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(123) R.A. O'Kiodan ✓
VOY-CI-TR16A

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SUPPLEMENT TO
VOYAGER RTG SPACECRAFT DESIGN DEFINITION
RTG STUDY - VOYAGER TASK C

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A. PREFACE

This document serves as a supplement to the RTG Spacecraft Design Definition Report (VOY-C1-TR16). Section B considers the spacecraft as a solar powered version using the baseline configuration generated for the RTG powered spacecraft. Section C presents an updating of the information contained in the parent report (VOY-C1-TR16).

B. SOLAR POWERED SPACECRAFT

SECTION 1

INTRODUCTION

This portion of the document considers the spacecraft design described in Report No. VOY-C1-TR16 as a solar powered version. The purpose of this consideration is twofold. First, the solar powered version serves as the model against which the RTG spacecraft will be compared on an overall basis. Secondly, it serves as the basis for determining the degree to which the design is convertible from a solar powered to an RTG powered spacecraft. This latter subject is treated in Section 7.0 of this supplement.

The solar powered version described herein is for the most part functionally identical to the Task B design (Ref. GE Document No. DIN 65SD4514). Differences are principally confined to the arrangement of spacecraft equipment resulting from the objective of attaining an interchangeable design.

SECTION 2

SUMMARY DESCRIPTION

2.1 CONFIGURATION

An isometric view of the solar powered spacecraft together with the flight capsule is shown in Figure 2-1.

The spacecraft general arrangement is shown in Figure 2-2. Antenna and sensor locations along with fields of view are shown in Figure 2-3. The principal differences with respect to the RTG powered spacecraft are as follows:

- a. Solar panels are substituted for the RTG's and associated mounting panels. A fixed projected area of 229 square feet in the minus Z direction (solar direction) is available for solar cell mounting. Solar array output is conservatively estimated to be 720 watts at 1.52 A. U. (estimated value at planetary encounter for the 1973 opportunity). Output at Mars aphelion (1.66 A. U.) is estimated to be 605 watts.
- b. The 90 inch high-gain antenna is articulated for earth-pointing, and is mounted at the periphery of the truss support structure in the region of the minus Y axis.
- c. A backup medium-gain antenna is mounted with its axis fixed at the encounter cone angle and is located at the inner diameter of the support structure in the region of the plus X axis.
- d. A Sun-Canopus attitude reference system is used identical to that of the Task B design. The need for the equipment bay sun shade, shown in the RTG version, is eliminated because of the sun-pointing of the Z axis during all non-maneuver periods.
- e. Electronic equipment changes are principally those in the Power Subsystem and are discussed later. Equipment bay locations and volume allocations are practically identical to that of the RTG spacecraft design.

2.2 FLIGHT OPERATION

Operation of the spacecraft during the various mission phases is identical to that of the Task B design excepting for possible revision subsequent to planetary encounter. The Task B design was principally based on the 1971 opportunity with encounter at 1.42 A. U. The array power capability at encounter was 820 watts. The design described herein assumes application to the 1973 opportunity with an estimated encounter at 1.52 A. U. and an associated array encounter output of 720 watts. The 100 watt difference is based on similar solar cell rating factors for the two cases taking array temperature differences into account because of the configuration change. Since power requirements for the missions are typically highest in the post-encounter period prior to the Flight Capsule separation, the difference in array output

Figure 2-1. Solar Powered Spacecraft

Figure 2-2. Solar Array Spacecraft --
General Arrangement

Figure 2-3. Antenna and Sensor Fields of View

for the two cases points to a possible revision to the post encounter events planned for the Task B design. Such a revision must take into account the possibility of solar occultations just after encounter, and the degree to which the available power can be adequately allocated to the needs of the Flight Capsule and Science. The sensitivity of the post encounter sequence to the A. U. distance at encounter in one comparison aspect of solar and RTG powered spacecraft and will be considered in greater depth in the later Comparison Report.

2.3 MASS PROPERTIES SUMMARY

A weight summary of the planetary vehicle is presented in Table 2-1.

The weight breakdown is by vehicle function with sub-totals shown for the spacecraft bus.

Weight, center of gravity, moments and products of inertia are presented in summary form in Table 2-2 for specific times during the mission. Longitudinal centers of gravity are referenced to Sta 0 shown in Figure 2-2 with positive distances in the direction of +Z. Lateral CG locations are referenced to the roll axis in compliance with the sign convention in Figure 2-1. Each inertia is about the center of gravity.

Table 2-1. Weight Summary

Description		Weight (lb)
Bay No. 1	Power System	83.8
Bay No. 2	Power System	84.3
Bay No. 3	Science Electronics	96.0
Bay No. 4	Science Dae	61.0
Bay No. 5	Guidance and Control	55.5
Bay No. 6	Computer and Sequencer	50.1
Bay No. 7	Command System	38.9
Bay No. 8	Telemetry & Power System	67.7
Bay No. 9	Data Storage	68.4
Bay No. 10	Data Storage	68.4
Bay No. 11	Radio System	54.0
Bay No. 12	Radio System	60.4
Primary Structure		395.6
Scan Platform		150.0
Solar Array System		298.0

Table 2-1. Weight Summary (Continued)

Description		Weight (lb)
A. C. Gas System		247.0
A. C. Independently Mtd. Sensors		16.6
Science Sensors		105.4
Antenna Assemblies		92.1
Thermal Control		151.4
PYROTECHNIC		4.4
Harness		51.5
Total No. 1	Subtotal Bus Weight	2300.5
Retro Propulsion		9285.5
Mid-Course Propulsion		2138.8
Meteroid Protection		71.1
Total No. 2	Subtotal Propulsion Weight	11495.4
Capsule (includes Biobarrier)		3000.0
Total No. 3	Subtotal Capsule Weight	3000.0
Total No. 1 & No. 2 & No. 3	Separated Over-All Planetary Vehicle	16795.9
Adapter		113.3
Planetary Vehicle		16909.2

Table 2-2. Mass Property Summary

Flight Sequence Description	Case No.	Weight (lbs)	$\bar{Z}$ (in.)	$\bar{X}$ (in.)	$\bar{Y}$ (in.)	I_{zx} (slug ft ²)	I_{zy} (slug ft ²)	I_{xz} (slug ft ²)	I_{oz} (slug ft ²)	I_{ox} (slug ft ²)	I_{oy} (slug ft ²)
Launch Cond. (On PAD)	1	16908.9	114.7	0.22	0.32	-26.0	-0.4	26.0	7536	21855	21592
Transit	2	16795.6	114.5	0.17	0.34	-28.2	0.8	6.5	7326	21716	21440
After Mid Course Correction	3	15427.6	121.3	0.19	0.37	-32.5	-7.6	6.5	6925	19569	19343
After Retro Burn (Orbiting)	4	7290.6	151.9	0.40	0.79	-51.6	-45.5	6.2	6344	15718	15492
P. S. P. Deployed	5	7290.6	152.5	0.89	2.50	-52.2	-43.7	159.2	6949	16209	15462
After Capsule Separation	6	4790.6	89.1	1.33	3.80	34.6	205.3	157.4	3411	2950	2207
After Orbit Adjust. Ant. Depid.	7	4484.6	93.1	1.19	3.14	10.9	117.5	215.1	3558	2961	2005

Notes 1 NOM C.G. on 3000 pound capsule

SECTION 3

DESIGN SOLUTION

3.1 GUIDELINES AND DESIGN CRITERIA

The guidelines used for the solar powered spacecraft are identical to those for the RTG powered spacecraft. Array power requirements were assumed to be those developed for the Task B design with its associated mission profile.

3.2 DESIGN EVOLUTION

Design evolution proceeded simultaneously with that of the RTG spacecraft as discussed in the Design Definition Report (VOY-C1-TR16).

SECTION 4

FLIGHT SEQUENCE OF EVENTS

The flight sequence of events is identical to that of Task B (See VC 220 FD 112, Vol A (book 1 of 2) of the Task B report).

SECTION 5

VEHICLE DESIGN

5.1 STRUCTURE

The structural design is shown in the general arrangement drawing Figure 2-2. The primary structure, selection rationale, and supporting analyses are identical to those of the RTG design.

5.2 EQUIPMENT BAYS

The spacecraft electronic equipment arrangement is shown in Figure 5-1. The arrangement is identical with the RTG design with the exception of the weight and power levels of bays 1, 2 and 8 due to additional batteries and power conditioning equipment.

The elliptical sun shade shelf at STA 37 on the RTG version is not required. The cylinder which encloses the MC & OA engine provides equipment bay shading up to two degrees of Z axis misalignment with the sun line.

5.3 WEIGHT STATEMENT

A detailed weight statement is presented in Table 5-1.

5.4 THERMAL ANALYSIS

The solar powered spacecraft design will perform thermally as described in the Task B thermal report both on-the-pad and in parking orbit. The primary difference from a thermal standpoint between the Task B design and the design described herein will occur when the array is subjected to a solar input. For this reason, thermal analysis was performed for phases of the mission after shroud ejection.

5.4.1 SOLAR ARRAY TEMPERATURE DISTRIBUTION

The design was modeled as shown in Figure 5-2 and analyzed on the digital computer in order to evaluate the solar array temperature distribution. Included in the analysis are the effects of conduction along and through the array, radiation from the back of the array to the flight capsule and space, and radiation from the array surface to the electronics bays and space.

Figure 5-3 shows the steady-state solar array temperatures that will result at different locations on the array during various phases of the mission. Figure 5-4 shows the radial temperature distribution along the solar array as a function of sun distance. A comparison of these

Figure 5-1. Electronic Equipment Bay Arrangement

Table 5-1. Detailed Weight Statement

Item	Weight (lb)	Item	Weight (lb)
BAY 1		BAY 5	
POWER SUBSYT		G + C SUBSYTM	
BATTERY	43.00	AC ELECTRONIC	8.60
MAIN REG.	11.50	AUTO PILOT	6.00
CHARGE REG.	1.50	ACCELMO GYRO	14.50
3 PHSE INVERT	3.50	ACCELMO + GYO	14.50
2.4 KC INVERT	4.50	BAY NO 5 WIRE	5.70
1 PHSE INVERT	1.50	EQUIPMENT CHS	5.47
PWR SW + LOGC	5.50	GYRO TEMP SEN	0.10
BAY NO 1 WIRE	6.70	ACCEL TMP SEN	0.10
EQUIPMENT CHS	5.47	BAY TEMP SEN	0.10
BATT TEMP SEN	0.10	SHUTT ANG DET	0.40
BAY TEMP SEN	0.10		
SHUTT ANG DET	0.40	BAY 6	
BAY 2		COMP + SEQCER	
POWER SUBSYS		COMPUTING + SQ	35.00
BATTERY	43.00	SOLAR ASPECT	5.00
MAIN REG.	11.50	EARTH NULLS	2.00
CHARGE REG.	1.50	BAY NO. 6 WIRE	2.10
3 PHSE INVERT	3.50	EQUIPMENT CHS	5.47
2.4 KC INVERT	4.50	BAY TEMP SEN	0.10
1 PHSE INVERT	1.50	SHUTT ANG DET	0.40
SYNCHRO + CLK	6.00		
BAY NO 2 WIRE	6.70	BAY 7	
EQUIPMENT CHS	5.47	COMMAND SSY	
BAY TEMP SEN	0.10	COMM DET P C	4.20
BAY TEMP SEN	0.10	COMM DET P C	4.20
SHUTT ANG DET	0.40	COMM DET P C	4.20
BAY 3		COMM DECODER	5.60
SCIENCE ELECT		COMM DECODER	5.60
SCIENCE ELEC	90.00	DEC ACES SWIT	3.00
EQUIPMENT CHS	5.47	POWER SUPPLY	3.10
BAY TEMP SEN	0.10	BAY NO. 7 WIRE	3.00
SHUTT ANG DET	0.40	EQUIPMENT CHS	5.47
BAY 4		BAY TEMP SEN	0.10
SCIENCE DAE		SHUTT ANG DET	0.40
DATA AUTO EQ	55.00		
EQUIPMENT CHS	5.47		
BAY TEMP SEN	0.10		
SHUTT ANG DET	0.40		

Table 5-1. Detailed Weight Statement (Continued)

Item	Weight (lb)	Item	Weight (lb)
BAY 8		BAY 11	
TELEMETRY & POWER		RADIO SUBSYTM	
PYRO CONTROLER	7.50	PWR AMP TWT	7.50
TELEMETRY ELE	16.60	PWR AMP TWT	7.50
BAY NO. 8 WIRE	1.10	POWER AMP TWT	3.00
EQUIPMENT CHS	5.47	TRANSPONDER	13.30
BAY TEMP SEN	0.10	DIPLEX + SW	13.40
SHUTT ANG DET	0.40	BAY NO. 11 WIRE	3.70
BATTERY	35.00	EQUIPMENT CHS	5.47
CHARGE REG.	1.50	BAY TEMP SEN	0.10
BAY 9		BAY 12	
DATA STORAGE		RADIO SUBSYS	
TAPE RECORDER	18.00	TRANSPONDER	13.30
TAPE RECORDER	18.00	TRANSPONDER	13.30
TAPE RECORDER	18.00	RELAY RADIO	25.00
CONTROL LOGIC	0.40	BAY NO. 12 WIRE	2.80
CONTROL LOGIC	0.40	EQUIPMENT CHS	5.47
CONTROL LOGIC	0.40	BAY TEMP SEN	0.10
PLAYBACK SEQ	0.60	SHUTT ANG DET	0.40
POWER SUPPLY	2.00		
BAY NO. 9 WIRE	4.60	PRIMARY STR	
EQUIPMENT CHS	5.47	SHEAR PANELS	89.10
BAY TEMP SEN	0.10	CAP SEP RING	9.30
SHUTT ANG DET	0.40	TAP LONGERON	35.00
BAY 10		KICK FRAME	38.90
DATA STORAGE		EQUIP CHANNEL	26.40
TAPE RECORDER	18.00	SUPPT ANGLE	1.04
TAPE RECORDER	18.00	SUPPT ANGLE	0.44
TAPE RECORDER	18.00	OUTER CAP	6.90
CONTROL LOGIC	0.40	TAPERED STRUT	94.00
CONTROL LOGIC	0.40	MISC	4.13
CONTROL LOGIC	0.40	FITTINGS STR	31.40
PLAYBACK SEQ	0.60	HARDWARE STR	38.00
POWER SUPPLY	2.00	SEPARATION FG	21.00
BAY NO. 10 WIRE	4.60		
EQUIPMENT CHS	5.47	METEROID PROT	71.10
BAY TEMP SEN	0.10		
SHUTT ANG DET	0.40	PSP STOWED	150.00

Table 5-1. Detailed Weight Statement (Continued)

Item	Weight (lb)	Item	Weight (lb)
SOLAR ARRAY		SCIENCE SENOR	
BAY 1	11.25	MISC SCIENCE	95.00
1	11.25	MAGNETOMETER	10.00
2	11.58	TEMP SENSORS	0.40
2	11.58	ANT ASSEMBLIES	
3	11.58	RELAY ANTENNA	2.00
3	11.58	ASSOCIATED ST	2.00
4	11.58	HI GAIN ANT S	46.50
4	11.58	ACTU + STR	33.50
5	11.58	LO GAIN ANT P	0.80
5	11.58	LO GAIN ANT S	0.80
6	11.58	MEDIUM GAIN	4.40
6	11.58	MTG STR	2.00
7	11.58	TEMP SENSORS	0.10
7	11.58		
8	11.58	THERMAL CONTL	
8	10.55	SHUTTER FACE	10.80
9	10.55	SHUTTER FACE	10.80
9	10.55	SHUTTER FACE	10.80
10	10.55	INSUL + FACE	10.80
10	11.58	SHUTTER FACE	10.80
11	11.58	SHUTTER FACE	10.80
11	11.58	SHUTTER FACE	10.80
12	11.58	SHUTTER FACE	10.83
12	11.25	SHUTTER FACE	10.83
PANEL TEMP SN	0.20	SHUTTER FACE	10.80
ARRAY HARNESS	25.00	SHUTTER FACE	10.83
		INSUL + FACE	10.83
AC GAS SYSTEM		INSUL AC SYS	2.50
GAS + TANK	46.40	PAINT	4.00
GAS + TANK	46.40	INSUL ORBITER	15.20
GAS + TANK	46.40		
GAS + TANK	46.40	PYROTECHNIC	
HARDWARE	47.40	PIN PULLERS	2.20
GAS SYS MTG SEN	14.00	SQUIBS	2.20
AC IND MTG S		HARNESS	
SUN SENSORS	2.60	MAIN HAR RING	35.00
CANOPUS TRACK	6.30	RELAY CABLE	1.00
CANOPUS TRACK	6.30	HI GAIN CABLE	4.00
MISC MTG STR	1.20	LO GAIN CABLE	5.00
TEMP SENSORS	0.20	MEDIUM G CABLE	2.00
		UMBILICAL	4.50

Table 5-1. Detailed Weight Statement (Continued)

Item	Weight (lb)
RETRO PROPULSION	
RETRO ENGINE	9192.00
TAPERED STRUT	60.80
KICK RING	12.50
TUBLAR STRUT	16.40
KICK RING	3.80
MID COURSE	
MONO ENGINE	516.75
TANK SUPPORT	12.95
MONO ENGINE	516.75
TANK SUPPORT	12.95
MONO ENGINE	516.75
TANK SUPPORT	12.95
MONO ENGINE	516.75
TANK SUPPORT	12.95
FRAMES	20.00

Figure 5-2. Thermal Analysis Nodal Breakdown

Figure 5-3. Solar Array Temperature Variation vs. Distance from Sun

Figure 5-4. Temperature Distribution Along Solar Array

results with the solar array temperatures calculated in the Task B report reveals an average increase in array temperature of about 20 degrees F, more or less independent of the A. U. distance. This results from reduced panel backface radiation to space.

5.4.2 EFFECT OF SOLAR ARRAY ON EQUIPMENT BAYS

The arrangement of the solar array allows the equipment bay heat rejection surfaces to receive heat directly from the array. This absorbed heat will cause a change in the shutter performance, particularly in the partially open position. An analysis was performed to determine the effect of the energy from the array on the heat rejection surfaces. The results of this analysis are shown in Figure 5-5 in which the heat absorbed by the equipment bays is shown as a function of shutter angle for blades pivoted both toward and away from the solar array. Inspection of this figure indicates that for the condition where the blades are pivoted from the edge away from the array (Case 1), a large heat input will be absorbed by the bays at small shutter angles. Pivoting the shutters from the edge closer to the array (Case 2) has the advantage of absorbing less heat from the array at small shutter angles. When the shutters are fully open, performance is equivalent for either approach. During space flight, the shutters of most of the bays will be in the partially open position. Therefore, to minimize additional heating of the bays, a shutter system with the shutters pivoted from the edge closer to the array was selected.

5.4.3 CRUISE AND ORBIT

Steady-state spacecraft temperature profiles have been calculated both near Earth and near Mars and are shown in Figures 5-6 and 5-7. Inspection of these figures reveals that spacecraft temperatures are highest near Earth. These higher temperatures are a direct result of thermal radiation from the solar array on the spacecraft. Since the solar array temperatures will decrease as the spacecraft approaches Mars, the effect of the solar array on the spacecraft will diminish and the overall spacecraft temperature profile will be lower as shown in Figure 5-6. During the entire cruise phase of the mission, all spacecraft temperatures are tolerable and no thermal problems are anticipated.

During Mars orbit, the effect of the solar array on the spacecraft is quite small and it is therefore expected that during this phase of the mission, the spacecraft will perform thermally as described in the Task B thermal report.

Figure 5-5. Effect of Solar Array on Equipment Bays

Figure 5-6. Voyager Spacecraft Temperatures Near Earth

Figure 5-7. Voyager Spacecraft Temperatures Near Mars

SECTION 6

SUBSYSTEM DESIGN

This section describes the significant differences in subsystem design required for the solar powered spacecraft version.

6.1 POWER SUBSYSTEM

6.1.1 SCOPE

The primary differences between the Task B power subsystem and that described herein occur in the solar array and battery designs. Secondary differences also occur in the detailed design of the power conditioning units. The subsystem arrangement is identical to Task B. A simplified diagram is shown in Figure 6-1 along with revised bus voltage levels reflecting changes discussed below.

6.1.2 SOLAR ARRAY

The solar array for the interchangeable spacecraft has more cells than the Task B array, but approximately the same power output, since the array is mounted at a solar incidence angle of 24 degrees. The predicted power output of the solar array, as a function of Sun-spacecraft distance, is shown in Figure 6-2. Combining this data with trajectory data yields the relationship between array power output and time during the cruise and orbit phases as shown in Figure 6-3. Detailed voltage — current and voltage — power curves are given for the complete range of Sun-spacecraft distances in Figure 6-4 and 6-5 respectively.

Power is drawn from the solar array at 43 volts or higher for those conditions when the array output is higher than the load demand. This represents a decrease from the 52 volt level used on the Task B design and was selected for two main reasons. First, layout studies of the twenty-four half-panels produced the most efficient packaging with solar cell strings having a number of series elements which yield optimum power capability at 43 volts. This occurs during late phases of the mission when the power margin is least. Second, the reduced level results in a comparable reduction in the regulated output of the main regulator to 20 volts DC. This is compatible with RTG output and is therefore a desired level from an interchangeability standpoint. This consideration is discussed in greater detail later.

Figure 6-1. Simplified Power Subsystem Schematic

Figure 6-2. Solar Array Power Output vs. Sun Distance

Figure 6-3. Solar Array Output vs Mission Time

Figure 6-4. Solar Array Voltage-Current Characteristics

Figure 6-5. Solar Array Power-Voltage Characteristics

Considering an associated change in the number of battery cells in series, also discussed later, the unregulated bus voltage (array-battery bus) range is reduced from 32-62 volts, for the Task B design, to 26-50 VDC.

The layout of the solar cells is shown for typical panel sections in Figure 6-6. The solar cell string arrangement is summarized below:

String type:	A	B	C	D	E
Cells in series:	103	103	112	112	115
Cells in parallel:	5	7-1/2	8	9	6
Number of strings:	51	17	3	3	8
Number of cells:	26,265	13,132	2688	3024	5520

The total number of cells (2 cm x 2 cm) is 50,629. The net active area of cells is 207.1 ft², and taking into account the angle of solar incidence, the net projected active area of solar cells is 189.1 ft². In spite of the fact that the projected area of the array is greater than the Task B array, its power output is approximately the same because of the higher temperature resulting from the increased thermal radiation blockage caused by locating the array closer to the lander.

The solar array performance characteristics (voltage-current and power-voltage) are based on performance factors used for the Task B design. A computerized technique is used to predict the overall array characteristics and takes into account the series and parallel arrangement of solar cells, panel temperature distributions plus initial cell efficiency and derating factors. These are listed below for reference purposes:

Bare cell efficiency (N-P type) @ air mass = zero
and cell temperature = 25°C 11%

SHORT CIRCUIT CURRENT DEGRADATION FACTORS

Filter transmission loss	.94
Micrometeoroid affects and random failures	.95
Radiation	.879
Uncertainties, effect of	.9265

(uncertainties # is calculated from rms of
mismatch and manufacturing 2%
measurement uncertain 5%
U. V. 5%)

SHORT CIRCUIT CURRENT DEGRADATION FACTORS (Continued)

Variation in solar intensity

$$=(A. U.)^{-2}$$

VOLTAGE DEGRADATION FACTOR

Radiation

.981

6.1.3 BATTERY

The reduction in unregulated bus voltage requires a reduction in the number of cells in series. For the same power profile as in Task B, the ampere-hour capacity of the cells must be increased. The revised battery size is summarized:

	<u>Task B</u>	<u>This Design</u>
Ag-Cd Batteries	2	2
Cells in Series	33	27
Cell Capacity	20 A-hr.	25 A-hr.
Charge Voltage Limits	A: 50.4 V B: 48.8 V C: 47.3 V	A : 40.5 V (charge) A': 38.3 V (float) B : 39.0 V (charge) B': 36.9 V (float)
Ag-Zn Batteries	1	1
Cells in Series	24	20
Cell Capacity	50 A-hr.	60 A-hr.
Charge Voltage Limits	A: 46.5 V B: 44.6 V C: 42.7 V	A : 38.8 V (charge) A': 37.4 V (float) B : 36.9 V (charge) B': 35.5 V (float)
Charge Current Limit (sum)	3.75 amp.	4.6 amp.

The battery energy capacity, weight, volume, and general description will be approximately the same as the Task B batteries.

Figure 6-6. Solar Panel Layouts

6.1.4 BATTERY CHARGE REGULATORS

The battery charge regulators will be changed from that for Task B in three areas: the general voltage limit range will be lowered to accommodate the reduced number of cells in series in both types of batteries; a provision will be included to reduce the charge voltage to the open-circuit voltage after charging is complete; and the charge current limits will be increased as noted previously.

The results of tests made during MSD's Battery Performance Test Program (DA L05-31) have shown that gas evolution in both Ag-Cd and Ag-Zn cells is reduced to negligible levels by lowering the charge voltage imposed on the cells to their open-circuit voltage after the cells are completely charged. The voltage combinations found to be most advantageous for securing full charge with minimum gassing are:

Cell type	Ag-Cd	Ag-Zn
Charge voltage per cell	1.50 v	1.94 v
Float voltage per cell	1.42 v	1.87 v

A two-step charger to perform this function was developed, built, and is being utilized as part of this test program. The charger is still basically a current-limit, voltage-limit charger as described for Task B. Using this type of charger, the battery voltage early in charge is low, while the current is being limited. Later in the charging period, the voltage limit is imposed, and the charge current will taper off as a consequence. The two-step charger senses this current tapering, and when the current has fallen to half of its limit value, the charger will automatically switch to the lower float voltage limit. A subsequent discharge will reset the charger for the next charge cycle through the action of the charge current interruption.

The voltage limits imposed by the charger are indicated in the battery description, A and B being the charge voltage limits, A' and B', the float voltage limits. A and A' are for the full complement of cells, and B and B' are provisions to allow for the short failure of one cell in series, to be set by ground command.

6.1.5 POWER CONDITIONING EQUIPMENT

A major parameter in selecting the design of the voltage regulator and inverters is the voltage level chosen for the regulated bus. One solution calls for the use of as high a regulated voltage as possible, in this case 24 volts, limited by the minimum battery discharge voltage, 26 volts. An alternative is to consider a 20-volt bus, to make the inverters interchangeable between the RTG and PV/Battery power systems. The trade off between these two voltages is affected by efficiency considerations.

Lowering the output voltage of the series switching regulator from 24 volts to 20 volts will result in a net loss of regulator efficiency estimated to be slightly less than 2 percent. Lowering the input voltage to the inverters would also result in a loss in efficiency for the same basic design. However, where a 24-volt input would be marginally too high, the 20 volt input will permit the use of germanium switching transistors in the output stages of the inverters, instead of the silicon transistors selected for the Task B study. This change would result in a net increase in inverter efficiency of approximately 3 percent in going from a 24-volt regulated bus to a 20-volt regulated bus. Thus, the overall efficiency of the regulator-inverter combination would be slightly higher using a 20-volt bus.

The regulated bus voltage was selected to be 20 volts, since this also allows interchangeability of the inverters between the RTG power subsystem and the PV/Battery power subsystem. The main (series switching) regulator would be essentially the same as for Task B, with design changes in the voltage reference and output filter to accommodate the lower 20-volt regulated output.

The inverters transistor output stages would be redesigned from the Task B designs to utilize germanium switching transistors. Design changes would also be required in the transformers to accept the lower input voltage.

It is estimated that the overall main regulator - inverter efficiency will be the same as for Task B.

6.2 OTHER SUBSYSTEMS

Aside from the Power Subsystem, other subsystems are essentially identical to those described in the Task B report.

SECTION 7

INTERCHANGEABILITY SUMMARY

The items described below are those principally affected in converting the spacecraft from a solar powered version to an RTG powered version. No consideration is given to items outside of the spacecraft, e. g. the need for additional ground cooling or modification to OSE. These have been previously discussed in the parent report (VOY-C1-TR16).

7.1 POWER

a. Power source - Replace solar panels with RTG's and associated mounting panels.

b. Bay No. 1 Equipment.

1. Replace 43 lb. Ag-Cd battery with 14 lb. Ag-Cd battery.
2. Modify battery charge regulator for smaller battery.
3. Replace Main Regulator (Buch) with battery discharge boost regulator.
4. Install RTG shunt circuits and partial shunt controls.

c. Bay No. 2 Equipment

1. Replace 43 lb. Ag-Cd battery with 14 lb. Ag-Cd battery.
2. Modify battery charge regulator for smaller battery.
3. Replace Main Regulator (Buch) with battery discharge boost regulator.
4. Install RTG shunt circuits.

d. Bay No. 8 Equipment

1. Remove 35 lb. Ag-Zn battery.
2. Remove battery charge regulator.

e. Load Interface

The following chart compares bus voltages for the solar and RTG powered spacecraft and indicates the range for which user equipment must be designed to achieve interchangeability:

Bus	User	RTG Spacecraft	Voltage Solar Powered Spacecraft	User Design Range
B	Radio, Capsule	20 VDC $\pm$ 1%	25-50 VDC	20-50 VDC
A	Intermittent loads: Solenoids, thrust vector control, etc.	25-40 VDC	25-40 VDC	25-40 VDC
C	Spacecraft and Science Loads	2.4 KHz, 1 ϕ 50V rms
D	Spacecraft and Science Loads	2.4 KHz, 1 ϕ 50V rms
E	Gyros	400 Hz, 3 ϕ 26V rms
F	Gyros	400 Hz, 3 ϕ 26V rms
G	Science	400 Hz, 1 ϕ 28V rms

7.2 GUIDANCE & CONTROL

- a. Replace null Sun sensors with digital Sun sensors.
- b. Bay No. 5 Equipment - Modify Attitude Control Electronics to accept digital Sun sensor input and C&S Sun-bias input.
- c. Relocate sensor mounting block from +X to -X location.

7.3 COMPUTER & SEQUENCER

- a. Incorporate logic modifications for Sun-bias mode of operation.

7.4 COMMUNICATIONS

- a. Bay Equipment (bays 7 to 11) — Logic and circuit modifications to accommodate:
 1. Communications modes of operation associated with Sun-biased attitude control system.
 2. Revised telemetry format (e. g. , RTG inputs replace solar panel and battery inputs).

3. Revised command requirements.

- b. High-gain antenna - Replace articulated HGA with fixed HGA.
- c. Medium-gain antenna - Replace back-up antenna used at encounter with mid-course medium gain antenna.

7.5 STRUCTURE

Install shelf to provide equipment bay shading during Sun-biased operational mode.

7.6 SCIENCE

Incorporate radiation shielding, extension booms, or thermal shielding as may be required for specific instruments for compatibility with the RTG radiation environment.

7.7 HARNESS

Modification required for above items.

C. REVISIONS TO TASK C DOCUMENT NO. VOY-C1-TR16

C.1 POWER SUBSYSTEM

The subsystem design of both the RTG and the photovoltaic/battery power subsystems for the Task C spacecraft have been reviewed, and some modifications have been made in the design of the RTG power subsystem.

The RTG power subsystem, as originally conceived, contained a topping battery to supply load peaks, such as solenoids, and the thrust vector control. The voltage level (consequently, the number of cells in series) was selected so the battery could be easily charged from the main bus using a simple dissipative charge regulator. Since the battery was there, it was decided to use it also as a main-bus back-up in the event of an RTG failure, and a switching boost regulator was specified to boost the battery discharge voltage up to maintain a regulated bus, though less tightly regulated than when the RTG's alone were powering the main bus.

In view of the desirability for maximum interchangeability between the RTG and PV/Battery power subsystems, the battery voltage selection was reviewed. Numerous parts and units in several subsystems are affected by the choice of battery bus voltage: solenoid valves, the antenna drive, control valves, heaters, and the thrust vector control. Most of these loads are designed to use an unregulated source, but must have a reasonable limit placed on the range of voltage which they will accept. To require this equipment to accommodate a change of battery bus voltage from a 26-40 v range (for the PV/Battery supply) down to a 12-18 v range (for the RTG supply) would result in extensive design changes. In order to avoid the necessity for design changes, at least in this equipment, it has been decided to increase the battery bus voltage, and the number of cells in series, in the RTG supply, to be the same as for the PV/Battery supply described in Section B, paragraph 6.1 of this supplement.

The increase in the battery bus voltage requires two major changes in the RTG supply. The battery charge regulator must be relocated; and the battery discharge regulator must be changed to a buck (series-switching) regulator.

The changes in Section 6, Subsystem Design, of Document No. VOY-C1-TR16 which are affected by these revisions in subsystem design are identified by paragraph.

Figure 6-1. The changed portion of the subsystem block diagram is shown in Figure 6-1A. Note that battery charging power is drawn from the 2400 cps AC bus instead of the regulated bus.

6.1.3.1 Power Flow

A. Bus A.

Supplies raw battery power at approximately 26 to 40 VDC, corresponding to the extremes of battery charge and discharge voltage, for momentary peak loads. DC power, instead of AC power, is distributed to these loads, since their associated peak power levels would require unnecessarily high ratings for the AC invertors. The loads are identified as solenoid valves, stepper motors, and the thrust vector control. Two silver-cadmium batteries of 27 cells in series supply this bus.

6.1.3.3 Battery Charging

The battery charge regulators are the current-limiting, voltage-limiting, series - dissipative type, identical to the BCR's for the Ag-Cd batteries of the PV/Battery power subsystem with the exception of a lower current limit setting. Thus it should be possible to use the same qualified battery charger hardware for both types of power subsystem. The battery charger obtains its power through an external 43-volt transformer/rectifier from the 2400 cps AC bus. It is possible to use this relatively less efficient source of conditioned power for battery charging since the total battery charge energy requirements are low.

There are two charge voltage settings (A and B) obtainable by ground command, and two associated float voltage settings (A' and B') obtained automatically when the charge current falls near the end of the charging period:

Setting A : 40.5 v (charge at 1.5 v/cell, 27 cells)
A': 38.34 v (float at 1.42 v/cell, 27 cells)
B : 39.0 v (charge at 1.5 v/cell, 26 cells)
B': 36.92 v (float at 1.42 v/cell, 26 cells)

The lower, float voltage setting minimizes gas evolution in the batteries during the long period of non-use until the Mars orbit insertion maneuver.

Figure 6-1A. Alterations to Voyager RTG Power Subsystem Functional Block Diagram

6.1.3.6 RTG/Battery Load Sharing

Normally, there will be no load sharing between the RTG's and the battery, since the RTG's produce a power output large enough to supply all the needs of the spacecraft. The only power taken from the battery will be drawn directly from the separate battery bus for the thrust vector control, and for small loads such as solenoids and heaters. The battery discharge regulator will be used for high power, low duty cycle science loads, if such are installed, or in the event of failure of one or more RTG's.

The voltage reference of the buck regulator is set slightly below the main bus. Thus, inability of the RTG's to supply the load will cause a small drop in the main bus voltage, allowing the battery to discharge onto the main bus through the buck regulator. Load management must be such as to limit the battery depth of discharge to about 50 percent and to permit periodic recharge.

6.1.4.1 Batteries

Two separate silver-cadmium batteries are used, discharged in parallel, and charged independently. Each battery consists of 27 cells in series of 5 ampere-hour capacity, weighing about 14 pounds, and occupying about 0.12 cubic feet of volume. The cells are hermetically sealed in stainless steel cases, each with a positive terminal isolated from the case by a ceramic bushing. The case of the cell is the negative terminal. The battery case is of aluminum.

The nominal battery discharge voltage is 29 volts; maximum charge voltage is 40.5 volts. The batteries are sized to provide (with 100 percent redundancy) sufficient stored energy to operate the science package for up to two hours at a time in the event of an RTG failure.

6.1.4.2 Battery Charge Regulator

The battery charge regulator (BCR) is provided its power from the 2400 cps AC bus through a separate transformer/rectifier. It will be a current-limit, voltage-limit, series-dissipative regulator, similar to the BCR of Task B, capable of providing four charge voltage limits to the battery. Two of these voltage limits can be set by command, and correspond to a charge voltage of 1.5 volts/cell, for 27 or 26 cells. The other two are float voltage settings of 1.42 volts/cell, and are switched in automatically when the batteries are fully charged.

The 26-cell voltage settings are to provide for a continued operation of a battery in the event of a short circuit failure of a single cell.

There is an interlock between the BCR and the battery discharge regulator (BDR) so that when the battery discharges through the BDR, the BCR is turned off, thus preventing the inefficient circulation of power through the BDR, the 2400-cps inverter, the T/R, and the BCR.

6.1.4.7 Battery Discharge Regulator

The battery discharge regulator is a pulse-width, modulated series – switching regulator which reduces the battery discharge voltage down to the main regulated DC bus voltage.

To prevent battery discharge when the RTG's are fully capable of providing the required power, an interlock is incorporated in the BDR. The Main (partial shunt) Regulator and the BDR are operated from the same reference supply, a small differential being subtracted from the Main Regulator reference to obtain the BDR reference. When the main DC bus voltage is above the BDR reference, the BDR will supply no power to the main bus. The main regulator must go slightly out of regulation before the BDR permits the battery to discharge onto the main bus.

Since the BDR is intended to be a backup device itself, no redundant unit is used.

C.2 OTHER REVISIONS

Additional revisions to VOY-C1-TR16 are described below by page & paragraph number.

Pg. 6-21

delete first para.

add: "If such failure probabilities are sufficient high, it will be necessary to provide the RTG's with omnidirectional re-entry capability. In the absence of specific probability goals, the separable approach has been adopted for reference design purposes.

Pg. 7-1

near bottom of page: delete "Nominal 6 KW thermal."
add: "Nominal power level is 1.75 kw thermal per fuel capsule."

Pg. 7-19 last line: delete "(require cryogenic cooling)

Pg. 7-20 top of page: delete "Glass dosimeter" and "(requires cryogenic cooling)"

Para. b.: delete the entire paragraph down to section 7.4.3.5

add: "These instruments will serve as safety monitors during isotope-fueled ground operation and will not be used after launch."

Pg. 7-22 Subpara. d.: delete "Detection of unsatisfactory radiation levels, which could result from secondary radiation from contaminants, or from leaks."

Add: "Detection of unsatisfactory radiation levels

Pg. 7-22 middle of page: delete entire paragraph starting with "radiological safety of personnel"

Pg. 7-23 Table 7-3: delete the heading "PAD"

Add heading "Permissible Quarterly Dose"

Pg. 7-23 middle of page, Note 1 & 2: delete note 1 & 2

add: "1. Can be increased to 3.0 rems provided the lifetime PAD < 5(N-18) where N is the age in years.
2. A restricted area is one to which access is strictly controlled."

Pg. 7-31 Table 7-4 In "Insertion/Removal of Fuel Capsule" under the Remarks column, add: "Must also be capable of reverse operation."