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BASELINE CONFIGURATION SELECTION
RTG STUDY - VOYAGER TASK C

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SECTION 1

CONFIGURATION SELECTION

In Bi-Monthly Progress Report No. 3, it was stated that the goal of the selection process would be the identification of two versions of the spacecraft configuration, one an optimized RTG design, the other an interchangeable solar array/RTG design. These two goals have now been combined and met by the configuration shown in Figure 1-1, which has been selected as best fulfilling RTG spacecraft system requirements; all future vehicle integration study and analysis will be performed on this baseline configuration.

Figure 1-1. Baseline Configuration

SECTION 2

METHOD OF SELECTION

The design has evolved from a configuration selected from the comparison matrix discussed in the bi-monthly report. The various rating and weighting factors have been assigned to the designs and criteria to complete the matrix (Figure 2-1), with results indicating that Configuration No. 4 best meets the overall criteria.

The two major uncertainties which existed before this selection could be made were:

- a. The choice of a cone or truss spacecraft support, upright or inverted, in terms of which structure provided a greater fraction of useful payload in Mars orbit.
- b. Concern with tip-off rates that can be anticipated during shroud and spacecraft separation for the Task B spacecraft design.

These items have been investigated and detailed reports are presented in Appendices A and

B. Briefly, the results indicate that:

- a. There is no significant gain or loss in payload for either method of spacecraft support.
- b. A definite spacecraft separation clearance problem exists with the Task B design.

DESIGN CRITERIA	WEIGHTING FACTOR w_i	RATING FACTOR r_i	$w_i \times r_i$ SCORE	COMMENTS	RATING FACTOR	SCORE	COMMENTS	RATING FACTOR	SCORE	COMMENTS	RATING FACTOR	SCORE	COMMENTS	RATING FACTOR	SCORE	COMMENTS	RATING FACTOR	SCORE	COMMENTS
1. THERMAL INTEGRATION																			
a. SOLAR INPUT SENSITIVITY	1.0	1.0	1.0	SHADED ELEC. & RTG	0.9	0.9	SHADED RTG	0.7	0.7	NO SHADE	0.7	0.7	NO SHADE	0.7	0.7	NO SHADE (RTG IN WAY)	0.7	0.7	NO SHADE (RTG IN WAY)
b. LANDER THERMAL ENVIRONMENT	4.0	1.0	4.0	CONST. RTG HEAT INPUT	1.0	4.0	CONST. RTG HEAT INPUT	1.0	4.0	CONST. RTG HEAT INPUT	1.0	4.0	CONST. RTG HEAT INPUT	0.7	2.8	NO RTG HEAT INPUT	0.7	2.8	NO RTG HEAT INPUT
c. INTERACTION SENSITIVITY	9.0	0.6	5.4	RTG CLOSE TO ELEC.	1.0	9.0	MAX DIST OF RTG/ELEC.	1.0	9.0	MAX DIST OF RTG/ELEC.	0.8	7.2	TRUSS DISTORTION	0.4	3.6	TRUSS DISTORTION	0.6	5.4	RTG CLOSE TO ELEC.
d. RTG HEAT UTILIZATION	1.0	0	0	N/A	0	0	N/A	0	0	N/A	1.0	1.0	A/C GAS	1.0	1.0	A/C GAS	0	0	N/A
e. RTG CONVECTIVE HEATING	2.0	0.9	1.8	RTG HIGH	1.0	2.0	ISOLATED ELEC./RTG	0.9	1.8	RTG HIGH	0.9	1.8	RTG HIGH	0	0	RTG LOW	0.1	0.2	RTG LOW
2. SHROUD AND SPACECRAFT SEPARATION																			
a. SHROUD TRANSLATION DISTANCE	9.0	0.9	8.1		0.9	8.1		0.6	7.2		1.0	9.0	MINIMUM	0	0	MAXIMUM	0.9	8.1	
b. SPACECRAFT/SHROUD CLEARANCE	8.0	0.9	7.2		0.9	7.2		0.6	6.4		1.0	8.0	MAXIMUM	0	0	MINIMUM	0.9	7.2	
3. ARTICULATION																			
a. PLANET SCAN PLATFORM VIEWING	6.0	0.4	2.4		0.4	2.4		0.9	5.4		1.0	6.0	OPTIMUM	0.8	4.8		0.4	2.4	
b. ANTENNA GIMBALING COMPLEXITY	4.0	0.6	2.4		0.7	2.8		0.5	2.0		1.0	4.0	MINIMIZED	0	0	MOST COMPLEX	0.8	3.2	
4. EQUIPMENT LOCATION																			
a. ANTENNA REQUIREMENTS	5.0	0	0	BLOCKS RTG & ELEC.	0.8	4.0	BLOCKS RTG	0	0	EXHAUST PLUME	1.0	5.0	NO ELEC. HEAT REJ. BLOCK	0.6	3.0		0.7	3.5	
b. SUN SENSOR REQUIREMENTS	9.0	0	0	BOOMS REQUIRED	0	0	BOOMS REQUIRED	1.0	9.0		1.0	9.0		1.0	9.0	BOOMS REQUIRED	0	0	BOOMS REQUIRED
5. RTG CONSIDERATIONS																			
a. HEAT REJECTION VIEW FACTOR	6.0	1.0	6.0		1.0	6.0		0.9	5.4		0.3	1.8	TRUSS, CAPSULE	0.5	3.0		1.0	6.0	
b. SAFETY	6.0	1.0	6.0		1.0	6.0		0.9	5.4		0	0	RTG MUST CLEAR TRUSS	0	0	RTG MUST CLEAR TRUSS	1.0	6.0	
c. DAMAGING RADIATION	8.0	0	0	ELEC., CAPSULE PROXIMITY	0.5	4.0		0.5	4.0		0.5	4.0		0.8	6.4		1.0	8.0	
d. DESIGN AND POWER LEVEL FLEXIBILITY	7.0	0.2	1.4		0.2	1.4		0.8	5.6		0.8	5.6		1.0	8.0	MOST GROWTH CAPABILITY	0.2	1.4	
6. STRUCTURAL CONSIDERATIONS																			
a. LENGTH OF PV AND SHROUD	3.0	0.5	1.5	CONE SUPPORT	0.5	1.5	STIFFER CONE SUPPORT	1.0	3.0		1.0	3.0		1.0	3.0		0.5	1.5	CONE SUPPORT
b. MARS ORBIT PAYLOAD	4.0	0.9	3.6		1.0	4.0	DIRECT LOAD PATHS	0.5	2.0		0.8	3.2	DIRECT LOAD PATHS	0.8	3.2		0.9	3.6	
c. MASS AND INERTIA PROPERTIES	1.0	1.0	1.0	COMPACT MASS DISTRIBUTION	0.9	0.9		0.8	0.8		0.8	0.8		0.8	0.8		0.9	0.9	
7. SERVICEABILITY	5.0	0.8	4.0		0.7	3.5	POOR ACCESS TO ELEC.	0.9	4.5		1.0	5.0	EXCELLENT ACCESS	0.6	3.0		0.6	3.0	
8. PRODUCIBILITY	2.0	0.8	1.6	A/C GAS STERILIZATION	0.8	1.6	A/C GAS STERILIZATION	0.5	1.0	TRUSS MORE COMPLEX	0.5	1.0	TRUSS MORE COMPLEX	0.5	1.0	TRUSS MORE COMPLEX	0.8	1.6	PROBLEM WITH A/C GAS STERILIZATION
	$\Sigma w_i = 100$	Σr_i	57.4			69.3			72.8			80.1			53.3			65.5	

Figure 2-1. Configuration Selection Matrix

SECTION 3

INTERCHANGEABLE SOLAR ARRAY/RTG SPACECRAFT

The matrix also indicates that designs 5 & 6, intended as interchangeable versions, are generally inferior to the "non-interchangeable" candidates. Because of the desirability of the interchangeability feature, Configuration No. 4 was further examined with this end in view. The main problem, the provision of a fixed solar array of adequate area, is dealt with by decreasing the diameter of the electronic equipment torus, necessitating an increase in its length and also relocation of midcourse fuel tankage. Solar cells on honeycomb substrates may then be located on the inclined support truss members, thus providing torsional shear capability for the structure. This arrangement provides an array projected area of approximately 264 square feet. For further growth, deployable panels may be provided as shown (Figure 3-1).

The vertical location of the spacecraft support point at the shroud (Launch Vehicle Interface) is selected to satisfy several competing characteristics:

- a. A lower support point results in less array surface area (for constant value of projected area), and provides ideal space for body-mounted science instruments and room for the aforementioned auxiliary panel deployment.
- b. A lower support aggravates the clearance problem by increasing the separation distance.

The conversion of solar array to RTG versions is accomplished by removing solar cells on their honeycomb substrates and replacing with similar panels to which the RTG's are attached, plus some secondary support structure to handle the point loading of the RTG itself.

Figure 3-1. Solar Array Growth

SECTION 4

POSSIBLE DESIGN EVOLUTIONS

Two sketches of possible evolutions of the design are presented. The first (Figure 4-1) shows a liquid propulsion system (modified LEMDE). One of the problems with this type of system is the thermal control of the fuel and oxidizer in its tanks of large surface area, and attendant critical high and low temperature limitations.

Preliminary calculations indicate that adequate thermal control may be provided utilizing the heat source from the radiating back surface of the solar cells (in the solar array version), or the waste heat from the RTG's (in the RTG version). The compartment containing the tankage must be suitably insulated.

The second sketch (Figure 4-2) shows an alternative design utilizing a fixed high gain antenna which remains pointed to Earth, which will require the spacecraft roll axis to be biased off the sun line by (typically) 30 degrees. This biased Sun-Canopus reference system is being evaluated elsewhere in the RTG Task C Study as one of the alternate orientation systems.

The RTG version is a simple adaptation of the baseline design, with a sun shade provided for the electronic equipment module. For the solar array version of this design, the considerable shadow which would be cast by the body of the spacecraft on the array of the baseline design necessitates extra area for solar cells; this may be provided on the shelf as indicated, which also serves as a sun shade for the electronic equipment module.

Figure 4-1. Liquid Propulsion System

Figure 4-2. Biased Sun - Canopus Design

SECTION 5

FUTURE WORK

Study and analysis of the selected baseline configuration has been initiated on both the solar array and RTG versions and will be reported in the Design Definition Report to be issued March 1, 1967.

APPENDIX A

SPACECRAFT SEPARATION STUDY

A.1 INTRODUCTION

The most salient feature of an over-the-nose separation which might have a significant influence on Planetary Vehicle design is the problem of clearing the Planetary Vehicle without impact. For the Task B design (similar to Configuration 5 of Figure 2-1), the shroud separation plane could reasonably be at or near the solar panel plane. In such a case the shroud must travel approximately 12 feet in the axial direction before clearing the capsule with an initial radial clearance measuring a few inches. If the shroud separation plane is raised to the vicinity of the capsule, then that portion of the shroud left behind forms a "gun barrel" out of which the Planetary Vehicle must be ejected without impact.

To examine the significant features of this problem in a preliminary way, we assume the separation event occurs during a coast phase of the trajectory (earth orbit) and conservatively take the relative motion as planar. It is further assumed that the mass and moments of inertia of the one separating item are much smaller than that of the remaining vehicle and also that the launch vehicle has no angular velocity at the initiation of separation.

A.2 KINEMATICS OF CLEARANCE

Figure A-1 illustrates the model used for shroud separation. We take the case where the center of mass of the shroud has been given axial and angular separation velocities, u , ω , relative to the planetary vehicle at time $t = 0$. Our interest is the condition under which Point A, the lower inside corner of the shroud, will contact Point B, the upper projection of the Planetary Vehicle.

Figure A-1. Separation Model

The kinematic relations giving the axial and transverse displacement of Point A, relative to its initial position, are:

$$x = L (1 - \cos \omega t) - R \sin \omega t + ut$$

$$y = R (1 - \cos \omega t) + L \sin \omega t$$

where (as shown on Figure A-1) L is the axial distance between the separation plane and the shroud center of mass, and R is the radius to the initial position of Point A.

For the case where $\omega t \ll 1$:

$$\cos \omega t \approx 1$$

$$\sin \omega t \approx \omega t$$

so that to this approximation:

$$x = ut - R\omega t$$

$$y = L\omega t$$

The critical condition is given by:

$$x(\bar{t}) = \ell$$

$$y(\bar{t}) = d$$

where ℓ is the axial distance between the separation plane and the obstruction point, d is the radial clearance and $\bar{t}$ is the time of contact. Then:

$$\ell = u\bar{t} - R\omega\bar{t}$$

$$d = L\omega\bar{t}$$

Solving the first of these equations for $\omega\bar{t}$ and eliminating $\omega\bar{t}$ from the second equation gives:

$$\omega\bar{t} = d/L$$

$$\frac{\omega}{u} = \left(\frac{L\ell}{d} + R \right)^{-1}$$

The first of these equations allows one to check the approximation that $\omega t \ll 1$. The second equation gives the critical tip-off rate in radians of tip per foot of axial travel. If the tip-off rate experienced is greater than the critical value, contact will occur. If it is less, clearance will be accomplished. These same equations apply to Planetary Vehicle separation when R is taken as the radius of the solar panels and L as the axial distance between the vehicle separation plane and Planetary Vehicle center of mass; ℓ remains the axial travel to the obstruction and d the initial radial clearance.

Figures A-2 and A-3 show the critical tip-off rates for the shroud and Planetary Vehicle as functions of ℓ and d . The curves are nonlinear for the shroud since, as the separation plane is raised, both L and ℓ vary.

It is of interest to note the relative tip-off rates obtained from flight data of the Nimbus A and Nimbus C spacecrafts during separation:

	Weight (lb)	I (Slug-in ²)	u (ft/sec)	ω/u (°/ft)
Nimbus A	800	29,000	4	0.12
Nimbus C	900	31,000	4	0.05

These relatively low tip-off rates were a result of very careful balancing of the Nimbus spacecrafts. The proposed Voyager Planetary Vehicle separation velocity of 2 feet per second and maximum tipping rate of 3 degrees per second imply a tip-off rate of 1.5 degrees per foot.

A.3 RELATION OF TIP-OFF RATE TO DESIGN PARAMETERS

The separation motion is taken to be caused by impulsive forces, so that:

$$P = Mu$$

$$T = I\omega$$

where P , T , M and I are the axial impulsive force, the impulsive turning moment, and the mass and moment of inertia, respectively. Dividing the second of these equations by the first gives:

$$\frac{T}{P} = \frac{I}{M} \left(\frac{\omega}{u} \right)$$

Figure A-2. Shroud-Clearance Relationship

Figure A-3. Planetary Vehicle Relationship

There are a number of off-design conditions that will give rise to an impulsive torque, T . We consider here two sources of torque.

The first torque considered is that due to the eccentricity, $e'R$, of the center of mass from the centerline. This gives rise to a torque, T' , of:

$$T' = e'RP$$

where P is assumed to act through the centerline.

The second source of torque considered is the nonuniform distribution of impulse around the separation joint. We take the distributed impulse per unit angle to be:

$$p(\varphi) = \frac{P}{2\pi} + e'' \frac{P}{2\pi} \sin\varphi$$

where φ is angular position as measured from a reference radial line, and e'' is zero to peak variation in terms of fraction of the average impulse.

This nonuniform impulse distribution gives rise to a torque:

$$T'' = \frac{e''R}{2} P$$

Taking both torques to act collinearly leads to:

$$R \left(e' + \frac{e''}{2} \right) = \left(\frac{I}{M} \right) \frac{\omega}{u}$$

which relates these eccentricity factors to the tip-off ratio they produce. It should be noted that for uniform density, the factor I/M does not depend on the material used, but rather is a geometric factor related to the radius of gyration.

Assuming a 3-inch center of mass offset and a peak-to-peak impulse variation of 10 percent gives the following results for the tip-off rate and clearance ratios of the shroud and planetary vehicle:

	$\frac{\omega}{u}$	$\frac{d}{\ell}$
Shroud	$\sim 0.2^\circ/\text{Ft}$	~ 0.045
Planetary Vehicle	$\sim 0.7^\circ/\text{Ft}$	~ 0.08

The equivalent Nimbus center of mass offset for a tip-off rate of 0.12 degree per foot is:

$$e'R \approx 0.2 \text{ inch}$$

A.4 SUMMARY

The results indicate that over-the-nose shroud separation may be feasible only if the shroud (and/or the spacecraft) is very carefully balanced, along with careful control of the separation forces. Even under ideal circumstances the very nature of the dimensions involved leads one to be cautious. It appears advisable to consider configurations in which the axial distance between the solar panels and maximum capsule diameter is reduced.

APPENDIX B

SPACECRAFT SUPPORT WEIGHT TRADEOFF STUDY

In order to determine if there is a significant weight advantage favoring a particular planetary vehicle support concept, a weight tradeoff study was performed. The configurations considered were: (1) Configuration 1 of Figure 2-1 with 50 degree and 60 degree support cones, (2) Configuration 4 of Figure 2-1, and (3) Configuration 4 mounted upside-down in the launch vehicle. In addition to changes in weight of the support structure and the primary structural elements, the different configurations result in different Planetary Vehicle lengths, and therefore in different shroud weights. Since the shroud is not carried into Mars orbit but ejected either in Earth orbit or after the second SIVB burn, changes in its weight do not have the same effect on payload weight as changes in support weight. It becomes necessary to first establish the effect on payload weight in Mars orbit of changes in the various mass items so that they can be compared properly.

B-1 EFFECT OF VARIOUS MASS CHANGES ON THE MARS ORBIT PAYLOAD MASS FOR CONSTANT TRAJECTORY PROFILE

This problem can be considered a type of staging where we hypothesize the velocity and trajectory profiles to be nearly unchanged from one mass staging profile to an alternative mass staging profile.

The differences in vehicle mass as a consequence of different staging profiles are compensated for by predesigned fuel loadings and, in the ultimate payoff, by variations in the final deployed payload.

Consider more than one alternative mass staging profile. For each such profile consider n possible staging events where some mass is discarded. These events may be either during propelled or nonpropelled flight.

Let F stand for expendable propulsion mass, P stand for payload and M stand for all of the remaining vehicle mass. Let the subscripts $1, 2, \dots, n$ on F and M represent the conditions immediately prior to a mass staging event. Let the superscripts $A, B, C, \dots$ represent

alternative staging profiles just prior to the first staging event; the masses for alternative profiles will be equal.

$$F_1^A + M_1^A + P^A = F_1^B + M_1^B + P^B$$

If, however, that part of M which is staged at event 1 is not equal for both alternatives A and B, then just prior to the next staging event the total masses will not be equal:

$$F_2^A + M_2^A + P^A \neq F_2^B + M_2^B + P^B$$

Likewise, for subsequent events:

$$F_n^A + M_n^A + P^A \neq F_n^B + M_n^B + P^B$$

Consider next the acceleration vector equation describing the vehicle response between staging events, where staging events are either events where some mass is deleted or where propulsion is changed from one propulsion system to the next.

$$-\left(\frac{dF}{dt}\right)(I_{sp}) (g) = \left[F(t) + M + P \right] \left[\frac{dv}{dt} + g(t) \right]$$

where

$$g = \text{constant} = 32.2 \text{ ft/sec}^2$$

$$g(t) = \text{local gravitational field intensity}$$

and where the constants and variables $\frac{dF}{dt}$, I_{sp} and M have the values appropriate to the staging sequence.

Let $m = F(t) + M + P$. Then:

$$\frac{dF}{dt} = \frac{dm}{dt}$$

This equation can be rewritten as:

$$dv + g(t) dt = -I_{sp} g \frac{1}{m} dm$$

or as:

$$v_n - v_{n-1} + \overline{g(t)} \times (t_n - t_{n-1}) = -I_{sp} g \ln\left(\frac{m_n}{m_{n-1}}\right)$$

where $\overline{g(t)}$ is the average value of $g(t)$, taking the antilog:

$$\frac{m_{n-1}}{m_n} = \exp \left[\frac{v_n - v_{n-1} + \overline{g(t)} \times (t_n - t_{n-1})}{I_{sp} g} \right]$$

or as:

$$\frac{m_{n-1}}{m_n} = \exp \left[\frac{\Delta v + \overline{g(t)} \times \Delta t}{I_{sp} g} \right]$$

The second term in the exponential, $\overline{g(t)} \times \Delta t$, can probably be neglected for some of the thrusting intervals.

Consider now, in a reverse sequence of events, how variations in vehicle mass staging will affect the final payload.

- a. Inert Weight in Orbit - This can be considered a unity mass ratio. It is evident that any inert weight which must be placed in orbit will replace payload weight on a pound for pound basis.

- b. Mass Staged Subsequent to Midcourse Maneuvers but Prior to Mars Orbit Insertion -
Here the mass ratio will be:

$$\left(\frac{\text{Total Vehicle Mass prior to Orbit Insertion}}{\text{Mass in Mars Orbit}} \right) = \exp \left[\frac{\Delta v + \overline{g(t)} \times \Delta t}{I_{sp} g} \right]$$

Here we assume:

$$\Delta v = 2.45 \text{ km/sec} \approx 7,600 \text{ ft/sec}$$

We also assume that this Δv is intended to be the specific force integral rather than an inertial velocity. Hence we need not include the $g(t)$ in the mass ratio.

$$I_{sp} g = \left(\frac{292 \text{ lb}}{\text{lb/sec}} \right) \left(32.2 \text{ ft/sec}^2 \right) = 9,080 \text{ ft/sec}$$

$$\left(\frac{\text{Vehicle Total Mass prior to Orbit Insertion}}{\text{Change in Mass in Mars Orbit}} \right) = \exp \frac{9,600}{9,080} = 2.32$$

The interpretation of the above discussion is that, for every pound that we can save by jettisoning subsequent to the last midcourse maneuvers but prior to orbit insertion, we can increase our payload in orbit by a factor of $\frac{1}{2.32}$ or about 43 percent.

Repeated applications working backward through the mission, allows calculation of the mass ratios across each event. This gives the following values relating to change in payload in Mars orbit.

$$\frac{\text{Change in Payload in Mars Orbit}}{\text{"Inert" Mass in Mars Orbit}} = 100\% \text{ (Trivial)}$$

$$\frac{\text{Change in Payload in Mars Orbit}}{\text{Part of Shroud Mass Inserted into Mars Transfer}} = 43\%$$

$$\frac{\text{Change in Payload In Mars Orbit}}{\text{Nose Fairing Mass Ejected in Parking Orbit}} = 14\%$$

$$\frac{\text{Change in Payload in Mars Orbit}}{\text{Nose Fairing Mass Ejected Between S-II and S-IV Stages}} = 9\%$$

$$\frac{\text{Change in Payload in Mars Orbit}}{\text{Nose Fairing Mass carried to 500,000 ft.}} = 3.3\%$$

$$\frac{\text{Change in Payload in Mars Orbit}}{\text{Nose Fairing Mass Carried to 350,000 ft}} = 2.9\%$$

B-2 EFFECTS OF VARIOUS SUPPORT CONCEPTS ON MARS ORBIT PAYLOAD MASS

Using the Task B design as a reference, the effect on payload weight in Mars orbit (for two vehicles) was estimated for specific support configurations. The supports and primary structural elements were sized to sustain a 10g vertical loading, 2g lateral loading and a 50 radian per second² torsional loading. The results are shown in tabular form below:

	<u>Change in Mars Orbit Payload Weight (Pounds)</u>	
	RTG Design	Solar Panel Design
Configuration 1 60 degree cone	86	-164
Configuration 1 50 degree cone	8	-242
Configuration 4	2	-137
Configuration 4 Inverted in launch vehicle	-19	-160

Comparisons directly with the Task B design should not be made, since that design has undergone more detailed dynamic and stress analyses while the designs tabulated here have not. Comparisons between the RTG and solar panel designs should not be made without adding in the weight of the RTG and its support structure.

There appears to be no significant difference in payload weight in Mars orbit between the various support concepts examined.