-band		
Antenna length (slotted waveguide array) ft.	25		
height, ft.	2.6*		
Swath width mapped, n.mi.	45 → 33 } can do by		
Depression angle from horizontal, degrees	48 → 60 } vehicle roll		
Range Resolution, ft.	30	60	90
Prime power, kW(e)	5.8	3.2	1.8
Average transmitter power, kW(e)	1.33	0.67	.33
Transmitter weight, lb.	470	240	120
Transmitter volume, cu.ft.	4	2	2
Other equipment weight, lb.	370		

* Altitude and antenna height can be varied proportionally; all other parameters remaining unchanged.

⁺⁺ Information courtesy of Westinghouse Systems Development Division

Table 5.10 is based on the following assumptions:

1. Signal is transmitted over data link in real time; storage and processing are done on ground.
2. Prime power does not include data link or other housekeeping
3. No redundancy has been included.

5.5.2 MECHANICAL CONFIGURATION

Figure 5-35 shows the SLR mission in a reactor spacecraft. Part of the mission equipment and all of the housekeeping electronics are mounted in an eight-bay annular structure just aft of the reactor radiator; separated from the radiator by a thermal blanket. This is followed by a truss structure supporting the long radar antenna, and a space-viewing radiator for the major heat dissipating equipment, such as the radar transmitter, which is mounted near the mid-plane of the antenna. The entire structure will fit within the shuttle payload bay, without requiring in-orbit deployables.

5.5.3 THERMAL CONTROL

A group of heat-dissipating shunt resistors is mounted on the equipment radiator. When the transmitter is switched off, the power is switched onto the shunt resistors. The reactor is slightly reduced in output to stabilize the equipment radiator to its operating temperature condition. This maintains equipment temperature during periods of non-transmission and also helps to maintain dimensional stability of the antenna array.

As currently conceived, the housekeeping module would contain its own thermal control subsystem, possibly utilizing louvers, etc., similar to most current spacecraft. It is possible that this module could be thermally stabilized by a controlled heat bleed from the reactor radiator. However, time did not permit the evaluation of this possibility in this phase of the study.

5.5.4 ATTITUDE AND ORBIT CONTROL

For the communications satellite mission (Ref 5-3), the

PAYLOAD

ITEM	NAME
30	TRANSMITTER
31	MICROWAVE UNIT
32	RECEIVER
33	MONITOR
34	ECCM
35	SHUNT

FIGURE 5-35 SIDE LOOK RADAR MISSION ON REACTOR SPACECRAFT

control system will be similar to the system previously discussed. The configuration (large length/diameter ratio) dictated by the power plant should be utilized to take advantage of the gravity gradient torques; thus the longitudinal axis should be radial (with the reactor end pointing away from Earth). These gravity gradient torques will of course only supplement the active control system. As discussed in Section 5.2 this control system could be in the form of three reaction wheels (similar to the RTG- and Brayton-vehicle control schemes), or it could consist of two or three double-gimbal control moment gyros.

A major amount of propulsion is required for both the transfer orbit impulse and the apogee impulse, to insert 5000 to 6000 lb into geosynchronous orbit. An attractive concept might be the use of electric propulsion (Resistojet, ion or MHD thrusters), utilizing the reactor electrical output to achieve and maintain the desired orbit. Station keeping requirements will be identical to the previously discussed missions (East/West: maximum of 7 ft/sec/year; North-South: ¹²⁰~~150~~ ft/sec/yr requiring hydrazine propellant usage at the rate of 0.0222 lbs/lb of spacecraft/year), unless electric propulsion is used.

For the low altitude side-look-radar mission orbit achievement propulsion may not be required: the shuttle can place the vehicle directly into low-earth orbit. As before (low altitude near polar missions) provision for approximately 87 ft/sec impulse (.0123 lbs of hydrazine per pound of spacecraft) can be provided for fine orbit adjustment if required. The mission requirement of side-look-radar mapping with a 25-to-30 ft antenna (long dimension along the vehicle path) dictates a flight attitude with vehicle axis tangent to the path. Either a three-reaction-wheel configuration or a set of 2 or 3 double-gimbal control moment gyros should be employed to stabilize and control the vehicle, the detailed design depending primarily on the attitude stability requirement for the mapping mission.

Although the mission analysis established the basic spacecraft capabilities needed to accommodate economically the largest possible number of missions, application of the standard nuclear-powered spacecraft to a specific purpose requires adaptation by the user in order to reach an effective system design. This application survey is aimed at gaining an appreciation of adaptation problems and providing a technical basis for cost estimates. This involves knowledge of the characteristics of the payload and of the ancillary equipment, and acquaintance with the spacecraft operations.

A representative cross section of the community of potential users was therefore contacted, using the mission model reference documents as a starting point. In almost every case it was found that the specific complement of sensors for the mission was in a state of flux, and, in many cases, the individual sensor requirements were being changed radically. Under these circumstances, it was necessary to sample more near-term missions for which information is available in order to proceed. It is hoped thereby that the designers of future spacecraft would be exposed to the basic configurations implicit in the use of the standard spacecraft, and to the alternative approaches which these may suggest. For example, the elimination of large solar array appendages offers some inducement to take advantage of larger aperture antennas, thereby reducing the power requirement for communications relay or broadcast satellites. This use, in turn, will impose further constraints on the design of the standard spacecraft, since it may require a more precise pointing accuracy than has been provided. However, vernier gimbaling systems are beginning to come into prominence, as evidenced in currently-proceeding studies on the Tracking and Data Relay Satellite System. (TDRSS)

Figure 5-36 is a sketch of the TDRS payload mounted on RTG spacecraft with the antennas deployed. Two of the antennas, the K_u -band and S-band, are mechanically steered while the VHF antenna array is electronically steerable. The trend, as the TDRS study progresses, is to lean more towards systems of

FIGURE 5-36 TRACKING & DATA RELAY SATELLITE SYSTEM (TDRSS) DEPLOYED ON RTG S/C

of antenna elements. The advantage of eliminating deployed solar array panels, therefore, becomes more prominent as the problem of beam interference becomes more severe. Figure 5-37 is a sketch of the same payload stowed on the spacecraft for the launch and injection sequence.

The advent of a synchronous data relay service for low orbiting observation satellites presents another perturbation to the nuclear-powered spacecraft users and designers. As ground coverage rate and resolution element size requirements become more severe, the data transmission rate increases significantly. The data bandwidth pressure can be met either by resorting to higher and higher carrier frequencies, with attendant space losses leading to a requirement for greater transmitter power, or the data can be sent via space-relay link. The extension in communications contact time permitted by the space relay can be used to alleviate the power and bandwidth problem.

One example of this is shown in Figure 5-38 which illustrates the application of the standard Brayton spacecraft to an Earth Observation Satellite. This sketch shows a large multi-element narrow beam antenna required because of the high data transmission load. Introduction of the TDRS system, would probably eliminate the large antenna in favor of the smaller-aperture higher-frequency steerable dish shown on the top side of the spacecraft for sending data through the relay satellite. The absence of a deployed solar array permits this link to one of the TDRS satellites to be established at any time. Therefore, the nuclear-powered standard spacecraft may alleviate the ever-present growth in power level requirements. This in turn helps reduce the cost of the nuclear powered spacecraft, by reducing the required fuel inventory.

Synchronous-altitude applications missions also show an upward trend in their power requirement. Application of the standard Brayton spacecraft to one such system, the ATS-F experimental communications mission, results in the configuration shown in Figure 5-39. The stowed configuration is shown in Figure 5-40. Comparison of this configuration with the ATS-F satellite now nearing completion shows that the former has a significantly greater volumetric efficiency for stowage. This occurs because the solar array boom stowage, which

RTG (3 REQ'D)

4.7 FT. DIA.
K_u BAND HIGH GAIN
ANTENNA (STOWED)

13 FT. DIA.
VHF ANTENNA
(STOWED)

K_u BAND EARTH
COVERAGE HORN

132.8

4 25 #

4 25 # HYDRAZINE
THRUSTERS

S BAND
HIGH GAIN ANTENNA
(STOWED)

TT&C
OMNI-ANTENNA

FIGURE 5-37 ILLUSTRATIVE APPLICATION OF STANDARD RTG S/C:
STOWED CONFIGURATION OF TDRSS

FIGURE 5-38 ILLUSTRATIVE APPLICATION OF STANDARD BRAYTON S/C TO THE EOS MISSION

FIGURE 5-39

ILLUSTRATIVE APPLICATION OF STD. BRAYTON S/C: ATS MISSION IN DEPLOYED CONFIGURATION

FIGURE 5-40 ILLUSTRATIVE APPLICATION OF STANDARD BRAYTON S/C:
ATS MISSION IN STOWED CONFIGURATION

results in a large truss support structure, is eliminated. This also makes it practical to compact the feed system for ATS-F to the Cassegrainian type shown. One can hypothesize that, had the Brayton standard spacecraft been available at the time ATS-F was initiated, the program would have been more focused on its primary mission and the result would have been a simpler, less costly, spacecraft with a shorter delivery time. A large percentage of the development program would have been eliminated due to the much simpler structural and deployment concepts made possible by use of the nuclear-powered spacecraft.

It was found that the standard Brayton configuration provided an excellent platform for deployment of antenna structures. Both the low-orbiting earth-observation-mission-type with large directional multifeed elements stowed in a folded configuration, as well as the large-aperture parabolic dishes which must be unfurled, could be given a more ideal hard mount as opposed to the extended-structure mounting which is frequently employed today. The softer mounts can give rise to structural resonances and dynamic feedback problems during attitude control and station-keeping maneuvers, which are largely avoided with the nuclear-powered standard spacecraft.

An analysis of spacecraft packaging concepts has illustrated a reasonable match between the volumetric requirements for components and propulsion fuel storage on the one hand, and the RTG and Brayton spacecraft external envelope sizes on the other. Both thermal power dissipation levels and packaging volume analyses were performed to establish the envelope dimensions which were chosen. These choices were validated by the limited mission studies which were performed herein. An illustration of this is given by the 300-watt Advanced Tiros Meteorological Satellite configuration of Figure 5-41. The rather compact spacecraft can easily handle the thermal radiation requirements of the two RTG's and that of the included electronic equipment by utilizing both the earth-viewing surface and a portion of the opposite surface as equipment mounting area. The mounting of payload sensors

FIGURE 5-41

ILLUSTRATIVE APPLICATION OF STD RTG S/C: ADVANCED TIROS METEOROLOGICAL SATELLITE

on the current Tiros satellite calls for 12 square feet of earth viewing surface. The standardized spacecraft design affords an almost 50% increase in this important satellite feature.

A similar packaging advantage is maintained with the TDRS configuration, although it dissipates more power, utilizing three RTG units. Because it is at synchronous altitude its thermal-radiation ability is enhanced by reduction of earth albedo effects.

On the Earth Observation Satellite (Figure 5-38), over 40 square feet of sensor mounting area is available--approximately 100% margin in this capability-- even though a portion of the earth-viewing face is obscured by the multi-element beam antenna array. Internal packaging volume is more than ample and permits a later growth version to autonomous operation by providing both the power capability and packaging volume to install redundant general-purpose computers.

MASTER SCHEDULE

As a guide to lead-time requirements for operational development of the standard spacecraft, a master schedule has been prepared (Figure 5-42). Although this schedule was derived for the Brayton spacecraft specifically, it is believed to be representative of that required for the RTG spacecraft as well.

The schedule does not include development of the Brayton power plant itself, which is assumed to run concurrently with or prior to this development program.

References

- Ref. 5-1 Proceedings of the Symposium on Advanced Aircraft Electrical Systems, Naval Air Development Center, Warminster, PA, April 20-22, 1971.
- Ref. 5-2 NASA TND-5925, "Transit Characteristic of a Voltage Regulator and a Parasitic Speed Controller on a 14.3 Kilovolt-ampere 1200-Hertz Modified Lundell Alternator", Wimmer and Ingle, November 1970.
- Ref. 5-3 NASA CR-120873. "Applications Technology Satellite Advanced Missions Study", Final Report, Contract NAS 3-14360, Fairchild Industries. Appendix A.

PHASE

ENGINEERING

MANUFACTURING

TESTING

OPERATIONS

SPACECRAFT MASTER SCHEDULE (R.I.B. CONFIG)

FIGURE 5-42

6.0 GROUND HANDLING CONSIDERATIONS

Special ground-handling facilities are typically required for radio-isotope fueled systems. Such facilities must be available for launch as well as recovery of the systems. After the initiation of routine Shuttle operations, it can be anticipated that recovery of the entire spacecraft, or at the very least of the fuel capsules, will be programmed following the active period of low-orbit spacecraft. Such recovery would permit the use of the standard nuclear spacecraft in orbits which would be otherwise prohibited because of excessively short orbital lifetimes. This would also permit re-use of the costly nuclear fuel, as well as other subsystems of the spacecraft or the entire spacecraft itself.

The special ground-handling facilities must account for both thermal and nuclear radiation from the radioisotope units. Although the detailed design of such facilities is premature at this point, a general description of anticipated requirements is presented in this section, together with a description of the operations specific to each of the two radioisotope power systems which were incorporated in the standard spacecraft designs presented herein.

It should be noted that experience in launching radioisotope generators has been acquired in numerous flights since 1961, so that the requirements summarized in this section are not conjectural but reflect this experience. The use of nuclear-powered standard spacecraft can only serve to make these operations more routine and less particular than has been the case heretofore.

6.1 NUCLEAR FACILITY

A certified facility for the handling and storage of encapsulated nuclear materials must be provided. Access to this facility must be limited to persons under strict health physics control. The facility must be instrumented to detect the failure of the encapsulation of any radioactive material. It must be equipped with emergency gear and be designed to facilitate decontamination in the event of a nuclear incident.

The following equipment is typical of that required for this facility:

- a) Emergency secondary container for either an entire RTG or an MHW heat source.
- b) Anticontamination suits, including boots, gloves, coveralls, and hoods.
- c) Pressure demand air packs.
- d) Neutron and gamma survey instruments.
- e) Alpha air monitor operating with clean background level established.
- f) Alpha air monitor alarm set at twice clean background count level.
- g) Neutron and gamma detectors with ambient background levels established.
- h) Film badges and dosimeters.
- i) Suitable building with provision for pedestrian traffic control and locked exclusion area.
- j) Suitable dollies, transfer casks, and remote handling equipment.

6.2 HEAT SOURCE RECEIPT AND INSPECTION

The heat source (or fueled RTG) is received at the nuclear facility in a DOT-approved shipping cask. All personnel connected with the receiving inspection portion of the operation will be issued the necessary dosimeters.

An external dose rate survey of the truck and cask is performed upon arrival. This survey is compared with the shippers survey. All discrepancies shall be noted and evaluated. The approximate shape and strength of the neutron and gamma fields around a MHW heat source are shown in Figure 6-1 and Figure 6-2. The truck interior atmosphere is monitored for airborne alpha activity. The airborne activity shall not exceed twice ambient background. If twice ambient background is monitored, the truck shall be sealed and the area secured by the site health physicist until the contingency plans for disposition of the contaminated generator or heat source are executed.

FIGURE 6-1 APPROXIMATE NEUTRON FIELD AROUND A MHW SOURCE

FIGURE 6-2 APPROXIMATE GAMMA FIELD AROUND A MHW HEAT SOURCE

If the airborne alpha survey is within limits, the truck floor and shipping cask shall be monitored with portable alpha survey instruments. If no readings greater than 150 cpm/probe are found, smears shall be taken from the van floor and shipping cask. Alpha contamination shall be less than ~~150~~ 150 dpm/cm².

If all above restrictions are met, the shipping cask may be removed from the truck and transferred to the nuclear facility. The area monitors, exclusion area, personnel access control, and dosimeter programs must be initiated in the facility prior to this time.

6.3 POWER TEST OPERATION

Two different sets of test procedures are required for the RTG and Brayton systems. This will be necessary because the RTG will probably arrive at the site tested and producing power across a shorting bar, while the Brayton system will probably arrive unfueled. In general, power test operations will be performed at reduced levels with both units. Both systems will have heat losses in air which are higher than would be experienced in space. This will be particularly true for the MHW-RTG because of the gas-fill, described in Section 2. The Brayton system should be capable of near full-power testing because of the use of Min-K fibrous insulation in the power-plant bay. This insulation exhibits good performance in air, although not as good as in vacuum. As a result the full system design temperatures are not obtained during the ground checkout tests. Operation of a foil-insulated system in air would result in even lower temperatures. If full-power testing at the launch site is required, such as for recovered and refurbished units, the use of a vacuum chamber would probably be necessary.

6.3.1 RTG POWER OPERATION

Each generator will be connected to the test console by means of a cabling harness which mates with the flight hardware connectors and which permits testing for power output while operating on a dummy load. Fin and case temperatures would be monitored automatically at preselected rates and time intervals by the test console data acquisition system. Generator internal pressure can also be monitored, via the vent tube and valve, if this can be reclosed.

The test console would be equipped with alarms set to indicate operation outside allowable limits. The RTG's and the test console must be separated by sufficient distance and shielding to minimize the radiation dose received by the operators. Access to the RTG's and the test monitor area would be strictly controlled. Operation within the test area would require personnel nuclear radiation monitors. Personnel exposure in this area would be timed, and cumulative records kept of personnel exposure time. The RTG must not be left unattended in its unshielded condition for any reason.

Upon completion of the tests, the RTG would be checked for contamination using the receipt procedures, returned to the shipping cask and placed either in a storage/holding area or prepared for spacecraft mounting. If any contamination is found the area shall be secured and decontamination procedures initiated.

The storage/holding area must be guarded, shielded, and air-conditioned to the extent necessary for the number of heat sources to be stored.

6.3.2 BRAYTON POWER OPERATIONS

As previously indicated, the Brayton cycle system differs from the RTG in this connection in that it would most probably be shipped unfueled. The heat source would be received separately at the launch site nuclear facility and mated to the power conversion equipment for the power test. The fueling operation will take place in the controlled access test area under strict personnel control. Health physics tests (air monitors, swipes, etc) of the type used when the shipping cask was transferred from the delivery vehicle to the facility will be employed throughout the fueling operation. A health physicist would be in attendance during all transfer operations.

Actual fueling operations are carried out using specially designed handling tools. These tools will be used to remove the fuel canister from the shipping cask and index it for insertion into the heat exchanger. Special attention is given in the design of the tools to minimize the radiation dose received by the operator.

All subsystems within the power conversion module are checked for function in the nuclear test facility.

Dummy power, control, and telemetry cabling are used to connect the power unit to the test console. The module is then subjected to the static and dynamic electrical load tests required by the nuclear facility test procedure manuals.

Upon completion of the operational tests, the power module is defueled. The fuel capsule is returned to the shipping cask and prepared for delivery to the Vertical Assembly area, or placed in the secure storage area. The power module is checked for contamination before and after defueling using the health physics procedure described for initial receipt at the nuclear test facility. Although the probability of fuel capsule failure is small, each operation must be checked to assure that no radioactivity is released. In the event of the detection of radioactivity, the fuel capsule must be placed in a decontamination cask and the contaminated area must be secured. If the power module shows contamination after defueling, decontamination procedures must be immediately initiated. These procedures are designed to minimize the spread of airborne alpha activity and are required to protect site personnel from potential radiation hazards.

The defueled power module is cleared for transfer from the nuclear test facility to the Vertical Assembly area after being shown to be contamination free. Transfer can be accomplished at any time thereafter under the rules governing sensitive but non-radioactive equipment.

6.4 INSTALLATION OF HEAT SOURCE WITHIN VEHICLE

The safety procedures required for handling an RTG or a Brayton heat source are similar, although the basic handling equipment is different because of the different hard-point and orientation geometries. The functional requirements are intended to apply in general terms to both the RTG and Brayton systems.

The safety requirements of the nuclear facility are transferred to the shuttle assembly area just prior to loading the radiation sources. All emergency equipment (air packs, emergency radioactive containers, area monitor survey instruments, etc.) is transferred also and controlled access film badging and dosimeter procedures are instituted.

The heat sources are transferred in shielded containers from the nuclear facility to the Vertical Assembly area under base security guard. All personnel not connected with the sources are prohibited from the Vertical Assembly area. The final assembly and installation of the sources are performed as late as possible to minimize the impact of the presence of a radiation source on the preflight timeline.

The sources are removed from their shielded canisters and installed on the spacecraft using the specially designed handling tools. All operations during this period must be closely monitored by health physicists to guard launch personnel against excessive radiation dosage.

The radiation field (neutron and gamma) around the source is mapped to determine the exclusion area after installation on the spacecraft. All personnel entering this area must observe the film badge, dosimeter, and accumulated dose limitations. Smears will be taken from the source transfer cask and the handling tools, and checked for alpha contamination. If contamination is found, emergency action would be initiated.

6.5 PREFLIGHT CHECKOUT

After installation, each power unit is generally given a preflight operational check to establish that all system interfaces are functioning. The GSE cooling systems are activated to prevent overheating of the various enclosed areas which receive waste heat from the power module.

7.0 COST STUDIES

7.1 SUMMARY

Cost studies were carried out to compare standardized nuclear-powered spacecraft with mission-specialized (or "dedicated") solar-powered spacecraft. Attention was limited to ten missions from the NASA 1972 mission model which are now planned for the post 1979 period: five below 500 watts and five above 500 watts, i.e. in the RTG and Brayton system power ranges.

Launch costs, operational costs, and specialized mission equipment costs were not considered in this phase of the study. No detailed analysis was made for the cost increases for handling nuclear fuel, nor for reduction in launch and operational costs due to standardization. These two factors were assumed to offset each other. Mission equipment costs should be relatively unaffected by the nature of the spacecraft. In view of the relative imprecision of cost models, it was assumed that these particular costs are the same for both types of spacecraft, and attention was limited to the actual spacecraft costs and to the costs of the standard equipment provided for all missions.

The cost model was drawn from the section of the USAF Space Planners Guide dealing with space vehicle system costs. It divides the costs into non-recurring costs, involving development, test, and evaluation (DTE), facilities, and aerospace ground equipment (AGE), and recurring costs of production and operations. The operations costs are not included.

Within each category, costs are broken down by subsystems, and estimated from a set of curves which usually plot cost against weight. In most cases several different curves are furnished to cover different types of each particular subsystem. While the subsystem cost estimates may not be accurate, the total system costs do provide a reasonable reflection of existing system costs. The model is thus useful for quick estimates of total system costs during early stages of conceptual design.

Since the model was prepared in 1965, costs were first increased by 30 percent to reflect inflation. The non-recurring DTE costs were then decreased by 25 percent to reflect the savings due to the NASA subsystem standardization program. The estimated weight of the dedicated spacecraft was assumed to be less than that of the standardized designs, since the former could presumably be more efficient in weight utilization by virtue of addressing only one mission rather than providing

capability for a variety of missions. In the USAF cost models, this weight difference imposed a penalty on the standardized spacecraft, since all the spacecraft subsystem cost curves showed increasing cost with increasing weight.

Recurring costs were also estimated as a function of weight (or power, for the power subsystem only), with an initial unit cost modified by a set of "learning curves" to reflect the effects of the size of production runs. In general, the recurring costs for the standardized spacecraft are higher than those for the dedicated spacecraft, while the non-recurring costs are lower since most of the DTE spacecraft costs were eliminated by the use of a standard design.

Overall cost savings of approximately \$⁴²⁰~~440~~ million are estimated to be achievable over the limited group of ten NASA missions analyzed, by the use of the standard spacecraft. This amortizes the estimated development costs of the two standard spacecraft (approximately \$200 million) over very few such missions.

7.2 THE COST MODEL

The Cost Model used to develop estimates for the standardized spacecraft is contained in the USAF "Space Planners Guide", issued by the Air Force Systems Command in July 1965. This model is in wide use today by advanced mission planners. Although the material contained is not sufficiently accurate and detailed to be used for budget projection and administration, the cost prediction accuracy for conceptual systems, such as contained in this study, is as high as many computerized cost models now in existence.

Due to lack of standardized cost categorization and costing methods throughout the aerospace industry, less correlation should be expected between the cost data for detailed elements of the spacecraft system and quoted costs for existing equipment than between estimated total system or program costs and real program costs. While many individual hardware item quotations include prorated costs for Development, Test and Evaluation (DT & E), Aerospace Ground Support Equipment, Technical Data and other items, the Cost Model develops those costs individually.

A large portion, approximately 50-70% of a total program budget, is typically spent on non-hardware items, such as DT & E and supporting documentation. Although a detailed breakdown of these costs would be desirable from a cost analysis standpoint, concept evaluation (overall potential cost-effectiveness), not analysis, is the object of this study. The Guide has been produced with this as the prime objective and should, therefore, prove effective in projecting system costs for the standardized spacecraft and especially cost comparison figures between these and dedicated spacecraft.

7.3 INPUTS TO THE COST MODEL

The performance and weight inputs to the Cost Model were developed in Section 5 of this report for the standardized spacecraft. For comparison with dedicated spacecraft, the NASA Payloads Study (1972 Mission Model) was used to obtain performance level and weight estimates to the subsystem level.

For each of the cost input areas, the calculations given in Section 7.4 reference that part of the cost model from which the costs were obtained.

The following assumptions were made in order to use the model and enhance its accuracy:

1. The Air Force cost models (AFCM) are valid,
2. The NASA Mission Model weight and power estimated are valid inputs to the AFCM,
3. Since the AFCM are in terms of 1965 dollars, a 30% inflation factor is added to the costs derived from them,
4. The costs of the specialized mission equipment are assumed to be essentially the same for both the bus and dedicated spacecraft approach.

Therefore, the mission equipment costs are not included in these cost analyses.

5. There will be added acceptance test costs associated with the integration of each individual flight payload to the standard spacecraft. This is because the individual payload bus interface has not achieved prior qualification. This cost is estimated as 1-million dollars for RTG missions and 1.5 million dollars for Brayton missions.
6. The one-time system costs (AGE, systems integration, and tooling) are based on dry weight of the spacecraft less the weight of the mission equipment for both the dedicated and standard approaches.
7. For the dedicated spacecraft approach, the DT & E costs derived from the AFCM are decreased by 25% in order to account for the cost advantage of using off-the-shelf hardware. This factor is not applied to the standard bus DT & E costs since the hardware there will undergo essentially new design, development and qualification.
8. In the area of non-recurring costs of the nuclear power systems, only the costs of integration are considered. The non-recurring costs for developing the nuclear power subsystems are assumed to be borne elsewhere. A one-time non-recurring cost for integration of the developed power subsystem with the standard spacecraft is estimated at 6-million dollars for the Brayton system and 3 million dollars for the RTG system.
9. Based on AEC-supplied data, recurring cost of the hardware for the power subsystems is estimated at \$650K for each 150-W RTG unit and \$360K for the Brayton hardware. Recurring fuel costs are estimated \$1,900K per 2400-W fuel capsule.

10. The AFCM has all environmental control subsystems in one group. The thermal control subsystem for unmanned spacecraft is not typical of the environmental control subsystem, since these include all pressurization and temperature control systems for complex manned missions. Since the thermal control subsystem for an unmanned, unpressurized spacecraft usually has about the same cost-to-weight ratio as a structure and mechanism subsystem, the same curve was used for estimating the cost for both the structure and thermal control subsystems.
11. With respect to propulsion subsystem costs, it is assumed that the dedicated spacecraft will be launched the same way (from circular low altitude shuttle orbit) as the standard spacecraft. Thus the costs of the propulsion systems will be essentially the same in both cases. Non-recurring and recurring costs are assumed to be as follows (in 1965 dollars):

	<u>Non-Recurring</u> \$	<u>Recurring</u> \$
Spin	2,000 K	200 K
3-axis	3,000 K	300 K

The differences are based on the number of catalytic thrusters in either system, the 3-axis system requiring generally 12 more thrusters than the spinning system for full redundancy. These numbers are not part of the AFCM because monopropellant N_2H_4 systems of the current type had not then been developed. The costs of current auxiliary propulsion systems are not very dependent on fuel loading but rather on the number of thrusters.

12. All recurring hardware costs are assumed to be first-unit costs and a learning factor based on a 95% learning curve is applied for more than one unit. This means that the cost of the n^{th} unit is $(0.95)^n$ of the cost of the first unit.

7.4 COST CALCULATIONS AND EVALUATIONS

7.4.1 RTG STANDARD SPACECRAFT

Based on the above assumptions, the cost of developing and manufacturing the RTG standard spacecraft is shown in Table 7.1. Based on these numbers, and on the power system and integration costs described in Section 7.3, the recurring costs of performing the five selected missions on the RTG "bus" is shown in Table 7.2. The non-recurring costs of these missions on the standard spacecraft are shown in Table 7.3. The missions were selected on the basis of power, weight, and environmental compatibility. Only those flights scheduled for the 1978-1990 time frame were counted in the number of flights.

The costs of these five missions performed on dedicated solar powered spacecraft have been calculated, as shown in Tables 7.4 through 7.8. It can be seen that for the TDRS mission the dedicated spacecraft approach is less costly than performing this mission on the standard RTG bus, mainly because of the high recurring costs of the RTG power systems. Therefore, this mission was removed to the Brayton bus, where it is more cost-effective.

Table 7.9 summarizes the results of these calculations, showing a savings of 64% resulting from the availability and use of a standard spacecraft. A net savings of 27% is achieved if the cost of development of the bus is amortized over these four missions only.

7.4.2 BRAYTON STANDARD SPACECRAFT

The costs of developing and manufacturing the Brayton bus are shown in Table 7.10. The recurring and non-recurring costs of the five selected missions performed on this bus are shown in Tables 7.11 and 7.12, respectively. The calculated costs of performing these missions on dedicated solar-powered spacecraft are shown in Tables 7.13 through 7.17.

These calculations are summarized in Table 7.18, including the TDRS. This shows a savings of 43% or almost 275⁹-million dollars from the use of the standard spacecraft. A net savings of 27% is achieved if the cost of development of the Brayton bus is amortized over these six missions only.

TABLE 7.1

RTG SPACECRAFT COST

	SUBSYSTEM	WEIGHT	COST MODEL FIG. NO.	DT & E COST (Thousands of Dollars)	COST MODEL FIG NO.	FIRST UNIT COST (Thousands of Dollars)	
	STRUCTURE	215	VII B-11	7,500	VII B-24	500	
	THERMAL CONTROL	78	VII B-11	5,000	VII B-24	150	
	GUIDANCE & CON- TROL	149	VII B-6	11,000	VII B-20	450	
	AUX PROPULSION (DRY)	25		3,000	VII B-25	300	
	FUEL	372					
	TT & C	79	VII B-2	8,000	VII B-15	500	
	ELECTRICAL POWER (INCL. 3 RTGs)	320		3,000			
	TOTAL WT (EXCL. MISSION EQPT)	1238					
	LESS FUEL	372					
	DRY WEIGHT	866					
	TOOLING			3,800			
	SYSTEMS INTEGRA- TION			7,000			
	AGE			14,000			
	BASIC NON-RECURRING COST			62,300			
	+ 30% INFLATION FACTOR			18,690			
	TOTAL NON RECURRING COST			80,990			
	BASIC FIRST UNIT COST (EXCLUDING POWER)					1900	
	+ 30% INFLATION					570	
	TOTAL FIRST UNIT COST (EXCLUDING POWER)					2470	
	X LEARNING FACTOR (17 FLIGHTS - 95% LEARNING CURVE)					x .82	
	AVERAGE RECURRING COST (EXCLUDING POWER)					2,025	
	+ SYSTEM ACCEPTANCE TEST					1,000	
				BASIC BUS COST		3,025	
	POWER SUBSYSTEM COST						
	FIRST UNIT COST (HDWRE)					650	
	X LEARNING FACTOR (30 UNITS - 95% LEARNING CURVE)					x .78	
	AVE RECURRING COST (HDWRE)					507	
	+ FUEL COST					1,900	
	AVE RECURRING COST PER 150 W RTG UNIT					2,407	
					THREE UNITS	7,221	
					TWO UNITS	4,814	

TABLE 7.2

RECURRING COSTS WITH RTG SPACECRAFT

[illegible]

TABLE 7.3

NON-RECURRING MISSION COSTS WITH RTG SPACECRAFT

[illegible]

TABLE 7.4

UPPER ATMOS. EXPLORER (NP2-13) MISSION COSTS USING DEDICATED S/C

SUBSYSTEM	WEIGHT	COST (THOUSANDS OF DOLLARS)			
		DT & E		FIRST UNIT	
STRUCTURE	240	8,000		500	
THERMAL CONTROL	30	3,000		130	
GUIDANCE & CONTROL	80 (SPIN)	10,000		300	
AUX PROPULSION (DRY)	70	2,000		200	
FUEL	450				
TT & G	100	9,100		600	
ELECTRICAL POWER	90 (100W)	3,000		500	
MISSION EQPT	100			60 (ACCEPTANCE TEST)	
TOTAL WT	1160				
LESS MISSION EQPT & FUEL	550				
"DRY" WT	610				
TOOLING		2,900			
SYSTEMS INTEGRATION		5,800			
AGE		11,000			
BASIC NON RECURRING COST		54,800			
PLUS 30% INFLATION FACTOR		16,440			
SUBTOTAL		71,240			
LESS 25% QUALIFICATION FACTOR		17,810			
TOTAL NONRECURRING COST		53,430			
BASIC FIRST UNIT COST				2,290	
PLUS 30% INFLATION FACTOR				687	
FIRST UNIT COST TOTAL				2,977	
X LEARNING FACTOR (6 FLIGHTS - 95% LEARNING CURVE)				x .87	
AVERAGE RECURRING COST				2,590	
X NUMBER OF FLIGHTS				x 6	
TOTAL RECURRING COSTS				15,540	

TABLE 7.5

SYNC MET SAT (NE 2-41) MISSION - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	125		6,000		350	
	THERMAL CONTROL	20		2,700		10	
	GUIDANCE & CONTROL	10 (SPIN)		5,800		100	
	AUX PROPULSION (DRY)	25		2,000		200	
	FUEL	35					
	TT & G	40		4,500		250	
	ELECTRICAL POWER	120 (150 W)		3,800		550	
	MISSION EQPT	160				50 (ACCEPTANCE TEST)	
	TOTAL WT	535					
	LESS MISSION EQPT & FUEL	195					
	"DRY" WT	340					
	TOOLING			1,900			
	SYSTEMS INTEGRATION			3,800			
	AGE			7,600			
	BASIC NON RECURRING COST			38,100			
	PLUS 30% INFLATION FACTOR			11,430			
	SUBTOTAL			49,530			
	LESS 25% QUALIFICATION FACTOR			12,383			
	TOTAL NONRECURRING COST			37,147			
	BASIC FIRST UNIT COST					1,510	
	PLUS 30% INFLATION FACTOR					453	
	FIRST UNIT COST TOTAL					1,963	
	X LEARNING FACTOR (2 FLIGHTS - 95% LEARNING CURVE)					x .95	
	AVERAGE RECURRING COST					1,865	
	X NUMBER OF FLIGHTS					x 2	
	TOTAL RECURRING COSTS					3,730	

TABLE 7.6

TIROS (NE 2-40) MISSION - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	300		8,500		600	
	THERMAL CONTROL	--					
	GUIDANCE & CONTROL	60 (3-AXIS)		9,000		250	
	AUX PROPULSION (DRY)	50		3,000		300	
	FUEL	90					
	TT & G	50		5,500		300	
	ELECTRICAL POWER	250 (250 W)		4,500		1,300	
	MISSION EQPT	580					
						60 (ACCEPTANCE TEST)	
	TOTAL WT	1380					
	LESS MISSION EQPT & FUEL	670					
	"DRY" WT	710					
	TOOLING			3,100			
	SYSTEMS INTEGRATION			6,200			
	AGE			12,000			
	BASIC NON RECURRING COST			51,800			
	PLUS 30% INFLATION FACTOR			15,540			
	SUBTOTAL			67,340			
	LESS 25% QUALIFICATION FACTOR			16,835			
	TOTAL NONRECURRING COST			50,505			
	BASIC FIRST UNIT COST					2,810	
	PLUS 30% INFLATION FACTOR					843	
	FIRST UNIT COST TOTAL					3,653	
	X LEARNING FACTOR (1 FLIGHT-95% LEARNING CURVE)					x 1.0	
	AVERAGE RECURRING COST					3,653	
	X NUMBER OF FLIGHTS					x 1	
	TOTAL RECURRING COSTS					3,653	

TABLE 7.7

GEOPAUSE (NE 2-45) MISSION -- COSTS USING DEDICATED S/C

SUBSYSTEM	WT	COST (THOUSANDS OF DOLLARS)				FIRST UNIT	
			DT & E				
STRUCTURE	200		7,300			450	
THERMAL CONTROL	15		2,300			10	
GUIDANCE & CONTROL	100 (3-AXIS)		10,500			350	
AUX PROPULSION (DRY)	--						
FUEL	--						
TT & G	70		6,800			420	
ELECTRICAL POWER	200 (125 W)		3,200			1,000	
MISSION EQPT	125						
						60 (ACCEPTANCE TEST)	
TOTAL WT	710						
LESS MISSION EQPT & FUEL	125						
"DRY" WT	585						
TOOLING			2,800				
SYSTEMS INTEGRATION			5,800				
AGE			11,000				
BASIC NON RECURRING COST			49,700				
PLUS 30% INFLATION FACTOR			14,910				
SUBTOTAL			64,610				
LESS 25% QUALIFICATION FACTOR			16,153				
TOTAL NONRECURRING COST			48,457				
BASIC FIRST UNIT COST						2,290	
PLUS 30% INFLATION FACTOR						687	
FIRST UNIT COST TOTAL						2,977	
X LEARNING FACTOR (2 FLIGHTS - 95% LEARNING CURVE)						x .95	
AVERAGE RECURRING COST						2,828	
X NUMBER OF FLIGHTS						x 2	
TOTAL RECURRING COSTS						5,656	

TABLE 7.8

TDRS MISSION (NC2-49) - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT (LBS)	COST (THOUSANDS OF DOLLARS)			
			DT & E		FIRST UNIT	
	STRUCTURE	155	6,500		350	
	THERMAL CONTROL	7	1,700		10	
	GUIDANCE & CONTROL	34 (SPIN)	7,500		170	
	AUX PROPULSION (DRY)	30	2,000		200	
	FUEL	70				
	TT & G	--	--		--	
	ELECTRICAL POWER	200 (430W)	6,000		800	
	MISSION EQPT	219				
	TOTAL WT	715				50 (ACCEPTANCE TEST)
	LESS MISSION EQPT & FUEL	289				
	"DRY" WT	426				
	TOOLING		2,200			
	SYSTEMS INTEGRATION		4,500			
	AGE		9,000			
	BASIC NON RECURRING COST		39,400			
	PLUS 30% INFLATION FACTOR		11,820			
	SUBTOTAL		51,220			
	LESS 25% QUALIFICATION FACTOR		12,805			
	TOTAL NONRECURRING COST		38,415			
	BASIC FIRST UNIT COST				1,580	
	PLUS 30% INFLATION FACTOR				474	
	FIRST UNIT COST TOTAL				2,054	
	X LEARNING FACTOR (6 FLIGHTS - 95% LEARNING CURVE)				x .87	
	AVERAGE RECURRING COST				1,787	
	X NUMBER OF FLIGHTS				x 6	
	TOTAL RECURRING COSTS				10,722	

TABLE 7.9

COST COMPARISON FOR FOUR LOW-POWER MISSIONS*
STANDARDIZED RTG S/C VERSUS SPECIALIZED SOLAR-POWERED S/C
In Millions of Dollars**

MISSION	Upper Atmos. Explor.	Synch. Meteor. Sat.	TIROS	Geopause	TOTAL
FLIGHTS	6	2	1	2	11
STANDARD S/C (Nuclear)					
Non-Recurring	1.1	1.8	5.9	7.1	15.9
Recurring	<u>32.6</u>	<u>10.9</u>	<u>7.8</u>	<u>10.9</u>	<u>62.2</u>
TOTAL	33.7	12.7	13.7	18.0	78.1
SPECIALIZED S/C (Solar)					
Non-Recurring	53.4	37.1	50.5	48.5	189.5
Recurring	<u>15.5</u>	<u>3.7</u>	<u>3.7</u>	<u>5.7</u>	<u>28.6</u>
TOTAL	68.9	40.8	54.2	54.2	218.1
RTG S/C SAVINGS	² 35.4	¹ 28.4	^{40.5} 42.0	^{36.2} 38.0	140.0
	51%	⁶⁹ 70 %	⁷⁵ 77 %	⁶⁷ 70 %	64%
COST OF DEVELOPING STANDARD S/C					81.0
NET SAVINGS, After Full Amortization Over These Four Missions of the Standard S/C Development Cost					59.0 27%

* Costs Include: Standard S/C Subsystems (Structure, Thermal, Power, Attitude Control, Telemetry, Command), AGE, and DTE

Costs Exclude: Specialized Mission Equipment, Launch, and Operation

** Based on USAF Space Planners Guide Cost Formula, Plus 30% for Inflation, Component and Subsystem Standardization Assumed to Reduce DTE Costs by 25% for specialized S/C only.

TABLE 7.10

TOTAL COST - BRAYTON SPACECRAFT

	SUBSYSTEM	WEIGHT (LBS)		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	439		10,500		800	
	THERMAL CONTROL	122		6,000		320	
	GUIDANCE & CONTROL	166		11,000		500	
	AUX PROPULSION (DRY)	32		3,000		300	
	FUEL	560					
	TT & C	79		8,000		500	
	ELECTRICAL POWER	705		6,000		---	
	TOTAL WT (EXC MISSION EQPT)	2103					
	LESS FUEL	560					
	DRY WEIGHT	1543					
	TOOLING			5,500			
	SYSTEMS INTEGRATION			11,000			
	AGE			20,000			
	BASIC NON RECURRING COST			81,000			
	PLUS 30% INFLATION FACTOR			24,300			
	TOTAL NON RECURRING COST			105,300			
	BASIC FIRST UNIT COST (EXCLUDING POWER)					2,420	
	+ 30% INFLATION FACTOR					726	
	SUBTOTAL EXCLUDING POWER					3,146	
	PLUS POWER FIRST UNIT COST (HARDWARE EXC FUEL)					360	
	TOTAL FIRST UNIT COST (EXCLUDING FUEL)					3,610	
	X LEARNING FACTOR (30 FLIGHTS - 95% LEARNING CURVE)					.78	
	AVERAGE RECURRING COST (EXC FUEL)					2,816	
	+ SYSTEM ACCEPTANCE TEST					1,500	
	BASIC BUS COST					4,316	

TABLE 7.11

RECURRING COSTS FOR BRAYTON SPACECRAFT
(Thousands of Dollars)

[illegible]

TABLE 7.12

NON-RECURRING MISSION COSTS WITH BRAYTON SPACECRAFT

[illegible]

TABLE 7.13

SYSTEM TEST SATELLITE (NC 2-51) MISSION - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT (LBS)		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	1070		16,000		1,500	
	THERMAL CONTROL	130		6,000		350	
	GUIDANCE & CONTROL	345 (3-AXIS)		14,000		850	
	AUX PROPULSION (DRY)	100		3,000		300	
	FUEL	210					
	TT & G	200		15,000		1,100	
	ELECTRICAL POWER	490 (600 W)		6,500		1,900	
	MISSION EQPT	315					
	TOTAL WT	2,860				80 (ACCEPT- ANCE TEST)	
	LESS MISSION EQPT & FUEL	525					
	"DRY" WT	2,235					
	TOOLING			7,000			
	SYSTEMS INTEGRATION			14,000			
	AGE			25,000			
	BASIC NON RECURRING COST			106,500			
	PLUS 30% INFLATION FACTOR			31,950			
	SUBTOTAL			138,450			
	LESS 25% QUALIFICATION FACTOR			34,613			
	TOTAL NONRECURRING COST			103,837			
	BASIC FIRST UNIT COST					6,080	
	PLUS 30% INFLATION FACTOR					1,824	
	FIRST UNIT COST TOTAL					7,904	
	X LEARNING FACTOR (8 FLIGHTS - 87% LEARNING CURVE)					x .86	
	AVERAGE RECURRING COST					6,797	
	X NUMBER OF FLIGHTS					x 8	
	TOTAL RECURRING COSTS					54,376	

TABLE 7.14

EOS (N E2-38) MISSION - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	400		10,000		800	
	THERMAL CONTROL	--					
	GUIDANCE & CONTROL	150 (3-AXIS)		11,000		450	
	AUX PROPULSION (DRY)	20		3,000		300	
	FUEL	40					
	TT & G	180		14,000		1,000	
	ELECTRICAL POWER	700 (1200 W)		9,000		2,200	
	MISSION EQPT	910					
						70 (ACCEPTANCE TEST)	
	TOTAL WT	2,400					
	LESS MISSION EQPT & FUEL	950					
	"DRY" WT	1,450					
	TOOLING			5,500			
	SYSTEMS INTEGRATION			10,000			
	AGE			19,000			
	BASIC NON RECURRING COST			81,500			
	PLUS 30% INFLATION FACTOR			24,450			
	SUBTOTAL			105,950			
	LESS 25% QUALIFICATION FACTOR			26,488			
	TOTAL NON RECURRING COST			79,462			
	BASIC FIRST UNIT COST					4,820	
	PLUS 30% INFLATION FACTOR					1,446	
FIRST	FIRST UNIT COST TOTAL					6,266	
	X LEARNING FACTOR (7 FLIGHTS - 95% LEARNING CURVE)					x .86	
	AVERAGE RECURRING COST					5,389	
	X NUMBER OF FLIGHTS					x 7	
	TOTAL RECURRING COSTS					37,723	

TABLE 7.15

SEOS (NE2-39) MISSION - COSTS USING DEDICATED S/C

	SUBSYSTEM	WEIGHT (LBS)		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	350		9,000		700	
	THERMAL CONTROL	20		2,700		10	
	GUIDANCE & CONTROL	65 (3-AXIS)		9,000		250	
	AUX PROPULSION (DRY)	40		3,000		300	
	FUEL	80					
	TT & G	95		10,500		580	
	ELECTRICAL POWER	950 (1500 W)		10,500		2,500	
	MISSION EQPT	900					
	TOTAL WT	2500				70 (ACCEPTANCE	
	LESS MISSION EQPT & FUEL	980				TEST)	
	"DRY" WT	1520					
	TOOLING			6,000			
	SYSTEMS INTEGRATION			11,000			
	AGE			20,000			
	BASIC NON RECURRING COST			80,700			
	PLUS 30% INFLATION FACTOR			24,210			
	SUBTOTAL			104,910			
	LESS 25% QUALIFICATION FACTOR			26,228			
	TOTAL NONRECURRING COST			78,682			
	BASIC FIRST UNIT COST					4,410	
	PLUS 30% INFLATION FACTOR					1,323	
	FIRST UNIT COST TOTAL					5,733	
	X LEARNING FACTOR (5 FLIGHTS - 95% LEARNING CURVE)					x .89	
	AVERAGE RECURRING COST					5,102	
	X NUMBER OF FLIGHTS					x 5	
	TOTAL RECURRING COSTS					25,510	

TABLE 7.16

ERS (NE 2-42) MISSION - DEDICATED VEHICLE COST

	SUBSYSTEM	WEIGHT'		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	400		10,000		800	
	THERMAL CONTROL	--					
	GUIDANCE & CONTROL	150 (3-AXIS)		11,000		450	
	AUX PROPULSION (DRY)	20		3,000		300	
	FUEL	40					
	TT & G ELECTRICAL	180		14,000		1,000	
	POWER	700 (1200 W)		9,000		2,200	
	MISSION EQPT	310					
	TOTAL WT	1800				70 (ACCEPTANCE TEST)	
	LESS MISSION EQPT & FUEL	350					
	"DRY" WT	1450					
	TOOLING			5,500			
	SYSTEMS INTEGRATION			10,000			
	AGE			19,000			
	BASIC NON RECURRING COST			81,500			
	PLUS 30% INFLATION FACTOR			24,450			
	SUBTOTAL			105,950			
	LESS 25% QUALIFICATION FACTOR			26,488			
	TOTAL NONRECURRING COST			79,462			
	BASIC FIRST UNIT COST					4,820	
	PLUS 30% INFLATION FACTOR					1,446	
	FIRST UNIT COST TOTAL					6,266	
	X LEARNING FACTOR (8 FLIGHTS - 95% LEARNING CURVE)					x .86	
	AVERAGE RECURRING COST					5,389	
	X NUMBER OF FLIGHTS					x 8	
	TOTAL RECURRING COSTS					43,112	

TABLE 7.17

DISASTER WARNING (NC 2-50) MISSION - DEDICATED VEHICLE COST

	SUBSYSTEM	WEIGHT		COST (THOUSANDS OF DOLLARS)			
				DT & E		FIRST UNIT	
	STRUCTURE	370		9,200		750	
	THERMAL CONTROL	50		4,000		150	
	GUIDANCE & CONTROL	120 (3-AXIS)		10,500		400	
	AUX PROPULSION (DRY)	60		3,000		300	
	FUEL	120					
	TT & G	120		10,000		700	
	ELECTRICAL POWER	420 (900 W)		8,000		2,100	
	MISSION EQPT	500					
						60 (ACCEPTANCE TEST)	
	TOTAL WT	1,760					
	LESS MISSION EQPT & FUEL	620					
	"DRY" WT	1,140					
	TOOLING			4,500			
	SYSTEMS INTEGRATION			9,000			
	AGE			18,000			
	BASIC NON RECURRING COST			76,200			
	PLUS 30 % INFLATION FACTOR			22,860			
	SUBTOTAL			99,060			
	LESS 25% QUALIFICATION FACTOR			24,765			
	TOTAL NONRECURRING COST			74,295			
	BASIC FIRST UNIT COST					4,460	
	PLUS 30% INFLATION FACTOR					1,338	
	FIRST UNIT COST TOTAL					5,798	
	X LEARNING FACTOR (2 FLIGHTS - 95% LEARNING CURVE)					x .95	
	AVERAGE RECURRING COST					5,508	
	X NUMBER OF FLIGHTS					x 2	
	TOTAL RECURRING COSTS					11,016	

TABLE 7.18

COST COMPARISON FOR SIX MEDIUM-POWER MISSIONS*
STANDARDIZED BRAYTON S/C VERSUS SPECIALIZED SOLAR-POWERED S/C
In Millions of Dollars**

MISSION	EOS	SEOS	ERS	Disaster Warning Satellite	System Test Sat.	TDRS	TOTAL
FLIGHTS	7	5	8	2	8	6	36
STANDARD S/C (Nuclear)							
Non-Recurring	25.2	25.0	8.2	13.9	8.3	2.2	82.8
Recurring	56.8	50.1	64.9	16.2	49.7	37.2	274.9
TOTAL	82.0	75.1	73.1	30.1	58.0	39.4	357.7
SPECIALIZED S/C (Solar)							
Non-Recurring	79.5	78.7	79.4	74.3	103.8	38.4	454.1
Recurring	37.7	25.5	43.1	11.0	54.4	10.7	182.4
TOTAL	117.2	104.2	122.5	85.3	158.2	49.1	636.5
BRAYTON S/C SAVINGS	53.8 35.2 33 46%	33.9 29.1 28 33%	37.6 49.4 40 31%	62.6 55.2 65 73%	97.4 100.2 64 61%	9.7 20%	274.8 43%
COST OF DEVELOPING STANDARD S/C							105.0
NET SAVINGS, After Full Amortization Over These Six Missions of the Standard S/C Development Cost							169.8 173.8 27%

* Costs Include: Standard S/C Subsystems (Structure, Thermal, Power, Attitude Control, Telemetry, Command). AGE, and DTE

Costs Exclude: Specialized Mission Equipment, Launch, and Operation

** Based on USAF Space Planners Guide Cost Formula, Plus 30% for Inflation, Component and Subsystem Standardization Assumed to Reduce DTE Costs by 25% for specialized S/C only.

8. RECOMMENDATIONS FOR FUTURE EFFORT

This program was intended and conducted as a feasibility study. Although the designs which have been described represent feasible approaches, none of the concepts has been optimized or sufficiently iterated to be able to claim to represent the best solutions to the various design problems which are posed by spacecraft standardization. The following areas of further analysis and experimental verification are recommended in order to carry the concepts to the point of detailed design and development.

8.1 RECOMMENDED ANALYTICAL & DESIGN EFFORTS

8.1.1 TEST MISSION ANALYSES

To a major extent, the overall usefulness and the technical and economic feasibility of a standard spacecraft will hinge on its ability to actually perform most of the missions in its power and weight range without imposing significant constraints on the mission plans. This can only properly be evaluated by performing detailed studies on the integration of various specific missions with the standard spacecraft. Unfortunately, in view of the long lead-time involved, most of these missions are only defined superficially at this point, so that this task must include some detailed mission modelling as a precursor. This was done to a limited extent in the course of this preliminary study, as reported in section 5.6. However, further, more detailed effort along these lines, is called for.

8.1.2 THERMAL DESIGN

Further study is required to evaluate the proposed thermal control solutions in detail. In the RTG spacecraft, a temperature mapping of the spacecraft interior is required with one, two or three RTG's mounted, with typical equipment inventories and worst-case orbits. Locations and effects of parasitic loads should be determined. Effects of direct RTG radiation on externally-mounted mission equipment, e.g. antennas, should be evaluated.

For the Brayton spacecraft, ~~alt~~er native thermal control solutions can be investigated in greater detail. The possible use of a non-isothermal radiator and its effect on spacecraft size and powerplant efficiency should be evaluated.

For both spacecraft, a more detailed consideration of the thermal interaction of the radioisotope heat sources with the Shuttle vehicle during the assembly, prelaunch and launch phases is required.

8.1.3 MECHANICAL DESIGN

The overall spacecraft configuration is dictated to a large extent by the thermal requirements and by the requirements for integration with the orbiter, tug, and/or pallet. As the thermal requirements are better defined, and as the vehicle interface requirements become better known, the mechanical design can proceed to a more definitive phase. Current NASA-sponsored studies on payload-orbiter/tug interfaces are likely to result in the design of standard pallets or other mounting/deployment devices for multiple-payload launches. These would provide the mechanical interface constraints to the standard spacecraft, and would permit proceeding with an up-dated calculation of launch loads on the spacecraft and included equipment.

8.1.4 COMMAND AND CONTROL SYSTEMS

Although the telemetry systems as described in section 5.2.3 appear to be fairly straightforward, the command and control functions require further consideration in any future effort. In particular, the design-investigation of a flexible on-board computer system capable of handling the large variety of potential missions, as well as the standard housekeeping functions, appears warranted. Ideally, the different missions and their specific equipments should interface with the command and control computer in a simple way without requiring major engineering effort. However, this can only be achieved, if at all, by transferring the engineering effort to the spacecraft design phase, i. e., by designing a centralized command/control system at the start, with sufficient versatility and sophistication to satisfy the majority of anticipated mission requirements.

Alternatively, following further study, it might be recommended that the command and control function be de-centralized, so that each mission design would provide its own control-logic system to interface with the on-board telemetry system.

8.1.5 DOD MISSIONS

Future efforts should include an assessment of the applicability and economic benefit of the standard spacecraft to future Defense Department missions, assuming such mission planning information is made available to the study team.

8.1.6 ECONOMIC ANALYSIS

Further economic analysis should be based on detailed subsystem cost estimates rather than on correlative models. However, such an analysis could not be done for the dedicated spacecraft without actually performing the detailed design of these spacecraft, so that comparative costs would still only be estimated from a correlative model. Nevertheless, such cost models tend to be more valid for dedicated spacecraft from which they have been generally derived. The projected cost of the standard spacecraft, however, and its recurring mission costs, can be more directly determined. In addition, any further economic analysis should take into account the results of the NASA efforts on cost-reduction of space payloads.

8.2 RECOMMENDED EXPERIMENTAL PROGRAMS

The experimental requirements at this point center mainly about the thermal control concepts. A full-size working model of the thermal shutter system of the RTG spacecraft should be built and tested for operation, effectiveness, and reliability.

For the Brayton spacecraft, a test of the counterflow radiator concept is called for, to verify its operating temperature profile.

At a later period, during the spacecraft development phase, both of these thermal-control concepts must be tested for service-operation with full-size spacecraft mock-ups in an exhaustive series of thermal-vacuum tests. Once verified in this manner, however, thermal-vacuum tests for most individual missions should no longer be necessary.

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APPENDIX A
PRELIMINARY STRUCTURAL ANALYSIS

A.1 RTG Spacecraft Structural Sizing

• Access Panels

The access panels were analyzed for their ability to withstand the 145 db acoustic environment. The equivalent pressure for 145 db is

$$p = 5.4 \text{ psi}$$

A basic 36" x 42" panel was divided into 6 panels by using stiffeners, resulting in a panel size of 12" x 21" for analysis. For a honeycomb panel supported on 4 edges, the maximum stress is

$$f_{\max} = \beta \frac{p b^2}{6h t_s}$$

For ultimate stress levels of 64,000 psi and 1/4" core, a face sheet thickness of $t_s = 0.010$ " is required. The stiffeners can be eliminated if the core depth and face sheets are increased without any significant weight change.

• RTG Support Panels

The RTG mounting panels were analyzed for a cantilevered RTG module.

It was idealized as a simply supported beam with an applied concentrated shear and moment.

It was determined that the same size panels and stiffeners used in the access panels provided the necessary structural stiffness.

- Equipment Mounting Panels

The equipment mounting panels were sized assuming the 2000 pound payload (without RTG modules) would be divided equally into the six bays and that all the equipment would be mounted on one wall. Also, the added stiffness due to the heat pipes was neglected.

Assuming the panel is supported by the radial ribs only and idealizing it to a beam of unit width: $(1350\# / 675 \text{ in}^2 \times 1 = 2\#/\text{in})$

For a 1/2" core (required for heat pipes) a face sheet thickness of $t = 0.008''$ is required. This size was checked for the 5.4 psi acoustic load and was found to be satisfactory.

The three equipment mounting panels on the bays where the RTG modules aren't attached will have two 12 1/2" x 6" holes for equipment which requires viewing area.

The structural stiffness of the continuous honeycomb panel was replaced by inserting channel sections along the edges of the holes. View AA would look as follows:

This is the direction with the least bending stiffness because the holes are larger in this direction and there is no added stiffness from the heat pipes. The channels needed to replace the lost stiffness are $1/2'' \times 1'' \times .050''$ channels. These channels would also be useful for mounting equipment or covers.

- Radial Ribs

The loads from the equipment mounting panels and RTG mounting panel are transferred to the radial ribs. These loads and the manner in which they are applied to the ribs are shown in the following sketch:

The ribs are truss structures which allow access to adjacent compartments and in addition provide thermal radiation between compartments for good thermal control.

Members AC and BC are sized based on the 1350 pound load in the x direction from the equipment mounting panels:

The load in member BC is 828 pounds. Sizing this member for critical column buckling load and stress:

$$P_{CR} = \frac{\pi^2 EI}{L'^2} \quad (L' = .7L)$$

A 1" round tube with an .035" wall provides the required stiffness.

Members AB and CD are the rib caps and must carry the loads due to a) reactions from 1350 pound x load on the rib (422#); b) loads from the RTG mounting panels (156#); and c) radial components of the Y and Z equipment panel loads which are determined as follows:

The total load is 1100#. The cross section of a representative cap may look like a double T to allow for the attachment of the equipment mounting panels and rib truss tubes or a standard T section with a special fitting for the attachment of the truss tubes. These loads in the rib caps will be reacted by the inner cylinder support ring. But this is not the critical loading condition for the rings and so they will be examined later.

• Inner Cylinder

The inner cylinder must carry the 1350 pound shear load from the ribs to the horizontal support ribs.

Sizing panel ABCD for critical shear stress

$$F_{SCR} = KE (t/b)^2$$

results in a panel thickness of $t = .040''$ with $1'' \times 1'' \times .064''$ angle stiffeners.

- External Support System

- End Bulkheads

The external support system, consisting of the 2 horizontal ribs and the inner cylinder support ring, was sized based on the maximum vertical loading (N_z) as defined previously.

The horizontal beam, which is the rib cap, between the inner cylinder ring and external support point was analyzed by simplifying it to a cantilever beam and determining the section properties required.

The moment at the ring attachment point is 55,000 in - lb. One way to obtain the required support stiffness is to increase the rib cap double T section to 4" x 2" x .25" and add stiffness at the ring attachment point and external support point.

The inner cylinder ring was idealized as a circular ring with a 20" radius, with the following combined loading conditions:

The maximum moment occurs at the point where the horizontal ribs attach and is 25,500 in - lb. This resulted in a preliminary ring size of 3" x 2" x .25" angle section for stress levels of 40,000 psi.

Horizontal Reaction Frame

The horizontal rib truss will carry the equipment mounting panel loads like the other ribs but this is not the critical loading condition for these two ribs. Their critical loading occurs when they are considered part of the external support system. This loading condition is the maximum thrust loading (N_x) as defined previously. The ribs are loaded as follows:

The load in member BC is 5583#. Members AC and BC were sized for critical column buckling load and stress, in a similar manner as the other ribs. It was determined that a 1-1/4" tube with an .063" wall provides the required stiffness. Members AB and CD were sized previously.

The results of this analysis were used to refine the initial structural weight estimate.

A.2 Brayton-Cycle Spacecraft Structural Sizing

The structural loads due to the maximum N_x and N_z loading conditions were applied to the Brayton structural system for preliminary sizing.

• Equipment Mounting Panels

The equipment mounting panel of a typical compartment was idealized to a 54" x 40" rectangular plate, supported on two edges by the radial ribs, with the maximum N_x load concentrated in the center:

The load in the face sheets is determined from the maximum moment and core thickness, h :

$$P = M/h$$

$$f = P/A = \frac{M/h}{54'' \times t}$$

For $F_{tu} = 64,000$ psi (2024 alum.)

and 1/2" core

The required face sheet thickness is

$$t = .012''$$

This panel was checked for the 145 db acoustic load. An equivalent pressure of 5.4 psi was applied to the panel and the resulting stress was 53 ksi.

• Radial Ribs

The three load components, N_x , N_y , and N_z , from the equipment mounting panel are transferred to the ribs. A typical compartment loading is shown in the following sketch:

Members AD, BD, BE, BF and CF are sized based on the X component of the loading:

The maximum load is in member BD and is 1630#. This member was sized for critical column buckling load and stress:

$$P_{cr} = \frac{\pi^2 EI}{L'^2} \quad (L' = .7L)$$

The required I for this member is

$$I = 0.022 \text{ in}^4$$

For a 1 1/4" tube, 0.035" wall

$$I = .024 \text{ in}^4$$

$$A = .1336 \text{ in}^2$$

This gives the required stiffness with minimum weight.

The rib caps, members AB, BC, DE, and EF must carry the load due to a) the kicks from the X load component (1092#), b) radial components of the Y and Z loads (278#) and c) the kicks from the circumferential components of the Y and Z loads (654#). The total load in member AB is 2024#. The cross section of a representative cap may look like a double T to allow for the attachment of the equipment mounting panels and the rib truss tubes.

• Inner Cylinder

The shear load from the radial ribs is transferred to the inner cylinder. This load then must be sheared around to the two horizontal support ribs.

Panel ABCD would look as follows:

The critical buckling shear stress for a curved panel is

$$F_{SCR} = K_S E (t/b)^2$$

where K_S is a function of the aspect ratio and curvature. For .040" skins and one 1" x 1" x .064" angle stiffener, the critical buckling shear stress is

$$F_{SCR} = 3000 \text{ psi}$$

The shear stress in the panel is 2275 psi, providing a satisfactory shell structure.

- External Support System

- End Bulkheads

The external support system consists of the two horizontal ribs and the inner cylinder support ring. The inner cylinder support ring and rib caps compose the end bulkhead system and they were sized based on the maximum N_z loading previously discussed. The bulkhead loading would appear as follows:

The horizontal beam (rib cap) between the inner cylinder support ring and external support point was analyzed by the following simplifying assumptions:

The moment at the interface with the cylinder support ring is 235,000 in - lb. One possible way of obtaining the required stiffness is to increase the double T rib cap section to 4" x 2" x .25" and add some stiffeners at the interface point.

The critical loading condition for the inner cylinder support ring is the applied moment from the previous discussion.

The maximum moment in the ring is 117,500 in - lb which would require a 4" angle, .25" thick to provide the required stiffness.

Reaction Ribs

The two horizontal reaction ribs will carry loads due to the equipment mounting panel just like the other radial ribs. However, the critical loading for these two ribs occurs because they are acting as part of the external support system. The critical loading for these ribs is shown in the following sketch:

Members AB, BC, DE, and EF have already been analyzed as part of the end bulkhead system. Members AB, BD, BE, BF, and CF were resized based on the loading shown above. The maximum load is 17,750# in member BD. Sizing for critical column buckling load and stress as before

$$P_{CR} = \frac{\pi^2 EI}{L'^2}$$

requires a 2" tube, 0.083" wall.

The results of this analysis were used to refine the initial structural weight estimate.

APPENDIX B

ATTITUDE AND ORBIT CONTROL ANALYSIS

Section 5.2.2 discussed design philosophy and requirements for the spacecraft orbit and attitude control systems. Some specific control system parameters are derived in this appendix to demonstrate feasibility of the design concepts.

B.1 RTG Spacecraft

B.1.1 Propulsion for Orbit Adjustment and Maintenance

Section 5.2.2 set a design requirement of 1170 ft/sec total impulse, to be achieved by a liquid propellant system capable of multiple firings. A hydrazine monopropellant system with specific impulse of 220 seconds is selected. The propellant requirement is given by:

$$\frac{m_f}{m_i} = 1 - e^{-\alpha} \quad \begin{array}{l} m_f = \text{mass of fuel} \\ m_i = \text{initial mass of spacecraft} \end{array}$$

$$\alpha = \frac{\Delta V_r}{g I_{sp}} \quad \begin{array}{l} \Delta V_r = \text{total impulse} = 1170 \text{ ft/sec} \\ I_{sp} = \text{specific impulse} = 220 \text{ sec.} \end{array}$$

resulting in: $m_f/m_i = 0.152$

To meet this requirement for spacecraft of various weights requires fuel as shown in the following table:

Total spacecraft weight (lbs)	1000	1500	2000	2500
Hydrazine (lbs)	152	228	304	380
Dry weight (lbs)	848	1272	1696	2120

Assume a total of 100 pounds of thrust provided by four fixed nozzle 25-pound thrusters. The small thrusters save weight and the relatively long time required to provide the total impulse is of no particular disadvantage for the missions considered.

For the case where the impulse is used to circularize a 300 - 900 n.mi. orbit, the total impulse can be spread over two or three successive apogee passages, providing additional opportunity for ground monitoring.

For the 2,500 lb spacecraft, the required impulse of $380 \times 220 = 83600$ lb sec will require 836 sec of firing at 100 lb thrust level. The

900 n. mi. orbit period being two hours, the thrust will require approximately 41.8 degrees of arc. Thrusting could cover two periods of approximately 360 seconds each (± 9 degrees about apogee) on two successive passes, with the remaining 116 seconds expended for fine correction on the third pass.

The spacecraft must be spun up prior to this maneuver. The need for spin originates in the requirement that induced nutation should not exceed 1° .

Assuming for the set of four nozzles a net thrust misalignment of 0.5 degree with a two-foot moment arm and a 0.1 inch uncertainty in net CG and in thrust axis the net moment arm will be:

$$r = \sqrt{[2 \times 0.5 \times .01745]^2 + (.0083)^2 + (.0083)^2} = 0.021 \text{ ft.}$$

If the spacecraft spin is Ω rad/sec the momentum change ΔH in one half the spin period is

$$\Delta H = \frac{Fr}{\Omega} \int_{-\pi/2}^{\pi/2} \cos \theta d\theta = \frac{2 Fr}{\Omega}$$

Where F is the thrust of 100 pounds, the total momentum being $H = I\Omega$, where I is the moment of inertia about the spin axis. The spin axis deviation due to thrust offset is

$$\frac{\Delta H}{H} = \frac{2 Fr}{I \Omega^2}; \text{ this quantity is to be kept less than } 1^\circ (.01745).$$

$$\text{Thus } \Omega^2 > \frac{2F r}{.01745 I}$$

For $F = 100$ lb thrust

$$I = 600 \text{ slug ft}^2$$

$$r = .021 \text{ ft.}$$

one obtains

$$\Omega > 0.64 \text{ rad/sec}$$

$$\frac{\Omega}{2\pi} > 0.102 \text{ rev/sec} = 6.1 \text{ RPM}$$

Typically, one may wish to spin at 10 RPM = 1.05 rad/sec, resulting in a net momentum of $600 (1.05) = 630$ lb ft sec, and g-loading at

the 4.8 ft. extreme outboard point of 0.165 g.

The hydrazine requirement for spinup (yaw thrusters with 3.5 ft moment arm) is

$$\frac{1 \Omega}{I_{sp} r_z} = \frac{(600)(1.05)}{220(3.5)} = 0.82 \text{ lb;}$$

and a like quantity is required for despin.

The spinning spacecraft must be positioned to the proper angle for thruster firing; this involves the exertion of torques to precess the 630 lb ft sec spin vector. The out-of-plane angle for a worst case of 1.5 degrees inclination correction is

Thus

$$H = (0.54)(630) = 340 \text{ lb ft sec}$$

$$\text{Moment arm} = 3.5 \text{ ft}$$

$$\text{Impulse} = \frac{340}{3.5} = 97.3 \text{ lb sec}$$

$$\text{Hydrazine required} = \frac{97.3}{220} = 0.44 \text{ lb}$$

The system should be capable of correcting errors in the initial orbit insertion. For example, to achieve a sun-synchronous orbit at 700 - 900 n. mi. altitude, the altitude (semimajor axis) and inclination are chosen so that the orbit precesses at exactly $360^\circ/\text{year}$:

$$\begin{aligned} \dot{\Omega} &= -3636 \left(\frac{R}{a} \right)^{3.5} \cos i \quad \text{degrees/year} \\ &= 360 \text{ degr/year} \end{aligned}$$

R = Earth radius

a = orbit semi-major axis

i = orbit inclination

$\dot{\Omega}$ = nodal precession rate

At a nominal 700 n. mi. altitude, insertion error coefficients are:

- a) 1 degree error in inclination results in an additional precession of ± 36.2 degr/yr.
- b) 1 n. mi. error in semi-major axis results in an additional precession of ± 0.305 degree/year.

A generous error allowance in orbit insertion may be a drift of 9.1 degrees/year = 0.025 degree/day

= 0.1 minute of time/day;

This could be adjusted by means of a change in semi-major axis of ± 30 n. mi., requiring a velocity correction

$$\Delta V = \pm (0.5) V_o \frac{\Delta a}{a} = 87 \text{ ft/sec};$$

and involving hydrazine expenditure of 12.3 lbs per 1000 lb of spacecraft.

For spacecraft in synchronous equatorial orbit, the propellant supply allocated to orbit insertion in the 900 n. mi. orbit case will be available for orbit maintenance. The maintenance requirements are approximately

$$\Delta V_1 = 150 \text{ ft/sec/year (inclination control, north-south stationkeeping)}$$

$$\Delta V_2 < 7 \text{ ft/sec/year (east-west stationkeeping);}$$

suggesting that the provision for the low orbit vehicle is adequate for six years of synchronous vehicle operation.

B.1.2 Attitude Control Requirements

The primary disturbance torques will be solar pressure and magnetic torques. The large variety of external configurations of the spacecraft (depending on the payload appendages to be installed in the several applications of the "bus") make it difficult to calculate precise torque levels. Thus, a most conservative design must be evolved, providing for control of possibly large moments. A reasonable estimate of solar torques has been based on

area (including possible antennae): 40 ft^2 in each plane (xy, xz, yz)

Unbalance moment arm (distance from center of pressure to center of mass): 6 inches;
resulting in torques of the order of

$$(40 \text{ ft}^2) \times (1/2 \text{ ft}) \times (10^{-7} \text{ lbs/ft}^2) = 2.0 \times 10^{-6} \text{ lb/ft}$$

about each axis. The torques about the y axis will be cyclic, but the x and z axis torques may have secular components.

Magnetic torques arise from the interaction of spacecraft magnetic moments (permanent magnets, circulating currents) with the Earth's field.

Experience has shown that spacecraft moments of the order of 3×10^{-3} amp.-m² per pound of spacecraft may be expected; or (for the typical 2000 lb bus) a total moment in arbitrary direction of 6 amp.-m².

This moment interacts with the Earth's field, which is

$$\bar{B}_{\text{earth}} = B_o \left[\bar{P} - (3 \sin \psi_m) \bar{R} \right]$$

where $\bar{P}$ is a unit vector along the Earth's magnetic polar axis

$\bar{R}$ is a unit vector radially from the Earth's center to the spacecraft

ψ_m is the magnetic latitude of the vehicle

$$\text{and } B_o = \frac{8.1 \times 10^{15}}{R^3} \quad \text{weber/m}^2$$

R being the radius vector in meters.

Thus, the magnitude of the field in circular orbit is approximately B_o near the equator and a maximum of $2 B_o$ near the magnetic poles. Numerically, $B_o = 0.11 \times 10^{-6} \text{ web/m}^2$ at synchronous altitude

$$B_o = 21 \times 10^{-6} \text{ web/m}^2 \text{ at 500 n. mi. altitude}$$

(500 n. mi. altitude is taken as minimum altitude, corresponding to a maximum magnetic torque). Thus in a polar orbit, torques as high as

$$\begin{aligned} 2 B_o M &= 42 \times 10^{-6} \text{ weber/m}^2 \times 6 \text{ amp-m}^2 \\ &= 252 \times 10^{-6} \text{ newton-m} \\ &= 186 \times 10^{-6} \text{ lb-ft} \end{aligned}$$

might be expected, with a secular component of possibly 10% of this value (near polar orbit). While this could be compensated with relatively large inertia wheels, unloaded periodically with the hydrazine torques, it is far more economical to provide magnetic compensatory torquers. Thus, the recommended attitude control system for low altitude vehicles includes three orthogonal variable magnetic torquers. To provide for ample margin, 10 amp-m^2 (10^4 pole-cm) per axis is recommended, at an estimated weight of 2 lb per axis (6 lb total). This variable magnetic torquer system may be expected to effectively compensate the major part of the disturbance by creating a bucking moment opposite to the stray spacecraft magnetic moment, leaving a residual which is conservatively estimated at 10%.

Thus, for the low polar orbit vehicles, magnetic torques of the order of

$$(0.10) (186 \times 10^{-6}) = 18.6 \times 10^{-6} \text{ lb-ft}$$

should be absorbed in the wheel system. (Most of this is an alternating torque, the secular component along the X and/or Z axes being at most 2×10^{-6} lb-ft).

The vehicle in synchronous orbit will experience torques of the order of

$$\begin{aligned} B_o M &= 0.11 \times 10^{-6} \times 6 = 0.66 \times 10^{-6} \text{ newton-m} \\ &= 0.5 \times 10^{-6} \text{ lb-ft} \end{aligned}$$

No compensatory system is required. (In this case, however, since $\vec{B}_{\text{earth}}$ is essentially parallel to the vehicle Y axis at all times, all of the X and Z components of torque - possibly the entire 0.5×10^{-6} lb-ft - may be secular and thus be cumulative).

The momentum wheel system should be capable of absorbing both cyclic and secular torques without requiring too frequent unloading. Design for weekly unloading would require absorption of magnetic and solar pressure secular torque for 7 days, plus the cyclical part of the momentum, for 1/2 orbit. Conservatively, the maximum are experienced in the low earth orbit (in each axis)

Solar pressure (max. secular component)	2×10^{-6} lb ft
Magnetic secular torque	2×10^{-6} lb ft
Total	4×10^{-6} lb ft

Weekly momentum:

$$4 \times 10^{-6} \text{ lb ft} \times 7 \times 86400 \text{ sec} = 2.4 \text{ lb ft sec}$$

The cyclic momentum (2×10^{-6} lb ft solar pressure, 18.6×10^{-6} lb ft magnetic) in 1/2 orbit (3300 sec) will be 0.07 lb ft sec. Thus wheels of 2.5 lb ft sec capacity in each axis should be adequate. However, to provide adequate maneuvering power and additional control margin, 6 lb ft sec wheels are recommended.

The cumulative effect of secular torques will result in jet reaction unloading requirements. The estimated requirements are: each axis requiring, as shown above, 2.4 lb ft sec/week each, this being a most conservative requirement. (A more likely situation would be only 2 axis, (x, z); experiencing far less than this peak value.) Even this conservative estimate results in propellant requirements for 5 years of operation of only 3 axes x 5 years x 52 weeks/year x 2.4 lb ft sec/week = 1872 lb ft sec.

Thrusting with hydrazine ($I_{sp} = 220$ sec) with 3.5 ft moment arm requires

$$\frac{1872 \text{ lb ft sec}}{3.5 \text{ ft} \times 220 \text{ sec}} = 2.44 \text{ lbs of propellant.}$$

Some additional propellant is required to overcome torques induced during propulsion (station-keeping) in the stabilized mode. Typically, this will increase the station keeping propulsion requirement by 4%.

B.1.3 Summary

1) Propellant requirements.

Table B.1 below lists the required hydrazine for a conservatively designed spacecraft in a 5 year near polar mission. This propellant will also be adequate for a craft placed in synchronous equatorial orbit utilizing this fuel for control and orbit maintenance.

Table B.1

RTG Spacecraft Propellant Allowances (pounds)

Initial S/C weight	1000	1500	2000	2500
Dry weight	832.7	1251.3	1669.9	2088.5
Propellant allowances:				
1170 ft/sec ΔV (orbit insertion or 5 years N-S station keeping)	152	228	304	380
Spin up, Spin down, Spin Vector Precession	2	2	2	2
Orbit maintenance/correction (E-W)	10.4	15.6	20.8	26
Attitude Control	2.9	3.1	3.3	3.5
Total Propellant Weight	167.3	248.7	330.1	411.5

2) Attitude control components

The recommended system will contain the following components:

a) Transfer mode:	Estimated weight, pounds
2 Earth Sensors	4
2 Sun Sensors	2
Rate Gyro Assembly	4
b) Operations mode:	
3 Momentum Wheels	58 (total)
2 Digital Computers	45 "
2 Earth Sensors and Electronics	15
Sun Sensors	2
Actuator Electronics	20
Total	150 lbs

In addition, the low orbit vehicles will require:

c) Magnetic moment torquers	6 lbs
Magnetometer	3 lbs

B.2 Brayton Spacecraft

B.2.1 Orbit Insertion and Maintenance

Following the same reasoning as in section B.1.1, one obtains essentially the same relationships:

Insertion: Provision of $V = 1170$ ft/sec results in hydrazine requirement (W_f) of 15.2% of initial weight, as before:

$$W_f = 0.152 W_i$$

(W_i = weight in transfer orbit), resulting in weight in the operating orbit of $0.848 W_i$ lbs. Again, for the synchronous orbit spacecraft, the entire propellant supply is used for station keeping purposes.

The maneuver will still be carried out by 4 thrusters of 25 lb thrust each, this being the largest size hydrazine thrusters in manufacture. A vehicle with gross weight of 4000 lbs will use 608 lbs of hydrazine ($I_{sp} = 220$) for a total of 133760 lb sec of impulse. Therefore, the burn time is 1337.6 sec; or 67 degrees of the nominally 2 hour orbit. It is suggested that this thrusting be apportioned over 3 orbits, with corrections for 500 sec (± 12.5 degrees about apogee) for 2 orbits, and the remainder, including any fine corrections required, on the 3rd orbit.

The desired spin rate is calculated similarly to the RTG requirement with a thrust offset moment arm of $r = 0.025$ ft

$$\text{an inertia} \quad I_z = 2850 \text{ slug ft}^2$$

$$\text{and thrust} \quad F = 100 \text{ lb}$$

one obtains a desired spin rate (for 1° rotation, maximum) of

$$\begin{aligned} \frac{2Fr}{I_z} &= \frac{2 \times 100 \times .025}{2850 \times .01745} = 0.317 \text{ rad/sec} \\ &= .05 \text{ rev/sec} \\ &= 3 \text{ RPM} \end{aligned}$$

Typically one may wish to spin somewhat faster, say 5 RPM (0.52 rad/sec). The spinup (and spindown) fuel requirement is

$$\frac{(2850)(0.52)}{220(7.5)} = 0.9 \text{ lbs.}$$

The spin vector precession maneuver to attain desired thrusting attitude requires 0.5 lb of propellant.

For the "sun synchronous" near polar vehicle the same 87 ft/sec orbit correction provision should be made, again requiring 0.0123 lbs/lb of spacecraft.

B.2.2 Attitude Control Requirements

Considering the larger dimensions of the Brayton spacecraft, and the increased diversity of "payload" appendages (antenna), the control system has been sized to cope with solar pressure moments (in each axis) due to:

$$\begin{aligned} \text{area;} & \quad 200 \text{ ft}^2 \\ \text{force;} & \quad 200 \text{ ft}^2 \times 10^{-7} \text{ lb/ft}^2 = 20 \times 10^{-6} \text{ lb} \\ \text{moment arm;} & \quad 1 \text{ ft} \\ \text{moment (each axis);} & \quad 20 \times 10^{-6} \text{ lb ft} \end{aligned}$$

Again, the X and/or Z (roll and/or yaw) torques may have substantial secular components.

Magnetic torques are estimated (as with the RTG bus) on the basis of 3×10^{-3} amp m² per pound of spacecraft. The low altitude spacecraft will have controllable magnetic torques (20 amp m² per axis) to compensate most, if not all, of this torque, leaving a remainder of the order of 3×10^{-6} lb ft. (each axis). For the synchronous altitude vehicle the magnetic torque is significant (less than 10^{-6} lb ft) when compared with typical solar pressure torques.

The 3 wheels should be capable of absorbing the secular torques for 7 days, in addition to the contribution from the cyclic torques, before unloading. Thus, the wheel rating, is based on;

Solar pressure torque (max. secular component)	10×10^{-6} lb ft.
Magnetic torque (max. secular component)	3×10^{-6} lb ft.
	13×10^{-6} lb ft.
Weekly momentum	
13×10^{-6} lb ft x 7 x 86400 =	7.8 lb ft sec
Cyclic momentum (20×10^{-6} lb ft solar pressure, 30×10^{-6} lb ft magnetic torque) for 1/2 orbit:	0.17 lb ft sec.
Brayton spacecraft bias requirement	0.47 lb ft sec.
TOTAL	8.44 lb ft sec.

Thus, wheels of 10 lb ft sec rating in each axis will be adequate; slightly larger wheels (15 lb ft sec) might be chosen to provide additional margin (less frequent unloading).

The propellant requirements for 5 years of operation (wheel unloading) is estimated conservatively at no more than 4.4 lbs.

B.2.3 Summary

Propellant Requirements

Table B.2

Brayton Spacecraft Propellant Allowances (pounds)

Initial S/C weight	2000	3000	4000
Dry weight	1667.7	2504.9	3342.1
Propellant allowances:			
1170 ft/sec ΔV (5 years N-S station-keeping or orbit injection)	304	456	608
Spin-up, Spin-down, Spin Vector precession	2.3	2.3	2.3
Orbit maintenance/correction (EW)	20.8	31.2	41.6
Attitude control	5.2	5.6	6.0
Total propellant	332.3	495.1	657.9

Attitude Control Components

a) Transfer mode	wt, lbs.
Transfer Earth sensors (2)	4
Transfer Sun sensors (2)	2
Rate Gyro assembly	4
b) Operational Mode	
3 Momentum wheels	84
2 Digital computers	45
2 Earth sensors and electronics	15
Sun Sensors	2
Actuator Electronics	<u>20</u>
TOTAL	176 lbs.

In addition, the low orbit vehicles will require

c) Magnetic moment torques	12 lbs.
Magnetometer	3 lbs.

APPENDIX C

ISOTHERMAL RADIATOR FOR THE BRAYTON - CYCLE POWER PLANT

ISOTHERMAL RADIATOR FOR THE BRAYTON-CYCLE POWER PLANT

C. 1 INTRODUCTION

To simplify the thermal design of spacecraft for specific payloads, it would be desirable to adopt a design in which the spacecraft walls are automatically maintained in the range of acceptable payload temperatures, regardless of variations in solar input or electrical power consumption. This would create a "benign environment" for the enclosed payload components.

One possible way of meeting this goal in a nuclear-powered spacecraft is to use the spacecraft walls as the principal radiator for rejecting the waste heat of the nuclear power plant. However, since the desired payload temperatures are in the range of 0 to 40°C, this scheme would lead to an unnecessarily large radiator, which would be rejected as too inefficient in most designs.

In the case of a Brayton cycle power plant, however, the optimum radiator outlet temperature is approximately equal to the optimum payload temperature. Thus, since a large radiator is in any case required for thermodynamic efficiency, it may be desirable to make it isothermal, to gain the benefit of the benign payload environment.

It should be noted that even in the case of the Brayton system, making the radiator isothermal still entails a considerable size penalty, typically a factor of two. But since the radiator here acts as the spacecraft wall, which must in any case be large enough to provide the required payload volume, this may not be objectionable. In addition, the above size penalty could be largely eliminated by dumping about half the heat through a small, relatively high-temperature heterothermal radiator, and the other half through a low-temperature isothermal unit.

C. 2 DESIGN DESCRIPTION

Consider a spacecraft in which all the waste heat is rejected from a low-temperature cylindrical radiator of radius R and length L . The gas stream to be cooled enters the radiator at a temperature T_{in} and leaves at temperature T_{out} , with a mass flow rate $\dot{m}$ and specific heat C_p . The gas pressure drop Δp through the radiator is small compared to the mean pressure p .

The gas flows around the circumference of the radiator through tubes of radius r . These tubes are in thermal contact with axial heatpipes of length L , each of which has two axial fins. The set of finned heatpipes forms the sidewall of the cylindrical radiator. In the low-temperature regime of interest here, such heatpipes are highly reliable. The radiator contains enough redundancy to tolerate the loss of several heatpipes. Thus, the need for anti-meteorite armor can be eliminated.

The above configuration could operate with just a single gas tube passing only once around the circumference of the radiator. However, such a radiator would be far from isothermal, because of the large difference between T_{in} and T_{out} . In a typical Brayton cycle, gas enters the radiator at 300°F and leaves at 75°F .

To flatten the radiator temperature profile, we propose to split the gas stream into two halves, which pass around the circumference of the radiator in opposite directions. This counterflow arrangement should result in a relatively uniform radiator temperature, since at the point where one gas stream is hottest the other is coldest. However, some temperature non-uniformities will remain because of the non-linear relation between temperature and radiative emission.

The purpose of the following analysis is to determine whether the proposed scheme can work, and how far from isothermal such a radiator system would be. It is, of course, recognized that further improvements in temperature uniformity could be achieved by various stratagems, such as variations in emissivity coatings and/or heatpipe fin effectiveness.

C.3 ANALYSIS

The following analysis assumes that the dominant thermal resistance between the gas streams and the radiator heatpipes is the resistance of the gas film, which can be adequately represented by a constant heat transfer coefficient h . Let θ denote the angular position around the radiator circumference (measured from the inlet-outlet position), let T and T' denote the local temperatures of the gas streams moving in the $-\theta$ and $+\theta$ direction, respectively, and let t represent the local radiator temperature at θ . The temperature variation of the two gas streams is then given by:

$$(1/2)\dot{m} C_p dT = (2 \pi r) (Rd\theta) h (T-t), \quad (1)$$

$$-(1/2)\dot{m} C_p dT = (2 \pi r) (Rd\theta) h (T'-t), \quad (2)$$

and the three θ - dependent temperatures are coupled by the heat balance

$$2 \pi r h [(T-t) + (T'-t)] = L \sigma \epsilon t^4, \quad (3)$$

where σ is the Stephan-Boltzmann constant and ϵ the emissivity of the radiator surface. Solar radiation and earth reflection is neglected in this phase of the analysis.

To solve the set of three equations we first divide Eq. (2) by (1), yielding

$$\frac{dT'}{dT} = \frac{T'-t}{T-t}, \quad (4)$$

and differentiate Eq. (3) with respect to T , to obtain

$$\frac{dT'}{dT} = 2 \frac{dt}{dT} + 1 = \frac{2 L \sigma \epsilon}{r h} t^3 \frac{dt}{dT} \quad (5)$$

Eliminating dT'/dT between Eqs. (4) and (5), and using Eq. (3) to eliminate T' , we obtain a single differential equation,

$$\left(1 + \frac{L \sigma \epsilon}{\pi r h} t^3\right) \frac{dt}{dT} = 1 - \frac{L \sigma \epsilon}{4 \pi r h} \left(\frac{t^4}{T-t}\right), \quad (6)$$

which can be rewritten in dimensionless form as

$$\left(1 + \rho^3\right) \frac{d\rho}{d\gamma} = 1 - \frac{(1/4) \rho^4}{\gamma - \rho}, \quad (7)$$

where ρ represents a dimensionless radiator temperature,

$$\rho \equiv \left(\frac{L \sigma \epsilon}{\pi r h}\right)^{1/3} t, \quad (8)$$

and γ denotes a dimensionless gas temperature,

$$\gamma \equiv \left(\frac{L \sigma \epsilon}{\pi r h}\right)^{1/3} T \quad (9)$$

to solve Eq. (7), we rearrange it in the form

$$4(\gamma - \rho) \rho^3 d\rho + \rho^4 d(\gamma - \rho) = 4(\gamma - \rho) d(\gamma - \rho) - \rho^4 d\rho, \quad (10)$$

which can be integrated to yield

$$(\gamma - \rho) \rho^4 = 2(\gamma - \rho)^2 - (1/5) \rho^5 + C. \quad (11)$$

To define the integration constant C, we note that at $\theta = \pi$, i.e. at the midpoint of the two gas ducts, symmetry requires that the two gas streams have the same temperature, $T_\pi = T'_\pi$. Hence, from Eq. (3),

$$4\pi r h (T_\pi - t_\pi) = L \sigma \epsilon t_\pi^4, \quad (12)$$

or in dimensionless form,

$$\gamma_\pi = \rho_\pi + (1/4) \rho_\pi^4. \quad (13)$$

Inserting this in Eq. (11), the integration constant C can be expressed in terms of ρ_π , the radiator temperature at $\theta = \pi$;

$$C = (1/8) \rho_\pi^8 + (1/5) \rho_\pi^5. \quad (14)$$

Substituting for C in Eq. (11) and solving for γ yields:

$$\gamma = \rho + (1/4) \left\{ \rho^4 \pm [(\rho^8 - \rho_\pi^8) + (8/5)(\rho^5 - \rho_\pi^5)]^{1/2} \right\}. \quad (15)$$

As shown by the $\pm$ sign in Eq. (15), the gas temperature γ is a double-valued function of the radiator temperature ρ , which is symmetrical about $\theta = \pi$. Thus, the dimensionless inlet and outlet temperatures and their difference can be expressed in terms of the radiator temperature ρ_o at $\theta = 0$:

$$\gamma_{in} = \rho_o + (1/4) \left\{ \rho_o^4 + [(\rho_o^8 - \rho_\pi^8) + (8/5)(\rho_o^5 - \rho_\pi^5)]^{1/2} \right\}. \quad (16)$$

$$\gamma_{out} = \rho_o + (1/4) \left\{ \rho_o^4 - [(\rho_o^8 - \rho_\pi^8) + (8/5)(\rho_o^5 - \rho_\pi^5)]^{1/2} \right\}. \quad (17)$$

Adding and subtracting Eqs. (16 and (17) yields

$$\gamma_{in} + \gamma_{out} = 2 \rho_o + (1/2) \rho_o^4, \quad (18)$$

$$(1/2)(\gamma_{in} - \gamma_{out})^2 = (1/8)(\rho_o^8 - \rho_\pi^8) + (1/5)(\rho_o^5 - \rho_\pi^5), \quad (19)$$

or in dimensional form:

$$T_{in} + T_{out} = 2 t_o + \frac{L \sigma \epsilon}{2 \pi \gamma h} t_o^4, \quad (20)$$

$$\begin{aligned} (1/2) (T_{in} - T_{out})^2 &= (1/8) \left(\frac{L \sigma \epsilon}{\pi \gamma h} \right)^2 (t_o^8 - t_\pi^8) + \\ (1/5) \left(\frac{L \sigma \epsilon}{\pi \gamma h} \right) (t_o^5 - t_\pi^5) \end{aligned} \quad (21)$$

Equation (20), of course, could have been obtained directly from Eq. (3). Given the inlet and outlet temperatures, and the value of the parameter $L \sigma \epsilon / \pi \gamma h$, Equation (20) can be solved for the maximum radiator temperature t_o , after which the minimum radiator temperature t_π is obtained from Equation (21). Alternatively, Eq. (20) can be used to compute the ratio $L \sigma \epsilon / \pi \gamma h$ for a given temperature t_o .

In designing an isothermal radiator for a Brayton system, we simultaneously seek to minimize the gas outlet temperature (for thermodynamic efficiency) and to maximize the radiator temperature (to reduce its size). These are conflicting aims. While, with the proposed arrangement, it would be theoretically possible to have a gas outlet temperature T_{out} lower than the maximum radiator temperature t_o , this would not be desirable since part of the radiator would then act to heat rather than cool the gas. The optimum design point appears to be the case where the gas outlet temperature equals the radiator temperature at $\theta = 0$ and 2π .

Letting $\rho_o = \gamma_{out}$ in Eq. (17), we obtain:

$$\rho_\pi^5 + (5/8) \rho_\pi^8 = \gamma_{out}^5 \quad (22)$$

or in dimensional terms:

$$t_\pi^5 + (5/8) \left(\frac{L \sigma \epsilon}{\pi \gamma h} \right) t_\pi^8 = T_{out}^5 \quad (23)$$

Similarly, since $t_o = T_{out}$, Eq. (20) yields

$$\frac{L \sigma \epsilon}{\pi \gamma h} = \frac{2 (T_{in} - T_{out})}{T_{out}^4} \quad (24)$$

Combining Eqs. (23) and (24), we obtain an equation relating the minimum radiator temperature t_π to the gas inlet and outlet temperatures:

$$\left(\frac{t_{\pi}}{T_{out}}\right)^5 + (5/4)\left(\frac{T_{in}}{T_{out}} - 1\right)\left(\frac{t_{\pi}}{T_{out}}\right)^8 = 1. \quad (25)$$

C. 4 RESULTS

To illustrate the use of the above equations, consider a specific case in which gas enters the radiator at $149^{\circ}\text{C} = 422^{\circ}\text{K}$ and leaves at $45^{\circ}\text{C} = 318^{\circ}\text{K}$. For this case, Eq. (25) yields a minimum radiator temperature $t_{\pi} = 0.943 T_{out} = 300^{\circ}\text{K} = 27^{\circ}\text{C}$. Thus, the radiator varies in temperature between 27°C and 45°C , which is close enough to isothermal for the intended application.

It is also of interest to examine whether Eq. (24) can be satisfied in a practical design. In our illustrative case, a gas mixture of 40 mol-% helium and 60 mol-% xenon flows counter-currently around the radiator through tubes of 2.1 cm radius, at a mass flow rate of 0.15 lb/sec in each direction, leading to a predicted heat transfer coefficient of $0.0030 \text{ w/cm}^2\text{K}$. The above numbers yield a value of $31.4 \times 10^{-9} \text{ K}^{-3}$ for the parameter $L\sigma\epsilon / \pi\gamma h$. For the same parameter, Eq. (24) demands a value of $20.4 \times 10^{-9} \text{ K}^{-3}$, which is 35% lower. In practice, this lower value could easily be achieved, either by adding turbulence promoters to increase the heat transfer coefficient h , or by adding internal fins to increase the effective tube radius γ .

C. 5 CONCLUSIONS

1. A spacecraft using a Brayton system power plant can be designed with an essentially isothermal low-temperature radiator serving as the principal wall of the payload compartment. This greatly simplifies control of payload temperature, making it insensitive to variations in load power, solar input, or earth reflection.
2. Such a low-temperature radiator can be designed for high reliability, by employing two circumferential gas ducts, in thermal contact with a large number of axial radiator heatpipes. Reliability is achieved by heatpipe redundancy, elimination of leakage-prone tube-to-header joints present in conventional radiators, virtual elimination of thermal stresses, and great reduction of areas requiring meteorite protection.
3. The use of counter-current circumferential gas flow results in a radiator with only very modest temperature variations.
4. This arrangement is compatible with practical heat transfer, fluid flow, and geometric constraints.

ADDENDUM

Temperature Profiles

The equations derived earlier are sufficient to define the temperature range (ρ_π, ρ_0) of the radiator, to determine how close to isothermal a given design actually is. However, to clarify the whole concept, it is desirable to illustrate a specific case by displaying the temperature profiles of the radiator and the two coolant streams as a function of the angular position θ . To derive equations for the radiator temperature profile, let us first rewrite Eqs. (1, 2, 3) in dimensionless form:

$$d\gamma / d\phi = \gamma - \rho, \quad (26)$$

$$d\gamma' / d\phi = -\gamma' + \rho, \quad (27)$$

$$\gamma + \gamma' - 2\rho = 1/2 \rho^4, \quad (28)$$

where ϕ is a function θ , defined by

$$\phi \equiv (4\pi\gamma RH / m C \rho) (\theta - \pi). \quad (29)$$

Adding and subtracting Eqs. (26) and (27) gives:

$$d(\gamma + \gamma') / d\phi = \gamma - \gamma', \quad (30)$$

$$d(\gamma - \gamma') / d\phi = \gamma + \gamma' - 2\rho. \quad (31)$$

Differentiating Eq. (30) with respect to ϕ , and combining the result with Eq. (31), we obtain

$$d^2(\gamma + \gamma') / d\phi^2 = \gamma + \gamma' - 2\rho. \quad (32)$$

Equation (28) can now be used to eliminate γ and γ' , leaving a second-order equation for the radiator temperature ρ :

$$d^2(\rho^4 + 4\rho) / d\phi^2 = \rho^4. \quad (33)$$

Multiplying both sides of Eq. (33) by $d(\rho^4 + 4\rho) / d\theta$, we obtain the integrable equation

$$\int \left[\frac{d}{d\phi} (\rho^4 + 4\rho) \right] d \left[\frac{d}{d\phi} (\rho^4 + 4\rho) \right] = \int \rho^4 d(\rho^4 + 4\rho), \quad (34)$$

the boundary condition of which is obtained by noting that at

$\theta = \pi$, $\phi = 0$, $\rho = \rho_{\pi}$, $d\rho / d\phi = 0$. Therefore,

$$(1/2) \left[\frac{d}{d\phi} (\rho^4 + 4\rho) \right]^2 = (1/2)(\rho^8 - \rho_{\pi}^8) + (4/5)(\rho^5 - \rho_{\pi}^5). \quad (35)$$

Finally, performing the second integration, we obtain an integral relation between ϕ and ρ :

$$\phi = \pm 4 \int_{\rho_{\pi}}^{\rho} [(\rho^8 - \rho_{\pi}^8) + (8/5)(\rho^5 - \rho_{\pi}^5)]^{-1/2} (\rho^3 + 1) d\rho. \quad (36)$$

Equation (36), which must be integrated numerically, not only defines the radiator temperature profile, but also allows us to compute the corresponding gas temperature profiles, by inserting the values of ρ into Eq. (15). Typical temperature profiles are presented in Figure C-1. These are for the same illustrative case as previously described.

Lastly, Eqs. (29) and (36) can be used to calculate the radius R of the cylindrical radiator:

$$R = \frac{\dot{m} C_p}{\pi^2 \gamma h} \int_{\rho_{\pi}}^{\rho_0} [(\rho^8 - \rho_{\pi}^8) + (8/5)(\rho^5 - \rho_{\pi}^5)]^{1/2} (\rho^3 + 1) d\rho. \quad (37)$$

FIGURE C-1. TEMPERATURES AROUND COUNTERFLOW RADIATOR