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**COMPARISON STUDY OF RTG
& SOLAR POWERED VOYAGER SPACECRAFT**

RTG STUDY - VOYAGER TASK C

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1.0 INTRODUCTION

One of the end objectives of the Task C RTG Spacecraft Study is to compare the relative merits of solar and RTG powered Voyager spacecraft. In preparation for such a comparison, separate baseline spacecraft designs were developed and are described in Task C Document Numbers VOY-C1-TR16 (RTG Spacecraft) and VOY-C1-TR16A (Solar Spacecraft). As presented in these documents the designs are fixed in the sense that a single spacecraft weight is identified for fixed levels of installed RTG or solar power capability. In this comparison report, capabilities are presented on a variable basis using the baseline designs as a datum base. This permits a wider latitude of comparisons to be made: power capability may be compared for RTG and solar spacecraft of identical weight or, alternately, weight may be compared for equivalent power output. Besides performance comparison based on weight and power, other comparisons consider such items as reliability, mission flexibility, design implementation complexity, and schedule implementation.

Some of the more significant findings of their comparison study are summarized below:

- a. RTG's have the following advantages over solar power:
 - They have higher overall reliability because of constant operating conditions throughout life. removal of high duty-cycle batteries.
 - They provide greater mission flexibility because of independence of solar energy.
 - They permit improved ground test verification.
- b. RTG's have the following disadvantages with respect to solar power:
 - They are heavier - an improvement of about 20 to 25 percent in specific power (watts per pound) is needed to make them comparable with solar power for Mars missions.
 - Development is required relative to isotope fuel containment reentry protection.
 - Certain radiation detection instruments are incompatible with the RTG radiation environment.
 - Additional shroud cooling is required during prelaunch operation.

2.0 COMPARISON BASIS

2.1 RTG POWERED SPACECRAFT

The baseline RTG powered design used in this comparison report is described in detail in Task C Document No. VOY-C1-TR16. Figure 2-1 shows a general view of the spacecraft and is included here for reference purposes.

2.2 SOLAR POWERED SPACECRAFT

The baseline solar powered design is described in detail in Task C Document VOY-C1-TR16A and shown in Figure 2-2.

Figure 2-1. RTG Spacecraft

Figure 2-2. Solar Powered Spacecraft

3.0 WEIGHT AND POWER COMPARISON

3.1 BASELINE DESIGNS

A summary weight comparison of the baseline RTG and solar powered spacecraft is shown below:

	<u>Solar S/C</u>	<u>RTG S/C</u>
Bus Weight, lbs.	2300	2493
Planetary Vehicle Weight, lbs.	17,000	17,450
Power System		
Source, lbs.	Array: 298 (Area: 229 ft ²)	RTG's: 496
Battery, lbs.	117	28
Source Regulation	26	22

A comparison of power capability for the two baseline designs is shown on Figure 3-1 for the four Mars mission opportunities (1973, 1975, 1977, and 1979). The effect of solar occultation is not shown, but is treated in a subsequent section.

Figure 3-1. Power After Encounter

3.2 VARIABLE SPACECRAFT DESIGNS

The performance capabilities are analyzed for three sizes each of an RTG and a photovoltaic/battery (PV/B) power system – the nominal systems described earlier, and systems with primary power sources (RTG or solar array) 20 and 40 percent larger than nominal. In the case of the PV/B system, the silver-cadmium portion of the battery was also made proportionately larger to supply the larger power requirements.

The primary measures of power system technical performance are subsystem weight, (and total bus and planetary vehicle weight) and the amount of conditioned power (2400 cps regulated AC) available for the science package. Of interest also for the first 30 days is the trade-off between science power and the average amount of power available for capsule functions. In all cases, the basic spacecraft housekeeping loads were assumed to be constant, the science and capsule being the only variable loads.

A housekeeping base load of 326 watts was determined from Table 6-1 of the RTG Spacecraft Definition Report, VOY-C1-TR16. Under the column "Orbit with Capsule - Full Science" the required RTG power is given as 538.9 watts. Subtracting out all science power reflected at the RTG output reduces the RTG requirement for housekeeping loads to 381.2 watts. As conditioned power, this is reduced to 326 watts using a net conversion efficiency of 0.856 which takes all conditioning losses into account. A similar conditioned power base load is used for the solar powered spacecraft.

The power output of the PV/B power subsystem is a function not only of the array area (and weight), but also of the distance of the spacecraft from the sun, while the RTG output is sun-distance independent. The dependence of conditioned power on power subsystem weight and bus weight is given in Figure 3-2 for both PV/B and RTG power systems, nominal and augmented.

For the RTG power system a specific power of 0.9 pounds per watt is used for direct RTG output. Together with a conditioning efficiency of 0.856 percent this results in a specific power at the conditioned bus of 0.77 watts per pound. The 20 and 40 percent augmented systems refer to power increases in the basic 600 watt RTG output of the baseline spacecraft. For the PV/B power system the 20 and 40 percent augmented systems refer to area increases in the baseline solar array area of 229 square feet. The power output of the PV/B power system includes a nominal penalty for solar eclipses, as discussed in Section 3.6.

Figure 3-2. Conditioned Power Versus Weight

The bus weight can be related to planetary vehicle weight by including the effects of the weight of the midcourse propulsion and retropropulsion systems. Figure 3-3 shows the estimated dependence of the two propulsion package weights on the bus weight. Including fixed weight, such as a 3000 pound capsule and various adapters and support structures, the relationship between bus weight and estimated planetary vehicle weight is given in Figure 3-4.

Figure 3-3. Estimated Propulsion Weight Versus Bus Weight

Figure 3-4. Estimated Planetary Vehicle Weight Versus S/C Bus Weight

3.3 MISSION PERFORMANCE

Analyses were made of the subsystem performance on each of four opportunities, 1973 to 1979. In each case, a nominal arrival date was used:

<u>Opportunity</u>	<u>Nominal Arrival Date</u>
1973	24 January 1974
1974	16 April 1976
1975	15 June 1978
1977	26 July 1980

The amount of conditioned power available for science is shown as it varies with time in orbit for the 1973 mission in Figure 3-5(a). Curves are shown for the nominal PV/B power system, and for systems with 20 and 40 percent larger solar arrays. The eclipse penalty is assessed against the solar array after the first 30 days in orbit (see Section 3.4). For comparison, the capabilities of the RTG power supplies are superimposed.

The science power shown for the first 30 days in this figure has limited meaning, since the capsule must also be supplied during this period. The trade-off between science power and capsule power averaged for the first 30 days is shown in Figure 3-5(b). It is evident that the more power used for science, the less is available for capsule functions. Even these trade-off curves do not reflect any battery losses, i. e. , it assumes the capsule or science loads are manipulated so that the power output of the primary power source is not exceeded and battery/prime source load sharing does not occur. If the batteries are used during this period, the average power available will decrease.

Figure 3-5(c) shows the cumulative conditioned energy available to the science subsystem after capsule separation (assumed to occur at 30 days in orbit), expressed in watt-months of 30 days each. These curves indicate to some extent the amount of scientific data which can be accumulated by the bus over the life of the mission.

Performance curves similar to these three are also given for the 1975 to 1979 missions. These are indexed below.

Performance Curve	Opportunity			
	1973	1975	1977	1979
	Figure 3-5	Figure 3-6	Figure 3-7	Figure 3-8
Time History of Science Power in Orbit	(a)	(a)	(a)	(a)
Trade-off Between Science Power and Capsule Power for First 30 Days in Orbit	(b)	(b)	(b)	(b)
Cumulative Science Power Available	(c)	(c)	(c)	(c)

Figure 3-9 summarizes the cumulative energy available for science for the 150 days after capsule separation. This is shown as a function of spacecraft weight for each of the mission opportunities.

3.4 ECLIPSE DURATION AND EFFECT ON AVERAGE ARRAY POWER OUTPUT

An analysis was made of the total duration of solar eclipses for a 1000 km by 10,000 km orbit about Mars at inclinations of 40° and 60° using eclipse duration data given in Table 4-3, Occultation Times, in Section VC2200FD101 of the Task B report. The fraction of the total time the spacecraft will be eclipsed was estimated to be 6.1 percent for a 40° inclined orbit, and 5.1 percent for a 60° orbit. A nominal value of 5.6 percent eclipse was selected as being representative of the total eclipsed fraction for a nominal 50° inclined orbit.

The solar array power subsystem was analyzed to determine the effect of eclipses on average power availability. The net efficiency for energy passing through the battery system is 50 percent, allowing for:

- a. Basic battery charge-discharge efficiency (71 percent)
- b. Losses in the battery charge regulator (5 percent)
- c. Battery discharge diode voltage drop (2 percent)
- d. Contingency for overcharge required for a degraded battery (about 20 percent)
- e. Charge current tapering at end of charge (about 10 percent).

Using this value of overall battery system efficiency, the relationship between eclipse fraction and an "array derating factor" was developed and is presented in Figure 3-10. Based on an eclipse fraction of 0.056, the resulting array derating factor, representing the fraction of array power usable for the load, is 0.894. In other terms, if an array is sized based on a given load without eclipses, then for a 5.6 percent eclipse time fraction, the array size must be increased by a factor of $1/0.894$, an increase of 11.8 percent.

This derating factor was not applied to the array power output during the first 30 days in Mars orbit in accordance with the guideline that no eclipses occur during this period.

Figure 3-5. RTG and PV/B Power Performance 1973 Mission

BLANK

(a)

(b)

(c)

Figure 3-6. RTG and PV/B Power Performance 1975 Mission

BLANK

(a)

(b)

(c)

Figure 3-7. RTG and PV/B Power Performance 1977 Mission

BLANK

(a)

(b)

(c)

Figure 3-8. RTG and PV/B Power Performance 1979 Mission

BLANK

(a)

(b)

(c)

Figure 3-8. RTG and PV/B Power Performance 1979 Mission

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Figure 3-9. Accumulated Energy Comparison

Figure 3-10. Usable Array Power Versus Fractional Eclipse Time

4.0 RELIABILITY

The relative reliability of the solar and RTG powered spacecraft are judged below in paragraph 4.1 on a qualitative basis. Insufficient design definition and failure rate data ruled out a quantitative approach originally considered. The qualitative data is sufficiently strong to identify the RTG spacecraft as the more reliable system. The results of a reliability apportionment calculation are presented in paragraph 4.2 to serve as a future goal for specific Voyager RTG designs. An estimate of RTG reliability based on the design considered for the spacecraft defined in Document No. VOY-C1-TR16 is presented in paragraph 4.3.

4.1 COMPARISON

The RTG powered spacecraft is considered to be more reliable than the solar powered spacecraft for the following reasons:

4.1.1 Independence of Solar Orientation

Because of the removal of battery energy limitation, RTG power results in benefit to several aspects of spacecraft operation:

- a. Midcourse Maneuvers - Under abnormal conditions, such as failure to verify the proper vehicle attitude before engine firing, it may be necessary to reacquire the sun, recharge the batteries, in the case of a solar powered vehicle, and reattempt the midcourse correction sequence. RTG's would not require the reacquisition sequence, at least for reason of battery recharging and would permit longer periods for troubleshooting and corrective action. Reliability is enhanced by the potential avoidance of additional maneuvers, and the associated saving of attitude control gas.
- b. Orbit Insertion Maneuver - As differentiated from midcourse correction maneuvers, there is a very limited period during which this maneuver can be executed. It is estimated that with solar power a maximum of 5 hours is available for this maneuver because of battery energy limitations. About two attempts of achieving the proper spacecraft attitude can be made. An analysis has indicated that the solar panels are typically pointed 150 to 175 degrees from the sun so that no battery charging is possible. With RTG power this period can be increased to permit additional attitude attempts. The exact number depends on gyro drift and errors in orbit insertion parameters that would not result in violation of the planetary quarantine constraint. A gross estimate indicates that a ten hour period might be acceptable. Thus the RTG's provide the potential for increased mission reliability by permitting greater flexibility with respect to this critical maneuver.
- c. Loss of Solar Reference - As in the solar powered spacecraft, the RTG powered spacecraft uses a Sun-Canopus attitude reference system. In the case of the solar powered spacecraft, failure of the fine Sun sensors or acquisition Sun sensors could result in a variety of failure modes ranging from minor effects to total mission failure. The question of failure effects is complex and depends on the nature of the failure and conditions at the time of failure. The use of redundant sensors potentially relieves this problem and

the question is then one of how to implement such redundancy. Logic to distinguish between good and bad sensors is not always straightforward and majority vote techniques employing additional sensors may be required. The use of RTG's through their ability to sustain power may permit reductions in such complexity by permitting ground control switchover to redundant sensors by interpretation of telemetry information.

4.1.2 Elimination of High Duty Cycle Batteries

As mentioned earlier spacecraft reliability increases with the removal of battery energy limitations. Besides this, however, batteries themselves, unless properly accommodated, can provide further sources of unreliability. Both silver-cadmium and silver-zinc batteries, the types considered thus far for Voyager, require careful charge and thermal control. The sensitivity of these control parameters is such that failure may occur if their limits are slightly exceeded. Thus very conservative design approaches are used backed up by extensive testing under simulated flight conditions. Even so, there is usually remaining doubt that all possible conditions have been sufficiently simulated. Both silver-cadmium and silver-zinc batteries exhibit limited cycle life capability. This, along with the above reasons, indicates improved reliability with RTG's. Total elimination of batteries is not feasible because of high power momentary loads. Such batteries however experience very few charge/discharge cycles and would therefore be more reliable.

4.1.3 Constant Operating Condition

RTG's operate under constant temperature conditions throughout their life. Elements of the solar power system on the other hand experience varying temperature conditions. The solar array temperature decreases gradually from 160°F to 50°F in going from Earth to Mars. During maneuvers or occultations its temperature may drop to minus 200°F. During eclipse seasons in Mars orbit such temperature cycles (50°F to -200°F) will occur for many successive orbits. The batteries similarly experience variations in dissipated heat. The batteries require close temperature control and therefore the dissipated heat variations will be reflected in cyclic operation of the thermal controllers.

4.1.4 Fixed High-Gain Antenna

The independence of RTG's for solar orientation permits the use of a fixed high-gain antenna. The method for achieving this is described in Document No. VOY-C1-TR16. In effect digital Sun sensors are substituted for a 2-motion antenna gimbal and this is considered to increase spacecraft reliability. This could be achieved with solar power but would require deployed solar panels to provide sufficient power during sun-bias operation.

4.2 RELIABILITY APPORTIONMENT

Mission reliability, R , is defined by the following expression:

$$R = e^{-\lambda t}$$

where λ = the equivalent system failure rate

t = system operating time

The apportionment of reliability to various subsystems is determined by the percentage that each subsystem contributes to the equivalent failure rate, λ . Thus if p_1 is the percentage for subsystem No. 1, then its reliability apportionment is determined by

$$R_{\text{subsystem 1}} = e^{-p_1 \lambda t}$$

The problem is then one of judiciously allocating the various percentages contributing to the net system failure rate. This is done by considering the number of functions performed by each subsystem (F_i) along with the number of dependencies on it of other subsystems (N_i). By proper use of F_i and N_i it is possible to arrive at a reasonable allocation of failure rate percentages. This methodology was used in the Task B report. (Ref. GE Document No. DIN 65SD4514). With slight adjustment resulting from consideration of a fixed high gain antenna the methodology results in an almost identical subsystem reliability apportionment.

This is shown in Figure 4-1 considering the overall system reliability as a variable for a total mission time of about 230 days. Table 4-1 shows the allocations by various mission phases for an overall reliability goal of 0.65.

The subsystems may be investigated using the same methodology. The RTG power subsystem is arbitrarily divided into RTG, regulator, battery and inverter groups. Relative complexity and interaction factors are assigned to each group to result in the percentages contributed by each to the net power system failure rate. Figure 4-2 shows the resulting reliability goal for the RTG group above including shunt elements indicating a value of 0.9904 corresponding to the mission goal of 0.65. With eight RTG's comprising the group, each RTG has a reliability goal of 0.9989.

4.3 RTG RELIABILITY ESTIMATE

The eight-RTG configuration for powering the spacecraft defined in VOY-C1-TR16 was examined for its reliability in delivering adequate power over a 10,000 hour space flight. The results show that the probability of losing more than 1 percent of the power capability via

Figure 4-1. System Reliability Design Goal Versus Subsystem Reliability Design Goal (Sheet 1 of 2)

open and short circuited couples and shorting to the RTG frame of two couples was found to be low. The reliability for delivering in excess of 99 percent power was calculated at slightly greater than 0.993 for the 10,000 hour mission.

The system analyzed herein consists of eight RTG's arranged in a parallel configuration and delivering a total of 600 watts at beginning of life (BOL). To allow for margin and degradation, there is approximately 20 to 25 percent additional power at BOL.

Each RTG employs approximately 500 thermocouples, in a configuration, of the type used in the SNAP-27 generator.

Figure 4-1. System Reliability Design Goal Versus
Subsystem Reliability Design Goal (Sheet 2 of 2)

Table 4-1. RTG Powered Spacecraft

Reliability Design Goals					
Mission Phase →	Launch	Cruise	Orbit Injection and Trim	$R_{\text{system}} = 0.65$	
				Capsule Separation	Orbit Operation
Subsystem Time Duration →	2 Hours	185 Days	15 Days	3 Hours	30 Days
Radio	Design	0.9693	0.9975	0.9999	0.9947
Command	goal	0.9743	0.9979		0.9958
C&S	for	0.9656	0.9972		0.9944
DH&S	Launch	0.9724	0.9977		0.9955
Power	and	0.9773	0.9988		0.9963
Thermal	Injection	0.9812	0.9985		0.9939
G&C Vehicle	= 0.9763	0.9675	0.9974		0.9946
Propulsion (MC)		0.9812	0.9985	0.9999	0.9939
Propulsion (Retro)		0.9724	0.9983	-	-
Autopilot		0.9773	0.9982	-	0.9963
PSP		-	0.9970	0.9999	0.9788
Pyro Control		-	0.9977		0.9831
Vehicle		0.9724	0.9981	0.9999	-
Reliability by Phase	0.9763	0.7211	0.9758	0.9999	0.9484

The objective of this analysis was to assess the eight-RTG configuration for conditions other than zero failures. Four modes by which a single RTG may exhibit failure, catastrophic or degrading, have been considered. The modes are:

- Structural failure resulting in total loss of power.
- Combinations of shorts and non-adjacent open-circuited thermocouples resulting in partial power loss.
- Adjacent open-circuited thermocouples resulting in total loss of power.
- Short-circuiting of two thermocouples to the cold or hot frames resulting in power loss dependent upon the number of couples separating the short-circuited couples.

Figure 4-2. System Reliability Design Goal Versus
RTG Group Design Goal (Including Shunts)

Reliability data used for the analysis are as follows:

a. Structure

The structural reliability for the SNAP-27 generator has been assessed for a 10,000 hour lunar surface mission. This same value will be used for the Voyager generator.

$$R (\text{RTG Structure}) = 0.99998$$

b. Thermocouple

The SNAP-27 thermocouple has been assessed for failure rate on the basis of life test results. The following failure rates have been estimated for the thermocouple:

$$\lambda (\text{open circuit}) = 0.07 \times 10^{-6} \text{ failures/hour}$$

$$\lambda (\text{short circuit}) = 0.07 \times 10^{-6} \text{ failures/hour}$$

$$\lambda (\text{short-to-cold (hot) frame}) = 0.007 \times 10^{-6} \text{ failures/hour}$$

4.3.1 Reliability Analysis

a. Mathematical Models

1. Structure

Structural failure has been assumed as resulting in total loss of the RTG power.

The probability that none of the RTG's will fail structurally is given by:

$$R(\text{RTG Configuration}) = R^8(\text{RTG})$$

Each RTG contributes 12.5 percent to the total output power. Structural failure of one RTG, therefore, represents a loss of 12.5 percent of the delivered power. The reliability of the eight-RTG configuration as a function of delivered power is given by:

$$P(N, S) = \sum_{x=S}^N \binom{N}{x} R^x (1-R)^{N-x}$$

where:

N = Number of RTG's = 8

S = Number of surviving RTG's

R = Structural Reliability of the RTG = 0.99998

P(N, S) = Probability of S survivors of N units

2. Combinations of Short and Non-adjacent Open-Circuited Thermocouples

The total number of couples in the eight-RTG configuration is approximately 4000 or 2000 ladder pairs (N). The probability of 2N-n thermocouples surviving is given by:

$$P(2N, S) = \sum_{X=N-\text{Maxk}}^N \binom{N}{X} P_1^X \cdot \left\{ \sum_y^{N-X} \binom{N-X}{y} P_2^y \cdot P_3^{N-X-y} \right\}$$

where: $0 \leq y \leq N-x$ when $\text{Maxk} \geq 2(N-X)$

$$2 \left(\frac{N-\text{Maxk}-X}{2} \right) \leq y \leq N-x \text{ when } \text{Maxk} < 2(N-X)$$

N = Total number of ladder pairs

Maxk = Maximum allowable open-circuited couples (non-adjacent)

$\frac{\text{Maxk}}{2}$ = Maximum allowable short-circuited couples (1 per ladder pair possible)

$$P_1 = R_o^2 \cdot R_s^2$$

$$P_2 = 2 R_o \cdot R_s^2 \cdot (1-R_o)$$

$$P_3 = 2 R_o^2 \cdot R_S \cdot (1-R_S)$$

R_o = Probability of non-open circuit

R_S = Probability of non-short circuit

$$R_o = e^{-\lambda_o t}$$

$$R_S = e^{-\lambda_S t}$$

$$\lambda_S = \text{Short-circuit failure rate} = 0.07 \times 10^{-6} \frac{\text{Failures}}{\text{hour}}$$

$$\lambda_o = \text{Open-circuit failure rate} = 0.07 \times 10^{-6} \frac{\text{Failures}}{\text{hour}}$$

t = mission time (hours)

3. Adjacent Open-Circuited Thermocouples

The probability of no adjacent open-circuited thermocouples is included in the foregoing expression for short and open combinations.

4. Short-to-Hot or Cold Frame

The probability that an RTG will not suffer a power loss due to two thermocouples short-circuiting to the hot or cold frame is given by:

$$R(\text{Short-to-Cold (Hot) Frame}) = 1 - W \cdot Q_{sc}^2$$

where: Q_{sc} = Probability of couple short-circuiting =

$$(0.007 \times 10^{-6} \times t) = 1 - R_{sc}$$

W = Number of ways two couples can short to cold or hot frame to result in a given power loss.

$$W = 4 \cdot \sum_{X=1}^{N-n} X$$

N = Number of couple pairs representing allowable power loss

N = number of ladder pairs

b. Reliability Calculations

The open, short, and short-to-hot (cold) frame reliability for a thermocouple for various space flight mission times are tabulated below in Table 4-2.

Table 4-2. Reliability Calculations

Mission (hours)	Reliability		
	Open Circuit	Short Circuit	Short-to-Hot (Cold) Frame
1,000	0.99993	0.99993	0.999993
3,000	0.99979	0.99979	0.999979
5,000	0.99965	0.99965	0.999965
8,000	0.99944	0.99944	0.999944
9,000	0.99937	0.99937	0.999937
10,000	0.99930	0.99930	0.999930

Substituting the above reliability values into the appropriate mathematical model the reliability over various mission times for different degrees of power loss may be obtained. The results of these calculations are tabulated in Table 4-3.

Table 4-3. Reliability Summary, Voyager RTG System

Mission (hours)	Probability of Delivering Given Power or Greater					
	99%	98%	97%	96%	95%	94%
10,000	0.99314	0.99316	0.99314	0.99317	0.99320	0.99321
9,000	0.99444	0.99446	0.99447	0.99449	0.99449	0.99450
8,000	0.99562	0.99563	0.99564	0.99565	0.99566	0.99567
5,000	0.99829	0.99829	0.99830	0.99830	0.99830	0.99830
3,000	0.99939	0.99939	0.99939	0.99939	0.99939	0.99939
1,000	0.99993	0.99993	0.99993	0.99993	0.99993	0.99993

Figure 4-3 is a plot of the above values for 94 percent power delivery reliability over the 10,000 hour space flight.

4.3.2 Results and Conclusions

The results of the analysis shows that the eight-RTG configuration possesses a high probability of delivering adequate power over the 10,000 hours mission.

Figure 4-3. Reliability Versus Time for Eight-RTG Configuration

The principle cause of power loss was found to be that of combinations of open and shorted thermocouples. During the process of calculating the reliability for this failure mode it was observed that the reliability approached a limiting value as the allowable power loss, represented by failed couples, was increased. However, this limit was practically reached at approximately 0.75 percent power loss.

As would be expected, the short-to-cold (or hot) frame reliability increased as the allowable power loss, represented by loss of thermocouple groups via two shorted couples, was increased.

The reliability for this mode was minimum at something less than 0.01 percent power loss due to the large number of ways in which failure could occur. From the results, it may therefore be concluded that the probability of power loss, via the three modes of failure, in excess of 1 percent is quite low over the 10,000 hour mission.

Therefore with a margin of approximately 4 percent, in couples, to provide for such contingencies as described herein, the reliability of the eight-RTG configuration would be more than adequate for the intended mission.

5.0 MISSION FLEXIBILITY

RTG and solar powered spacecraft are compared below as they relate to several mission flexibility criteria:

5.1 EFFECT OF SOLAR DISTANCE

The ability of RTG's to operate independently of the sun distance provides a very significant advantage over solar power. Considering a mission to Jupiter it would be necessary to deploy an area of about 1500 square feet to provide 600 watts of raw power. This is about six times the fixed area for current Voyager solar powered designs. Solar array designs are presently being developed for weight of 0.35 to 0.5 pound per square foot and would result in a total array weight of 500 to 750 pounds. Assuming that about 120 pounds of batteries would be required, the solar power system would weigh at least several hundred pounds more than an RTG powered version. Complexity in the solar array design would result from the large variation in generated power in going from Earth to Jupiter and in the need to withstand mid-course correction thrust loads.

In missions toward the sun, solar arrays would ultimately encounter problems of excessive temperature operation. At Venus the problem can probably be adequately handled. In orbital eclipse operation array temperature excursions might be rather severe and would degrade array reliability.

5.2 ORBIT SELECTION

Constraints on planetary orbit selections may be related to avoiding or reducing the extent of battery discharge during solar occultations. Such constraints pertain to solar powered spacecraft and are removed with the use of RTG power.

5.3 LAUNCH AZIMUTHS

Safety questions relating to mission aborts and the possible need to confine the RTG isotope fuel within certain geographical zones may restrict the range of launch azimuths. The implications of such restrictions can only be assessed in terms of a specific mission and associated hazards analysis.

6.0 DESIGN IMPLEMENTATION

The items discussed below are those principally affected by RTG integration in the general area of spacecraft design implementation. Comparisons are made relative to solar spacecraft design implementation in the specific areas identified.

6.1 THERMAL INTEGRATION

The principal thermal problem identified for the RTG's concerns the need to provide under the shroud cooling during prelaunch phases. Such cooling will have to be provided continuously, with only short interruptions, from the time that the spacecraft is encapsulated in its shroud and end diaphragm dirt barriers. Although the solar powered spacecraft may also require cooling to remove ambient heat loads, the problem is much less severe since the RTG's dissipate about 15 kw (+) per spacecraft and the solar powered spacecraft will dissipate from 0 to 300 watts depending on particular checkout sequences.

Assuming that the spacecraft electronics and propulsion unit temperatures are about 80°F to 90°F at launch, no thermal problem has been identified for either spacecraft (solar or RTG) until they are separated and on a heliocentric transfer trajectory. This is based on a maximum 90 minute period from launch until separation.

Similarly, no thermal problem was identified for either spacecraft during free space flight.

Some advantage is seen for the RTG spacecraft with respect to thermal integration of the lander. Previous guidelines indicate that 200 watts of electrical power are needed by the lander more or less continuously during the cruise phase. Short interruptions are permissible during maneuver or occultation periods. Assuming that this power is used principally to maintain temperatures within the lander, it is conceivable that with the design of particular thermal paths in the lander, the RTG's can supply this power thermally rather than electrically. Figure 6-1 is used to illustrate this ability of the RTG's in comparison with a solar powered spacecraft. The left-hand sketch shows a cross-section of the baseline solar powered spacecraft and the right-hand sketch the baseline RTG powered spacecraft. In near-Earth space the solar array temperature is around 160°F. It is possible to imagine that the lander bio-carrier is designed to intercept radiated heat from the back solar array surface such that 200 watts of power, Q_E , is introduced into and distributed within the lander. Both the interception and distribution of this power require intricate thermal design but conceptually this is feasible. No electrical power, P , from the spacecraft would be required during this period.

Figure 6-1. Lander Thermal Integration

In near-Mars space the solar array temperature is around 70°F . Assuming the lander bio-barrier is around a similar temperature no radiative heat transfer is possible and the 200 watts must be supplied electrically.

With the RTG's, on the other hand, the amount of heat intercepted by the lander can be varied by adjusting the thermal impedance of the RTG thermal barrier as shown in the right-hand sketch. Once established, this heat input is constant, assuming the effects of the varying solar input are removed by properly insulating the back surface of the RTG mounting panels.

To summarize the above: With solar power, the transfer of power to the lander by thermal means is not attractive because of the varying solar constant. Since solar electrical power would be required anyway in near-Mars space and is in excess in near-Earth space, it is probably the best means of supplying the lander power as far as a solar powered spacecraft is concerned. Alternately, the constancy of the RTG thermal environment makes the thermal transfer of power to the lander more attractive. The alleviation of a 200 watt electrical requirement for the RTG's during the post-injection period with the lander still on-board would permit more power to be devoted to planetary science.

6.2 RADIATION

Nuclear radiation from the RTG's influence the following areas of spacecraft design:

6.2.1 Spacecraft Equipment

Threshold damage effects are predicted to occur in several types of electronic equipment. The extent of effects is documented in detail in Task C Document Nos. VOY-C1-TR3 and TR5. It is expected that component and circuit derating will adequately solve this problem. Testing of some of the components and circuits predicted to be more sensitive will establish the need for imposing the radiation environment as a general design requirement for spacecraft equipment. Similar though less severe requirements may be necessary for the solar powered spacecraft from two possible sources: (1) the use of RTG's in the lander; (2) solar flare protons. With the occurrence of solar storms, the equivalent neutron effect of the solar flare protons has been estimated to be about 1/4 to 1/2 as damaging as the RTG neutrons.

6.2.2 Science Integration

Twenty-two possible instruments for use on the orbiting spacecraft have been examined with respect to interaction with an RTG power source (See VOY-C1-TR4 and TR11). Of these, six would experience dynamic interference from the RTG radiation and are listed below:

- a. Gamma Ray Spectrometer
- b. Cosmic Ray Telescope
- c. Plasma Probe
- d. Mars Spectroscope
- e. Trapped Radiation Detector
- f. Ion Chamber

All of these instruments are radiation detection devices so that solutions aimed at permitting their use must in some way discriminate between the natural radiation of interest and those emanating from the RTG's. Solutions will in general combine the effects of shielding, distance and possible instrument redesign. If instruments in this category are considered for use on an RTG powered spacecraft, extensive analysis and testing will be required to solve the indicated incompatibility.

The possible use of magnetometers was examined in Document No. VOY-C1-TR7. The results indicated that background fluxes were sufficiently low for magnetometers mounted about 5 to 10 feet from the maximum diameter of the spacecraft. Of particular note is the fact that the magnetic field remains relatively stable throughout the mission independent of

solar distance and occultations. Contrasted with this the solar array contribution to the flux might be expected to vary to a greater extent since higher array current is required in near-Earth space because of lower array voltages and vice-versa in near-Mars space.

6.2.3 Personnel Radiation Exposure

Data contained in the RTG Spacecraft Design Definition Report, VOY-C1-TR16, indicates that unshielded fueled RTG's on the spacecraft will limit the amount of time that personnel can work around the spacecraft during prelaunch ground operations. By delaying fuel loading until the latest possible time and placing protective shielding near the RTG's the problem can be minimized. It does nevertheless constitute additional procedures that will require careful control and monitoring.

6.3 DEVELOPMENT

The principal development requirement for the RTG powered spacecraft concerns reentry protection to assure fuel containment in the event of a near-Earth mission abort. The problems relating to this have been discussed in the Mission Abort Profile Report, VOY-C1-TR14. Since the reentry protection intimately influences the overall RTG designs, both would undoubtedly be developed concurrently. However, the basic operating elements of the RTG could depend heavily on well-developed designs, such as SNAP-27, and in this sense these elements are within the category of available technology. All non-similar elements of the solar powered spacecraft, those principally associated with the power subsystem, also fall within the category of available technology.

6.4 GROWTH

Both the RTG and solar powered spacecraft versions can accommodate power growth. Such growth versions RTG and solar powered spacecraft are shown in Figures 6-2 and 6-3 respectively.

For the RTG version, 50 percent growth is achieved by mounting four additional RTG's around the midcourse propulsion system, at a weight increase of 232 pounds. Due to the proximity of the SIVB Instrument Unit, extra insulation and/or radiation shielding may be required to prevent undesirable interaction due to the RTG's of the lower spacecraft (Figure 6-2). Further growth may be provided by an increase in size of the RTG itself; the spacecraft can accommodate an overall RTG diameter of 45 inches; an estimated power output of 125 watts per RTG gives a total growth capability of 250 percent at a weight increase of 556 pounds.

Figure 6-2. RTG Spacecraft Growth Version

Figure 6-3. Solar Spacecraft Growth Version

For the solar array/version, growth is provided by the deployment of auxiliary panels. The placement and size of these panels is influenced by the following factors:

- a. Antennae viewing requirements
- b. Planet scan platform viewing requirements
- c. Sun sensor viewing requirements
- d. Packaging volume requirements for storage of array panels in the launch configuration
- c. Bending moment limitation of the deployed panel configuration.

Figure 6-3 shows two growth versions for the basic spacecraft configuration described in the Supplement to the Design Definition Report (VOY-C1-TR16A). The first version (b) uses a simple single panel deployment scheme similar to that employed on Mariner. Existing hardware such as hinges, springs, latches and deployment dampers would be utilized with minimum modifications required. The panels are attached by hinges to the closure ring bounding the fixed portion of the solar array and folded back to their stowed launch position behind it. The length of the panel is restricted during deployment by the capsule interface boundary. It is possible that more area than the present 3.9 square foot panel is obtainable by increasing this length. This depends on further definition of the adjacent portion of the capsule.

The second version (c) shown requires a compound deployment. Some of the fixed array panels are required for support of various spacecraft components, and to provide torsional shear capability for the spacecraft support structure. These fixed panels are located in the vicinity of the high gain antenna and the planet scanner. The remaining panels are hinged at the ends adjacent to the spacecraft body and are deployed to a position about 10 degrees past the plane normal to the sun line, allowing secondary panels, stowed underneath and hinged at the outboard edges of the primary panels, to deploy. The primary panels employ a bracing strut with an over dead center locking device to stiffen them against maneuver loads. A sequencing device must be used to prevent premature deployment of the secondary panels.

All of the growth versions discussed above have only been studied on a preliminary basis to establish that growth appears feasible for both the RTG and solar powered spacecraft.

6.5 TEST

The principal items of comparison identified in this area are as follows.

6.5.1 Power Subsystem

Since the heat output of the RTG fuel capsules can be simulated within several percent by electrical heater simulators, the overall power system performance can be accurately measured in ground tests. Of course, these tests could be performed with actual fuel capsules, but the simulators permit unlimited testing without radiation exposure to personnel. Contrasted with this, test of the total solar power system requires a source of solar simulation. This is complicated by the need for a varying solar input simulating the actual flight condition and the need to duplicate predicted panel temperature conditions. These considerations can lead to very sophisticated test equipment. The practice has been to predict solar array output on the basis of terrestrial measurements and careful extrapolation of solar intensity and panel temperatures. Limited power subsystem testing with full solar arrays may be conducted to verify interface compatibility but overall performance mapping is difficult and usually avoided. Solar array simulators are used to duplicate the solar array electrical characteristics for system testing, but here too the simulation is only as good as predicted analysis will permit. In an overall sense ground testing of an RTG power system provides a more accurate appraisal of space performance and therefore increased confidence in its operation.

6.5.2 Radiation Test

With respect to the use of radiation detecting instruments as part of the science payload, verification of their compatibility with the RTG powered spacecraft may result in the need for complicated test arrangements. Of particular concern are the possibilities of secondary radiations from surrounding materials not necessarily a part of the spacecraft – walls, test equipment, etc. The complexity of the problem will depend largely on the nature of the particular instruments being used.

6.5.3 Magnetic Tests

As in the case of power subsystem tests, light stimulation of the solar array would be required to induce currents to permit magnetic mapping of the spacecraft. A number of such tests might be required to determine the effects of varying sun distances and occultations.

With the RTG's these complexities are eliminated since the RTG contribution to the magnetic flux remains constant independent of solar intensity. With actual fuel capsules and proper compensation for the Earth magnetic flux contribution, ground test mapping should closely duplicate the spacecraft magnetic fields in space flight. Some additional compensation might be required if electrically simulated fuel capsules are used because of power leads. With careful attention to minimizing the areas of the enclosed current loops, this problem can be adequately handled.

7.0 PROGRAM IMPLEMENTATION

This portion of the report identifies RTG hardware requirements in relation to the principal project activities. The project schedule was assumed to be identical to that established in the Task B report. It is also assumed that a power system selection, either RTG's or solar power, has been made at the time that the program associated with a particular mission opportunity is underway. Such a selection in favor of RTG's is contingent on the following:

- a. Proof of RTG isotope fuel containment which meets safety criteria objectives. It is estimated that development efforts to prove this capability would take approximately 24 months.
- b. Safety concept approval, and acceptability of possible compromises on mission flexibility, e.g., restrictions on launch azimuths. It is estimated that 12 months would be required for these studies.
- c. Assurance of sufficient isotope fuel availability.
- d. Program acceptance of possible compromises in science instrument selection resulting from the RTG radiation environment. Substantiation of these possible compromises should be based on experimental investigations of science instrument interactions with actual or simulated RTG's.

7.1 PROJECT SCHEDULE

The project schedule developed during the Task B study has been taken as representative of the development cycle for the Voyager system using a conventional solar power source. This schedule is illustrated on the upper portion of Figure 7-1. Calendar times have been removed and only months prior to launch have been shown. It is recognized that specific dates in the schedule were based upon an assumed starting date and a 1971 launch opportunity, but the general time periods apply for purposes of this study.

The lower part of Figure 7-1 indicates the probable activities related to the RTG's which would precede the overall spacecraft project activity. These earlier RTG activities are considered to be necessary based on the present evaluation of RTG applicability to a Voyager spacecraft. The results of other R & D programs, particularly related to the reentry protection problem, could reduce and possibly eliminate the earlier activities shown.

7.2 SPACECRAFT DEVELOPMENT

7.2.1 Development Phase

For spacecraft development, the key point in the program is the Critical Design Review (CDR). Figure 7-2 relates the activities in the development phase leading up to the CDR.

The typical design approach is to provide design and performance requirements for the design engineer who transforms these requirements into drawings and specifications. Up through the development tests on engineering model components and assemblies, this work is accomplished on a fairly independent basis. It is necessary to follow this with a series of tests to integrate these components and assemblies into subsystems and the system. In this activity, the compatibility of the components with other system elements is verified and the subsystem and system performance is evaluated.

In order to reduce risk, it is necessary to make a comprehensive system evaluation before the CDR. In the program shown in Figure 7-1, three system models are used. The Structural Test Model is used to evaluate the structure and the dynamic response of it to such expected inputs as vibration, shock, acoustics, etc. Particular emphasis will be placed upon evaluation of the dynamic environment at various locations on the structure. The Thermal Test Model will be tested to validate the thermal control devices and to determine the thermal environment throughout the system. The Engineering Development Model spacecraft will be used for spacecraft system integration and performance evaluation excluding mechanical and thermal environments.

Integration of the RTG into the system has been considered within this general framework.

Because of the necessity to design the RTG for entry conditions, it is felt that the mechanical integration of the RTG into the system will be straightforward. The RTG assembly will be relatively stiff and it is felt that there is a minimum probability that the spacecraft system will induce loads in the assembly beyond the specified design requirements which will be derived analytically.

Preliminary thermal analysis has shown that no major problems exist for the thermal interactions of the RTG with the Spacecraft; however, due to the relatively high operating temperature of the RTG, its influence on the spacecraft may be significant. A relatively sophisticated thermal model and test program will be required to validate the analysis.

The requirements for development hardware are summarized in Figure 7-3 and discussed below:

a. Development Components

The most critical development component will be the RTG re-entry protection and, therefore, earlier efforts should be initiated for this element. The extent to which

flight testing is required would be determined in the initial phases of such a development effort. Other components of the RTG, such as the thermopile and heat rejection system will be based largely on the SNAP-27 technology and earlier development efforts need not be devoted to these.

b. Subsystem and System Integration

Eight electrically operated RTG's are required to evaluate subsystem and system performance in all modes of operation. It is also expected that two or three fuel capsules will be required to determine the radiation levels at specific spacecraft hardware locations. This will be done by making measurements with the RTG's in various locations and summing the totals. The RTG's must be representative of the design from the electrical performance point of view and in the sense that the proper mass, C.G. location, etc., is present; however, materials can be substituted to reduce cost if desirable.

c. Thermal Models

Eight thermally representative models will be required for use in Thermal Test Model testing. These models will be thermal masses with heaters.

d. Structural Models

Eight structurally representative models will be required for use in tests of the Structural Test Model. These models would be representative in terms of mass properties, center of gravity, and attachment hardware. At least one or two should be suitable for RTG separation system testing if safety studies result in RTG separability as a requirement.

7.2.2 Type Approval Phase

Hardware identified for the Type Approval phase in Figure 7-3 is that principally associated with the thermopile and heat rejection system. Fuel capsules are also required insofar as they would affect the thermopile and heat rejection system during Type Approval tests. It is assumed that qualification of the fuel capsules will have been carried out in conjunction with earlier proof of fuel containment reentry capability. The validity of this assumption depends largely on the method of reentry protection finally selected. If the reentry protection is principally in the form of fuel capsule cladding, the assumption is reasonable. If the thermopile and heat rejection system provide a major part of this protection, additional Type Approval models may be required for specific reentry protection proof tests.

Proof Test Model (PTM) testing for system type approval will require eight RTG's of the flight design which are electrically operated.

7.2.3 Acceptance Testing

The schedule in Figure 7-1 shows the time when acceptance tested RTG assemblies are required for assembly and check-out of the Flight Spacecraft. It is expected that electrical heaters will be used in the assembly and checkout tests and that the actual fuel capsules will be assembled into the RTG during the launch preparations at KSC. Twenty-four RTG's plus two spares are required for the flight spacecraft.

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Figure 7-1. Spacecraft Project Cycle

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Figure 7-2. Project Activities – Development Phase

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HARDWARE USE		USE PERIOD (MONTHS)								LAUNCH	FUEL CAPSULES	ELECTRICAL FUEL CAPSULE SIMULATORS	THERMOELECTRIC & FIN ASSEMBLY	RE-ENTRY PROTECTION UNITS	SIMULATED RTGS	O.S.E.	
		70	60	50	40	30	20	10	0							TEST CONSOLES	HANDLING TOOLS & FUEL CASKS
COMPONENT DEVELOPMENT (INCL. BREADBOARD)											4 - FUELLED 20 - NON FUELLED	4	4	10	—	✓	✓
ENGINEERING DEVELOPMENT MODELS	MOCK UP										—	—	—	—	8	—	—
	SYSTEM										—	—	—	—	8 (ELECTRICAL)	—	—
	STRUCTURAL										—	—	—	—	8 (MASS, C.G.)	—	—
	THERMAL										—	—	—	—	8 (THERMAL)	—	—
COMPONENT TYPE APPROVAL (QUAL.)											2	2	2	2	—	✓	✓
PROOF TEST MODEL (QUAL.)											3 (INCLUDING 2 ABOVE)	8	8	8	—	✓	✓
PRIME											26	24	26	26	—	✓	✓

Figure 7-3. RTG Development Schedule and Hardware Requirements

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