NUCLEAR-POWERED GPS SPACECRAFT DESIGN STUDY

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SECTION I

INTRODUCTION

1.1 OBJECTIVES

This is the final report of a study to investigate the potential benefits of a nuclear (radioisotope) -powered satellite for advanced phases of the Global Positioning System (GPS) program. The study extended over approximately 11 months and 6000 hours of technical effort, and was jointly sponsored and managed by the SAMSO GPS Joint Program Office (YE) and ERDA/NRA Office of Space Applications.

A number of "critical parameters" were suggested by the SAMSO study manager (Capt. M. Marcucci) as parameters in which improvement is desired in advanced phases of the program. These parameters are not all comparable in terms of importance, scope, or relevance to nuclear-powered designs. Nevertheless, as ultimately elaborated and modified, they formed the bases for most of the design decisions which were made in the study. The critical parameters were defined as follows:

1. Power to user
2. Mean mission duration of spacecraft
3. Spacecraft orbital predictability
4. Thermal control of on-board frequency standards
5. Spacecraft vulnerability

The first four of these provided the major influences to the study, with parameters 1, 3 and 4 becoming most relevant to the question of nuclear power. The study was dedicated solely to an elaboration of the capability and consequences of the nuclear-powered approach.

The first phase of the effort was devoted to a study of the capability of a retrofit design approach, i.e., where nuclear power replaces solar power in the phase 1 navigation development satellite (NDS) of the GPS program, without major changes to the remainder of the spacecraft. This exercise enabled a detailed evaluation to be made of the factors which influence the critical parameters in the NDS satellite as well as in the nuclear-retrofit cases. This led to certain modifications in the study ground rules and suggested the design approaches which would be most fruitful in the second phase of the study. The second phase was devoted to the preliminary design of all-new reference nuclear-powered GPS spacecraft. The nuclear-powered spacecraft which were so evolved would be totally compatible with the existing NDS spacecraft, so that a mixed system of solar-and nuclear-powered spacecraft can exist within the program simultaneously.

The bulk of this report is devoted to the all-new nuclear-powered version. The retrofit designs are described in the Appendix (Appendix K).

1.2 GROUND RULES

The ground rules which were established at the start of the study are as follows:

1. Spacecraft weight limit of 1200 lbs. without AKM.
2. Spacecraft volume limit based on Atlas-F capability.
3. Orbit is 12-hour GPS orbit; modifications for operation to 120-hour orbit to be described, if possible.
4. Study to emphasize improvements in critical parameters, as defined.
5. Spacecraft vulnerability considerations limited to x-rays; survival level = JCS standard.
6. Weight analysis to be based on $\phi 1$ NDS where applicable.

The ground rules were added to and modified following the first phase briefing on 17 November 1976, as follows:

7. All new reference designs to be based on both Organic-Rankine and Brayton-cycle power systems, with no further consideration to be given to thermoelectric systems for the advanced (non-retrofit) designs.
8. Power system heat sources to be based on 2400 Watt_T and 3200 Watt_T versions of MHW heat source.
9. Configuration and weight ground rules to be compatible with Shuttle/IUS launch with four spacecraft per launch.
10. Retain individual spacecraft interface ring at 36-inch diameter.
11. Retain possible back-up launch vehicle capability of Titan III/IUS.

Converting to the Shuttle/IUS as the baseline launch vehicle has the effect of extending the weight cutoff above 1200 pounds per spacecraft. However, a lower weight is beneficial since larger plane changes can be accomplished by the IUS as a continuous inverse function of payload weight (Figure 1.2-1). This, in turn, allows a greater number of missions to share the launch, and presumably the cost of the shuttle flight, with the GPS spacecraft on a two-stage IUS.

1.3 SUMMARY AND CONCLUSIONS

A nuclear-powered retrofit version of the phase 1 NDS appears feasible, using either advanced selenide radioisotope thermoelectric generators, an Organic-Rankine-cycle generator, or a Brayton-cycle generator. For each case, a custom-designed generator configuration is required which differs in varying levels of detail from the generators currently in development. The following benefits would result, however, from the retrofit approach:

1. Increased power-to-user: enables achievement of high-power mode throughout satellite design life of 7 years.
2. Improves frequency standard temperature stability.

Figure 1.2-1. Shuttle/IUS Performance

3. Improves spacecraft orbital predictability somewhat.
4. Results in extended mean-mission-duration, exceeding 5.5 years.
5. Minimizes development costs of new spacecraft if nuclear power is desired

The limited improvements in critical parameters which would result from the retrofit approach, and the requirement for custom-designed power systems with the increased development costs for these, led to consideration of new spacecraft designs specifically tailored to make use of the best features of the advanced nuclear power systems, with such systems configured as closely as possible to those in current development.

Reference designs were developed for both Organic-Rankine and Brayton-cycle generators. Both spacecraft are similar in external features and overall design approach, differing only in various internal details of the power systems modules. To maximize orbit predictability, the spacecraft were designed to

approach the ideal uniform sphere as closely as possible in a practical cylindrical shape. All external surfaces are covered with the same material with highly predictable thermal and reflective optical properties. To further enhance ephemerides predictability, the spacecraft slowly rotate in orbit, thereby minimizing the effects of solar-force variations.

To assure thermal uniformity and minimum temperature variation in orbit, all internal spacecraft equipment is cooled via the power system fluid loop. Despite the close thermal and mechanical coupling of spacecraft electronic equipment with the power system, the two may be fabricated, assembled, and tested completely separately, and joined after the separate assembly and test processes are completed. Further testing can be accomplished with electrical heater(s) in place of nuclear heat source(s), which need not be mated to the spacecraft until both heat sources and spacecraft are delivered to the launch site.

The resulting values for the critical parameters which are predicted for the reference nuclear GPS (N/GPS) spacecraft are compared to those of the NDS spacecraft in Table 1.3-1. Substantial improvements in power-to-user and in orbit predictability are achieved, and significant improvements in each of the other critical parameters are predicted.

Table 1.3-1. Summary of Improvements in Critical Parameters

PARAMETER	N/GPS	Φ 1 NDS	Δ
1. POWER TO USER, dBw	L1:-154 L2:-163	-160 -166	+6 +3
2. MEAN MISSION DURATION, YRS.	5.5+	4.5	+1+
3. ORBIT PREDICTABILITY, M/DAY	0.03-0.3	2.5	+ x10+
4. MAXIMUM CLOCK TEMPERATURE VARIATION, °C/ORBIT WITHOUT CONTROL	2	2	—
WITH HEATER CONTROL	0.2	—	+x10
5. VULNERABILITY, JCS	1+*	0.01	N/A

*POWER SYSTEM ONLY

A spacecraft weight of slightly over 1000 pounds is calculated. The IUS payload weight for four spacecraft plus adapter is approximately 4400 pounds.

These results apply to both Brayton and Organic Rankine -powered versions of the reference design. A significant conclusion of the study is that neither design differs substantially from the other in any of the major parameters. However, if the superalloy Brayton system is used instead of the baselined columbium system, a reduction of some 5lb in the L1(P) link results. This may be partially or fully recovered by an increase in the weight and volume of the Brayton system reference design.

Significant legacy from earlier phases of the GPS program is achieved in the navigation, attitude control, reaction control and telemetry subsystems. The reference design may be readily adapted to higher orbit operation with no apparent penalty, requiring only modifications to the antennas and to the Earth sensor.

Other than the power systems themselves no advanced or non-flight-qualified technology is required or postulated for the reference nuclear-powered GPS spacecraft. One of the two radioisotope dynamic power systems is now planned to be space qualified in the Air Force Space Test Program in calendar year 1981-2.

SECTION II

CRITICAL PARAMETERS AND PROGRAM GOALS

2.1 POWER-TO-USER

The current Phase I GPS navigation development satellite ($\phi 1$ NDS) is designed for a "normal" and a "high-power" mode. The link power calculations for these modes are shown in Table 2.1-1. The $\phi 1$ NDS cannot maintain the "high-power" mode throughout its 5-year design life due to degradation of the solar arrays.

Table 2.1-1. NAV Subsystem Link Budgets, $\phi 1$ NDS

	L-1 Normal Mode		L-1 High Power Mode		L-2 Normal Mode	
	P	C/A	P	C/A	P	C/A
HPA Total RF Output (dBW)	+16.43		+17.99		+9.52	
Circuit Losses (dB)	.46		.46		.45	
Variations: Voltage (dB)	.27		.27		.28	
Temperature (dB)	.16		.40		.10	
Drive (dB)	.10		.10		.10	
VSWR Uncertainty (dB)	.10		.10		.20	
Measurement Error (dB)	—		—		.15	
PRNSA Output (dBW)	+15.34		+16.66		+8.24	
P Signal Ratio, Worst Case (dB)	-4.65		-6.00		0.0	
C/A Signal Ratio, Worst Case (dB)		-2.09		-1.4		+0.44
Antenna Gain (dB)	13.7		13.7		11.3	
Net Link Losses (dB)	186.8		186.8		185.1	
User Signal (dBW)	-162.41	-159.85	-162.44	-157.84	-165.56	-165.12
Required Signal (dBW)	-163.00	-160.00	-163.00	-158.00	-166.00	-165.00
Margin	+.59	+.15	+.56	+.16	+.44	+.88

Improvements in this critical parameter can be made by distributing excess power equally among the four signals in the two links, or can be made

preferentially in whatever order is desired by the user community. As a guideline for this study, the following sequence of improvements (Table 2.1-2) was suggested by the SAMSO program manager.

Table 2.1-2. Power-to-User Improvement Options

Option Nos.	Description
1.	Maintenance of high power mode throughout spacecraft life.
2.	Increase of L-1 C/A link power to maximum allowable flux density limits.
3.	Increase of L-1 P link power to 3 dB below revised C/A link level
4.	Use of any remaining power to increase the L-2 links by 3 dB.

As indicated in Appendix B, maximum allowable user power for the L-1 C/A link is computed to be -154 dBW based on the current allowable flux density limit of -152 dBW/m²/4 kHz, established by international convention. The -154 dBW value takes into account the spectrum spreading factor of the L-1 C/A link and also considers atmospheric attenuation to be zero, whereas the worst case link calculations allow 2 dB for atmospheric losses.

Table 2.1-3 shows the link power budget which would be required for option 2. Similarly Table 2.1-4 projects the power budgets of options 3 and 4. The four increased power options and the baseline normal mode are summarized graphically in Figure 2.1-5.

For initial estimating purposes, DC-RF conversion efficiencies, and housekeeping power requirements similar to those of the $\phi 1$ NDS were used to calculate the approximate required spacecraft DC power for the four options, shown in Table 2.1-6.

Table 2.1-3. NAV Subsystem Link Budgets - Option (2)

		MAX. LEGAL L-1 POWER		UNCHANGED L-2 MODE	
		(P)	(C/A)	(P)	(C/A)
HPA TOTAL RF OUTPUT	(dBW)	20.9		9.5	
FILTER AND FEED LOSSES	(dB)	1.3		1.3	
PRNSA OUTPUT	(dBW)	19.6		8.2	
P SIGNAL RATIO (WORST CASE)	(dB)	-9.5	—	0.0	
C/A SIGNAL RATIO (WORST CASE)	(dB)	—	-0.5	—	+0.4
ANTENNA GAIN	(dB)	13.7		11.3	
NET LINK LOSSES	(dB)	186.8		185.1	
USER SIGNAL	(dBW)	-163.0	-154.0	-165.6	-165.6

Table 2.1-4. NAV Subsystem Link Budgets - Options (3) & (4)

		OPTION 3 L-1 MODE		OPTION 4 L-2 MODE	
		(P)	(C/A)	(P)	(C/A)
HPA TOTAL RF OUTPUT	(dBW)	22.2		12.5	
FILTER AND FEED LOSSES	(dB)	1.3		1.3	
PRNSA OUTPUT	(dBW)	20.9		11.2	
P SIGNAL RATIO (WORST CASE)	(dB)	-4.8		0.0	
C/A SIGNAL RATIO (WORST CASE)	(dB)		-1.8		-0.4
ANTENNA GAIN	(dB)	13.7		11.3	
NET LINK LOSSES	(dB)	-186.8		-185.1	
USER SIGNAL	(dBW)	-157.0	-154.0	-163.0	-163.0

Figure 2.1-5. Power-to-User Growth Options Suggested by SAMSO/YE

Table 2.1-6. Power to User Options Versus D.C. Loads*

OPTION	NAV SUBSYS. D.C. POWER (Watts)	SPACECRAFT D.C. POWER (Watts)
1. MAINTAIN HIGH POWER MODE THROUGHOUT SPACECRAFT LIFE	306	378
2. INCREASE L-1 C/A LINK POWER TO ALLOWABLE POWER FLUX DENSITY LIMITS	498	565
3. INCREASE L-1 P LINK POWER TO 3 dB BELOW REVISED C/A LINK	637	704
4. INCREASE L-2 LINKS BY 3 dB	664	731

*Numerical Values are Approximate; Used for Scoping Purposes

The requirement for some 700 W_E of spacecraft prime power led to the specification of nuclear-dynamic power systems for the nonretrofit nuclear spacecraft designs. Although these systems are scheduled to be ground-demonstrated at the 1300 W_E output power level, they are projected to be capable of operating under derated conditions with good efficiency, by off-loading nuclear-isotope fuel. The characteristics of the power systems used in the new-design phase of the study are discussed in Section IV.

2.2 MEAN-MISSION-DURATION

The mean mission duration (MMD) of a satellite is a function of both its reliability R and its truncated lifetime or wear-out limit T_L , as given by the relationship:

$$MMD = \int_0^{T_L} R dt = \frac{1}{\lambda} \left[1 - e^{-(\sum_i \lambda_i) T_L} \right]$$

Typically, R is derived from the failure rates, λ_i , of the various components of the spacecraft:

$$R = e^{-(\lambda_1 + \lambda_2 + \dots) t}$$

Over the range of values expected in spacecraft programs, the mean-mission-duration is much more sensitive to truncated lifetime than it is to failure rate. An example of these sensitivities is given by Figure 2.2-1 which shows that for a constant truncated lifetime of 5 years, and failure rates ranging from 0.05 failures per year (5 year reliability of 0.779) to 0.010 failures per year (5 year reliability of 0.951), the MMD improves by only 11% while for a constant failure rate of 0.05 and truncated lifetimes ranging from 5 years to 8 years, the MMD improves by 49%.

The $\phi 1$ NDS has an estimated mission duration of approximately 4.5 years, based on the reliability values shown in Table 2.2-2. Design lifetime is given as 6 years, governed by solar array degradation.

Figure 2.2-1. MMS Sensitivity Characteristics

Table 2.2-2. GPS-NDS Phase I Reliability

<u>SUBSYSTEM</u>	<u>5-YEAR RELIABILITY *</u>
T, T&C	.8927
AVCS	.9847
EPS	.9628
RCS	.9945
THERMAL	.9999
STRUCTURE	.9992
NAV	.9407
OIS	.9968
<hr/>	
NDS SPACECRAFT	.7885

* July 1976 predictions

The MMD is a critical parameter of the entire GPS program because of the large number of operational satellites eventually required. Ultimately, the satellite replacement rate and hence the logistics costs of the space segment, will be determined by the MMD of the satellites in orbit. Over a period of tens of years, hundreds of millions of dollars can be expended in this replacement program, so that even small changes in satellite MMD can have substantial economic impact.

A nuclear spacecraft may extend the design lifetime and hence MMD by virtue of the lower predicted degradation rate of the power subsystem, and by the elimination of the battery failure mode as a limiting factor. Reliability values currently assigned to the nuclear power systems are based on subsystem and component test histories of similar, if not identical, components. Ground and space testing now planned as part of the development programs and follow-on test programs will, it is expected, generate such numbers with adequate confidence levels. At the present time, no wearout or degradation modes for the dynamic generators have been identified, other than degradation due to radioisotope decay.

For the purpose of this study, a range of possible reliability values is considered for the nuclear power systems. Design lifetime is assumed to be at least 7 years.

The question of orbital stability relates to the accuracy of the ranging calculations done by the user equipment. Table 2.3-1 below shows the one-sigma user equivalent range error (UERE) budget in meters for the Phase I and Phase III systems.

Table 2.3-1. One Sigma Error Budget, Meters

	Phase I	Phase III
Space Vehicle Ephemeris	3.9	1.6
Atmospheric Delay	2.6 to 5.6 *	2.6 to 5.6 *
Space Vehicle Group Delay	2.8	1.0
Receiver Noise	1.6	1.6
Multipath	1.3 to 3.0 *	1.3 to 3.0 *
	<hr/>	<hr/>
Total, Root-Sum-Squared	5.9 to 8.2	3.9 to 6.9

*varies with propagation path

These are initiation errors, valid for 12 hours following satellite upload with fresh data. For later times, accuracy will degrade at a rate determined by the combination of 1) clock drift, and 2) satellite drift.

The space vehicle group delay in Table 2.3-1 includes the contribution of clock drift. Table 2.3-2 summarizes, to an order of magnitude, the expected stability of the three frequency standards which are loosely associated with the three phases of the GPS program. Clearly, achievement of the phase I error budget will require the introduction of cesium clocks in phase I, as is now planned for the later spacecraft in this phase.

With the successful introduction of cesium clocks the errors due to clock drift and to unpredictable ephemeris drift will be of the same order of magnitude. Subsequent introduction of a more stable hydrogen-master frequency standard will not improve system degradation rate significantly without the simultaneous introduction of a spacecraft with a more accurately predictable vehicle ephemeris. This is illustrated

Table 2.3-2. On-Board Clock Frequency Stability

GPS PHASE	FREQUENCY STANDARD	FREQUENCY STABILITY $\frac{\Delta f}{f}$ /day	Range Error Buildup	
			m/hr	m/day
I	Rb	10^{-12}	1.1	26
II	Cs	10^{-13}	0.1	2.6
III	H ₂ -maser	10^{-14}	0.01	0.26

in Figure 2.3-3, where the user-calculated position error is shown as a function of time from last update, assuming the geometric arrangement of a complete 3x8 satellite constellation. The upper pair of curves assumes a spacecraft of the NDS type, with an unpredicted ephemeris drift of 1.25 m/day, as currently claimed by NSWC*. The lower curve assumes the introduction of a spacecraft whose ephemeris can be predicted to within 0.25 m/day.

The factors which generally influence ephemeris predictability are discussed below, with mathematical detail in Appendix C. Those which apply to the reference nuclear-powered spacecraft have been calculated -- and will be discussed in Section X, following the description of the reference designs.

To the extent that vehicle ephemeris drift can be modelled and predicted, the ground user will be able to compensate for this drift by the application of perturbation coefficients which will be transmitted by the satellite along with the Keplerian orbit elements. Such important disturbances as geodetic variations in the gravitational field of the Earth, and the major solar-pressure effects, can be modelled and predicted with good accuracy. However, there are significant disturbances which

*Naval Surface Weapons Center, the program element charged with software development for GPS upload.

GPS - NAVSTAR

ASSUMES

- 2% SOLAR-PRESSURE MODEL ERROR (1.25 M/DAY) FOR SOLAR-POWERED SPACE VEHICLE
- 0.25 METER/DAY ERROR BUILD-UP FOR NUCLEAR-POWERED VEHICLE
- GDOP=2 (MINIMUM GEOMETRIC DILUTION OF PERFORMANCE FOR 98+% OF CASES IN OPERATIONAL PHASE)

Figure 2.3-3. System Accuracy Degradation with Time from Last Upload

cannot be modelled or which vary so much or so unpredictably from day to day that they will result in unpredictable changes in satellite ephemerides, and hence time-dependent degradation in system accuracy. Among such unpredictable disturbances are:

1. Solar pressure variations due to non-repetitive orientation of spacecraft surfaces with different α and ϵ values to the sun.
2. Solar pressure variations due to non-repetitive shadowing of spacecraft surfaces by the antennas, thrust cone, etc.
3. Solar pressure variations due to apparently random changes in thermal-control louver blade angle (i.e. not synchronized with orbital position).
4. Impulse variations due to thermal radiation variations controlled by the louvers (or due to misestimated temperatures on various surfaces).
5. Impulses due to gas leakage from the reaction control system or outgassing from any surface.

The phenomena noted in items 1. and 2. above do, of course, repeat on an annual basis and only change slowly from day to day. However, the extent to which they repeat is governed by the accuracy of the control system, and the extent to which the satellite can be physically modelled. For example, Figure 2.3-4 shows the error build-up in both radial and lateral components of the predicted orbit due to a 1% error in modelling the solar pressure force on a spacecraft of NDS size for the worst case (noon) orbit (Appendix C). If this "error" is fixed in time, it will eventually be corrected by the ground-based computer. However, if it varies from day to day, the computer model will not identify it and correct it. Clearly, errors of a few meters/day are possible from this source.

The significance of item 3 arises from the fact that the louvers are mounted on the sun-parallel faces in the NDS satellites. Consequently, when they open they almost always reflect the sun and when closed they are essentially shadowed.

Figure 2.3-4. Relative Displacement of GPS/NDS Spacecraft due to 1% Increment of Total Solar Pressure Force (Case $i=0$; Sun Vector in Orbit Plane)

Item 4 can be significant because the louvers on both sides of the spacecraft operate independently. A certain amount of skepticism that thermal radiation from within the spacecraft can result in significant orbital offset is inevitable. However, it is noted that for radiant energy, the ratio of energy to momentum is the constant c (speed of light in vacuum). Therefore, if the energy radiated is Q , then the momentum imparted by this radiation is Q/c , and the total force applied is simply $(1/c)dQ/dt = \text{radiated power}/c$.

An unbalanced radiation of 500 watts can, therefore, result in a force of 1.7 micronewtons, which if applied in the same direction for 12 hours to a 1000 pound spacecraft, can result in a displacement of 3.5 meters, clearly within the range of interest. The fact that the louvers operate independently and in response to baseplate temperature would cause this force to be applied at various portions of the orbit, and hence, since spacecraft yaw angle is constantly changing, in unpredictable directions.

The simple calculation illustrates by inference the care in design and assembly which is necessary to minimize the effects of item 5.

It is obvious that by eliminating the solar arrays, a nuclear spacecraft would have a positive impact on item 1 above simply by reducing the projected area and hence the absolute magnitude of the total solar force. Perhaps not as obviously, however, a nuclear spacecraft can have a substantial impact on all of the above disturbances. Elimination of the sun-pointing requirement of the solar arrays introduces a number of design options which can substantially improve or eliminate most or all of the listed disturbances. This requirement strongly influenced the design approaches which were explored and that which was ultimately selected, as discussed in Section III.

The relative value of radial and transverse errors in satellite position to ground user position errors is illustrated in Figure 2.3-5 (Appendix D). It is assumed that the positions of satellites 1 and 2 are known exactly and that the position of satellite 3 is known in error. The error is either radial, δ_R , or transverse, $\delta_\perp$. The user position error resulting from this simplified analysis is shown on the figure

as a function of satellite position above the horizon. Clearly, for all useful satellite positions, the radial error is more important, by a factor of 4 or more, than the transverse error in satellite position.

Figure 2.3-5. Effect of Radial and Lateral Errors of User Position Error

2.4 THERMAL CONTROL OF ON-BOARD CLOCKS

The stability of the on-board frequency standard is closely allied to the question of ephemeris drift, as was discussed above. Unpredictable clock drift from the time of last upload will cause an error in the ranging calculation, given by $\delta R = c (\Delta f/f)$, where c is the speed of light, equivalent to a radial error in satellite position.

The order-of-magnitude stability characteristics of the various frequency standards is given in Table 2.3-2, based on various random noise factors controlling the standards. In addition, these are all temperature sensitive to one degree or another. Although they typically are temperature-controlled by internal heaters, the achievable control is a function of the temperature stability of the heat sink.

A rule-of-thumb which was applied in this study is that the baseplate temperature coefficient of frequency stability, $\Delta f/f$ per $^{\circ}\text{C}$, is the same order of magnitude as the random frequency drift per day as given in Table 2.3-2. Therefore, baseplate temperature control of $1^{\circ}\text{C}/\text{day}$ for the worst-case orbit was set as a design goal for the reference spacecraft.

2.5 SURVIVABILITY

Because Phase III of the GPS program is specified to meet JCS survivability standards, the study ground rules were that the power systems themselves were to be evaluated for JCS-level survivability. Two areas of possible concern in the nuclear power systems are the electronic control systems and the external thermal-control surfaces. These were to be evaluated for vulnerability insofar as possible. However, it became apparent in the course of the study that, although spacecraft survivability is a critical parameter to the GPS program, it is not primarily a power systems issue at the present time.

SECTION III

APPROACH TO THE REFERENCE DESIGN

3.1 SELECTION OF ATTITUDE CONTROL APPROACH

The attitude control approach has a substantial impact on many of the spacecraft subsystems and on the method of operating the spacecraft in space, as well as on weight, power, reliability, and orbit predictability.

With regard to the important orbit predictability problem, the reference design strives to approximate the ideal spherical satellite. Because this can only be approximated by an exterior profile with a minimum number of variations in surface reflectivity/absorptivity, the selected attitude control approach will have a significant modifying effect on the orbit predictability. This comes about through two influences:

1. The attitude control approach determines which spacecraft surfaces face the sun for how long and in which sequence.
2. The attitude control approach and system design determines the magnitude, frequency, and direction of jet firings required to unload stored angular momentum. The magnitude and possible methods of applying the jet impulses will determine the errors which this action can introduce into the predicted ephemeris.

Three different attitude-control approaches were considered, with two versions of one of them forming a total set of four:

1. Dual-Spin - In this approach, the rotation of the spacecraft body is used to provide a relatively large angular momentum normal to the orbit plane, thus making the spacecraft very insensitive to torques about the other two axes. This system has been highly

developed and is in wide use for synchronous-orbit communications spacecraft. With nuclear power, it can be used in other orbits as well, since power is not generated by body-mounted solar cells. It requires that directional Earth-oriented antennas be despun via a bearing and power-transfer assembly. The antenna assembly rotates at orbit frequency, while the spacecraft typically rotates 30 - 60 rpm. This configuration is shown below.

2. Bias Momentum - In this approach, some of the benefits of the spinner are retained by providing the biasing angular momentum via an internal wheel in constant rotation. The major advantage is elimination of the rotating bearing and RF joint to the antennas. Because of practical limitations on size, weight, and rotation speed of the wheel, however, the biasing momentum is typically much less than that of the spinner. Two choices for orientation of the momentum vector are possible:
 - 2a. H along spacecraft Z axis -- This essentially reproduces the spinner attitude orientation. Antenna array must be oriented in the x or y direction, as in the spinner, but without the rotational bearing. The following illustration shows this configuration.

- 2b. H normal to spacecraft Z axis -- This retains the attitude orientation of the phase I space vehicle, although without the possibility for yaw steering. This is shown in the drawing below.

3. Zero-Momentum Yaw Rotator - In this approach the spacecraft is fully stabilized in pitch and roll by a redundant set of momentum wheels, but allowed to rotate slowly in yaw (about the spacecraft axis). This is shown in the drawing below.

An internal momentum wheel is used to counterbalance the yaw momentum so that there is no net momentum to precess in orbit. In the case of the Organic Rankine power system the angular

momentum of the rotating unit alone is adequate (and ideal) to provide the counterbalancing momentum. The yaw rotation rate is normally uncontrolled, but can be corrected by jets if necessary. Since the antenna pattern is axially symmetric about the Earth-pointing axis, no harm is done to the transmitted signal by the slow (1/2-to-1 rpm) rotation.

The remaining obvious candidate for attitude control consideration is the fully 3-axis-controlled vehicle. This was believed to offer no significant advantage over a bias momentum system unless yaw steering was to be performed in order to assist in thermal control. In that case, the spacecraft would resemble the retrofit nuclear-powered spacecraft already considered in the first phase of the study.

The various factors which entered into the decision on attitude control approach are listed in Table 3.1-1, with factors rated or estimated for the four approaches which were considered. The zero-momentum yaw rotator (no. 4) was ultimately selected, with the classical dual-spinner (no. 1) a close second. Although in early estimates the latter produced the least orbit prediction error, as well as a low weight, low power, and space-proven approach, the former scored quite acceptably in orbit predictability and was judged to be equal or superior to the dual spinner in every other respect, with the possible exception of the peak power dissipation of the attitude control system. This peak power difference (ultimately it vanished) was not considered significant compared to most of the more basic factors of operation and performance on the list. In addition, it was suspected that the orbit predictability error could be made very much smaller for the yaw rotator than this early estimate. This ultimately proved to be the case.

The various factors are each described below, and the bases for the estimates and judgments are discussed.

1. Estimated Orbit-Prediction Error

The dual spin and bias-momentum systems with spacecraft Z-axis normal to the orbit plane (cases 1 and 2) both share a common advantage in the significant orbit-predictability area: they maintain a fixed body orientation to the sun as they orbit the earth. Over a period of time, therefore, the prediction error with regard to

Table 3.1-1. Candidate Attitude Orientation and Control Approaches

CHARACTERISTIC	1 DUAL SPIN, Z NORMAL TO ORBIT PLANS	2 BIAS MOMENTUM, Z NORMAL TO	3 BIAS MOMENTUM, Z TO NADIR	4 ZERO MOMENTUM, Z TO NADIR, YAW ROTATOR
1. ESTIMATED ORBIT PREDICTION ERROR, M/DAY				
RADIAL	0.1	0.6	> 0.6	0.2
TRANSVERSE	0.3	1.5	> 1.5	0.8
2. { ATTITUDE CONTROL SYSTEM WEIGHT, LBS. (5 DAY/10 DAY UNLOAD PERIOD)	39/39	140/200	74/99	61/71
RADOME REQUIRED	YES	YES	YES	NO
WEIGHT OF ACS WITH RADOME, LBS.	82/82	183/243	112/137	61/71
PEAK ACS POWER, WATTS (E)	20	15	20	20-40
3. RF POWER LOSS (RADOME & ROT. JOINT), dB	0.2	0.1	0.1	0
4. RELIABILITY AND SPACE EXPERIENCE RATING	A	B	B	A
5. THERMAL CONTROL RATING	A	B	B	A
6. VEHICLE SHAPE CONSTRAINT RATING	B	C	C	A
7. GPS LEGACY RATING	C	C	A	A
8. EASE OF AUTONOMOUS UNLOADING OF WHEELS	C	B	B	A
9. HEAT PIPE APPLICABILITY FOR THERMAL CONTROL	D	A	A	B
10. COMPATIBLE WITH DYNAMIC POWER SYSTEMS	B	B	A	A

DECISION: OPTION 4, YAW ROTATOR

solar pressure effects on the vehicle body approaches a very low value, estimated at some 0.1 m/day in the radial direction and 0.2 m/day in the normal-to-radial direction (Appendix E).

The remaining error is that associated with reaction jet firings. Because of its large angular momentum, the spinner is particularly insensitive to external torques; therefore, internal momentum wheels are not provided. However, because it maintains a fixed orientation in inertial space, the spacecraft orientation to the Earth changes a little each day as the vehicle orbit plane precesses with respect to Earth. The orbit precession rate due to Earth oblateness has been calculated to be 0.0272 degrees/day (Appendix E). Therefore, if an antenna pointing error of ± 0.5 degree is specified, and if half this error is assigned to the results of orbital precession, then this correction must be made at approximately 9-day intervals, by jet firing.

A matched pair of jets are fired to precess the vehicle angular momentum vector by the required 0.25 degree. However, imperfections in jet orientation and thrust matching can lead to a net translation of the vehicle, thus upsetting its orbit, and hence obviously its predictability.

The major part of this error can be cancelled out by only applying half the correction at each of two opposite positions in the vehicle orbit, 6 hours apart. In this way, total errors of only 0.1 meter/day in the radial direction and 0.3 meter/day in the transverse direction are predicted (Appendix E).

The bias momentum systems are both more easily precessed by solar-pressure torques since the biasing angular momenta which can be achieved by internal wheels are much smaller than those which can be achieved by a rotating spacecraft. Consequently more frequent thrusting is required, estimated at a 1.5-day

interval rather than the 9.2 days for the spinner. Because the solar pressure uncertainties are also somewhat greater for the non-rotating spacecraft, the estimated orbit prediction error is as shown for the Z-normal-to-orbit-plane, case 2 in Table 3.1-1. Case 3 in Table 3.1-1 involves the additional uncertainty of a spacecraft which changes its body angle to the sun as it orbits the Earth, so that its uncertainty would be greater than that of case 2, and was not estimated separately.

The zero-momentum case is entirely different from the other three. Because it includes reaction wheels to compensate for imposed torques about two axes, with the third (Z) axis essentially free-spinning, jet thrusting intervals are tradeable against momentum wheel storage capacity. It has been estimated (Section VII) that a very modest wheel size of 0.5 ft-lb-sec would extend this interval to 10 days. Moreover, jet unloading requires only very small thrust levels, thus significantly reducing the orbit ephemeris disturbance which this can introduce. In this case, jet thrusting may be the limiting uncertainty. As in the phase 1 (solar-powered) spacecraft the modelability of the spacecraft was initially thought to be limiting. However, because of the factors of reduced size, simplified shape, simplified and uniform surface finish, passive and uniform thermal radiation, and continuous rotation in orbit, the solar force uncertainty was initially estimated to be at least an order of magnitude less than that of the existing phase 1 NDS design. Following completion of the reference design, this factor was re-estimated in more detail, and the initial estimate was found to be conservative by a significant factor (see Section X).

2. Attitude Control System Weight and Power

These weight figures have been estimated (Appendix F) for both 5-day and 10-day unload periods. Case 1 (spinner) exhibits the

lowest weight of this subsystem. Because of the fixed high angular momentum of the satellite, the ACS weight does not depend on the unload period. As explained earlier, the jet firing period depends only on the orbit precession rate.

In the spinner case, as well as in cases 2 and 3, the antenna array is fixed with respect to the Earth and exhibits a continually changing view to the sun. Projected areas, angles, and shadows change continuously over each orbit, but also from day to day. This provides an element of error in estimating solar pressure forces which would be undesirable in a system designed for high predictability. Therefore cases 1, 2, and 3 would very likely require a radome-type dielectric cover over the antennas, thus substantially reducing this uncertainty factor. The weight of such a cover has been estimated for the three cases and added to that of the ACS in Table 3.1-1.

The zero-momentum yaw rotator, case 4, continually rotates about the Z-axis, so that projected area, shadows, and other variations are reproduced many times and averaged out over each orbit. This case, therefore, is judged not to require a dielectric cover, so that in total weight of ACS plus antenna cover, it is the lightest system in the group.

In principle, case 4 may be expected to exhibit the largest peak power consumption. This comes about because the actuators of the momentum wheels are continually transferring momentum from one wheel to another as the vehicle rotates. During a one-quarter (90°) rotation about the spacecraft Z-axis, the yaw and roll axes are completely interchanged, together with momentum wheels that are fixed to the body of the spacecraft. Any momentum stored in one wheel, therefore, must be transferred to the other and vice-versa. During the 5-to-10-day period between unloading of the

wheels, the power consumed by the actuators gradually rises until the peak power is reached. Following unload this falls abruptly to the low initial value and begins the cycle again.

In a more detailed study of this control approach, it was determined that a peak power consumption of ten watts per wheel is adequate, leading to a total peak power of less than 20 watts for a 10-day unload period (Section VII). This could be reduced still further by unloading more frequently, or by pre-loading the wheels in the opposite direction, or by some combination of the two approaches.

3. RF Power Loss

This line illustrates the fact that cases 1, 2, and 3 all suffer some RF power loss through the radome, while case 1 suffers an additional loss in the rotating joint. Values are order-of-magnitude estimates.

4. Reliability and Space Experience

Spin stabilization historically has been the most commonly applied attitude control technique, leading to a high experience rating and good expectation for high reliability for this system. Bias momentum systems have been rarely flown, probably because the limited angular momenta which can be practically achieved lead to a preference for a fully 3-axis controlled vehicle.

A zero-momentum yaw-rotator satellite is not very common, but considerable practical space experience has been accumulated on such a system over the past several years by an Air Force satellite program with classified details.

5. Thermal Control

In general, rotating spacecraft may be said to enjoy a thermal control advantage because the solar input is averaged over the

entire surface of the spacecraft. Bias momentum systems in non-equatorial or non-sum synchronous orbits have more restrictive geometric conditions leading to more complex thermal control solutions, in general. (In the selected reference design, with its forced-convection cooling of equipment, this difference would not be applicable.)

6. Vehicle Shape Factor

This factor relates to the ease of approximating an ideal shape for orbit predictability: a sphere with a uniform surface coating. The two bias-momentum approaches would be significantly penalized because the thermal control requirements would introduce a variety of surface coatings and treatments. The spinner is constrained mainly by the need for an "erected" antenna system. The yaw rotator, however, offers the maximum freedom of selection of external shape because of the benefits of the rotating action on thermal control, while retaining zero net angular momentum and the more favorable antenna orientation. (Again, in the reference design, this difference would be minimized because of the selected thermal-control approach.)

7. GPS Program Legacy

Cases 1 and 2 would benefit least from the phase 1 GPS satellite program since they involve a completely different attitude orientation in space. Cases 3 and 4 retain the same navigation system/antenna arrangement, and similar attitude and reaction control system.

8. Ease of Autonomous Unloading of Wheels

Although each of the attitude control approaches can, in principle, be mechanized for completely autonomous operation (i. e., without ground command), this item rates the ease or degree of complication involved in doing so. The spinner is given the poorest rating in this category because not only is a relatively high level of thrusting involved, but it must be applied at opposing positions of the same

orbit, 180° apart. Also, the input information, which determines the proper time to initiate thrusting corrections, must be derived from the scanning Earth sensor, as must be the feedback measurement on which the control of the thrusters is based.

The bias momentum systems involve moderately high thrust levels to precess the spin vector and are also dependent on information from the scanning Earth sensor. By contrast, the zero-momentum yaw-rotator has the lowest thrust levels, and derives its control information from a tachometer which measures the speed of the momentum wheels. When these reach their maximum preset values, the thrusters may be programmed to fire until the speed reduces to its preset value. This is judged to be the simplest of the possible autonomous control schemes. (In addition, the low correction torques which are required here make the use of magnetic unloading most feasible. This would eliminate the possibility of any orbit perturbation from this source.)

9. Heat Pipe Applicability for Thermal Control

This item merely identifies the fact that a spinner satellite would find it generally impossible to use heat pipes to assist in thermal control. This is due to the centrifugal gravity of the satellite which would severely restrict the permissible orientation and direction of heat flow of the heat pipes. The yaw rotator could have a similar potential problem, if its rotation rate were such as to create a troublesome magnitude of centrifugal force. This is not the case in the reference design (Section VIII).

10. Compatible with Dynamic Power Systems

This item expresses the fact that the rotating units of the power systems would prefer to be oriented in the spacecraft Z-direction for launch (or IUS) operation. For case 3, this

would result in an angular momentum in the wrong direction on orbit, thus requiring either a less-preferred orientation during launch or a counterbalancing momentum wheel on orbit. In any case, the problem is not controlling, and more important for the higher-angular-momentum Organic Rankine power system than for the Brayton power system.

As indicated previously, these considerations led to the selection of the two-axis-controlled, zero-net-momentum yaw rotator as the reference design approach.

3.2 THERMAL CONTROL APPROACH

In their most common conceptions, nuclear power systems, like solar-power systems, are treated as appendages to the main body of the spacecraft. This is the so-called "separated" design approach in which the power system is interfaced electrically and mechanically, but isolated thermally from the remainder of the spacecraft. With the preferred attitude-control approach the slow axial rotation of the spacecraft would simplify thermal control of the spacecraft, but excursions of 10-20°C during eclipse orbits may be expected, even with the application of louvers and heaters.

The desire for closer thermal control of the frequency standards, and the knowledge that the power system radiator requirements result in a largely "empty" radiator internally, led to the consideration of the so-called "integrated" design. In this approach, the spacecraft equipment is contained within the power system radiator and all (or most) of the equipment thermal dissipation is rejected through the radiator.

In earlier studies of an integrated design *, the spacecraft dissipative equipment was mounted to the internal surfaces of a flat sided, isothermal, power-system radiator of low overall temperature. However, the power systems now under development do not have isothermal or low-average-temperature (relative to required

*AEC SNSO 3063-2, Standardization and Economics of Nuclear Spacecraft, Fairchild Report No. 217-053-2, March 1973

equipment temperature) radiators. Radiator temperatures typically vary from approximately 100°C at the inlet to approximately $5\text{--}50^{\circ}\text{C}$ at the outlet, depending on the power system, for the Brayton and Organic-Rankine systems respectively. In addition, it was known from the earlier studies that mounting equipment to the radiator panels directly is mechanically intricate and presents accessibility and testability problems.

The thermally-integrated design approach which was finally selected is shown in Figure 3.2-1. An oversized radiator is used to overcool the radiator fluid, in the Organic Rankine case, to approximately 30°C . From the radiator outlet the liquid Dowtherm-A working fluid is routed to an equipment cooling section internally. Spacecraft equipment is cooled by conduction to the fluid loop, so as not to penetrate the hermetic integrity of the loop. This should raise the loop temperature by some 10°C , from whence it is returned to the power system.

Figure 3.2-1. Thermally Integrated Design Approach

The Organic Rankine system is not affected by the temperature rise introduced by the equipment since it normally requires approximately 70°C as the inlet to the jet condenser, and will provide a radiator by-pass loop to control this temperature by mixing the radiator flow with uncooled by-pass flow. The Brayton system, however, requires a lower inlet temperature to its compressor, and so a modification of this approach is required. This is shown in Figure 3.2-2. In this case, the He/Xe working fluid is removed from the radiator at what would be its normal outlet temperature and routed through the equipment section. Its temperature rise through the equipment, however, is approximately 35°C . Therefore, it is then returned to a separate section of the radiator and allowed to recool to $5\text{-to-}10^{\circ}\text{C}$ before returning to the Brayton rotating unit. If this is implemented as shown in Figure 3.2-2, the radiator temperature will show a variation as illustrated in Figure 3.2-3. The skin can be continuous, therefore, and no insulator section need be interspersed between the two radiator sections.

Figure 3.2-2. Thermally-Integrated Design Approach

Figure 3.2-3. Brayton Radiator Temperature Profile

In this design approach, the entire spacecraft, with minor exceptions, is enclosed by the power system radiator and is cooled by the power system fluid loop. It is a good approximation, therefore, to say that the spacecraft is thermally interconnected during eclipse, and will cool down as a single body. Internal gradients must be kept small because of the relatively high sink temperature (i. e., of the fluid loop) into which the equipment must reject heat.

Therefore, a good approximation of the cool-down rate during eclipse can be gotten by calculating

$$\frac{dT}{dt} = \frac{\Delta Q}{MC_p}$$

For $\Delta Q = 500$ watts as the approximate heat loss during eclipse, $M = 1200$ pounds as the overall spacecraft weight and $C_p = 0.2$ cal/gm $^{\circ}$ C as a representative specific heat for solid structure,

$$\frac{dT}{dt} = 4^{\circ}\text{C/hr.}$$

It will be seen that this is a good approximation for the spacecraft as a whole (Section VIII) and is achieved without any active thermal control action whatsoever. Internal equipment, e.g., the frequency standards, will tend naturally to vary less during eclipse, and can be finely controlled by active means because of the small base variation of the heat sink. This verifies the main advantage of the thermally integrated design approach which was adopted.

3.3 MECHANICAL DESIGN APPROACH

A mechanical design which would allow the spacecraft equipment to be assembled and tested separately from the power system, despite their eventual close integration, was desired. In concept, this is shown in Figure 3.3-1. The two major subassemblies would be manufactured and assembled separately, but with mechanical and thermal interfaces provided for their eventual mating. As may be anticipated, the thermal interfaces provide the major design problems in this approach, but are shown to be amendable to solution for both candidate power systems.

Figure 3.3-1. Mechanical-Design Approach

Before embarking on detailed design efforts, it is necessary to verify that the overall size of the power system radiator is in keeping with the general desired spacecraft size, and can be accommodated with the desire to mount four spacecraft atop the IUS.

Figure 3.4-1 shows three alternative cross sections for clustering four spacecraft within the shuttle bay. Flat-sided spacecraft might be preferred from the viewpoints of fabricability and the mounting of louvers, equipment radiators, etc., in the typical spacecraft. However, in this case, the use of louvers or directly-radiating equipment is not anticipated. Some small advantage in external surface area per unit length is also evident for flat-sided spacecraft, but from the viewpoint of simplicity and orbit predictability, the cylindrical shape was selected. For a shuttle bay diameter of 15 feet, a spacecraft diameter of 72 inches allows 2 inches of clearance between adjacent spacecraft and 3 inches between spacecraft and shuttle bay wall, in the four spacecraft cluster.

ALTERNATIVE APPROACHES:

DECISION: RETAIN CYLINDER FOR SIMPLICITY AND ORBIT PREDICTABILITY; PAY SLIGHT PENALTY IN LENGTH OF RADIATOR

Figure 3.4-1. Ground Rule: Four Spacecraft per Shuttle/IUS Launch

The nominal spacecraft length, governed mainly by required radiated area, is calculated approximately in Table 3.4-2 for the two power systems. Although the two systems have substantially differing thermal power outputs to reject (3050 and 4600 watts_t) the effective radiating temperatures differ in the right direction to compensate for this difference, resulting in a required radiator area, and spacecraft length, which is remarkably similar for both spacecraft. After considerations of realistic radiator fin effectiveness factors and fluid-to-wall heat transfer coefficients, the radiator sizes increased to approximately 130 square feet in both cases. However, radiator length only increased to approximately 6 feet, as a result of including a portion of one end face as part of the radiator (Section V).

Table 3.4-2. Approximate Radiator Areas - Integrated Design

	BRAYTON	ORGANIC RANKINE	UNIT
TOTAL THERMAL POWER, P_T	3200	4800	WATTS _T
APPROX. ELECTRICAL POWER, P_E	600	750	WATTS _E
APPROX. RADIATED RF PWR, P_{RF}	150	200	WATTS _{RF}
RADIATED THERMAL PWR, $P_T - P_{RF}$	3050	4600	WATTS _T
NOMINAL RADIATOR INLET TEMP.	235	212	°F
NOMINAL RADIATOR OUTLET TEMP.	10	100	°F
EFFECTIVE RADIATOR TEMP, T_{EFF}	100	152	°F
P_T/A @ T_{EFF} ($\epsilon = 0.8$)	39.3	56.0	WATTS _T /FT ²
MAX. SOLAR INPUT ($\alpha = 0.2$)	8.3	8.3	WATTS _T /FT ²
NOMINAL RADIATOR P_T/A	31.0	47.7	WATTS _T /FT ²
NOMINAL RADIATOR AREA	98.4	96.4	FT ²
NOMINAL RADIATOR LENGTH (D=6FT)	5.2	5.1	FT

SECTION IV

POWER SYSTEMS INPUT DATA

4.1 CANDIDATE POWER SYSTEMS

Two dynamic turbine-alternator power systems are currently under development by ERDA/NRA: the Organic Rankine Kilowatt Isotope Power System (KIPS), and the Brayton Isotope Power System (BIPS). Both systems are following the same basic development schedule, shown in Figure 4.1-1. Following ground demonstration of both systems and the system selection process scheduled for FY 78, one of the two power systems will be continued on to flight system design, qualification and production.

DEVELOPMENT SCHEDULE

MILESTONES	FY75	FY76	FY77	FY78	FY79	FY80	FY81	
DEV. FLT. CONCEPT		■						
SYS. GROUND DEMO.		■	■	■				
SYS. SELECTION				▲				
FLT. SYS. DESIGN AND QUAL				■	■	■		
FLT. SYS. PRODUCTION						■	■	
DELIVERY FOR FLT. TEST							▲	

Figure 4.1-1. Dynamic Power System Development Program

In the present ground demonstration phase, both power systems are designed for end-of-mission output power of 1300 watts_e. However, both incorporate the capability for operating in the range of 500-to-2000 watts_e with some modification of equipment and adjustment of fuel loading and cycle conditions.

Equipment modifications were to be kept to a minimum for the present study; however, the designations "KIPS" and "BIPS" were reserved for the 1300 w_e versions. The generic "Rankine" or "Brayton" designations are employed for the power systems used in this study.

4.2 ORGANIC RANKINE SYSTEM

The Organic Rankine cycle system is under development by Sundstrand Energy Systems of Rockford, Ill. The basic system schematic is shown in Figure 4.2-1.

Figure 4.2-1. Organic-Rankine Cycle Schematic

The radiator by-pass valve, normally used to control jet condensor inlet temperature, can be used here to control radiator outlet temperature, so as to assure a constant-temperature heat sink for the spacecraft equipment. However, as will be seen in Section VIII, the total maximum variation of the liquid temperature in this application will barely exceed the normal control range of this valve ($\sim 3^{\circ}\text{C}$); therefore, day-to-day control will be exercised in the normal manner. The flow control valve will serve to handle long term changes due to isotope decay or changes in radiator surface properties.

In order to achieve a radiator outlet temperature of 30°C , a radiator area of 129 square feet is required. The other major system parameters are shown in Table 4.2-2, based on the use of two standard MHW heat sources. The radiator design parameters are shown in Table 4.2-3.

Table 4.2-2. Organic-Rankine System Baseline Power System

(RADIATOR AREA = 129 SQUARE FEET, INCLUDING EQUIPMENT HEAT LOAD)

	BoL	7 YR
THERMAL INPUT, WATT_T	4800	4558
HEAT ABSORBED, WATT_T	4684	4442
OUTPUT POWER, WATT_E	820	768
CONVERSION EFFICIENCY (SYSTEM)	.17	.17
TOTAL WEIGHT (INCL. RADIATOR), LBS.		390
RADIATOR LIQUID OUTLET TEMP., $^{\circ}\text{C}$		30
TURBINE INLET TEMP, $^{\circ}\text{C}$		343
SYSTEM MASS FLOW RATE, LB/HR	73	69
RADIATOR MASS FLOW RATE, LB/HR		322
JET CONDENSER FLOW RATE, LB/HR		629

Table 4.2-3. Organic-Rankine System Radiator Design Parameters

TOTAL HEAT REJECTED, WATT _T	4242
LIQUID INLET-OUTLET TEMPERATURE, °C	93-30
LIQUID PRESSURE DROP, PSI	21
ABSORBITIVITY/EMISSIONIVITY	0.2/0.8
TOTAL AREA, SQ. FT.	129
NUMBER OF TUBES	1
TUBE LENGTH, FT.	174
TUBE INTERNAL DIAMETER, IN.	.281
TUBE PITCH, IN.	8.8
TUBE EXIT REYNOLDS NO.	2300
FIN THICKNESS, IN.	.025
FIN EFFECTIVENESS	.837
TUBE & FIN MATERIAL	6061 ALUMINUM
TUBE & FIN CONDUCTIVITY, BTU/HR-FT°F	100

The requirement for very cold outlet temperature from the radiator for cooling purposes leads to certain heat transfer problems because of the high resultant liquid viscosity and tendency to laminar flow. In order to minimize temperature drop from liquid to radiator fin, high heat transfer coefficients are needed while a low pressure drop is required to minimize system pump work.

In order to meet these constraints, a dimpled tube configuration has been selected using heat transfer and friction factor data based upon Kays and London* data table FTD-1 (Appendix G, Figure G-7). The dimpled surface would be formed into a thin walled stainless steel tube and then a thick walled aluminum tube would be cast around or shrunk onto this inner liner. The single radiator tube would be wound spirally around the cylindrical radiator fin at the given pitch. The use of a single tube would preclude potential problems of flow maldistribution or freezing associated with the cold outlet temperature. Freezing point of the DowTherm A working fluid is approximately 13°C.

* Kays and London, Compact Heat Exchangers, McGraw Hill)

End of mission (EOM) electrical output power is based on the decayed isotope output and the given radiator coating properties. The beginning of mission (BOM) output power is estimated based on the initial Heat Source Assembly (HSA) thermal output without using BOM radiator coating properties. The excess power will need to be dissipated while the higher heat input at BOM will result in higher radiator inlet temperature in order to reject the higher heat load. However, the control valve will maintain the required outlet temperature.

It is assumed that the working fluid will re-absorb all the electrical output power except for 200 watts of radiated electromagnetic (RF) power.

System weights are based upon a KIPS flight conceptual design using two heat sources and a radiator weight scaled from the KIPS current design. However, radiator weight was recalculated in the reference design.

The methods used to optimize the cycle for the conditions given is summarized in Appendix G, along with computer print-outs of the major cycle parameters.

4.3 BRAYTON-CYCLE SYSTEM

Initial scoping analysis established that the Brayton cycle system would require a stretched version of the MHW heat source, with four trays of fuel sphere assemblies totalling 3200 watts_t, instead of the usual three. In addition, a 2400 watt_t case was derived. Two versions of the 3200 w_t Brayton system were carried through to the cost analysis phase in order to achieve a cost/benefits comparison between them: a C-103 columbium alloy system and a superalloy system. The different materials appear only in the heat source heat exchanger, the turbine inlet scroll, and the connecting ducting. The superalloy system has lower output power due to the lower turbine inlet temperature capability of this material, but benefits from reduced hardware, assembly, and test costs.

The major system parameters for the three cases are shown in Table 4.3-4 based on a total radiator area including the equipment radiator section of 130 sq. ft. The cycle conditions for the three cases are shown in Figures 4.3-2, 4.3-3, and 4.3-4, with temperatures in °R, pressures in psia, and flow rates in lb/sec.

Table 4.3-4. Brayton-Cycle Systems Alternatives

(RADIATOR AREA = 130 SQUARE FEET, INCLUDING EQUIPMENT RADIATOR)

THERMAL INPUT, WATT _T	3200	3200	2400
MATERIAL	SUPERALLOY	C103	C103
BoL OUT PUT POWER (3φAC), WATT _E	665	735	490
BoL CONVERSION EFFICIENCY	0.21	0.23	0.20
7 YR OUTPUT POWER (3φAC), WATT _E	614	678	452
TOTAL WEIGHT (INCL. RADIATOR, ENCLOSURE), LBS	413	396	397
EQUIPMENT RADIATOR INLET TEMP., °C	32	38	27
OUTLET TEMP., °C	7.2	4.5	-9.5

(NOTE: Output Power Includes Penalty for Pressure Drops in Equipment Headers and Radiator of $\Delta P/P = .006$)

Figure 4.3-2. C103 Brayton System with 2400 Wt Heat Source

Figure 4.3-3. Superalloy Brayton System with 3200 Wt Heat Source

Figure 4.3-4. C103 Brayton System with 3200 Wt Heat Source

SECTION V

REFERENCE DESIGN DETAILS

5.1 ORGANIC-RANKINE-POWERED SPACECRAFT

The external profile and major dimensions of the Rankine-cycle space vehicle (RSV) are shown in Figure 5.1-1 as assembled. The spacecraft disassembles into two major subassemblies: The Power System Module (PSM) and the Equipment Module (EM), as shown in Figure 5.1-2.

Each module can be assembled and tested fully independently of the other. System-level tests can be performed following final assembly, but with electrical heat sources in place of the nuclear heat sources. The nuclear heat sources need not be inserted until the spacecraft is delivered to the launch site. Provision is made for access to the heat source container (heat exchanger or boiler) after final assembly of the spacecraft in both reference designs. In the RSV the radiator extensions which form part of the lower bulkhead would be attached following heat source insertion.

The inboard profile of the RSV/PSM is shown in Figure 5.1-3. In addition to the power system components and interfaces, the PSM carries the reaction control modules, consisting of two identical sets of hydrazine tanks and a thruster assembly, as shown.

The major support structure is the central conic section. The power conversion unit is mounted to the upper ring of this cone, with the heat sources, RCS modules and radiator supported from various points of this cone by ribs and spanners, as shown.

Figure 5.1-1. N/GPS Organic Rankine Cycle Spacecraft Profile

Figure 5.1-2. N/GPS Spacecraft Exploded View (Organic Rankine Cycle Version)

Figure 5.1-3. Organic-Rankine Power System Module Inboard Profile

The thermal interfaces are provided by three sections of radiator tubing, called "headers" although these are not plenums for radiator tubes in this single-tube radiator design. They are simple series extensions of the single radiator tube which are provided for thermal interface purposes. They are wound in flat circles in three different locations in the PSM. The lower or inlet header provides thermal contact for the radiator extension panels. The upper two headers provide similar heat-transfer surfaces for the upper and lower equipment shelves of the EM.

The method of heat transfer is similar for all three locations, and is best illustrated by reference to the radiator extensions which form part of the lower bulkhead. (Figure 5.1-4).

The panels are constructed of aluminum honeycomb with embedded ammonia-in-aluminum heat pipes. The boiler section of each heat pipe is bonded within the panel, within channels cut in the honeycomb. The condenser section is bent at right angles and fitted with a conducting saddle. The saddles are then bolted to a mating saddle on the header tube, with thermal grease applied to the smoothly fitting interface. This entire technology, including honeycomb-embedded heat pipes with right-angle bends mated via conduction contact to the heat sink, is taken directly from Fairchild ATS-6 experience. A total of 54 such heat pipes operating at similar temperature and heat flow, have performed for three years in space aboard the ATS-6 without failure.

The equipment module for the RSV is shown in Figure 5.1-5. The upper shelf, which consists of three sections (see Figure 5.1-2), contains the two cesium standards on each of two sections, and the momentum wheels and their control box on the third section.

Figure 5.1-4. Lower Closure Bulkhead-Radiator Extension (Both Spacecraft)

Figure 5.1-5. Equipment Module; Organic Rankine Spacecraft

The lower shelf contains the remainder of the electronic equipment complement, as shown in Figure 5.1-6. Half of this shelf contains the L_1 high-power amplifiers, designed to dissipate a total of 400 watts_t in the two sections. The only item not included in the equipment complement which is part of the Ø1 NDS is the load control unit. The remaining functions of this unit, which has many fewer functions to perform in this battery-less and essentially constant load spacecraft, was incorporated into the power system control box which is mounted to the underside of the support shelf of the PSM, and remains as part of the PSM.

5.2 BRAYTON-POWERED SPACECRAFT

The external profile and major dimensions of the Brayton-cycle space vehicle (BSV) are shown in Figure 5.2.1. The additional spacecraft height as compared to the RSV results from some added height required to accommodate the mid-radiator headers which delineate the two radiator sections, and from the desire to retain the same dimensions for the EM in both designs. In the RSV the antenna mounting plate is recessed some 5-inches below the lip of the radiator shell. In the BSV this plate is essentially flush with the top.

The inboard profile of the BSV/PSM is shown in Figure 5.2.2. In this case the entire power converter is contained within the central cone; now configured almost cylindrical. Upper and lower domes serve to hermetically seal the converter. This allows complete evacuation of the converter for ground testing of the C-103 version of the Brayton-cycle system. External access to the heat source container is still provided via a flange welded to the lower dome.

Mechanical attachment of the EM has been modified somewhat to account for the increased diameter of the mating ring. The thermal interface headers are similar in principle but differ in size and internal detail from those of the RSV (see Section VIII).

The BSV/EM (Figure 5.2-3) is identical to that of the RSV, except for the modification of the cross-braces to pick up the new attachment points, and for the inclusion of the fourth momentum wheel in the axial direction to provide the momentum cancellation of the spacecraft yaw rotation. This wheel is mounted at a very slight angle to the spacecraft z-axis to also cancel out the angular momentum of the horizontally-mounted Brayton rotating unit.

ITEM	NAME	SIZE LENGTH x WIDTH x HEIGHT	WEIGHT LBS (EA)	DISSIP WATTS	QTY USED
1	NAV SYS L ₁ HPA	16.0 x 40.0 x 2.0	21.5	200	1
2	L ₁ HPA FILTER	5.3 x 4.3 x 2.2	2.5	—	1
3	L ₂ HPA FILTER	4.8 x 4.0 x 2.0	1.6	—	1
4	NAV SYS L ₂ HPA	9.6 x 6.0 x 1.3	4.34	14.7	1
5	NAV SYS L ₁ /L ₂ SYN	10.4 x 10.3 x 2.5	7.43	10.6	1
6	NAV SYS L ₁ MOD IPA	11.25 x 4.68 x 1.65	3.08	3.4	1
7	NAV SYS L ₂ MOD IPA	6.7 x 3.6 x 1.65	1.7	1.5	1
8	NAV SYS DIG PROC/BB	11.7 x 10.2 x 5.8	20.8	21.1	1
9	PCM/TLM UNIT	10.65 x 8.28 x 5.75	12.0	8.0	1
10	NAV DC/DC CONVERTER	6.0 x 6.0 x 3.75	7.6	18.1	1
11	TT&C KIR-23	5.02 x 3.6 x 5.8	4.3	2.2	2
12	TT&C CMD DECODER	8.8 x 7.41 x 5.75	8.0	0.8	1
13	TT&C SIG COND	8.28 x 7.05 x 5.45	7.0	2.0	1
14	TT&C RF ASSY	8.0 x 6.5 x 6.5	5.3	2.7	1
15	TT&C TRANSMITTER	5.82 x 4.25 x 2.7	5.0	20.0	2
16	TT&C RCVR DEMOD	5.82 x 4.25 x 2.7	5.0	1.8	2
17	L ₁ HPA FINAL AMPLIFIER	16.0 x 40.0 x 2.0	15.0	200	1
18	CURRENT SENSOR	2.5 x 1.7 x 1.3	0.5	1.3	2
★19	PWR SYS CONT ELEC				
20					
◆21	AVCS CONTROL ELECT	16.0 x 10.0 x 6.0	17.0	8.5	1
◆22	MOMENTUM WHEEL	7.50 DIA x 4.15	11.0	10.0	3
◆23	CESIUM CLOCK	16.0 x 10.0 x 10.0	34.0	24.0	2
◆24	EARTH SENSOR	7.44 DIA x 122	13.5	3.0	1

◆ FOR EQUIP LOCATION See page 5-12.

Figure 5.1-6. Lower Equipment Shelf Reference Design (Both Spacecraft)

Figure 5.2-1. N/GPS Brayton Cycle Spacecraft Profile

Figure 5.2-2. Brayton Power System Module; Inboard Profile

Figure 5.2-3. Equipment Module - Brayton Spacecraft

The fluid-flow pattern in the BSV is shown in Figure 5.2-4.

An alternative equipment shelf heat transfer system is illustrated in Figure 5.2-5. This ducted shelf with coolant flowing through it was designed to be part of the RSV/PSM. The lower equipment shelf would bolt to this cooled shelf at the indicated locations between ducts. This design was found to be thermally superior to the reference design, resulting in a smaller total ΔT from equipment baseplate to coolant. However, it was ultimately discarded because the large number of welds required to fabricate the ducts and the duct-to-plenum connections were judged to result in a design of questionable reliability, particularly as the coolant fluid at this point is at a pressure of 76 psia. (Appendix G).

Figure 5.2-4. Working Fluid Loop; Brayton Spacecraft

5.3

OVERALL DIMENSIONS

Figure 5.3-1 shows the major dimensions and subsystems locations for the two reference designs. The BSV is shown in three-dimensional view in Figure 5.3-2.

CONCLUSION: THERMALLY SUPERIOR; MECHANICALLY UNRELIABLE

Figure 5.2-5. Alternate Equipment Shelf Design

ORGANIC RANKINE SPACECRAFT

BRAYTON SPACECRAFT

Figure 5.3-1. Major Dimensions

Figure 5.3-2. N/GPS External Views

5.4 FOUR SPACECRAFT CLUSTER

In order to enable a complete preliminary evaluation of the structural requirements for the spacecraft, an interface structure was designed to hold the four spacecraft cluster and mate with the IUS interface ring. This structure, shown in Figure 5.4-1, consists of welded channel beams which pick up the IUS ring at eight tie points. At the opposite face of the beams four cylindrical sections are attached which connect to the four SV thrust cones by marmon clamps. Beams and cylinders are each 8-inches high, and beam widths were adjusted for maximum combined loads as discussed in Section VI. The interface structure weighs 360 lbs.

The overall configuration of the four spacecraft cluster on the adapter and mounted on the two-stage IUS within the Shuttle cargo bay is shown in Figure 5.4-2. Overall length in the cargo bay is less than half the available length.

Figure 5.4-1. Four-Spacecraft Interface Structure

Figure 5.4-2. Spacecraft Mounted in Shuttle

SECTION VI

STRUCTURAL DESIGN AND ANALYSIS

6.1 SUMMARY OF ANALYSIS

A structural and dynamic study of the proposed nuclear-powered global position system satellite has been accomplished to assure its structural integrity for both the space shuttle and Titan IIIC environments.

The basic spacecraft concept is to provide a central cylinder (or truncated cone) to support the power system, with radial ribs and shelves to support the payload equipment. The central cylinder with the power conversion unit (PCU), heat source, and radiator is an independent lower structural system. The upper payload section is designed as a separate structural system which is inserted into the radiator and onto the central cylinder. When tied together, the combined system acts as one structural unit.

The structural integrity of this spacecraft is based on a strong central cylinder/cone structure. A marmon clamp interface connection to the IUS adapter structure assures the full load carrying capability of the cylindrical structure and also provides a simple and well-proven separation technique.

In determining the design criteria for this configuration, both the space shuttle and Titan IIIC are evaluated. The load factors for the shuttle crash conditions and Titan IIIC launch are applied here to a four-spacecraft grouping mounted to the IUS interface ring via a specially designed adapter structure.

The ascent and landing load factors for the shuttle require that the exact location of the spacecraft in the cargo compartment is determined. This is required because the specified angular accelerations would increase or decrease

the translational load factors on the spacecraft depending on its location. Therefore, for analytical purposes two cases were evaluated in which the spacecraft were located at the extreme opposite ends of the shuttle payload compartment. All design loads include a 1.5 safety factor to cover potential design and material changes.

Three mathematical models were developed to accomplish this study. The first one was a bar model of the spacecraft cantilevered at its interface. This model considered the translational and rotational mass inertias for loads and frequencies. This seven-mass bar model was synthesized to a one-mass model that exhibited the same frequencies. Finally, four synthesized spacecraft models were added to a beam/grid adapter structure which in turn was attached to the IUS interface points.

A separate calculation of the frequency modes of the power conversion unit as mounted in the reference-design spacecraft was also performed. It is concluded that the inclusion of shock-mounting isolators in this case is unnecessary and counter-productive.

6.2 BASIC SPACECRAFT MODEL

6.2.1 STRUCTURAL CONCEPT

Figures 6.2-1 and 6.2-2 depict the basic dimensions of the RSV and BSV configurations and the basic location of the equipment. An evaluation of the required structural members and the equipment weights resulted in a preliminary spacecraft weight estimate of 1120.0 pounds. The approximate distribution of this weight is shown in the figures. The IUS adapter weight of 360.0 pounds is used in the third analytical model which combines four spacecraft with the adapter and in turn is tied to the IUS interface.

The central cylinder of the power system modules is the prime structural element. It supports the PCU and heat source directly. The radiator is supported via radial ribs from the conical section. The equipment module is an independent structural system which has an upper and a lower shelf supporting equipment, frequency standards (clocks); inertia wheels, and antennas. Tied

Figure 6.2-1. Analytical Beam Model -- Organic Rankine Spacecraft

Figure 6.2-2. Analytical Beam Model -- Brayton Spacecraft

together by rings and radial ribs it forms a toroidal cylinder. During final spacecraft assembly it is lowered onto the upper central cylinder and is fastened to it as well as to the radiator structure.

Providing a central cylinder with high strength and resonant frequency allows changes in weight distribution and radial rib design. The assurance of good frequency separation of the shelves and ribs from the central cylinder is of primary importance. In obtaining this structural frequency separation, it is estimated that a central cylinder frequency of approximately 40.0 Hz would be satisfactory. The mass properties used for this study are given in Table 6.2-3.

Table 6.2-3. Mass Properties About Spacecraft CG

$$I_{x_0} = 134.98 \text{ slugs}$$

$$I_{y_0} = 149.64 \text{ slugs}$$

$$I_{z_0} = 164.00 \text{ slugs}$$

$$\text{Inertia Ratio} = \frac{I_{z_0}}{I_{y_0}} = 1.096$$

6.2.2 ANALYTICAL MODEL

Using the dimensions depicted in Figures 6.2-1 and 6.2-2, in conjunction with the weight table, a simple equivalent beam model was generated. In addition to transferring the weight laterally to the center line, the inertias about x, y, and z were also considered. Each of the six mass points shown in the model are analyzed for six degrees of freedom. The characteristics of a conical section

which basically increases the stiffness over that of an equivalent cylinder has been incorporated in the model. The primary objective of this model is to define a structurally sound central cylinder to carry all the static and dynamic loadings.

6.2.3 DESIGN CONDITIONS

The preliminary analysis of the N/GPS utilizes the design load criteria of both the Titan IIC launch vehicle and the space shuttle. Table 6.2-4 shows the conditions and load factors used for the Titan IIC criteria. Table 6.2-5 provides the translational and rotation factors for the space shuttle. Table 6.2-6 shows the emergency-landing condition. However, it will be seen that the 'normal' landing conditions provide the worst-case quasi-state loads in the critical lateral directions.

Table 6.2-4. Titan IIC Design Load Criteria

PHASE	ACCELERATION, g
LIFTOFF	
AXIAL	3.0*
LATERAL	1.5#
LATERAL	±5.0
MAXIMUM AIRLOADS	
AXIAL	3.0*
LATERAL	1.5#
TORSIONAL	±2.5
	2.5 @ 50-INCH RADIUS
STAGE I DEPLETION SHUTDOWN	
AXIAL	8.0*
LATERAL	6.5#
LATERAL	±3.0
STAGE II DEPLETION SHUTDOWN	
AXIAL	10.0*
LATERAL	8.0#
LATERAL	±5.0

* COMPRESSION

TENSION

Table 6.2-5. Space Shuttle Ascent/Descent Payload Design Load Factors★

EVENT DESCRIPTION	LOAD FACTORS (g's)*			ANGULAR ACCEL (RAD/SEC ²)		
	N _X (+ AFT)	N _Y (+ RIGHT)	N _Z (+ UP)	θ _X (+ RT ROLL)	θ _Y (+ PITCH UP)	θ _Z (+ NOSE LEFT)
LIFT-OFF & ASCENT						
LIFT-OFF	-2.54	±0.12	+0.09	+0.40	+0.10	+0.80
LIFT-OFF	-1.10	±0.44	-0.30	-0.50	+0.60	-2.20
LIFT-OFF	-1.40	±0.12	+1.20	+0.10	+0.60	+0.50
LIFT-OFF	-2.10	±0.06	-1.5	+0.30	-0.80	+0.50
MAX Q ENVELOPE	-1.90	±0.40	+0.25/0.5	+0.10	±0.02	±0.02
MAX g BOOST (STACK)	-2.87	±0.06	-0.10	-0.02	0.0	0.0
SRB POST STAGING	1.20	0.0	-0.70	TBD	TBD	TBD
MAX g (ORBITER ONLY)	-3.17	0.0	-0.60	0.0	0.0	0.0
DESCENT & LANDING						
HIGH-SPEED LANDING	+1.42 (+1.60)**	0.0	+0.60	0.0	1.74	0.0
HIGH-SPEED LANDING	-1.53 (-1.70)**	0.0	+4.92	0.0	+2.25	0.0
HIGH-SPEED LANDING	+1.28 (+1.44)**	0.0	0.23	0.0	-2.68	0.0
ONE-WHEEL LANDING	-0.55	±0.55	+2.69	±2.74	±1.38	±2.41
TAEM SYM PITCH MANEUVER	+0.44	0.0	+2.50	0.0	+0.08	0.0
TAEM UNSUM YAW MANEUVER	+0.87	±1.26	+1.00	0.0	0.0	0.0
ABRUPT SYMMETRICAL	+0.54	0.0	-1.00	0.0	+0.68	0.0
PITCH MANEUVERS	+0.49	0.0	-1.03	0.0	0.06	0.0

★ LOAD FACTOR IS DEFINED AS THE TOTAL EXTERNALLY APPLIED LOAD ON A PAYLOAD OR COMPONENT DIVIDED BY THE PAYLOAD OR COMPONENT WEIGHT, & CARRIES THE SIGN OF THAT APPLIED LOAD

* REFERENCED TO ORBITER STATION X = 1078, Y = 0, Z = 400

** FIRST VALUE FOR TOTAL PAYLOAD; PARENTHETICAL VALUE FOR PAYLOAD COMPONENTS

Table 6.2-6. Emergency Landing Ultimate Load Factors

LONGITUDINAL (+ AFT)	LATERAL (+ RIGHT)	VERTICAL (+ UP)
+9.0	± 1.5	+4.5
-1.5		-2.0

The Titan IIC load factors only considered combined axial and lateral translational effects. The space shuttle conditions are more complicated. Since the shuttle can launch and land payloads, the payloads will now be subjected to angular accelerations in addition to the translational accelerations as shown in Table 6.2-5. Therefore, the location of the spacecraft in the shuttle payload compartment is significant in determining the design load factors. All of the translational and angular accelerations are referenced to fuselage station 1078, the standard load and CG reference point in the shuttle cargo bay. A payload can be as far forward as 496 inches or as far aft as 224 inches from the reference station (See Figure 6.2-7.)

An example of the effect of location is gotten by considering a high speed landing condition with a spacecraft CG located 174. inches aft of station 1078, the $N_z = +4.92$ and $\theta_y = +2.25$. The load factors at station 1252 are shown in Figure 6.2-8.

The structural investigation for the shuttle covered the emergency landing condition, the high speed landing condition and a one-wheel landing condition. The lift-off and ascent criteria of the shuttle were less critical than that of the Titan IIC.

6.2.4 STATIC ANALYSIS

With the design conditions discussed in Section 6.2.3 and the analytical model discussed in Section 6.2.2, the NASA-NASTRAN program was used to determine the central cylinder loads as well as the interface loads. Table 6.2-9 provides the summary of the interface loads for the shuttle and Titan IIC.

Figure 6.2-7. Shuttle Installation Alternatives

Figure 6.2-8. Design Load Factors for Translation and Rotation

Table 6.2-9. Summary of S/C Interface Loads - Analytical Beam Model★

	SPACE SHUTTLE ORBITER						TITAN IIIC LAUNCH		
	EMERGENCY LANDING			DESCENT & LANDING					
	4.5g VERT	4.5g LAT.	-9.0 AXIAL	ONE-WHEEL LDG.		HIGH SPD. LDG.			
LBS IN.-LBS				FWD. PAYLOAD	AFT PAYLOAD	AFT PAYLOAD	y=5.0g z=3.0g	y=5.0g z = -10.g	y=5.0g z=8.0g
P _x , lbs.	-7476.	—	—	2456.	6123.	12478.	—	—	—
P _y	—	-7467.	—	6156.	-1510.	—	-8297.	-8297.	-8297.
P _z	—	—	-14935.	1460.	-1460.	-2605.	4978.	16594.	-13275.
M _x , in.-lbs.	—	222420.	—	-183375.	44978.	—	247133.	247134.	247134.
M _y	-222420.	—	—	73151.	182385.	371689.	—	—	—
M _z	—	—	—	—	—	—	—	—	—
δ, in.	.028	.028	—	.023	.024	.047	.032	.032	.032

★SEVEN BAR ELEMENTS
 SEVEN MASS POINTS - 6 DOF
 S/C WEIGHT = 1120.0 lbs.
 CG = 29.78 in.
 SAFETY FACTOR = 1.5
 δ - SPACECRAFT CG DEFLECTION

From Table 6.2-9, the high-speed landing of the shuttle is seen to result in the maximum load and bending moment. This provides the design condition for the central cylinder as the primary structural element. The conical section is .080 inches thick with an allowable bending moment of 484137 in. lbs. and an allowable axial load of 74223.0 pounds. This results in a margin of safety for the high speed landing of:

$$M.S. = \frac{1}{R_b + R_a} - 1 = \frac{1}{.77 + .17} - 1 = .064$$

over and above the safety factor of 1.5 applied in the calculation.

Table 6.2-9 also shows the maximum lateral deflection of the spacecraft under the calculated bending moment. When combined with the deflections of the entire spacecraft mounted on the IUS adapter (Section 6.3) this will determine the clearance requirements between adjacent spacecraft and between spacecraft and shuttle cargo bay walls.

6.2.5 DYNAMIC ANALYSIS

Using the same analytical model described above, a modal analysis was accomplished with the six concentrated masses, each having six degrees of freedom. The results of this analysis are given in Table 6.2-10 with the eigenvalues and mode characteristics as shown.

Table 6.2-10. Summary of S/C Frequencies Analytical Beam Model

MODE	MODE CHARACTERISTIC
$f_1 = 41.9 \text{ Hz}$	TORSION
$f_2 = 46.8 \text{ Hz}$	BENDING IN X DIRECTION
$f_3 = 46.84 \text{ Hz}$	BENDING IN Y DIRECTION
$f_4 = 93.9 \text{ Hz}$	2ND BENDING MODE IN X DIRECTION
$f_5 = 94.0 \text{ Hz}$	2ND BENDING MODE IN Y DIRECTION
$f_6 = 103.1 \text{ Hz}$	2ND TORSIONAL MODE
$f_7 = 152.8 \text{ Hz}$	1ST AXIAL MODE

6.3

SPACECRAFT-TO-IUS ADAPTER

This phase of the study evaluates the capability and requirements of the adapter which holds four spacecraft on to the interface ring of the IUS.

6.3.1

ANALYTICAL APPROACH

The primary effort is to define an adapter to support four spacecraft that will be supported by an IUS and fit into the space shuttle. The size of the analytical model will be dominated by the adapter. To keep the overall size of the model to a minimum, the six-mass bar model was synthesized to a one-mass system with an eight-bar structure, shown in Figure 6.3-1, that is equivalent to the bar model. This concept is valid for this purpose since the one-mass model will satisfy the load introduction into the adapter and provide the proper dynamic coupling. The comparison of resonant frequencies for the three primary modes of the two models is shown in Table 6.3-2.

Table 6.3-2. Model Frequency Comparisons
Frequency in Hz

Mode	Bending	Torsion	Axial
Bar Model	47.	42.	152.
One-Mass Model	45.	48.	144.

6.3.2

ADAPTER DESIGN AND ANALYTICAL MODEL

The adapter shown in Figure 6.3-1 is composed of six-inch channels around the periphery and spacecraft support areas and eight-inch channel cross beams tying the four spacecraft together.

Four cylindrical transition sections are attached to the adapter at eight points each. These provide an even load transfer to the spacecraft via a marman clamp at the other end. The eight spacecraft attachment points are shown in Figure 6.3-1, as are the attach fittings to the IUS interface ring. Clearly, the eight bar spacecraft model has been synthesized to properly introduce loads into the adapter structure.

Figure 6.3-1. Four-Spacecraft Adapter Structure

Figure 6.3-3 depicts the computer drawing of the analytical model. The spacecraft mass allows translational and rotational degrees of freedom while the eight adapter masses only consider translational degrees of freedom.

6.3.3 DESIGN CONDITIONS

The design conditions are the same as those used in Section 6.2 for the individual spacecraft. Since the four spacecraft are located off the shuttle longitudinal (x) axis, the landing conditions will produce different design load factors on each spacecraft. Figure 6.2-8 shows the design load factors used in analyzing the adapter.

6.3.4 STATIC ANALYSIS

Table 6.3-4 gives the maximum interface loads between the adapter and IUS. It was assumed that the eight interface attachment points were fully fixed. The interface bolt pattern is identical to that of the Titan IIIC interface. These interface loads will vary depending on the type of attachments and the flexural characteristics of the IUS interface ring with its supports. For the purpose of this analysis the interface ring is assumed to be rigidly mounted to the IUS. Figure 6.3-5 shows the CG displacement and satellite top displacement for the worst shuttle case (emergency landing).

With the current reference design clearances between adjacent spacecraft and between spacecraft and shuttle cargo bay wall are approximately two inches each way. Since the calculated deflections are unidirectional under the quasi-static loads applied, this clearance is adequate. Deflections under dynamic (vibration) loading can be in opposite directions, and hence additive between adjacent spacecraft, but these will be seen to be much lower in magnitude (next section).

It should be recognized that the Titan IIIC cases are included for reference only. The same four-spacecraft adapter structure would not be applicable to the Titan because of the diametral limitations of the Titan shroud. Preliminary indications are that this may be enlarged to a 147-inch dynamic diameter by 30-foot length in which case a two-over two type of adapter structure may be applicable.

3-D VIEW NP GPS STUDY

P

ELEMENTS

1, 1000

Figure 6.3-3. Four-Spacecraft Analytical Model

Table 6.3-4. Summary of Adapter - IUS Interface Analytical Model

P = LBS. M = IN.-LBS. δ = IN.	SPACE SHUTTLE ORBITER						TITAN IIIC LAUNCH		
	EMERGENCY LANDING			DESCENT & LANDING					
	4.5g VERT	4.5g LAT.	9.g AXIAL	ONE-WHEEL LDG.	HIGH SPD. LDG.		y=5.0g z=3.0g	y=5.0g z = -10.g	y=5.0g z=8.0g
				FWD. PAYLOAD	AFT PAYLOAD	AFT PAYLOAD			
P _x	4203.	9477.	1535.	5193.	2855.	4731.	10018.	12236.	9165.
P _y	-1661.	-4063.	-422.	2658.	1367.	1852.	-4374.	-4984.	-4139.
P _z	-4895.	-6024.	8317.	4332.	-2954.	6612.	-6327.	12796.	-10948.
M _x	-10975.	17933.	-24785.	-9008.	-7426.	15293.	28188.	47465.	41958.
M _y	-32646.	52768.	-73711.	-25153.	-22078.	-45885.	83201.	140533.	124152.
M _z	-11257.	-6751.	543.	-3703.	-5703.	-12695.	-7682.	-8105.	-7984.
S/C δ _x	-.380	—	±.016	-.26	.080	.180	.005	.018	.014
S/C δ _y	—	.443	±.074	.108	.020	.005	.157	.575	.558

RESULT: ADAPTER STRENGTHENED FOR MAXIMUM BENDING MOMENT

MAXIMUM SPACECRAFT DEFLECTIONS IN SHUTTLE EMERGENCY LANDING

Figure 6.3-5. Maximum Spacecraft Deflections in Shuttle Emergency Landing

6.3-5 DYNAMIC ANALYSIS

The same analytical model described above was used for the modal analysis of this integrated system. Again the IUS back-up structure was considered rigid for this analysis. The results of this modal analysis are given in Table 6.3-6 which gives the frequency values and the mode characteristics. Figures 6.3-7 and 6.3-8 depict the mode shapes of the two lowest frequency vibration modes. The displacements shown are amplified by a scale factor of 20.

Table 6.3-6. Summary of 4 S/C + Adapter to IUS Frequencies Analytical Model

$f_1 = 9.97 \text{ Hz}$	2 S/C BENDING TOWARDS EACH OTHER Y- DIRECTION
$f_2 = 15.73 \text{ Hz}$	2 S/C BENDING IN SAME DIRECTION Y-DIRECTION
$f_3 = 17.46 \text{ Hz}$	2 S/C BENDING IN SAME DIRECTION X-DIRECTION
$f_4 = 20.03 \text{ Hz}$	2 S/C BENDING TOWARDS EACH OTHER X-DIRECTION
$f_5 = 24.60 \text{ Hz}$	2 S/C BENDING OPPOSITE TO EACH OTHER X-DIRECTION
$f_6 = 27.29 \text{ Hz}$	2 Y AXIS S/C BENDING TOGETHER IN X-DIRECTION
$f_7 = 30.19 \text{ Hz}$	2 Y AXIS S/C BENDING OPPOSITE IN X-DIRECTION

PLOT 1 12 85 20 NP GPS

1ST. MODE

$f_1 = 9.97 \text{ Hz}$

DEFORMATION MAGNIFIED 20 TIMES

Figure 6.3-7. IUS Structure System, 1st Mode

PLOT 2 12 85 20 NP GPS IUS STRUCT. SYS.

2ND MODE
f = 15.73 Hz

The combined CG of the four spacecraft and its adapter when joined to the IUS was estimated.

If mounted in the cargo bay as shown in Figure 6.2-7 another payload of approximately 28,000 pounds must be mounted with its CG within a few feet of the location shown ($x = 915$), if a full 65,000-pound load is to be launched.

If it is to be launched alone, it can be mounted with its CG at any location between $x = 942$ and $x = 1133$. Two orientation options are shown in Figure 6.2-7 with the combined CG at $x = 1078$.

The RSV/PCU is an independent mass system tied to the upper ring of the conical structure at three points. To evaluate the dynamic characteristics of the PCU, Sundstrand's mathematical model of the PCU was used. The support legs were modified so that the regenerator supports were tied to the cone ring by "V" members. Identical shock mount characteristics were also used. Figure 6.5-1 depicts the analytical model while Figure 6.5-2 depicts the computer plot of this model.

The analytical model was evaluated with and without shock mounts and compared to the fundamental structural frequencies of the spacecraft and stowed shuttle system.

The results of the analysis is given in Table 6.5-3. With the shock mounts, the first frequency of 8.6 Hz is relatively close to that of the stowed shuttle system of 10 Hz (Table 6.3-6). This would be an undesirable condition in which the PCU mass would be highly amplified. On the other hand, the fundamental frequency of the PCU without the shock mounts is 33 Hz. This is sufficiently isolated from the spacecraft and stowed shuttle system frequencies that the chance of dynamic coupling is remote. This will assure that the loads applied to the PCU will be very close to the loads at the spacecraft interface. Hence, no shock mounts are recommended for this case, where the PCU is mounted to the rigid central cone-cylinder.

ANALYTICAL MODEL FOR MODAL ANALYSIS; ORGANIC RANKINE CONVERTER SYSTEM

Figure 6.5-1. Power Conversion Unit

Figure 6.5-2. 3 - D View KIPS Dynamic Model

Table 6.5-3. Organic-Rankine Converter System Summary of Model Analysis

MODE CHARACTERISTIC	MODE NO.	FREQUENCY, Hz	
		WITH ISOLATORS	WITHOUT ISOLATORS
LATERAL MOTION (REGENERATOR)	1-2	8.66	33.7
LATERAL MOTION (ACCUMULATOR)	3-4	21.70	69.9
THRUST MODE	5	22.40	88.6
TORSIONAL MODE	6	44.30	119.8

SECTION VII

ATTITUDE AND REACTION CONTROL SYSTEMS

7.1 EQUIPMENT COMPLEMENT SUMMARY

The Attitude and Velocity Control Subsystem, AVCS, comprised of the Attitude Control Subsystem (ACS) and Reaction Control Subsystem (RCS), serves to stabilize the nuclear-powered space vehicle (SV) following separation from the launch vehicle. The AVCS consists of sensors, controllers and torquers as required to satisfy the various mission operation requirements. These operations include initial stabilization and Earth acquisition after separation, orbit position control, on-orbit attitude control, momentum unload, and orbit maintenance.

The AVCS operates autonomously, with ground command override, during initial acquisition and normal on-orbit operations including unload. Orbit position control and orbit maintenance thrustings are ground initiated and ended while the concurrent attitude control is autonomous. Status and monitoring signals, for both ACS and RCS, are telemetered to the ground.

Performance is not impaired by any single point failure. The ACS is completely redundant and the RCS thrust capability is multiply redundant.

AVCS component characteristics for an Organic-Rankine-Cycle-powered SV are given in Table 7.1-1. Modifications caused by use of a Brayton cycle power plant are considered in Section 7.6. Component locations are shown in Figure 7.1-2.

7.2 REQUIREMENTS

7.2.1 EARTH ACQUISITION

After separation from the IUS launch vehicle with its concomitant rate ($\pm 0.5^\circ/\text{sec}$) and attitude errors ($\pm 2^\circ$), the AVCS shall bring the SV to Earth pointing within 1.0 degrees and $.05^\circ/\text{sec}$, using roll and pitch jet control.

Table 7.1-1. AVCS Weight and Power

COMPONENT		QUANTITY PER SV	WEIGHT PER SV (LB.)	MAXIMUM POWER (WATTS)
ACS	CONTROL ELECTRONICS ASSY	1*	18.5	7.5
	CONBINED EARTH SENSOR	1*	13.5	4
	REACTION WHEEL	3	33.0	20
	SPINNING SUN SENSOR	2	2.6	4
RCS	0.2 LB. THRUSTER	12	10.6	—
	TANKS	2	10.4	—
	REMAINDER OF COMPONENTS	—	13.1	0.75
	PROPELLANT	—	55.0	—

*INTERNALLY REDUNDANT

7.2.2 SPIN-RATE CONTROL

With the SV pointing at the nadir, yaw torquing jets shall bring the SV to its nominal yaw spin speed within 0.02 rpm.

7.2.3 VELOCITY CONTROL

The system must accept ground commands to initiate thruster firing during a) drift orbit circularization (4° /degree drift), b) correction of IUS launch velocity errors (80 ft/sec, 3σ), c) orbit position, and d) two orbit corrections ($\epsilon = .0.1$). Total delta-V impulse shall be oriented within TBD degree of nominal.

7.2.4 ON-ORBIT CONTROL

The SV Z axis shall point toward the Earth within ± 0.5 degrees and the SV shall rotate about the yaw axis at 0.5-1 rpm nominal with stored momentum control using reaction wheels controlling about pitch and roll axes. Momentum unload may be accomplished by spin-rate control or using wheel speed information to drive the appropriate jets.

OPTICAL AXIS

3° X 3° FOV

OPTICAL AXIS - X_{SV}

PARALLEL $\pm 0.5^\circ$

+X

SPIN EARTH SENSOR

OPTICAL AXIS

3° X 3° FOV $\pm 10\%$

10 $\pm .1^\circ$

10 $\pm .1^\circ$

OPTICAL AXIS

3° X 3° FOV $\pm 10\%$

+Z

PARALLEL $\pm .1^\circ$

STATIC EARTH SENSOR

OPTICAL AXIS

3° X 3° FOV $\pm 10\%$

10 $\pm .1^\circ$

10 $\pm .1^\circ$

OPTICAL AXIS

3° X 3° FOV $\pm 10\%$

$\phi - Z_{SV}$ PARALLEL $\pm .1^\circ$

RCS THRUSTER MODULE (2)

X_T

Y_T

Z_T

X_{SV}

Y_{SV}

Z_{SV}

$-X_{SV}$

$X_T - X_{SV}$

$Y_T - Y_{SV}$

$Z_T - Z_{SV}$

PARALLEL $\pm .1^\circ$

SPINNING SUN SENSOR (2)

OPTICAL FOV (β)

$-39.5 \leq \beta \leq +89.5$

DETECTION RANGE (β)

$.25^\circ \leq \theta \leq 28^\circ$

PULSE READ WIDTH = 2θ

2θ

θ

β

θ

2θ

β

$+Z$

$-X$

REACTION WHEELS

+Z_{SV}

$-X_{SV}$

120° $\pm 0.5^\circ$

7-3

7.3 DESCRIPTION

7.3.1 INITIAL ACQUISITION

Four GPS satellite vehicles are simultaneously lifted by an IUS into a nominal 12-hour orbit, inclined at 63° . These vehicles are eventually to be positioned at prescribed locations in precise circular 12-hour orbits; the locations can be anywhere in the orbit. To minimize fuel consumption, the SV's are dispensed from the IUS at the location which minimizes angular travel rate to the final orbital locations consistent with maximum allowable constellation deployment time.

Prior to separation of the SV's from the IUS, the IUS is positioned so that the on-board SV's look directly down at the Earth. One SV is dispensed, the IUS backs off, another is dispensed, and the process continues until all four SV's are dispensed and the IUS deploys to prevent impact with the SV's. (Alternately, each SV after separation can acquire the Earth and move itself relative to the IUS by firing its longitudinally directed thrusters.) (See Figure 7.3-1.)

At separation, each SV is pointing to the Earth within 2° (Ref. 1) and has 3σ inertial rates of about $\pm 0.5^{\circ}/\text{sec.}$ due to the combination of IUS errors and separation maneuvers. The 0.2-lb. pitch and roll attitude control jets with 3-ft. moment arms can null these rates of the vehicle, with lateral moment of inertia of 150 slug ft.², in about 1.1 seconds. Filtering of the SV's Earth sensor output is used to obtain pseudo-rate control. During this rate nulling period, the vehicle may rotate an additional 0.6 degrees from nadir. Because the Earth subtends an angle of about 27 deg from orbit altitude and because the Earth sensor has monotonically increasing outputs over a range of about 7 degrees and correct sense of attitude error over about 12 degrees, there is little likelihood that the described sequence of events will not occur. A back-up acquisition sequence is described in Appendix H.

1. Attachment 2 to Contract F04701-76-C

System Specification, Performance and Design Requirements for the Department of Defense Space Transportation System, Volume 3 Interim Upper Stage System Segment, Basic Requirements for, 76 January 15 SS-STS-100, Volume 3, Code Identification 07868.

ORBIT INSERTION SEQUENCE

Figure 7.3-1. Orbit Insertion Sequence

When the spacecraft Z axis is centered on the Earth within 1 degree, and vehicle rates are below .05 deg/sec, normal on-orbit control with reaction wheels is activated. In the normal operation the SV spins about the satellite-earth line with closed loop reaction wheel control about axes normal to that line. Vehicle spin rate is uncontrolled between limiting rates equivalent to wheel unload momentum. When attitude and rate control have improved to 0.5 degrees and .01 degrees/sec., the yaw torquing jets are pulsed to bring vehicle rate to 1.03 rpm. (At this rate, RSV angular momentum is just the negative of the nuclear engine's 17.6 ft.-lb.-sec. spin momentum.) Vehicle yaw rate is determined either by monitoring the vehicle's spin sun sensor and/or by

monitoring the time variation of the reaction wheel's speed. The vehicle is now in its normal on-orbit operating mode and orientation but probably in the wrong position in orbit.

The figure below illustrates the jet control mode when the Earth is in the field-of-view of the CES, after separation from the IUS. Pseudo-rate control is obtained by a deadband plus hysteresis circuit with lag feedback to drive the appropriate jets.

Yaw spin rate control is based on the time between successive transits of the sun by the spin sun sensor looking normal to the yaw axis. When, as shown below that time exceeds (or is less than) the nominal time the appropriate yaw jets are fired (as couples) for a fixed time (approx. 0.13 sec.) so that the spin speed is changed in increments of about 0.01 rpm.

7.3.2 POSITIONING SV's IN THE CONSTELLATION

After separation from IUS, the SV's will be in nominal drift orbits which provide TBD day/day longitudinal drift. This nominal drift rate is selected based on considerations of SV spread in the constellation, delta-V available for the maneuver,

allowable time for final positioning, and ground coverage during thrusting periods. After drift movement to the desired constellation position, the vehicle is brought to circular orbit speed by jet thrustings by the along-velocity change technique described below.

Deviations between nominal and actual drift orbits are determined after ground tracking. Corrections of IUS induced errors may be commanded after only a few orbits or may be delayed for each SV until the SV is at its proper position in the constellation and final orbit circularization is commenced.

Along-orbit velocity changes are achieved by firing the X axis thrusters (see Section 7.5) as the SV is rotating at about 1 rpm about the local vertical. Each pulse is started about 15 degrees before the X axis is in the orbit plane and ends the same number degrees after; less than 1% inefficiency, due to mis-directed thrusting, is incurred. Pulse on-times are about 5 seconds so that a high specific impulse (200 sec) is achieved. Body-mounted sun sensors are used to initiate and stop the thrusting. (As shown in Section 7.5, ΔV thrusting along the orbit is triply redundant, i. e., 3 jet failures can be tolerated without performance degradation.) Y torquing thrusters are used to counter the Y torquing effects of the X pointing thrusters. Total propellant requirements are given in Section 7.5.

IUS radial velocity errors are correctable by firing the Z pointing thrusters at the 12-hour orbit radius or by firing to achieve the necessary along-orbit velocity changes at drift orbit apogee or perigee.

All delta-V changes are achieved by ground command firing of the SV plus or minus X axis pointing jets. These are pulsed when the X axis is within 15° of the orbit plane. Pulse enables are obtained by using the spin sun sensor. The sensor looks in a plane containing the Y and Z axes.

Pulsing starts a ground-commanded delay time after the detector sees the sun and stops after that delay time plus the time for SV 30° rotation. Concurrent roll and pitch control by jets uses lead-lag compensation. The delay time varies as a function of the angle between the Earth-sun line and the orbit plane.

7.3.3 NORMAL ON-ORBIT OPERATION

During normal operation, the vehicle spins about the satellite-Earth line, with vehicle attitude controlled about axes normal to that line. Earth sensor information, after processing, is used to drive reaction wheels for lateral axis control.

Vehicle spin rate is uncontrolled; variations in rate occur primarily as the vehicle accumulates angular momentum due to external torques. Periodic, autonomous or ground-controlled, unloading of the accumulated angular momentum to re-establish the desired spin rate is accomplished by jet thrusting. Spin rate information indicating the unload requirement is obtained from tachometers monitoring wheel speed over a spin cycle or from body-mounted sun sensors when the vehicle spin axis is normal to the vehicle-sun line, i. e., twice each orbit.

Unload occurs no more frequently than every 10 days, 0.5 foot-pound-second being unloaded by jets firing as couples while the vehicle is rotating (Appendix H). Two 0.2 - pound thrusters, 6 feet apart, and firing for 0.42 second will unload 0.5 foot-pound-second.

Wheel torquing is based on attitude errors from the CES followed by filtering to achieve loop stabilization. Resolution must be made of desired pitch and roll torque outputs into wheel drives for the two operating wheels (the three wheels have their spin axes in the vehicle X-Y plane, the axes are 120° apart).

7.3.4 ORBIT MAINTENANCE

As a natural result of the forces on the SV, its orbit will become distorted. Ref. 2 indicates an ellipticity increase from zero to 0.01 in about 5 years. Ellipticity corrections are achieved by sequential firing along the orbit at orbit apogee (or perigee) and perigee (or apogee). Ref. 2 suggests reduction of ellipticity to 0.006 (not 0.0) requiring 30 ft/sec. (Ideal reduction to $\epsilon = 0$ requires 63.4 ft/sec.) Fuel for low orbit corrections is budgeted. The technique for along-velocity-vector firing described in Section 7.3.2 is used to achieve the ellipticity reduction. The on-orbit design options which were explored, the control laws, and the results of control simulations are given in Appendix H.

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2. Global Positioning System, Space Vehicle, Critical Design Review, Mission Analysis Working Group. Rockwell International, Space Division, 2-6 June 1975.

7.4.0 ACS COMPONENTS

The components used in the AVCS have been selected based on their suitability for the intended application; advantage is taken of their prior development and application in NDS vehicles.

7.4.1 COMBINED EARTH SENSOR (CES) BARNES 13-169

The static earth sensor portion of the CES is an internally redundant no-moving-part horizon sensor which is designed for operation at 11,000 nm altitude and operates on energy in the range $14-16\mu$. Radiant energy is converted into electrical signals by thermopiles which are sampled every quarter second and logically checked for "validity". Error signals due to sun, moon, and failed internal components are thereby rejected.

Pitch and roll sensor outputs are analog and errors at null are $0.1^\circ \ 3\sigma$. A linear output signal range exists for pitch/roll displacements of ± 4 degrees.

The spinning earth sensor part of the CES looks along the -X axis and senses intersections of its line-of-sight with the Earth. A redundant unit also looks along the -X axis.

Unit weight is 13.5 lbs. It consumes 4 watts max.

7.4.2 REACTION WHEELS (RW) - BENDIX

The reaction wheels are basically Bendix's OGO pitch and roll wheel design with increased motor torque. Components parameters are:

Unit Angular Momentum	0.75 ft. lb. sec. at 490 rpm
Stall torque	20 in. - oz.
Weight	11 lbs.
Maximum power	10 watts

7.4.3 SPIN SUN SENSOR - ADCOLE 18980

This unit, with digitized output, looks in the vehicle X-Z plane with total field-of-view of 180° . Angular resolution of sun location is $1/2^\circ$ and transition accuracy is $\pm 15^\circ$. The unit weight is 1.3 lbs. and consumes 2 watts.

CONTROL ELECTRONICS ASSEMBLY

The CEA contains all AVCS electronic functions listed below. Estimated unit weight is 18.5 lbs and estimated maximum power 7.5 watts. The performing characteristics of the individual CEA elements shall be:

- a. Mode Select Logic (MSL) - The MSL shall decode a 14-bit binary serial digital word received, via ground command from the TT&C subsystem and appropriately route the information to the desired CEA element.
- b. Jet Control Electronics (JCE) - The JCE shall be utilized to perform initial earth acquisition and, if necessary, earth reacquisition. In addition, the JCE shall be utilized for temporary backup attitude control.
 - (1) JCE Signal Interface -- The JCE shall have the following signal interfaces:
 - (a) Ground command signals from the MSL.
 - (b) An analog signal from each axis of the two-axis horizon sensor.
 - (c) An enable/disable signal to activate and deactivate the JCE.
 - (d) Output signals to the JSL commanding specific thruster firing.
 - (e) Signals from the spin earth sensor.
 - (2) JCE Functional Characteristics - The JCE shall contain jet control electronics including gains, shaping, and deadbands.
- c. Wheel Control Electronics (WCE) - The WCE shall be utilized to provide on-orbit space vehicle pointing control.

- (1) WCE Signal Interface - The WCE shall have the following signal interfaces:
 - (a) Override attitude bias command signal from the MSL.
 - (b) An analog signal from each axis of the two-axis horizon sensor.
 - (c) A ground command signal, from the MSL, indicating deactivation of a wheel causing appropriate compensation to be switched in.
 - (d) An enable/disable signal to activate and deactivate the WCE.
 - (e) Output control signals to the reaction wheels.
 - (2) WCE Functional Characteristics - The WCE shall contain control electronics to activate the reaction wheels thereby providing control torques to the vehicle. Gains, shaping, inverters and signal mixing shall be included in the control electronics. Logic shall also be provided that switches, upon ground command, a failed wheel out and appropriate compensation in.
- d. Momentum Dump Logic (MDL) - The MDL shall be utilized to desaturate (dump momentum) the reaction wheels periodically in order to maintain continuous vehicle control.
- (1) MDL Signal Interface - The MDL shall have the following signal interfaces:
 - (a) Signals from the reaction wheel tachometers indicating each reaction wheel spin speed.
 - (b) An enable/disable signal to activate and deactivate the MDL.
 - (c) Ground commanded attitude override commands from the MSL.

- (d) Output signals to the JSL commanding specific thruster firings. These signals shall also turn the thruster heaters on automatically 60 ± 5 minutes before thruster firing.
 - (e) Signals from the spin sun sensor.
- (2) MDL Functional Characteristics - The MDL shall contain control electronics to monitor the angular velocity of each reaction wheel and command thruster pulsing. Also, it monitors SV spin speed using spin sun sensor outputs. The threshold level at which an axis will be "dumped" shall be selectable by ground command.
- e. Attitude Bias Electronics (ABE) and Commandable Thruster Firing Logic (CTFL) - The ABE shall be utilized to bias the pitch and roll attitude via ground command. The CTFL shall provide for open loop thruster firing via ground command.
 - (1) ABE Signal Interfaces - The ABE shall have the following signal interfaces:
 - (a) Commands from TT&C
 - (b) Bias commands to WCE and JCE
 - (c) Enable/Disable discrete inputs from TT&C
 - (2) CTFL Signal Interfaces - The CTFL shall have the following signal interfaces:
 - (a) Commands from TT&C
 - (b) Output commands to the JSL
 - (c) Enable/Disable discrete inputs from TT&C
- f. Jet Select Logic (JSL) - The JSL shall be utilized to receive thruster commands from the other CEA units and route these commands to the required thruster logic and valve driver amplifiers to perform these functions. A Jet Watchdog Logic (JWL) shall be provided to insure

that a runaway jet due to an electronic failure is automatically shut down in less than one minute.

7.5 RCS DESCRIPTION

The Nuclear Powered GPS makes direct use of the propulsion components and modular assemblies from the solar-powered spacecraft with modifications to the extent suggested by reduced propellant and thrust level requirements and aft-end jet location.

Because the 13" tanks with an accumulator provide more than sufficient impulse margin, the 6 1/4" pressurant accumulator was eliminated and the propellant per tank reduced to a nominal load of 27.5 lb. The increased ullage volume of the propellant tank retains the blowdown ratio of 3 to 1. Table 7.5-1 details the propellant budget and indicates a 24% contingency reserve.

Table 7.5-1. Propellant Budget

Maneuver	ΔV Req'd ft/ sec, 3	Avg. Is (lbf. sec/ lb)	H ₂ H ₄ Req'd lb
Null, Roll, Pitch and Yaw Separation Rates	(0.5° / sec, each axis)	100	0.1
IUS Injection Error Correction	Radial = 75 Normal = 12 Tangential = 16	205 200 200	12.5 3.1 4.1
Circularization of Drift Orbit (4° / Day)	Tangential = 23.5	200	6.1
Momentum Dump	(0.5 ft. lbf. sec every 10 days)	125	0.3
Ellipticity Correction	Tangential = 60	200	15.5

Total Required	41.7 lb (75.8%)
Total Usable (2 Tanks)	55.0
Contingency/ Reserves	13.3 lb (24.2%)

The dual valve 0.2 - lbf thrusters for attitude control are retained in a modular configuration where eight thrusters in each of two modules provide three-axis attitude control as well as translational ΔV capability. The 5 lbf thrusters are eliminated since the nuclear-powered spacecraft is non-rotating during injection error correction and high torques for precession are not required. The 0.2 lbf is also sufficient for nodal correction of orbit non-circularity.

The thruster modules have been moved aft from the center of mass to the separation plane of the satellite for four reasons:

- 1) Electronic equipment occupies the central volume at the plane of the center of mass.
- 2) The cylindrical power system radiator is designed as an integral unit, preferably with no cutouts that would compromise its performance.
- 3) Installation of the RCS modules is facilitated by using the annular segments of space between the radiator and the thrust cone 90° from the two heat sources.
- 4) The penalty incurred by translational thrusting along the X axis using thrusters offset somewhat from the CM (requiring rotational stabilization with other jets) is small because of the larger moment arms in the stabilizing jets.

Figure 7-5-2 illustrates jet arrangement and utilization and indicates a necessary and sufficient level of redundancy. Added flexibility is provided by the fact that individual thrusters in either the backup or prime subset can be activated to replace single failed thrusters in the other subset.

The schematic arrangement of components shown in Figure 7.5-3 is essentially that used on the Solar Power GPS. The accumulators and 5 lbf thrusters with their latch valves have been removed and an interconnect latch valve has been added. The intermodule line is installed last, utilizing mechanical couplings with redundant seals.

VIEW INTO AFT END

	Controlled Axis	(+)		(-)	
		Prime Jets	Backup Combinations	Prime Jets	Backup Combinations
Body Fixed Torquing	Body X	3/ 11	4/ 11, 3/ 12	5/ 13	5/ 14, 6/ 13
	Body Y	7	16, 1, 2	15	8, 9, 10
	Body Z	3/ 13	4/ 14, 3/ 14, 4/ 13	5/ 11	6/ 12, 5/ 12, 6/ 11
	Along Vel Vector	7/ 16	8/ 15, 3/ 11, 4/ 12 3/ 12, 4/ 11, 5/ 13 6/ 14, 5/ 14, 6/ 13 7, 8, 15, 16	8/ 15	7/ 16, 3/ 11, 4/ 12 3/ 12, 4/ 11, 5/ 13 6/ 14, 5/ 14, 6/ 13 7, 8, 15, 16
Inertial Space Torquing with S/V Rotating	⊥ Orbit Plane	7/ 16	8/ 15, 3/ 11, 4/ 12 3/ 12, 4/ 11, 5/ 13 6/ 14, 5/ 14, 6/ 13	8/ 15	7/ 16, 3/ 11, 4/ 12 3/ 12, 4/ 11, 5/ 13 6/ 14, 5/ 14, 6/ 13
	Along Radius Vector	3/ 13	4/ 14, 3/ 14, 4/ 13	5/ 11	6/ 12, 6/ 11, 5/ 12
	ΔV (During Rotation)				
	Along Velocity Vector		Prime 1	Backup 2, 9, 10	
	⊥ Orbit Plane		1	2, 9, 10	
	Along Radius Vector		+8/ 16 -7/ 15	8, 16 7, 15	

/ = couples

Figure 7.5-2. Jet Orientation/Utilization

Figure 7.5-3. Reaction Control Fluid Schematic

The total loaded weight of both modules is 89.1 lb with the nominal propellant load of 55.0 lb of hydrazine. Table 7.5-4 itemizes the component weight and indicates nominal value and heater power levels.

Thrusters are activated through driver and logic circuits located in the AVCS Control Electronics Assembly.

7.6 CHANGES CAUSED BY USE OF THE BRAYTON POWER SYSTEM

The angular momentum of the Brayton power system (0.68 ft.lb. sec.) is very much less than that of the Rankine power system (17.6 ft.lb. sec.), so that it is impractical to use the former for counterbalancing the yaw rotation of the BSV. Therefore, a large fixed speed reaction wheel with an angular momentum of 10 ft. lb. sec. along the yaw axis is included for this purpose. This allows the RSV to rotate at 0.51 RPM with full momentum cancelling. Because the Brayton rotating unit has been mounted horizontally, with its angular momentum in the XZ plane, a slight canting of the reaction wheel, at an angle of 4-degrees to the yaw axis, will cancel its momentum as well.

A suitable unit is a Bendix manufactured wheel of the ATS type which has a unit weight of 19.5 lbs, AM of 8.5 ft. lb. sec, and consumes 10 watts at 1450 rpm. Because the SV is turning at 0.5 rpm, i. e., somewhat more slowly than an SV with an Organic Rankine Cycle engine, there is about a 10-watt savings in the reaction wheels used for attitude control, so no AVCS net power difference exists.

Table 7.5-4. RCS Weight and Power Summary

Component/ Assv	Number (Both Modules)	Unit Wt, lb	Sub Total, lb	Nominal Power, Watts
Tank (13" Dia)	2	5.22	10.4	
.2 lbf Thruster	16	.66	10.6	
Isolation Valves	3	1.04	3.1	
F/ D & Pressure Valves	4	.36	1.4	
Manifolds	AR	.50	.5	
Filter	2	.30	.6	
Harness & Electrical Interface Box	1	2.20	2.2	
Sensors & Heaters	AR	.7	.7	.75 watts
Module Structure & Brackets	AR	4.6	4.6	
			34.1 lb Dry	
Nominal Propellant Load: Requirements			41.7	
Contingency			<u>13.3</u>	
			89.1 lb Wet Total	

SECTION VIII

THERMAL DESIGN AND ANALYSIS

8.1 PROBLEMS ADDRESSED

The major thermal design problems which were addressed in the study are as follows:

- 1) Heat pipe performance in spinning mode.
- 2) Equipment shelf thermal performance.
- 3) Overall spacecraft thermal variations in eclipse orbit.
- 4) Temperature variation and control of frequency standards.

These are each discussed in the following paragraphs.

8.2 PERFORMANCE OF RADIAL HEAT PIPE IN SPINNING MODE

The reduction in pumping capacity of an ATS-type heat pipe spinning about an axis normal to its longitudinal axis was evaluated. For the GPS application it is proposed to orient heat pipes in a radial direction normal to the vehicle spin axis to provide thermal control of the frequency standard clocks mounted on the upper shelf. The pipe evaporators will be located 10 inches from the spin axis; the condensers will be 33 inches from the axis. (See sketch Figure 8.2-1). A grooved ATS heat pipe, which uses ammonia as the working substance, provides a capillary pumping head of about 1.4 psf. With the evaporator inboard, this pumping head will be reduced by the centrifugal forces acting on the fluid in accordance with

$$\Delta P = \frac{\rho \omega^2}{g} (r_2^2 - r_1^2)$$

where

- ρ = density of liquid ammonia, about 40 lbs/ft³
 ω = angular velocity
 r = distance to end points from spin axis

This equation was used to generate the curve shown in Figure 8.2-1, where it may be noted that the percentage loss in pumping capacity is negligible providing spin rates of about one RPM are not exceeded.

Figure 8.2-1. Pumping Capacity of ATS-6-Type Heat Pipe (Ammonia in Aluminum) Rotating About An Axis Normal To Its Axis

8.3

EQUIPMENT SHELF TEMPERATURE CONTROL

Dissipation of electronic equipment thermal power is achieved by transferring heat along heat pipes which are thermally coupled to a coolant header. The header is an extension of the power system fluid system. The heat pipes are imbedded in the honeycomb panel and connected to the header wall at the saddle region. Figures 8.3-1 and 8.3-2 show the heat pipe layout for the lower and upper shelves, respectively.

ITEM	NAME	SIZE LENGTH x WIDTH x HEIGHT	WEIGHT LBS (EA)	DISSIP WATTS	QTY USED
1	NAV SYS L ₁ HPA	16.0 x 40.0 x 2.0	21.5	200	1
2	L ₁ HPA FILTER	5.3 x 4.3 x 2.2	2.5	—	1
3	L ₂ HPA FILTER	4.8 x 4.0 x 2.0	1.6	—	1
4	NAV SYS L ₂ HPA	9.6 x 6.0 x 1.3	4.34	14.7	1
5	NAV SYS L ₁ /L ₂ SYN	10.4 x 10.3 x 2.5	7.43	10.6	1
6	NAV SYS L ₁ MOD IPA	11.25 x 4.68 x 1.65	3.08	3.4	1
7	NAV SYS L ₂ MOD IPA	6.7 x 3.6 x 1.65	1.7	1.5	1
8	NAV SYS DIG PROC/BB	11.7 x 10.2 x 5.8	20.8	21.1	1
9	PCM/TLM UNIT	10.65 x 8.28 x 5.75	12.0	8.0	1
10	NAV DC/DC CONVERTER	6.0 x 6.0 x 3.75	7.6	18.1	1
11	TT&C KIR-23	5.02 x 3.6 x 5.8	4.3	2.2	2
12	TT&C CMD DECODER	8.8 x 7.41 x 5.75	8.0	0.8	1
13	TT&C SIG COND	8.28 x 7.05 x 5.45	7.0	2.0	1
14	TT&C RF ASSY	8.0 x 6.5 x 6.5	5.3	2.7	1
15	TT&C TRANSMITTER	5.82 x 4.25 x 2.7	5.0	20.0	2
16	TT&C RCVR DEMOD	5.82 x 4.25 x 2.7	5.0	1.8	2
17	L ₁ HPA FINAL AMPLIFIER	16.0 x 40.0 x 2.0	15.0	200	1
18	CURRENT SENSOR	2.5 x 1.7 x 1.3	0.5	1.3	2
19	PWR SYS CONT ELEC				
20					
21	AVCS CONTROL ELEC	16.0 x 10.0 x 6.0	17.0	8.5	1
22	MOMENTUM WHEEL	7.50 DIA x 4.15	11.0	10.0	3
23	CESIUM CLOCK	16.0 x 10.0 x 10.0	34.0	24.0	2
24	EARTH SENSOR	7.44 DIA x 122	13.5	3.0	1

Figure 8.3-1. Lower Equipment Shelf Reference Design (Both Spacecraft)

Figure 8.3-2. Equipment Module - Brayton Spacecraft

Experimental and flight data from ATS-6 heat pipes (grooved aluminum tubing with ammonia for working fluid) reveals that the temperature gradient between the evaporation and condensation regions is small ($\sim 1^{\circ}\text{C}$) when the heating level is in the range of interest for this nuclear GPS design: about 30 watts along a 30" heat pipe. The largest changes in temperature occur across the interfaces which connect the heat pipe to the heat generating components and to the sink. In the current application, an additional gradient is realized across the coolant fluid film adhering to the header wall. ATS-6 experience shows that the contact resistance due to the mounting interfaces is reduced considerably by employing conductive thermal grease and mechanical mounting pressure. The convective heat transfer across the header film will be a function of the flow characteristics and available heat transfer area.

An analysis was conducted to evaluate various header configurations which will provide the required heat transfer from the lower equipment shelf at acceptable gradients. The selected header design for the Organic-Rankine space vehicle provides an annular flow area by a tube-in-tube arrangement. The outer tube has 1.5 in. outer diameter. The flow gap of 0.035 -inch is maintained by conductive wire spacers which run the length of the header. This design, shown in Figure 8.3-3 has the advantage of achieving high values of convection coefficients without sacrificing the heat-transfer surface area in contact with the fluid. This header is fabricated and assembled as part of the power system module, and undergoes the same rigid cleanliness and inspection procedures as the rest of the power system.

The fluid properties for the Dowtherm A working fluid of the RSV/PSM are given in Table 8.3-4. The rate of flow is a specified $322\text{ lb}_m/\text{hr}$. Following dissipation in the upper header the temperature at the inlet of the lower header is found to be 31.3°C (88.3°F). The temperature at the outlet is calculated based on a total components dissipation of 532.7 watts as indicated in Figure 8.3-5. It may be assumed that all component heating is transferred to the fluid leading to a temperature rise of 8.2°C (14.7°F) and hence a temperature at the outlet of 39.5°C (103°F). The relative location of the dissipators with respect to the header is selected such that the critical high-power amplifiers are nearest to the inlet region in order to take advantage of the cooler regime.

Figure 8.3-3. Heat Pipe/Header Interface

Table 8.3-4. Properties of Dowtherm - A

Temp (°F)	Density (lb _m /ft ³)	Viscosity (lb _m /hrft)	Conductivity (Btu/fthr°R)	Specific Heat (Btu/lb°R)
80	65.82	7.70	0.0814	0.388
100	65.27	6.36	0.0805	0.388
120	64.72	5.03	0.0797	0.396

Figure 8.3-5. Lower Equipment Shelf Component Dissipation (Watts_T)

The Reynolds number based on the hydraulic diameter ($D_o - D_i$) and average fluid properties is found to be

$$\text{Re} = \frac{745.5}{D_o + D_i}; \quad D_o, D_i \text{ in inches} \quad (1)$$

Hence, for the range of diameters under consideration, the flow is laminar and the friction factor may be assumed to be

$$F = 24/\text{Re} = 0.032 (D_o + D_i) \quad (2)$$

Thus, in terms of the annulus dimensions, the pressure drop P (psf) and the heat transfer coefficient H ($\text{Btu}/\text{ft}^2\text{hr}^\circ\text{R}$) are given by

$$\Delta P = \frac{0.4315}{(D_o + D_i) (D_o - D_i)} \quad (3)$$

and

$$H = \frac{49.86}{D_o - D_i} \left[\frac{1}{L} \frac{D_o - D_i}{D_o + D_i} \right]^{0.33} \quad (4)$$

where L is tube length in inches.

Equations (1) through (4) were used in a parametric evaluation of the annular header. The results are shown in Figures 8.3-6 and 8.3-7. The intent is to minimize the temperature gradient from the annulus walls to the fluid without resorting to excessively large diameters or high pressure drops. Structural and spacecraft envelop limitations indicate a suitable outer diameter of about 1.5" which yields a gradient of 2°C (3.6°F) in the portion of the header around the shelf quadrants supporting the high power amplifiers.

A thermal model was generated to determine the temperature profile on the lower mounting shelf and the results are shown in Figure 8.3-8. It is noted that the temperatures are in the vicinity of 40°F (104°F) due to the inherently high temperature of the fluid. Acceptance and qualification test levels for the components can be in excess of 40°C , as specified in MIL SPEC 1504A. This is possible since no batteries and, apparently, no other temperature-critical components are on the shelf.

Similar considerations for the Brayton spacecraft led to a header design which utilizes a tube with internal fins as shown in Figure 8.3-9. Comparison is made with a standard circular tube of the same diameter. The design parameters were selected to correspond to a 1.75" O.D tube but, as can be seen in Figure 8.7-9, only a slight increase in the diameter is needed to achieve allowable pressure drops without significantly affecting the temperature gradient from the tube wall to the gas. Figure 8.3-10 gives the components baseplate temperature.

A substantial reduction of the baseplate temperature of the high power amplifiers can be achieved by using an auxiliary heat pipe immediately below the radial heat pipes saddles and running continuously around the shelf periphery. The purpose of this heat pipe is to more evenly distribute the dissipation along the header and thus reduce the heat flux to the header in the region of the amplifiers. The temperature profile in this case is as shown in Figure 8.3-11. It is noted that the reduction in the amplifier temperature is achieved at the expense of higher baseplate temperatures in the vicinity of the fluid outlet. Clearly, a redistribution of components is required with this approach, so that the least temperature-sensitive boxes are located in the last

Figure 8.3-6. Annular Header - Equipment Shelf (Organic - Rankine)

Figure 8.3-7. Annular Header - Equipment Shelf (Organic - Rankine)

Figure 8.3-8. Organic - Rankine Spacecraft Lower Equipment Shelf Temperature Map

quadrant of the equipment shelf. This also illustrates the desirability of further reduction of the fluid temperature at the outlet of the radiator. A potential additional radiator area of some 25 square feet is available by use of the antenna mounting plate as a radiator extension area, which was not taken advantage of in this analysis.

Temperature control of the upper shelf on which the cesium clocks and momentum wheels are mounted is achieved by the same method as that of the lower shelf. The header, however, is a circular tube of 1-inch, I. D. for both RSV and BSV versions, and the design in both cases includes an auxiliary circular heat pipe for enhancing the thermal interface between the radial heat pipe saddles and the coolant header wall. The function of the auxiliary heat pipe will be to distribute the dissipation over the entire header area and hence eliminate local hot spots. Steady state temperatures based on the power listing shown in Figure 8.3-1 are shown in Figure 8.3-12.

Figure 8.3-9. Equipment Shelf Header Design Parameters (Brayton Spacecraft)

UPPER EQUIPMENT SHELF TEMPERATURE MAP

Figure 8.3-12a. Organic Rankine Spacecraft

UPPER EQUIPMENT SHELF TEMPERATURE MAP

Figure 8.3-12b. Brayton Spacecraft

Overall spacecraft temperature variations during the worst case eclipse orbits were determined for a variety of spacecraft surface finishes. Because the selected design approach results in essentially the entire spacecraft being cooled by the power system fluid loop, the vehicle was treated as a single-node isothermal body for the purpose of the calculation. Since eclipse transients have time constants of the order of an hour, and the fluid loop transit time through the spacecraft is of the order of seconds, this is believed to be a reasonable assumption for the spacecraft as a whole.

Minimization of the overall spacecraft temperature variation will have the beneficial effect of minimizing the variations of the critical frequency standards, discussed in the next subsection.

The resulting temperature profile for various surface finishes is given in Figure 8.4-1. An absorptance to emittance ratio (α/ϵ) of 0.4/0.76 refers to a silvered teflon coating which is available in the form of an adhesive tape. $\alpha/\epsilon = 0.3/0.9$ are standard degraded properties values for a limited class of white paints which include SG-13 and Z-93. The aluminum side of vapor-deposited aluminum of Kapton has an α/ϵ ratio of about 0.11/0.035. The Kapton side of a 2-mil sheet has an α/ϵ of 0.36/0.76.

Figure 8.4-1 indicates that a silvered teflon coating offers the most satisfactory selection. It combines a low overall temperature ($\sim 24^{\circ}\text{C}$) and a low worst-case orbit variation as seen in Figure 8.4-2 (4°C from maximum to minimum). In addition, it is superior to white paints in the stability of optical properties and shows only slight dependence of absorptance on the angle of incidence. These characteristics are important for accurate orbit predictability.

Figure 8.4-1. Overall Spacecraft Temperature Transient in Maximum Eclipse Orbit

Figure 8.4-2. N/GPS Orbital Temperature Variation

8.5

ACTIVE THERMAL CONTROL OF FREQUENCY STANDARDS

The design of the heat path between the clocks and the header is directed towards minimizing the gradient so as to maintain the clock temperatures in the neighborhood of 40°C. This approach to coupling results in a variation on the clock baseplate temperature which nearly parallels the orbital variation of the sink although reduced to approximately 2°C in the absence of active control (See Figure 8.5-1).

A heater/thermostat system can be employed at the clock baseplate for an active control which would minimize the orbital temperature variations still further. Calculations were made based on a best estimate for the conductance from the baseplate to the header fluid, which is found to be about 7.0 Btu/hr°R for the RSV, and a clock dissipation of 24 watts. A heater element rated at 30 watts and thermostatically controlled at 38°C (ON for $T < 38^{\circ}\text{C}$ and OFF for $T \geq 38^{\circ}\text{C}$) operating at 13% duty cycle

can maintain the temperature variation at less than 0.2°C . This is shown in Figure 8.5-1 for the maximum eclipse orbit. Also shown in Figure 8.5-1 is the effect a change in the clock-to-fluid conductance would have after the thermostat set point is fixed. This could occur during launch, or after a period of time at temperature in space. Clearly, the most effective design would be one for which the set point can be shifted by ground command during orbital operation.

Figure 8.5-1. CS Frequency Standard Temperature Variations with Heater/Thermostat Control

SECTION IX

HIGHER ALTITUDE VERSION

9.1 HIGHER ALTITUDE VERSION

The nuclear-powered reference designs are quite insensitive to operating orbit altitude, and could be flown in higher orbit without modification, except for the Earth sensor coverage angle. However, in order to preserve RF illumination levels on the Earth without substantial radiated power increases would require modification of the navigation antenna. Table 9.1-1 lists the variation in the angle subtended by Earth for various altitudes, together with a nominal antenna diameter for the L1 signal based on first order optical considerations. At some altitude between 48 and 120 hour orbital periods, a reflector-type antenna would become more effective for the L1 - 19 cm wavelength.

Table 9.1-1. Higher Altitude Versions

ORBIT PERIOD	ALTITUDE	EARTH ANGLE	NOMINAL ANTENNA DIAMETER
DAYS	N. MI.	DEGREES	METERS
½	10,900	26.9	0.49
1	19,350	17.2	0.77
2	32,700	10.9	1.22
5	62,950	5.92	2.24
10	102,000	3.74	3.56

ARRAY

↑

↓

REFLECTOR

$\lambda_{L1} = .19 \text{ M}$

For the 120-hour alternative orbit suggested by the original ground-rules, a reflector antenna with diameter only slightly smaller than the spacecraft diameter is adequate to maintain essentially the same illumination levels as are achieved from the 12-hour orbit. This illustrated in Table 9.1-2, conservatively

assuming an antenna efficiency of 40%. The half-power beamwidth of the 1.7 meter reflector is 8.0 degrees, compared to the Earth angle of 5.9 degrees, so that illumination should be fairly uniform over the surface of the Earth.

Table 9.1-2. Fixed Antenna Design for 120-Hour Orbit

SPACECRAFT DIAMETER = 1.82 M
 ANTENNA DIAMETER = 1.7 M
 HALF-POWER BEAMWIDTH = 8.0 DEG.
 ANTENNA GAIN (η) = .40 = 25.0 db

L₁ SIGNAL COMPARISON

	120 HR. ORBIT			12 HR. ORBIT		
	P	C/A	UNIT	P	C/A	
PRNSA OUTPUT	20.9		dBw	20.9		
WORST CASE SIGNAL RATIO	-4.8	-1.8	dB	-4.8	-1.8	
ANTENNA GAIN	25.0		dB	13.7		
NET LINK LOSSES	-197.3		dB	186.8		
USER SIGNAL	-156.2	-153.2	dBw	-157.0	-154.0	

Figure 9.1-3 illustrates the spacecraft configuration as modified for the 120 hour-orbit.

Figure 9.1-3. Spacecraft Design for 120-Hour Orbit

SECTION X

WEIGHT AND COST SUMMARY

10.1 WEIGHT AND COST SUMMARY

The weight breakdowns for the two reference spacecraft are given in Tables 10.1-1 and 10.1-2. As can be seen the higher converter weight of the BSV is almost exactly compensated by the somewhat heavier radiator, so that the weight of the two spacecraft are within a few pounds of each other. The weight of the four spacecraft cluster plus adapter would therefore come to approximately 4400 lbs, or a nominal 5000 pounds with adequate margin. Including the 32,000 lbs two-stage IUS, a total of 37,000 lbs, would be the shuttle-chargeable launch weight. Also chargeable to the mission would be the heat-source cooling kit required until the power system is started in parking orbit. This would consist of a water tank plus a pump, valves and controls. Based on a four-spacecraft heat load of 12.8 kw_T (BSV), the water consumption rate would be approximately 45 lb/hr, assuming that the resultant water vapor would be vented overboard through the shuttle auxiliary 4-inch vent port. This cooling kit would fit within the IUS support structure; however, it has not as yet been designed in detail.

Cost estimates for the two reference spacecraft were derived from the SAMSO cost model, 1973 update (Tables 10.1-3 and 10.1-4). These costs are relevant when compared to each other; however, their relevance to actual spacecraft costs may be questioned because of the dated nature of the data in the cost model. For example, no allowance for inflation was made in the cost estimating relationship (CER's) because this would have distorted them considerably beyond the level of commonly known costs.

Table 10.1-1. Weight Breakdown, Organic Rankine Spacecraft

TOTAL SPACECRAFT WEIGHT , LBS.		1008.2
ANTENNA ASSEMBLY		<u>38.1</u>
SHELF	8.1	
ANTENNAS	30.0	
EQUIPMENT MODULE		<u>359.4</u>
STRUCTURE		<u>78.8</u>
CYLINDER SYSTEM	24.0	
RINGS	5.1	
RIBS	9.2	
TOP SHELF	15.0	
BOTTOM SHELF	25.5	
EQUIPMENT		<u>280.6</u>
CS. FREQ. STDS.	68.0	
REACTION WHEELS (3)	33.0	
SENSORS	16.1	
ACS CONTROL BOX	18.5	
SHELF EQUIPMENT	145.0	
POWER SYSTEM MODULE		<u>610.7</u>
STRUCTURE		<u>129.5</u>
CYLINDER	28.0	
RING	34.0	
RIBS	30.8	
SUPPORT SHELF	12.2	
BOTTOM COVER	24.5	
REACTION CONTROL SYS.		<u>89.1</u>
TANKS & PROPELLANT	65.4	
THRUSTERS	10.6	
LINES & COMPONENTS	13.1	
RADIATOR SYSTEM		<u>72.1</u>
RINGS	15.6	
LONGERONS	9.8	
SHELL	33.9	
HEADERS	5.0	
SPIRAL TUBE	7.8	
POWER SYSTEM		<u>320.0</u>
PCU	160.0	
MHW	160.0	
4 S/C + ADAPTER = 4040 + 360 = 4400 LBS.		

Table 10.1-2. Weight Breakdown, Brayton Powered Spacecraft

TOTAL SPACECRAFT WEIGHT , LBS.		1001.6
ANTENNA SUBASSEMBLY		<u>38.1</u>
SHELF	8.1	
ANTENNAS	30.0	
EQUIPMENT MODULE		<u>356.6</u>
STRUCTURE		<u>65.0</u>
CYLINDER SYSTEM	12.7	
RINGS	4.0	
RIBS	8.8	
TOP SHELF	15.0	
BOTTOM SHELF	25.5	
EQUIPMENT		<u>291.6</u>
CS FREQ. STDS	68.0	
REACTION WHEELS (4)	44.0	
SENSORS	16.1	
ACS CONTROL BOX	18.5	
SHELF EQUIPMENT	145.0	
POWER SYSTEM MODULE		<u>606.9</u>
STRUCTURE		<u>139.4</u>
CYLINDER	45.0	
DOMES	17.4	
RINGS	27.7	
RIBS	22.0	
SUPPORT SHELF	12.8	
BOTTOM COVER	24.5	
INSULATION	2.8	
REACTION CONTROL SYS.		<u>89.1</u>
TANKS & PROPELLANT	65.4	
THRUSTERS	10.6	
LINES & COMPONENTS	13.1	
RADIATOR SYSTEM (AI)		<u>92.6</u>
RINGS	18.6	
LONGERONS	10.2	
SHELL	36.9	
HEADERS & TRANSFER	14.2	
VERT. TUBES	12.7	
POWER SYSTEM		<u>273.0</u>
HSA	101.0	
MINI-BRU	38.0	
RECUPERATOR	84.0	
DUCTING	15.0	
INSULATION	15.0	
CONTROL BOX	15.0	
PARASITIC LOAD	5.0	

Table 10.1-3. Recurring Costs -- Organic Rankine Spacecraft

(COSTS IN K\$)

SUBSYSTEM	Normalized CER Costs	Complexity Factor	Complexity Weighting	Technology Factor	Technology Weighting	Composite Factor	Adjusted Costs
Structure, TC	400	1.11	0.445	0.90	0.400	1.00	400
Commun/Nav	500	1.51	0.550	0.83	0.310	1.23	614
TT & C	275	1.17	0.445	0.65	0.435	0.92	254
EPS	950*	—	—	—	—	1.00	950*
ACS/RCS	458	1.20	0.550	0.69	0.350	1.00	459
Total Subsystems Direct Cost							2677
Program Level Cost @ .30							803
Burden + G&A @ 1.06							3689
Fee @ .13							932
Total Spacecraft Recurring Cost							8101
Nuclear Fuel @ \$400/W _T							1920
Total Unit Recurring Cost							10,021

* Power System Contractor Estimate +\$200K For Heat Sources and Other EPS Functions

Table 10.1-4. Recurring Costs -- Brayton Spacecraft

(COSTS IN K\$)

SUBSYSTEM	Normalized CER Costs	Complexity Factor	Complexity Weighting	Technology Factor	Technology Weighting	Composite Factor	Adjusted Costs		With Super-Alloy EPS
Structure, TC	400					1.00	400		
Commun/Nav	500					1.23	614		
TT & C	275					0.92	254		
EPS	1350 *					1.00	1350 *		
ACS/RCS	500					1.00	500	→	1100*
Total Subsystem Direct Cost							3118		2868
Program Level Cost @ 0.30							935		860
Burden + G&A @ 1.06							4297		3952
Fee @ 0.13							1085		998
Total Spacecraft Recurring Cost							9435		8678
Nuclear Fuel @ \$400 /W _T							1280		1280
Total Unit Recurring Cost							10,715		9958

*Power System Contractor Estimate +150K for Heat Source and Other EPS Functions

Costs of the power systems themselves were based on ERDA-supplied estimates, and adjusted as indicated in the Tables.

Nuclear fuel costs are based on quantity use which would be typical of the GPS program. Fuel costs were not burdened in the analysis, since the nuclear fuel itself would be part of an intergovernmental transfer and would not show up on the contractor's books. However, costs for heat source fabrication were included in the contractor cost estimates.

The nonrecurring development costs of the first nuclear-powered GPS spacecraft were similarly estimated (Table 10.1-5) although three of the five subsystem costs were arbitrarily assigned, and no claim can be made for their accuracy. It is assumed herein that the power system development is funded separately, and is not included herein.

Table 10.1-5. Brayton and Organic Rankine; Non-Recurring Costs *

(COSTS IN K\$)

	Normalized CER Costs	Complexity Factor	Complexity Weighting	Technology Factor	Technology Weighting	Composite Factor	Adjusted Costs
Structure, TC	1200	1.07	0.44	1.0	0.42	1.07	1284
Commun/Nav	2250	1.51	0.46	0.4	0.44	0.87	700 *
TT & C	1150	1.17	0.45	0.4	0.45	0.71	500 *
EPS	4000	—	—	—	—	1.00	4000 *
ACS/RCS	2272	1.20	0.53	0.4	0.37	0.78	1781
Total Direct							8265
Program Level Costs @ 0.4							3306
							11,571
Burden + G&A @ 1.06							12,265
Fee @ .13							3,099
Total Non-Recurring (RDT&E) Cost							26,934

* Based on SAMSO Cost Model of July 1973

* These Values Have Been Arbitrarily Assigned Based on Estimated Legacy From Earlier Phases of GPS Program

* Estimated Power System Integration Cost; Non-Recurring EPS Development Cost Is Not Included

SECTION XI

CRITICAL PARAMETERS -- RESULTS

11.1 POWER TO USER

The determination of user power for the two reference designs, including the two versions of the Brayton design, is dependent on the prime spacecraft power less the fixed power requirements, and taking into account the efficiency of conversion of DC-to-RF of the high power amplifiers.

The fixed power requirements (i. e., not dependent on specific design) are shown in Table 11.1-1. Because the reference-design vehicles are not equipped with batteries or other power storage capability, attention to peak as well as average power requirements is necessary.

The AVCS power consumption varies as a result of build-up in speed of the momentum wheels over the ten-day unload cycle. Following momentum unload, wheel power consumption is close to zero, gradually building to a maximum near 20 watts over 10 days. (In practice, wheel peak power consumption could be reduced to roughly half this value by pre-loading angular momentum in the direction opposite to the normal build-up, so that consumption would vary from 10 watts to zero and back to ten watts before the next unload period).

Telemetry and command power consumption is based on NDS values, with the peaks occurring during telemetry transmission periods.

Losses in the electric power subsystem are based on the prime power source of 3-phase AC (100 or 120 watts) from the generators. The assumed AC-to-DC conversion losses of 5% are considerably less than the typical losses experienced by DC-to-DC converters. The losses accounted for in Table 11.1-1 are based only on the power required by the fixed elements listed in the table.

Table 11.1-1. Fixed Power Budget

<u>HOUSEKEEPING</u>			<u>PEAK</u>	<u>AVG.</u>
AVCS	SPINNING SUN SENSORS	—		
	EARTH SENSOR	3.0		
	CONTROL UNIT	8.5		
	NOM. WHEELS	20/10		
			31.5	21.5
TTEC	PCM/TM	8.0		
	CDC	0.8		
	KIR-23	2.2		
	SIG COND.	2.0		
	RF ASSY	2.7		
	X-MITTER	20.0		
	RCVR/DEMODO	<u>1.8</u>		
			37.5	7.5
EPS	LCU (INCL IN PCU	—		
	AC-DC CONV (η = .95)		7.7	5.0
	HARNESS		<u>10.0</u>	<u>10.0</u>
	TOTAL HSKPNG		<u>86.7</u>	<u>44.0</u>
<u>NAV SYSTEM (LESS HPA)</u>				
	L1/L2 SYN	10.6		
	L1 MOD/IPA	3.4		
	L2 MOD/IPA	1.5		
	BB PROCESSOR	21.1		
	CS STANDARDS	<u>30.0</u>		
	TOTALS		<u>66.6</u>	<u>66.6</u>
			153.3	110.6

The fixed power requirements of the navigation system are also shown in Table 11.1-1, based on NDS values, and assuming that only one of the two frequency standards is operating at any given time. The remainder of the spacecraft prime power is assumed to be available for powering the final amplifiers of the navigation transmitters.

The operating efficiency of these solid state amplifiers is dependent on their junction temperatures, which depends in turn on their baseplate temperatures. An estimate of this temperature dependence was made * as shown in Figure 11.1-2. From the equipment shelf temperature calculations as shown in Figures 8.3-8 and 8.3-11 the efficiency values assumed for the two reference designs are shown in Figure 11.1-2. This was rounded off to 0.3 for the RSV and 0.4 for the BSV, in Table 11.1-3.

* Personal communication of Capt. Wayne LeCompte, SAMSO/YE

Figure 11.1-2. High Power Amplifier Efficiency Variation With Temperature

Table 11.1-3 . Total Power Budget (Watts)

SYSTEM	ORGANIC RANKINE	BRAYTON		
		Cb	SUPERALLOY	Cb
BOL THERMAL POWER	4800	3200	3200	2400
BOL AC POWER	875	735	665	490
7-YR AC POWER	823	678	614	452
HPA $\eta_{DC \rightarrow RF}$	0.3	0.4	0.4	0.5
HPA POWER INPUT				
OPTION (4)	644	483	—	—
OPTION (2) + 1 dB	—	—	430	—
OPTION (2)	—	—	—	277
TOTAL FIXED POWER	153/111	153/111	153/111	153/111
7-YR POWER MARGIN	26/68	42/84	31/73	23/64
BOL POWER MARGIN	78/120	99/141	82/124	60/102

Table 11.1-3 shows the resulting overall power budget. Both the 4800 watt_T RSV and the 3200 w_T BSV can achieve the so-called option (4) level of power-to-user (Ref. Section 2.1) with adequate power margin for seven years. The superalloy BSV, on the other hand, can only achieve option (2) plus 1 dB in the L₁ (P) link (i. e. - 154 dBw L₁ (C/A) and -162 dBw L₁ (P), although at a lower cost (Ref. Section 10).

Although not carried through in the design detail phase, a 2400 W_T Brayton vehicle of the same size would be capable of a baseplate temperature low enough to achieve a high-power - amplifier efficiency approaching 0.5. If feasible, option (2) might be achieved by such a vehicle with a power margin similar to that of the other designs, as shown in Table 11.1-3.

11.2 MEAN MISSION DURATION (MMD)

From the discussion of Section 2.2, it is apparent that the major gains to be achieved in MMD will come about from an extension of the design lifetime of the life-limiting subsystems. Because the power degradation of the solar array in the $\phi 1$ NDS is the current life-limiting element, replacement by a nuclear power system with a 7-year design lifetime will result in an evident gain in this critical parameter. The exact value of this gain will depend on the reliability achieved by the nuclear power system, as well as by the other subsystems in the spacecraft.

If it is assumed that the failure rates of the other subsystems are identical to those of the NDS and are constant for 7 years, then the dependence of MMD on the reliability of the nuclear power system can be calculated, as shown in Figure 11.2-1. Reliability design goals for the nuclear power systems have been set at 0.95, which would lead to an MMD of approximately 5.9 years. Even if this goal is not fully achieved, however, an MMD in excess of 5.5 years appears assured. No more precise claims can be made at this time, obviously, without full-system test data on the nuclear power system and a more detailed study of the other subsystems as applied in the reference designs.

Figure 11.2-1. Variation of Spacecraft MMD with Reliability of Power System

11.3 ORBIT PREDICTABILITY

Three major sources of errors in spacecraft ephemerides predictions which may not be modelable, and hence correctable by extended periods of ground observation, have been evaluated. These are:

1. Errors in predicting surface temperatures.
2. Errors in alignment or pairing of jet firings during momentum unload.
3. Errors in calculating solar forces due to spacecraft misalignment (i. e., spacecraft axis not precisely aligned with axis of rotation).

These are the major effects which are believed to apply to the reference nuclear-powered spacecraft; not necessarily to all spacecraft in general. In particular, item 3) will apply to rotating spacecraft only. Stationary spacecraft will have a counterpart effect due to attitude control errors or control deadband.

The magnitude of these effects have been estimated, as reported in Appendix J, and summarized in Table 11.3-1. In all cases errors in the more important (in terms of translating into user position errors) radial direction are very small, and errors in the transverse direction, also small, can probably be reduced still further.

Table 11.3-1. Maximum Orbit Ephemerides Errors Due to Unpredicted Spacecraft Conditions

ERROR SOURCE	ERROR RATES (ft/day)	
	R DIRECTION	TRANSVERSE TO R
<u>THERMAL EFFECTS</u>	0.03	0.30
1. PREDICTION ERROR IN TEMPERATURE OF EARTH-VIEWING FACE = 10°C	0.025	0.25
2. PREDICTION ERROR IN TEMPERATURE OF SPACE-VIEWING FACE = 2°C	0.005	0.05
3. MEAN TEMPERATURE DIFFERENCE ON OPPOSITE SIDES OF CYLINDER DUE TO HELICAL COOLING COIL (ORGANIC RANKINE) = 10°C	2.8×10^{-4}	2.8×10^{-3}
<u>UNLOAD PERTURBATIONS</u>	0.0132	1.3
4. JET MISALIGNMENT = 0.5°, ATTITUDE ERROR = 1°		
a. IMPULSE PERPENDICULAR TO ORBIT PLANE	0.0016	0.155
b. IMPULSE ALONG RADIUS VECTOR	0.0016	0.155
5. JET MISMATCH = 5% , ATTITUDE ERROR = 1°		
a. IMPULSE PERPENDICULAR TO ORBIT PLANE	0.005	0.5
b. IMPULSE ALONG RADIUS VECTOR	0.005	0.5
<u>SPACECRAFT MISALIGNMENT (CONICAL ROTATION MODE)</u>	0.065	0.13
6. ALIGNMENT ERROR = 0.5°, UNIFORM REFLECTIVITY	0.025	0.05
7. ALIGNMENT ERROR = 0.5°, NON-UNIFORM REFLECTIVITY	0.025	0.05
8. ALIGNMENT ERROR = 0.5°, SELF SHADOWING EFFECT	0.04	0.08
SIMPLE TOTAL, ALL EFFECTS (FT./DAY) (M/DAY)	0.11	1.73
	0.03	0.53

For example, the thermal effects are due mainly to the assumption that the temperature of the Earth-viewing face is not predictable to better than 10°C. This is based largely on the fact that this is the only surface of the spacecraft which is not in intimate thermal contact with the power system coolant loop. However, it is unlikely that the thermal model will be so much in error. In addition, this adds to the desirability of using this face as a radiator extension, thus controlling its temperature much more closely. Also, unload perturbations can be reduced to absolute zero by use of magnetic torqueing in place of jet firings. The small daily angular momentum build-up of the reference vehicle (~.02 ft-lb-sec) makes this a possible option (Ref. Appendix H).

Errors due to jet misalignment or mismatch are liberally estimated, resulting in the rather large attitude error which was postulated. In addition, impluses along the radius vector will be made much less frequently than those normal to the orbit plane, perhaps no more frequently than every 50-to-100 days. Therefore, the simple sum of these two errors results in a clear overestimate of the total error which may be expected from this source.

Errors due to the misestimation of solar pressure forces are likely to be the most significant errors which are encountered by most three-axis controlled spacecraft. The fact that these are so small in the reference case is due to the averaging effects of the spacecraft rotation, as detailed in Appendix J. The most significant residue of this effect is due to the variability of the self-shadowing effect of the thrust cone extension and the antennas because of spacecraft misalignment. Clearly shadowing due to the thrust cone can be minimized at very little cost by minimizing the length of this extension. Shadowing due to the antennas can be effectively eliminated by use of a radome-type cover, as was considered earlier in the study for the other design approaches. However, the magnitude of this effect is so small that the dielectric cover is judged to be unnecessary.

11.4 TEMPERATURE CONTROL OF FREQUENCY STANDARDS

The results of this analysis are discussed in Section 8. It must be noted here that these results are only preliminary. A complete multi-node transient thermal analysis which models the entire spacecraft is required before a definitive conclusion is possible. In particular, power transients as well as orbit-cycle transients will effect the temperature distribution of the spacecraft in ways which are not readily apparent in simplified models.

11.5 VULNERABILITY

The study ground rules specified that the power system only was to be evaluated for vulnerability to x-rays and the control system to nuclear radiation.

The selection of teflon-coated silver tape as the surface thermal-control material would appear to provide an adequate level of hardness as shown in

Table 11.5-1. This table, taken from the SAMSO Survivability Conference of 23 March 1977, summarizes the hardness of various materials on the basis of JCS multiples, with silvered teflon corresponding to "second-surface aluminized plastics" in physical response.

The two power systems differ somewhat in the design of their control systems. The Organic Rankine system uses discreet components in a frequency discriminator which drives a transistor comparator. This in turn generates control signals which switch parasitic load resistors in or out as required to maintain a fixed alternator speed. A check of the components used in the controller as currently designed revealed the switching transistors to be the only component of potential concern, although no hardness data on the specific devices (2N5927) could be located. In general, however, switching transistors which have been qualified to JCS standards are available and could be substituted for the specified devices.

The Brayton system uses a digital control system for greater control flexibility, which to the present time has only been designed to the subsystem block diagram level. Therefore, component checks could not be conducted.

In general, it may be concluded that the power system should be able to meet JCS survivability standards or greater with the proper selection of control system components.

Table 11.5-1. Nuclear Effects on Surfaces*

	SOLAR CELLS	TRANSPARENT MATERIALS	THERMAL CONTROL MATERIALS	MIRRORS
10 x JCS	HARDENED	FUZED SILICA QUARTZ	FUZED SILICA	
	HARDENED	LOW Z GLASSES	LOW Z PAINTS SECOND SURFACE ALUMINIZED PLASTICS	Al on Be
3 x JCS				
1 x JCS	STANDARD	GERMANIUM	HI Z PAINTS	Al on GLASS
	STANDARD	GERMANIUM	HI Z PAINTS	Au on GLASS

* FROM SAMSO SURVIVABILITY CONF. OF 23 MARCH 1977

A. SUMMARY OF GPS PROGRAM
AND PHASE I SPACE VEHICLE

The Global Positioning System (GPS) is a space-based radio positioning navigation system that will provide extremely accurate three-dimensional position and velocity information together with system time to suitably equipped users anywhere on or near the earth. The GPS consists of three major segments: Space Segment, Control Segment, and User Segment. Figures A-1, A-2, and A-3 provide a simplified overview of the GPS concept.

A functional flow diagram of the GPS system and its interfaces is given by figure A-4 (taken from figure 1 of IRN 14726). Note that the satellites are controlled by the Air Force Satellite Control Facility (AFSCF) which maintains

FIGURE A-1.

GLOBAL POSITIONING SYSTEM (GPS)

USER

- RECEIVES SYNCHRONIZED POSITIONING SIGNALS FROM SATELLITES (continuous, global coverage)
- COMPUTES 3 DIMENSIONAL POSITION AND VELOCITY
- OPERATES IN COMMON GRID WITH OTHER USERS

FIGURE A.2

GPS CONCEPT

FIGURE A.3

FIGURE A-4 FUNCTIONAL FLOW DIAGRAM GPS SYSTEM

S-band telemetry and command links with each satellite in the constellation. A separate Upload Station to be located in CONUS will transmit satellite ephemerides, system time, and propagation delay data to the satellites periodically as required to update the system. A description of the NAVSTAR system, taken from sections 1.5 and 2.2 of the Joint Program Management Plan, is attached for reference as Appendix A.

The system has been approved for Phase I implementation by DSARC I (Figure A-5). Phases II and III will be considered for approval by DSARC II and III as shown in Figure A-5.

A.2 Space Segment

The operational GPS will deploy satellites in circular, 10,900 nautical mile orbits inclined at approximately 63 degrees and having a twelve hour period. This deployment will provide the satellite coverage for continuous, three-dimensional positioning and velocity determination. Each satellite will transmit a composite signal at two L-band frequencies consisting of a protected navigation signal and a clear acquisition navigation signal. The signals contain navigation data such as satellite ephemeris, atmospheric propagation correction data, and satellite clock bias information. Using the L-band navigation signals permits the user to determine the ionospheric group delay or other electro magnetic disturbances in the atmosphere which may affect the transmitted signals.

The user is required to effectively synchronize his clock with system time through a combination of physical and computation corrections. Users with very precise clocks could range to three different space vehicles. However, to avoid requiring each user to be equipped with a very precise clock, an additional independent range measurement is utilized. Thus, measurement data from four space vehicles will be used to compute the user's three-dimensional position and system time (see Figure A-6). In addition, the user measures doppler shift of the four independent navigation signals to determine his three-dimensional velocity and the drift of his clock. When more than four (4) satellites are within the user's line of sight, the user will select the optimum subset of four satellites to use in his position determination. Some of the criteria that may be used to select this subset are:

- a. Relationships of satellites to user to insure optimum geometry.
- b. Minimum ionospheric and tropospheric delay uncertainty.
- c. Remaining view time of space vehicle.

NAVSTAR GLOBAL POSITIONING SYSTEM SCHEDULE

FIGURE A-5

System Technique

PSEUDO-RANGING TO FOUR SATELLITES

Figure A-6

A.3 Control Segment

Four widely separated Monitor Sets (MS), located on U.S. territory, will passively track all satellites in view, and accumulate ranging data from the navigation signals. This information will be processed at a Master Control Station (MCS), to be located in the Central Continental United States for use in satellite orbit determination and systematic error elimination. The orbit determination process derives progressively refined information defining the gravitational field and solar pressure parameters influencing the satellite, the clock drift of the satellite, the electronic delay characteristics of the signal, the location of the ground station, and other observable system influences.

An Upload Station (ULS), co-located with the Central Continental United States MCS, will transmit the satellite ephemerides, clock drifts, and propagation delay data to the satellites as required.

A.4 User Segment

Using the navigation signal from each of four satellites, the user's receiver will measure four independent pseudo-ranges and pseudo-range rates to the satellites. The user's receiver/processor will then convert these pseudo-range and pseudo-range rates to three-dimensional position, velocity, and system time. This position solution is in World Geodetic System - 72 coordinates, an earth-centered, earth fixed coordinate system, which can be converted to any coordinate frame in units of measure the user requires.

A.5 Concept

The GPS is a space-based radio navigation system that will provide GPS-equipped users with the capability to precisely determine their three-dimensional position, velocity, and time. Each GPS space vehicle (SV) will generate and transmit two GPS Navigation signals. Also, each GPS user receiver will generate replicas of the desired set of SV generated navigation signals. When the user's signal matches in time with the received SV signal, a measurement of signal transmit time and user clock asynchronization can be made. All signals emanating from the GPS space vehicles will be synchronized by the GPS Control Segment.

A.6

GPS Performance

The user's single position fix accuracy is a function of the user's position relative to the GPS SV constellation. The relationship between the user's navigation accuracy and his relative position to the SV constellation is referred to as Geometric Dilution of Precision (GDOP). User Equivalent Range Error (UERE) is defined as the uncorrelated portion of the observed user range error. Thus, the user's actual navigational accuracy is obtained by multiplying the UERE by the GDOP for the user's position. During Phase I with primarily six SVs available, the user's single position fix navigation accuracy will be approximately three times the UERE (average GDOP of 3). The components of UERE for Phase III and Phase I of the GPS are shown in Table A.1. The Space Vehicle Group Delay components of the Phase I Error Budget will be applicable for two hours after all SVs are updated. All other components of the Error Budget will be applicable for twenty-four hours after all SVs are updated. All values are one sigma (1σ) errors given in meters.

TABLE A.1
GPS ERROR BUDGET

	Phase III	Phase I
Space Vehicle Ephemeris	1.6	3.9
Atmospheric Delay	2.6 to 5.6*	2.6 to 5.6*
Space Vehicle Group Delay	1	2.8
Receiver Noise	1.6	1.6
Multipath	<u>1.3 to 3.0*</u>	<u>1.3 to 3.0*</u>
Total R.S.S.	3.9 to 6.9	5.9 to 8.2

*These error quantities represent the use of different atmospheric delay correction methods and different user environmental conditions (values in meters).

A.7

GPS Error AnalysisEphemeris

Space vehicle ephemeris error is defined as the difference in actual satellite

ephemeris and the satellite ephemeris computed by the user from the data provided in the Navigation Signal data frame. Since the GPS satellites will be in 12-hour orbits, the effect of satellite in-track and cross-track ephemeris errors will be reduced. For example, a three foot ephemeris in-track error projects to a one foot error in range. Thus, the primary error source that will effect the user's position determination is satellite radial position error. In addition, ephemeris errors common to the four range measurements cause a user clock calibration error, rather than a user position error because the user clock asynchronization is common to all four measurements.

The Naval Surface Weapons Center's (NSWC) operational experience with orbit determination for Transit satellites (600 nmi altitude) has produced the following conclusions:

- a. Observed versus predicted values in the orbit determination process check within 2 m (rms).
- b. Extremely accurate laser tracking by NASA of Transit satellites agrees with NSWC ephemerides within 1-2 m (rms).
- c. Long baseline interferometry experiments conducted by MIT agree with Transmit/Geoeiver surveys to within 1 m (rms).

Based upon NSWC's experience and the more favorable geometry of GPS satellites, there should be low risk in achieving the satellite ephemeris accuracies specified in the error budget.

Propagation Errors

Atmospheric delay error is defined as the difference between actual signal propagation delay through the ionosphere and troposphere and the delay determined by the user employing suitable ionosphere and troposphere models. Since the ionospheric delay is a function of frequency, two coherent signals can be used to calibrate the delay. In this dual frequency calibration mode, the ionospheric delay uncertainty is a function of the difference in noise in the receiver channels. The variation with frequency is shown in Figure A-7. For GPS, the ranging error is at most three times the interchannel noise. Interchannel noise can be minimized if a single measuring circuit is time-shared between channels. A

FREQUENCY DEPENDENCE

USER RECEIVER MEASUREMENT ERROR

FIGURE A-7

sophisticated modeling process can produce an equivalent error. It is currently anticipated that users demanding the best GPS accuracy will use dual frequency calibration. A less sophisticated ionospheric model will be available for medium accuracy users. The tropospheric propagation delay model will be contained in the user software. A conservative estimate for a user's tropospheric model requiring no external data indicates a vertical uncertainty of 1 meter (1σ) with satellites at the zenith increasing to $2\frac{1}{2}$ meters (1σ) for a satellite at 95-deg. elevation angle.

Space Vehicle Delays

Space vehicle group delay error is defined as the sum of delay uncertainty in the SV; such as unmodeled clock drift and uncalibrated delay in signal equipment. The satellite group delay is undistinguishable from clock errors and will be accounted for in the clock calibration process. Immediately after an orbit determination solution, the satellite clock uncertainties will be the same for each satellite and so highly correlated between satellites there is little or very slight effect on the user. In Phase I, after two hours, the magnitude of the accumulated clock drift will be approximately 7 nanoseconds (7 feet using a 1×10^{-12} clock in each satellite).

User Errors

Receiver noise and resolution errors are the contributions of noise and measurement errors in the user navigation receiver hardware and software which perturb the correct navigation solution. The User Equipment Definition and Experiments test program at White Sands Missile Range demonstrated that a receiver noise level of five feet is conservative and obtainable.

Multipath error is the effect of several propagation paths from the SV to the user which corrupt the measurement of line-of-sight distance. Operational environments may cause multipath arrivals that are comparable in amplitude to the line-of-sight signal. However, the GPS receiver searches for the first signal to arrive and not the most powerful signal. Therefore, the likelihood of tracking a multipath signal that is delayed more than $1\frac{1}{2}$ code chips (≈ 19 m for P code) is drastically reduced. If the time of arrival is sufficiently

close to that of the line-of-sight signal (less than $1\frac{1}{2}$ chips), the user receiver will not easily be able to discriminate between the signals. The magnitude of the resulting ranging error will depend strongly on the location and nature of reflecting surfaces in the user environment.

A-8

Signal Structure

The primary navigation signal of the GPS is a composite waveform consisting of a Precision (P) Signal and a Clear Acquisition (C/A) Signal transmitted in phase quadrature. The P signal is used by the precision military user and will resist jamming, spoofing, and multipath and will be deniable to unauthorized users by employing nonlinear encoding. The C/A signal will serve as an aid to the acquisition of the P Signal, and will also provide a navigation signal in the clear to both the military and civil user. This signal may also employ System techniques to deny the high accuracy of this signal to unauthorized users.

The navigation signals are transmitted on two channels: L 1 and L2. Channel L1, the Primary Navigation Signal, is nominally centered at 1575 MHz and shall carry both the P and the C/A Signals. Channel L 2, the Secondary Navigation Signal, is nominally centered at 1230 MHz and shall carry the P Signal or C/A Signal. On the L 2 channel, either P or C/A can be selected through a telemetry controlled switch.

Both the P and the C/A Signals are pseudorandom noise bi-phase shift keyed (PN/BPSK) continuous sinusoidal carriers. Both signals carry system data. The chipping rates of the PN codes are 10.23 Mbps and 1023 kbps for the P and the C/A Signals respectively. The system data rate is 50 bps. Each Space Vehicle is assigned a unique set of PN codes. The P code repeats each seven (7) days and the C/A code repeats each millisecond. This waveform is specifically designed to allow system time to be conveniently and directly extracted in terms of standard units of days, hours, minutes, and interger multiples and submultiples of the second.

The system data contains information which allows the user to navigate successfully with GPS. It will provide space vehicle ephemerides, system time-of-week, space vehicle clock behavior data, system status messages, and C/A to P signal handover information. The data stream will be common to both the P and the C/A Signal on both L 1 and L 2.

The P Signal is a continuous sinusoidal carrier, bi-phase modulated according to the modulo-2 sum of a PN code and a synchronous data bit stream. The P codes will be generated such that no space vehicle shall duplicate the output of any other space vehicle before seven (7) days have elapsed in system time. The period of the code will be sufficiently long (267 days) to accommodate 37 different initial code phases, each displaced by a minimum of seven (7) days ($i = 1, 2, \dots, 37$). The same basic code generator will be used by all GPS P Signal transmitters with each one assigned one of the 37 possible unique code phases.

The C/A Signal is a continuous sinusoidal carrier, bi-phase modulated according to the modulo-2 sum of a 1023 bit Gold code sequence and a synchronous data bit stream. All of the 36 legitimate codes within the 1023 bit Gold code family will be used for each C/A signal transmitter. The codes will be selected to have low envelope cross-correlation with any other code in the Gold family for all possible shifts of the code and doppler conditions.

A-9 GPS Design Goals

Tables A.2, A.3, and A.4 provide design goals for the GPS user equipment for Phase I, Phase II, and Phase III, respectively. These goals are subject to design-to-cost trade-offs and other considerations.

A-10 SPACE SEGMENT; NDS DATA

A-10.1 REQUIREMENTS

The GPS Space System Segment/User Systems Segment interfaces are defined in Rockwell Document IRN 14726 and in the NAVSTAR GPS Joint

TABLE A.2 PHASE 1 DESIGN GOALS FOR NAVSTAR/GPS

Eq. Class Appl. Design Parameters		A	B	C	D	E	F	G/M (No Dev)	S (No Dev)
		High Performance Aircraft w/ High AJ	High Performance Aircraft	Mission Support Vehicles	Land and Sea Vehicles w/ High AJ	Man Backpack	Submarine	Space Missile	Spartan, Very-Low Cost
Performance	(1) x y z	6-9 m	6-9 m	15-20 m	18-25 m	18-25 m	6-9 m	16	30-100 m (cep)
	(2) x y z	0.1 kt	0.1 kt	0.5 kt	No req	No req	0.1 kt	.012 m/sec	No req
	Time	18-25 nsec	18-25 nsec	18-25 nsec	1 sec displayed	1 sec displayed	18-25 nsec	TBD	1 sec displayed
	(3) Time-to-first-fix	80-180 sec	80-180 sec	200-300 sec	100 sec	100 sec	80-100 sec	400 sec	600 sec
	(4) AJ margin	P -54 C/A 34	P -54 C/A 34	P-NA C A 34	P -44 C A 34	P -44 C/A 34	P -44 C/A 34	(5) 29.8 29.8	P-NA C/A TBD
Size		.2 m ³	.2 m ³	(6) .11 m ³	.018 ³	.018 m ³	.2 m ³	.2 m ³	.04 m ³
Weight		.56.1	56.1 kg	22.5 kg	10.5 kg	10.5 kg	56.1	30 kg	18 kg
Power		632 w	632 w	151 w	30 w	30 w	632 w	80 w	30 w
MTBF		500	500	500	500	500	500	TBD	TBD
DTC		TBD	TBD	\$25,000	TBD	TBD	TBD	TBD	TBD

Assumptions:

- (1) UERE
(2) UERRE

- (3) Simultaneous conditions of Table II SS-US-101-B
(4) dB-Hz - Code Tracking Only

- (5) C/No dB-Hz
(6) Design-to-cost based upon 1974 dollars (14 units)

Table A.3 Phase II Design Goals for NAVSTAR /GPS

Periodic 3-D

Eq. Class. Appl.		A	B	C	D	E	F	G/M (No Dev)	S (No Dev)
		High Performance Aircraft w/High AJ	High Performance Aircraft	Missile Support Vehicles	Land and Sea Vehicles	Man Backpack	Submarine	Space Missile	Spartan, Very-Low Cost
Performance	(1) x				Cep-5 m	Cep-5 m			
	y	5-8 m	5-8 m	15-20 m	PE 10 m	PE 10 m	5-8 m	15 m	30-50
	z								
	(2) x								
	y	.08 kt	.08 kt	0.4 kt	No req	No req	.08 kt	.012 m/sec	No req
	z								
Performance	Time	15-25 nsec	15-25 nsec	46-60 nsec	1 sec displayed	1 sec displayed	15-25 nsec	TBD	1 sec displayed
	(3) Time-to-first-fix	80-100	80-100	200-300	60	60	80-100	60	450 sec
	(4) AJ margin	P-65	P-65	P-NA	P-50	P-50	P-50	(5) 29.8	P-NA
		C/A 34	C A 34	C A 34	C A 34	C/A 34	C A 34	29.8	C/A TBD
Performance	Size			(6) .11 m ³	1000 in. ³	1000 in. ³		400 in. ³	800 in. ³
	Weight	TBD	TBD	22.5 kg	200 lb	20 lb	TBD	16 lb	36 lb
	Power			151 w	20 w	20 w		12-16 w	14-18 w
	MTBF			500	1000	1000		1000	TBD
	DC			\$15,000	TBD	TBD		TBD	TBD

Assumptions:

- (1) UERE
(2) UERRE

- (3) Simultaneous conditions of Table H SS-US-101-B
(4) dB-Hz - Code Tracking Only

- (5) C/No dB-Hz

- (6) DTC based upon 1974 dollars (500 units)

Table A.4 Phase III Design Goals for NAVSTAR/GPS

World-Wide 3-D

Eq. Class. Appl. Design Parameters		A	B	C	D	E	F	G/M (No Dev)	S (No Dev)
		High Performance Aircraft w/High AJ	High Performance Aircraft	Mission Support Vehicles	Land and Sea Vehicles w/High AJ	Man Backpack	Submarine	Space Missile	Spartan, Very-Low Cost
Performance	(1) x y z	4-7 m	4-7 m	15-20 m	3 m Cep 3 m PE	3 m Cep 3 m PE	4-7 m	10 m	30 m
	(2) x y z	.05 kt	0.5 kt	.2 kt	No req	No req	.05 kt	1012 m/sec	No req
	Time	12-21 nsec	12-21 nsec	45-60 nsec	1 sec displayed	1 sec displayed	12-21 nsec	TBD	1 sec displayed
	(3) Time-to-first fix	80-180	80-180	200-300	30	30	80-180	60	300
	(4) AJ margin	P -74 C/A 34	P -74 C/A 34	P-NA C/A 34	P-58 C/A 34	P-58 C/A 34	P-58 C/A 34	(5) 29.8 29.8	P-NA C/A TBD
Size		TBD	TBD	(6)	500 in. ³	500 in. ³	TBD	250 in. ³	800 in. ³
Weight		TBD	TBD		8-12 lb	8-12 lb	TBD	10 lb	TBD
Power		TBD	TBD	TBD	10 w	10 w	TBD	8-12 w	TBD
MTBF		TBD	1200	1200	2000	2000	1200	2000	TBD
DTC		TBD	TBD	\$15,000	TBD	TBD	TBD		

Assumptions: (1) UERE (3) Simultaneous conditions of Table II SS-US-101-B
(2) UERRE (4) dB-Hz - Code Tracking Only

(5) C No dB-Hz
(6) DTC based upon 1974 dollars (500 units)

Program Management Plan. The key satellite performance requirements as they relate to the objectives of this study are:

- earth coverage right hand circularly polarized (RCP) beams at two frequencies: L-1 and L-2 (1575.42 MHz and 1227.6 MHz).
- generation of pseudo-random-noise (PRN) phase-modulated protected (P) signals and clear acquisition (C/A) signals which are BPSK (bi-phase shift key) modulated in phase quadrature with separate PRN codes.
- direct sequence spectrum spreading at chipping rates of 10.23 Mbps for the P signals and 1.023 Mbps for the C/A signals.
- code repetition periods of 7 days for the P signals and 1 milli-second for the C/A signals
- frame lengths of 1500 bits
- operating modes in accordance with the following table

Table A-5. Transmitter Operating Modes

Mode	L_1 Link		L_2 Link	
	P	C/A	P	C/A
1	Normal	Normal	Normal	Note 4
2	Normal	Normal	Note 4	Normal
3	Normal	High	Normal	Note 4
4	Normal	High	Note 4	Normal
5	Normal	Normal	Note 3	Note 3
6	Normal	High	Note 3	Note 3
7	Note 3	Note 3	Normal	Note 4
8	Note 3	Note 3	Note 4	Normal
Standby	Note 3	Note 3	Note 3	Note 3
Notes: 1. High power will be on the NDS space vehicle only. 2. Mode control shall be effected through the TT&C system. 3. No RF carrier is transmitted 4. Modulation not present. 5. Modes 7 and 8 are for ground checkout.				

- Minimum user carrier power levels of -163 dBW for L1 - P, -160 dBW for L-1 C/A (Normal), -158 dBW for L-1 C/A (high power), and -166 dBW for L-2 P or C/A.
- Maintenance of phase coherence between all signals within 5 nanoseconds (one sigma)
- Maintenance of equipment group delay variation (uncertainty) within 1.5 nanoseconds (one sigma) for either L-1 or L-2 and maintenance for either L-1 or L-2 and maintenance of differential group delay between L-1 and L-2 within 1.5 nanoseconds (one sigma).
- Provision of a highly accurate frequency standard adjustable in steps no larger than $4 \times 10^{-12} \Delta f/f$ over a range of $\pm 2 \times 10^{-9} \Delta f/f$ around a nominal output frequency of 10.23 MHz (10.22999999545 MHz $\pm 10^{-11}$ adjusted for relativistic effects) with a frequency drift not to exceed one part in 10^{12} per day.
- Provision of navigation signal data at 50 bps containing, as a minimum, space vehicle ephemerides, system time, space vehicle and clock behavior data, transmitter status information and C/A to P signal handover information. The data stream shall be common to both the P and C/A signals on both the L1 and L2 links.
- Maintenance of space vehicle attitude with the + Z axis pointed toward earth center with an accuracy of $\pm 0.5^\circ$ except during momentum dump when the accuracy may be $\pm 1.0^\circ$.

The major navigation signal is carried by the L-1 link. The L-2 link is provided at a different frequency so that users requiring greater precision can compare the two signals to correct for atmospheric propagation errors.

SPACECRAFT INTERFACES

External and internal interfaces for the GPS Space Vehicle (SV) are defined in the Prime Item Development Specification (CID-SV-101A) volumes 1 through 11). The SV subsystems are identified as shown by figure A-8 (block diagram) and, for the NDS, by figure A-9 (exploded view). External space vehicle interfaces are defined by figure A-10 and internal interfaces are defined by figure A-11.

The key interfaces are described briefly below.

A-11

Navigation Subsystem

The navigation subsystem RF signal characteristics are given by table A.6 and a (non-redundant) block diagram of the navigation subsystem is given by figure A-12. The LCU (Load Control Unit) and DC/DC Converter are part of the electrical power subsystem (EPS). Nominal DC current and voltage requirements and nominal DC power requirements are given by table A.7. The antenna gain for the L-1 band varies from +12.2 dB on-axis ($\theta = 0^\circ$) to +13.7 dB off-axis ($\theta = 14.3^\circ$). Corresponding gains for the L-2 band are +9.7 and +11.3 respectively. Diplexer losses can be inferred from the power levels shown on the block diagram.

It should be noted that the high-power mode specified in table 2.1-1, and many of the different mode combinations requested, are a result of a desire to use more power at the beginning of mission life, before solar array degradation has limited the available power, and a further desire to apportion the reduced power subsequently available according to priority.

Other mission requirements which have been identified for particular emphasis in this study lie in the areas of mean-mission duration, orbital stability, spacecraft vulnerability and the thermal stability of on-board clocks. Mean-mission duration for the phase I navigation development satellite (NDS) is currently estimated at 4.54 years against a minimum requirement of 5 years. Orbital stability, as reflected by the predictability of the space vehicle ephemerides has one-sigma error

A-21

FIGURE A-8. SV SUBSYSTEM DESCRIPTION

RCS

- THRUST FOR SV MANEUVERS AND ATTITUDE CONTROL

OIS

- THRUST FOR SV DRIFT ORBIT INJECTION

- STRUCTURE SUBSYSTEM
- SUBSYSTEM INSTALLATION AND DISTRIBUTION OF LAUNCH AND FLIGHT LOADS

- THERMAL CONTROL SUBSYSTEM
- PASSIVE AND ACTIVE ENERGY DISSIPATION AND CONTROL

- ELECTRICAL POWER SUBSYSTEM
- ELECTRICAL ENERGY GENERATION, STORAGE AND DISTRIBUTION

- NAVIGATION SUBSYSTEM
- GENERATE AND RADIATE PRN RF SIGNALS

AVCS

- MONITOR AND CONTROL OF SV THRUSTERS FOR ATTITUDE AND MANEUVERS

TT&C

- PROVIDE SV COMMAND AND CONTROL OPERATIONS COMPATIBLE WITH SGLS

FIGURE A-9. SV SUBSYSTEMS

NUMBER	TITLE
① ICD309-CD-2002	NAVSTAR GLOBAL POSITIONING SYSTEM NDS SPACE VEHICLE/STAGE VEHICLE
② ICD MH08-00002-400	SPACE VEHICLE NAV SUBSYSTEM AND NTS PRN NAV ASSY/USER AND MONITOR SYSTEM SEGMENTS
③ ICD MH08-00003-400	PRN NAV ASSY/NTS PHYSICAL AND FUNCTIONAL REQUIREMENTS
④ ICD MH08-00004-400	SPACE VEHICLE NAV SUBSYSTEM AND NTS PRN NAV ASSY/UPLOAD STATION
⑤ ICD MH08-00006-400	SIMULATORS/CONTROL AND USER SYSTEM SEGMENTS AND AFSCF
⑥ ICD TBD	BOOSTER-FAIRING/STAGE VEHICLE AND SPACE VEHICLE/SITE AGE
⑦ SD 75-GP-0007	AFSCF/MCS COMPUTER PROGRAMS
⑧ ICD MH08-00009-400	ORBIT INJECTION SYSTEM/NTS PHYSICAL AND FUNCTIONAL REQUIREMENTS
⑨ (PRD)	PROGRAM REQUIREMENTS DOCUMENT

FIGURE A-10. EXTERNAL SPACE VEHICLE INTERFACES

FIGURE A-11. INTERNAL SV INTERFACES

TABLE A.6. NAV SUBSYSTEM RF SIGNAL CHARACTERISTICS

Parameter	L_1	L_2
Channel Allocation	20.46 MHz	20.46 MHz
Nominal carrier frequency	1575.42 MHz (f_1)	1227.6 MHz (f_2)
Carrier frequency accuracy	2 part in 10^9	2 part in 10^9
Minimum EIRP at $f_0 \pm 10.23$ MHz to end of life (including eclipse operations)	P +23.8 dBW C/A +26.8 dBW	P +19.1 dBW C/A +19.1 dBW
Minimum EIRP at $f_0 \pm 10.23$ MHz for high power mode	P +23.8 dBW C/A +28.8 dBW	P +19.1 dBW C/A +19.1 dBW
Antenna polarization	Right hand circular polarization	Right hand circular polarization
Maximum axial ratio over the band	0.8 dB	1.5 dB
Phase noise measured by a P.L.L. receiver with 10 Hz single sided loop bandwidth.	100 mr	100 mr

objectives of 3.9 m for Phase I and 1.6 m for Phase II (Reference section 2.2.2 of the Joint Program Management Plan.) The spacecraft vulnerability objective is to meet JCS standards with the Phase III spacecraft. Thermal stability of the on-board clocks impose a temperature-change requirement for the mounting plate of the clocks of < 4 degrees/day and < 5 degrees/hour. Current phase I satellites achieve ± 2.5 degrees/day; however, even tighter control is thought to be desirable.

FIGURE A-12. NAVIGATION SUBSYSTEM PRNSA BLOCK DIAGRAM

TABLE A-7. NAVSTAR SUBSYSTEM/EPS INTERFACE

(a) Nominal DC Voltage/Current Requirements

Units ⁽¹⁾	Bus +26.5V	Regulated Voltages ⁽²⁾					Power Leads	Remarks
		+24V	+15V	+12V	+5V	-12V		
L ₁ Band HPA (Nom)	4.64						2	(3)
(INCR)	2.44							(3)
L ₂ Band HPA	0.85						2	
	0.90							
L ₁ Band MOD/IPA			0.14A		0.12A	0.15A	6	Group B*
L ₂ Band MOD/IPA			0.05A		0.045A	0.09A	6	Group C*
L ₁ /L ₂ Synthesizer		0.44A					2	Group A*
PRN Baseband/Processor				0.21A	2.74A	0.4A	8	Group D*
Frequency Standard	0.93A						3	(4)

- NOTES: * Selectable DC/DC Converter Primary and Back-up groups to each NAV Subsystem unit
- (1) See Paragraph 3.1.3
- (2) DC/DC Converter (Specified in Volume 4)
- (3) Selectable Primary and Back-up input to each unit
- (4) Selectable input to each of 3 units

(b) Nominal DC Power Requirements (Watts)

Units ⁽¹⁾	Operating Modes ⁽²⁾						Stdby ⁽³⁾
	1	2	3	4	5	6	
L ₁ Band HPA	123.0	123.0	187.6	187.6	123.0	187.6	--
L ₂ Band HPA	23.8	23.8	23.8	23.8	--	--	--
L ₁ Band MOD/IPA	3.4	3.4	3.4	3.4	3.4	3.4	--
L ₂ Band MOD/IPA	1.6	1.6	1.6	1.6	--	--	--
L ₁ /L ₂ Synthesizer	10.6	10.6	10.6	10.6	10.6	10.6	--
PRN Baseband/Processor	21.0	21.0	21.0	21.0	21.0	21.0	21.0
Frequency Standard	24.5	24.5	24.5	24.5	24.5	24.5	24.5

- NOTES: (1) See Paragraph 3.1.3
- (2) See Paragraph 3.2.1.2
- (3) See Paragraph 3.2.1.2

The primary EPS interface definitions are shown in Figure A-13 and the nominal electrical load budget is given by table A.7. Electrical power for vehicle loads is supplied from the main bus at 27 ± 1 volt d.c. The voltage at the load interfaces will be at 26.5 (+1.5, -1.0) volts d.c. Power is provided through a two-wire system (power and return) and there is no sharing of the power loads. Main bus dynamic impedance is shown in figure A-14 and transient voltage limits are shown by figure A-15.

Figure A-16 shows the AVCS component block diagram and figure A-17 defines the AVCS top level interfaces. These figures are taken from CID-SV-101A, Volume 5. (Prime Item Development Specification).

The figures are quite complete and self-descriptive. Note, however, the relatively small size of the reaction wheels, $H = 0.8$ ft-lb-sec each.

The two 5-lb thrusters are used for orbit correction during initial insertion, as well as for later orbit maneuvers as required. The smaller thrusters are also available for orbit corrections and maneuvers, but are used mainly for unloading of the momentum wheels.

The thermal control subsystem consists of the following major components:

- a. Louver Panel Assemblies (7)
- b. Thermal Doublers
- c. Space External Insulation Blanket Assemblies
- d. Spacecraft Tunnel Insulation Blanket Assembly
- e. Battery Radiators (3)
- f. Battery Heater/Thermoswitch Assembly (3)
- g. OIS Nozzle Cap
- h. OIS Plume (Support Skirt Close Out) Shield

FIGURE A-13 EPS INTERFACES

TABLE A-8. NOMINAL ELECTRICAL LOAD BUDGET

Subsystem	Mission Phase					
	(1) Prelaunch & Launch (avg)	(2) Transfer Orbit (avg)	(3) Orbit Phasing (avg)	On Orbit		
				(4) Normal NAV Non Eclipse Orbit (avg)	(5) Normal NAV Eclipse Orbit (avg)	(6) High-Power NAV Non- Eclipse Orbit (avg)
EPS	12	12	21	20	20	20
RCS	9	8	8	1	1	1
AVCS	—	10	33	33	33	33
TT&C	60	15	15	11	11	11
NAV	—	—	—	227	227	292
THERMAL	—	41	41	21	13	16
TOTAL (Watts)	81	86*	118	313	305	373
<p>* 133W for the first five hours of transfer orbit</p> <p>NOTE: Subsystem allocations are nominal values and are for reference only.</p>

FIGURE A-14. EPS MAIN BUS DYNAMIC SOURCE IMPEDANCE

FIGURE A-15. TRANSIENT VOLTAGE LIMITS

FIGURE A-16. AVCS TOP LEVEL INTERFACES

PROVIDES SUPPORT STRUCTURE
FOR WHEELS, CONTROL
ELECTRONICS & SENSORS

PROVIDES GROUND COMMAND (UPLINK)
DATA (14-BIT WORD) & DISCRETES FOR
SWITCHING FUNCTIONS. ACCEPTS
VARIOUS TELEMETRY SIGNALS PROVIDED
TO GROUND PLUS ROLL MOMENTUM DUMP
DISCRETE TO NAV

PROVIDES YAW-AXIS SUN
NULL POINTING INFORMATION
FOR YAW STEERING

PROVIDES TEMPERATURE
CONTROL WITHIN
SPECIFIED LIMITS

PROVIDES 26.5 VDC TO
ALL AVCS ELEMENTS

PROVIDES (TWO) 22.24 NEWTON
THRUSTERS FOR PRECESSION
CONTROL AND INITIAL ORBIT
CORRECTION ΔV MANEUVERS.
PROVIDES SIXTEEN 0.448
NEWTON THRUSTERS FOR CONTROL,
MOMENTUM DUMPING, AND ΔV
MANEUVERS.

FIGURE 3-17. AVCS TOP LEVEL INTERFACES

- i. OIS Nozzle Throat Heater, Vendor Supplied
- j. RCS Component Heater/Thermoswitch Assemblies (10)
- k. Surface coatings, finishes, tapes, etc.

Most of the thermal control requirements are those commonly applied to spacecraft components, with the critical exception of the navigation frequency standards. Although the baseplate temperature limits of these "on-board clocks" are 20 to 45°C, the permitted temperature variation is not to exceed 4°C/hour and 5°C/day.

The rubidium frequency standard in current use on the Phase I space vehicle (ϕ1 SV) is mounted (within a double oven) to one of the radiator shear panels on the "Y" face of the spacecraft. These panels are subject to control by the thermal louvers, which are located on the shear panels, 3 louvers on the +Y and 4 on the -Y panels. The louvers are mounted outboard, over silvered FEP Teflon radiators. The louvered area is approximately 1.318 m² (14.2 ft²). Maximum orbital heat rejection load capability is 454 watts.

The efficacy of the radiators in limiting temperature excursions of the clocks is substantially improved by the yaw steering of the spacecraft, which keeps the Y faces out of direct sunlight. Therefore, in the absence of variations of internal dissipation, the major remaining sources of time-varying thermal input are due to:

- 1.) reflected sunlight and thermal radiation from the solar array edges and panels at an angle,
- 2.) nominal direct sunlight due to control deadband, and due to temporary re-orientation required during noon orbit and when coming out of eclipse.

A-11.4 Telemetry and Command Subsystem (TT&C)

Figure A-18 shows a block diagram of the telemetry and command subsystem taken from the Command and Measurement Requirement List (Drawing # V 505-910006). The command and measurement requirements for the current GPS Phase I Spacecraft are summarized in table A.8. As may be seen from this table, the Electrical Power Subsystem (EPS) command and measurement budget is fairly generous and preliminary study has indicated that a nuclear power subsystem would be able to operate within this budget.

Figure A-19. TT&C Block Diagram

Table A-8.

NAVSTAR Command and Measurement Requirement Summary

<u>Subsystem</u>	<u>Number of Commands</u>		<u>Hardware (Umbilical)</u>
	<u>Discrete</u>	<u>Magnitude</u>	
AVCS	98	33*	2
EPS	68	—	9
NAV	88	—	—
OIS	4	—	—
RCS	22	—	12
TCS	18	—	—
T, T & C	53	127	16

<u>Subsystem</u>	<u>Number of Measurements</u>	
AVCS	114	20
EPS	57	17
NAV	65	—
OIS	1	—
RCS	30	8
TCS	16	—
T, T & C	73	—

* Plus 10 other AVCS command groups defined by changing the bit patterns within the data field of the uplink command word.

B. CALCULATION OF MAXIMUM
ALLOWABLE USER POWER

B.1 Case of Uniform Increases in All Links

The GPS load summary and navigation subsystem power requirements, as obtained from the Rockwell July 1976 Design Review, are given by Table B-1.

From the EPS Load Summary Chart, it is seen that average (low power/high power mode) total spacecraft power requirements are 300/372 watts and that the NAV subsystem uses 233/306 watts of this total. The LP/HP mode is utilized in the L1 links only. The L2 link power output is constant. It is first assumed that any increased spacecraft power capability resulting from use of a nuclear power supply would first be used to maintain the high power mode continuously. The analysis presented therefore concentrates on this mode. Additional increases in power capability would next be directed to increased output of the L1 and L2 High Power Amplifiers (HPA's). RF power output is given by the L1/L2 Link Budgets previously shown in Table 4.1-1. Note that the L1 HPA provides two simultaneous signals (a protected (P) signal and a clear acquisition (C/A) signal) and that L1 HPA power has to be shared between these two signals and self-generated noise.

Using the values for the net L-1 HPA and L-2 HPA RF output and adding the losses shown on the link margin charts results in the following HPA gross RF output values.

	<u>L-1 HPA</u>	<u>L-2 HPA</u>
Net RF Power	16.66 dBW	8.24 dBW
Losses	1.33 dB	1.28 dB
Gross RF Power	17.99 dBW	9.52 dBW
	(63.0 watts)	(9.0 watts)

Total RF output is therefore 72 watts which requires 232 watts for the HPA's and power filters and leaves 74 watts for the rest of the NAV subsystem with a power division between the L1 and L2 HPA's of .875/.125.

Table B-2 shows the effect of increased spacecraft power on the critical parameter (CP) of "power to user" assuming equal ratio increase of power in all four links. Note that a 3.0 dB (doubling) of spacecraft power capability results in a 4.3 dB increase in power to user. The 0.1 dB difference between the L-1 and L-2 links is not

Table B-1

<u>GPS Load Summary</u>		<u>Navigation Subsystem Power</u>	
<u>Subsystem</u>	<u>Power Demand (watts on orbit)</u>	<u>Component</u>	<u>DC Watts</u>
EPS	15.2	L 1 HPA	131.0
RCA	0.5	L 2 HPA	27.0
AVCS	21.3	Power Filters	1.0
TT&C	11.0	L 1 Mod 1 PA	2.8
NAV	233.0	L 2 Mod 1 PA	1.0
Thermal	<u>13.4</u>	L 1 /L2 Synthesizer	8.4
Normal Mode Load	299.4 Watts	Baseband Processor	14.4
High Power Load	372.0 Watts	Frequency Standard	31.8
		Power Converter	<u>15.5</u>
		Normal Mode Power	232.9 Watts
		Incremental HPA	72.0
		Power Filter	<u>0.6</u>
		High Power Mode Power	305.5 Watts

TABLE B-2 - USER POWER VERSUS TOTAL DC SPACECRAFT POWER

<u>Total DC Power (Watts)</u>	<u>Total HPA Power (Watts)</u>	<u>L-1 HPA RF Power (Watts) (dBW) net</u>		<u>L-1 P User Power (dBW)</u>	<u>L-1 C/A User Power (dBW)</u>	<u>L-2 HPA RF Power (Watts) (dBW) net</u>		<u>L-2 P User Power (dBW)</u>	<u>L-2 C/A user Power (dBW)</u>
370	230	63.0	16.7	-162.4	-157.8	9.0	8.2	-165.6	-165.1
400	260	71.5	17.3	-161.8	-157.2	10.2	8.8	-165.0	-164.5
450	310	85.8	18.0	-161.1	-156.5	12.3	9.6	-164.2	-163.7
500	360	100.1	18.7	-160.4	-155.8	14.4	10.3	-163.5	-163.0
550	410	114.4	19.3	-159.8	-155.2	16.4	10.8	-163.0	-162.5
600	460	128.7	19.8	-159.3	-154.7	18.4	11.3	-162.5	-162.0
650	510	143.0	20.2	-158.9	-154.3	20.5	11.8	-162.0	-161.5
700	560	157.3	20.7	-158.4	-153.8	22.6	12.2	-161.6	-161.1
750	610	171.6	21.0	-158.1	-153.5	24.6	12.6	-161.2	-160.7
increase				4.3 dB increase	4.3 dB increase			4.4 dB increase	4.4 dB increase

* Reference state

significant and is due to round-off in the calculations. Rounded off values have been used for spacecraft power and HPA power.

The rate of increase varies from 0.7 dB/50 watts at the low power end of the table to 0.3 dB/50 watts at the high power end of the table and further "diminishing returns" would be expected for additional power increases. Figure B-3 is a graphical presentation of this user power versus spacecraft power relationship.

B.2 Case of Power Increases in One Link Only

This analysis shows the effect on "power to user" when additional spacecraft power is used to enhance a single link instead of being proportionally divided between the L1 (P), L1 (C/A) and L2 (P or C/A) links as evaluated in the previous section.

The assumptions used in this analyses are the same as those stated previously i.e. Reference state total spacecraft DC power 370 watts, total HPA DC power 230 watts (203 W-L1, 27 W - L 2).

In table B-4, all DC power increments above the reference state are applied to the L 1 HPA and alternatively used to enhance the (P) link or (C/A) link. Resultant increments in user power as a function of DC spacecraft power are tabulated. As indicated by table 2, a 3.0 dB increase in spacecraft power can result in either a 9.6 dB increase in power to the L 1 (P) link user or a 5.8 dB increase in power to the L 1 (C/A) link user.

In Table B-5, all DC power increments above the reference state are applied to the L 2 HP and used to enhance the (P) link or (C/A) link. Resultant increments in user power as a function of DC spacecraft power are tabulated. As indicated by table B-5, a 3.0 dB increase in spacecraft power can result in an 11.8 dB increase in the L 2 (P) or (C/A) link. Note that 6.0 dB of increase is accomplished in the first 80 watts of DC power increment.

Figure B-6, shown below, illustrates the power division relationships. Nominal (reference state) values are shown. In the tabulations L 2 HPA DC input power is held constant while L 1 HPA DC input power is increased (Table B-4) then L-1 HPA DC input power is held constant while L 2 HPA DC input power is increased (Table B-5).

POWER TO USER (dbw)

-150

-155

-160

-165

Flux Density Limit

L-1 C/A

L-1 P

L-2 C/A

L-2 P

400

500

600

700

SPACECRAFT POWER (WATTS)

FIGURE B-3. POWER TO USER VS SPACECRAFT POWER

TABLE B-4. USER POWER VERSUS TOTAL SPACECRAFT POWER (Concentrated in one L 1 Link)

	<u>Total DC Power</u>	<u>Total HPA Power</u>	<u>L 1 HPA Power</u>			<u>L 1 Power to Antenna</u>		<u>L 1 (P) Tx Power</u>		<u>L 1 (P) User Power</u>	<u>L 1 (C/A) Tx Power</u>		<u>L 1 (C/A) User Power</u>	
	(Watts)	(Watts)	(DC Watts)	(RF Watts)	(dBW)	(dBW)	(Watts)	(dBW)	(dBW)	(dBW)	(Watts)	(dBW)	(dBW)	
B-6	370	230	193	630	18.0	16.7	46.8	11.7	10.7	-162.4	33.9	15.3	-157.8	(reference state)
	400	260	223	73.6	18.7	17.4	55.0	19.9	13.0	-160.1	42.1	16.2	-156.9	
	450	310	273	90.1	19.6	18.3	67.6	32.5	15.1	-158.0	54.7	17.4	-155.7	
	500	360	323	106.6	20.3	19.0	79.5	44.4	16.5	-156.6	66.6	18.2	-154.9	
	550	410	373	123.1	20.9	19.6	91.1	56.0	17.5	-155.6	78.2	18.9	-154.2	
	600	460	423	139.6	21.4	20.1	102	66.9	18.3	-154.8	89.1	19.5	-153.6	
	650	510	473	156.1	21.9	20.6	115	79.9	19.0	-154.1	102	20.1	-153.0	
	700	560	523	172.6	22.4	21.1	129	93.9	19.7	-153.4	116	20.6	-152.5	
	750	610	573	189.1	22.8	21.5	141	106	20.3	-152.8	128	21.1	-152.0	
	Increase 3.0 dB									Increase 9.6 dB			Increase 5.8 dB	

TABLE B-5. USER POWER VERSUS TOTAL SPACECRAFT POWER

<u>Total DC Power</u> (Watts)	<u>Total HPA Power</u> (Watts)	<u>L 2 HPA Power</u>			<u>L 2 Power to Antenna</u> (dBW)	<u>L 2 (P) (User Power)</u> (dBW)	<u>L 2 (C/A) (User Power)</u> (dBW)	(reference state)
		(DC Watts)	(RF Watts)	(dBW)				
370	230	27	9.0	9.5	8.2	-165.6	-165.1	
400	260	57	18.8	12.7	11.4	-162.4	-161.9	
450	310	107	35.3	15.5	14.2	-159.6	-159.1	
500	360	157	51.8	17.1	15.8	-158.0	-157.5	
550	410	207	68.3	18.3	17.0	-156.8	-156.3	
600	460	257	84.8	19.3	18.0	-155.8	-155.3	
650	510	307	101	20.0	18.7	-155.1	-154.6	
700	560	357	118	20.7	19.4	-154.4	-153.9	
750	610	407	134	21.3	20.0	-153.8	-153.3	
Increase 3.0 dB						Increase 11.8 dB	Increase 11.8 dB

FIGURE B-6

Figure B-7 shows a graphical representation of "power to user" versus spacecraft power for the condition where all incremental power increases are concentrated in one link.

The impact of increased "power to user" can be reflected in either increased jamming protection or decreased user terminal sensitivity. In a spread spectrum communication system, jamming protection, expressed as a jammer to signal ratio (J/S), is a function of the processing gain in the receiver correlator and the allowable error rate threshold expressed as an energy per bit to noise density ratio (E_b/N_o). Assuming a bit error rate (BER) threshold of 10^{-5} , coherent PSK modulation and allowing 2.0 dB for non-theoretical equipment performance, the E_b/N_o threshold is 11.6 dB. Standard curves showing the BER to E_b/N_o relationship given by Figure B-8. Correlator processing gain is equal to the chip rate (10.23 Mbps for P links, 1.023 Mbps for C/A links) divided by the bit rate (50 bps). Processing gains and J/S ratios for the nominal (reference state) and maximum improvement (750 watts of available spacecraft power) are tabulated below.

FIGURE B-7. POWER TO USER VERSUS SPACECRAFT POWER
(Increments Concentrated In One Link)

	<u>L 1 (P) Link</u>	<u>L 1 (C/A) Link</u>	<u>L 2 (P) Link</u>	<u>L 2 (C/A) Link</u>
PG (dB)	53.1	43.1	53.1	43.1
E_b/N_o threshold (dB)	<u>11.6</u>	<u>11.6</u>	<u>11.6</u>	<u>11.6</u>
J/S (reference) (dB)	41.5	31.5	41.5	31.5
Link Enhancement (dB)	<u>9.6</u>	<u>5.8</u>	<u>11.8</u>	<u>11.8</u>
J/S (enhanced) (dB)	51.1	37.3	53.3	43.3

$$J/S \text{ (dB)} = PG \text{ (dB)} - \frac{E_b}{N_o} \Bigg|_T \text{ (dB)}$$

Note that with incremental power concentrated in one link, only one of the L 1 or both of the L 2 J/S enhancements is possible.

B-3 Power Flux Density Limitations

The GPS system has legal power flux density limitations and increases in spacecraft power will be effective only insofar as these limitations are not exceeded. Attachments 1, 2 and 3 taken from the White House Office of Telecommunications Policy (OTP) Radio Frequency Management Regulations and Procedures show the spectrum sharing in the L-1 and L-2 bands and define a power flux density limitation for either band of $-152 \text{ dBW/m}^2/4\text{kHz}$. Current GPS signals at the best performance evaluation angle of 40° are estimated to have power flux densities of $-154.3 \text{ dBW/m}^2/4 \text{ kHz}$ for the L-1 (C/A) signal and $-156.6 \text{ dBW/m}^2/4 \text{ kHz}$ for the L-2 (C/A) signal. These values are taken from figures 2 and 3 of Interface Release Notice (IRN) #14726.

As may be seen, these values do not allow for much in the way of user power growth. User's are counting on an unjammed C/N_o of 34 dB-Hz the raw bit error rate is 10^{-3} and this raw rate is corrected by a Hamming Code (3226) with a concatenated algorithm to give 10^{-7} BER at 50 bps which is the correlated data rate. The projected system with 10 satellites

will not produce self-interference significantly above thermal noise (N_0 estimated at -200 dBW/Hz), but extensive receiver filtering is required to eliminate adjacent interference. It was commented that the users work primarily with the L-1 link using a 4-channel CDMA (Code Division Multiple Access) receiver with approximately 4% of the time spent on each visible satellite and 20% of the time at L-2 for ionosphere delay corrections. While other waveforms and other frequencies are being considered for later phases of its program, no attention is being given to frequency hopping as opposed to the current direct sequence method of spectrum spreading.

As a cross-check on the power flux density status, the power distribution of a direct sequence spread spectrum is of the form $\left(\frac{\sin X}{X}\right)^2$ as indicated by figure 2.8 of attachment 4 taken from "Spread Spectrum Systems" by Dixon (Wiley 1976). Figure 2.9 shows the effect on correlation function of the usual bandwidth restriction as used in GPS (Bandwidth equal to two times chip rate). The power distribution has a peak to average factor of 1.53 within the main lobe and 10% of the power lies outside the restricted bandwidth and is contained in the side lobes. Peak power density is therefore 1.53×0.9 or 1.38 times the density that is obtained assuming uniform spreading which equals a worst case increase of 1.4 dB. Calculations on the L-1 C/A link (worst case) are as follows:

specified user power	-158.0 dBW (HP mode)
uniform spreading factor	-63.1 dB (over 2.046 MHz)
gain of 1 M^2 aperture at L-1	+25.5 dB
peaking factor	+1.4 dB
4 kHz reference bandwidth	+36.0 dB-Hz
correction for difference between standard space loss and worst case space loss in Rockwell link budgets	+3.9 dB
	-154.3 dBW/ M^2 /4 kHz

which correlates with the value given previously.

The significance of the 34 dB-Hz carrier to noise density (C/N_o) ratio and the 10^{-3} raw bit error rate (code corrected to 10^{-7}) can be seen from the following calculations:

Specified L-1 (P) link user power	-163.0 dBW
Receive System Noise density (4 dB NF)	-200.0 dBW/Hz
C/N_o	37.0 dB-Hz (3 dB above stated requirement)
Processing gain $\frac{\text{chip rate}}{\text{bit rate}}$	53.1 dB
Chip rate (10.23 Mbps)	73.1 dB
E_b/N_o	17.0 dB
10^{-3} BER Threshold (PSK coherent modulation)	6.8 dB reference Figure B-8
correlation loss	1.0 dB reference IRN 17426
	<hr/> sec. 3.3.7.27(d)
Operating Margin	9.2 dB

As seen from the bit error rate vs E_b/N_o curves of Figure B-8, improvements in bit error rate by 3 orders of magnitude require only a 2 dB increase in E_b/N_o and the system could probably operate without coding corrections under its present link power budgets.

A 4 dB noise figure receiver at L-band is not excessively expensive; more cost involved in the receiver's filtering than in its low noise preamplifier.

Probability of Bit Error vs Signal to Noise Ratio

— A comparison of M-ary and binary signaling techniques relating signal to noise ratio to the probability of bit error; ex. 10^{-3} means one bit lost or in error per thousand bits.

Digital Modulation Methods	Required Power BER = 10^{-5}	Required Bandwidth
16-ary coded PSK	0 dB (reference)	4 x bit rate
16-ary MFSK	0.8 dB more	4.5 x bit rate
Quadrphase	2.5 dB more	1 x bit rate
Coherent PSK	2.5 dB more	2 x bit rate
PCM/FM	4.6 dB more	2 x bit rate
FSK	6.0 dB more	3 x bit rate

E_b/N_o and the terminal sensitivity factor C/KT_s are related by the bit rate R_b as expressed by:

$$\frac{C}{KT_s} = \frac{C_r}{N_o} \cdot \frac{1}{\tau_b} = \frac{E_b}{N_o} \cdot R_b$$

where $\tau_b = R_b^{-1}$, the bit duration time; $N_o = KT_s$, the noise power density; and $C_r = E_b$.

Courtesy of C.L.Cuccia, W.J.Gill, L.H.Wilson, Philco - Ford.

INTERNATIONAL			UNITED STATES				
REGION 1 MHz	REGION 2 MHz	REGION 3 MHz	BAND MHz	NATIONAL PROVISIONS	GOVERNMENT ALLOCATION	NON-GOVERNMENT ALLOCATION	REMARKS
606-790 BROADCASTING		610-890 FIXED MOBILE BROADCASTING	806-902	NG US36 US116		LAND MOBILE	806-881 Domestic public 881-902 (non-common carrier)
329 330 330A 331 332 332A		330B 332 332A 338 339					
790-890 FIXED BROADCASTING 329 331 333 334		890-942 FIXED MOBILE BROADCASTING Radiolocation	902-928	G US36 US115 US215 US218 (ISM 915 + 13 MHz)	RADIOLOCATION	NG30 NG43 NG63 NG65	
890-942 FIXED BROADCASTING Radiolocation	890-942 FIXED RADIOLOCATION	890-942 FIXED MOBILE BROADCASTING Radiolocation					
329 331 333 339A	339A 340	339 339A	928-947	NG US36 US116 US215		LAND MOBILE	(non-common carrier)
942-960 FIXED BROADCASTING	942-960 FIXED	942-960 FIXED MOBILE BROADCASTING	947-960	NG		NG64 FIXED NG9 NC10 NC40 NG58	
329 331 333 339A	339A	338 339 339A	960-1215	G, NG 341	AERONAUTICAL RADIONAVIGATION	AERONAUTICAL RADIONAVIGATION	
960-1215	AERONAUTICAL RADIONAVIGATION 341		1215-1300	G, NG US34	RADIOLOCATION	Amateur	
1215-1300	RADIOLOCATION Amateur 342 343 344 345		1300-1350	G, NG 346	G12 G55 G56 G111		
1300-1350	AERONAUTICAL RADIONAVIGATION 346 Radiolocation 347 348				AERONAUTICAL RADIONAVIGATION Radiolocation G2	AERONAUTICAL RADIONAVIGATION	

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Attachment 1

INTERNATIONAL			UNITED STATES				REMARKS
REGION 1 MHz	REGION 2 MHz	REGION 3 MHz	BAND MHz	NATIONAL PROVISIONS	GOVERNMENT ALLOCATION	NON-GOVERNMENT ALLOCATION	
1535-1542.5	MARITIME MOBILE-SATELLITE 352 352D 352E		1535-1542.5	G, NG 352E US39	MARITIME MOBILE-SATELLITE	MARITIME MOBILE-SATELLITE	
1542.5-1543.5	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE 352 352D 352F		1542.5-1543.5	G, NG 352F US39	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE	
1543.5-1558.5	AERONAUTICAL MOBILE-SATELLITE (R) 352 352D 352G		1543.5-1558.5	G, NG 352G US39	AERONAUTICAL MOBILE-SATELLITE (R)	AERONAUTICAL MOBILE-SATELLITE (R)	
1558.5-1636.5	AERONAUTICAL RADIONAVIGATION 352 352A 352B 352D 352K		1558.5-1636.5	G, NG 352A 352B US39 US39A US208	AERONAUTICAL RADIONAVIGATION G54 (G112)	AERONAUTICAL RADIONAVIGATION	
1636.5-1644	MARITIME MOBILE-SATELLITE 352 352D 352H		1636.5-1644	G, NG 352H US39	MARITIME MOBILE-SATELLITE	MARITIME MOBILE-SATELLITE	
1644-1645	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE 352 352D 352I		1644-1645	G, NG 352I US39	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE	AERONAUTICAL MOBILE-SATELLITE (R) MARITIME MOBILE-SATELLITE	
1645-1660	AERONAUTICAL MOBILE-SATELLITE (R) 352 352D 352J		1645-1660	G, NG 352J US39	AERONAUTICAL MOBILE-SATELLITE (R)	AERONAUTICAL MOBILE-SATELLITE (R)	

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(Government Footnotes Continued)

- G109 All assignments in the band 157.0375-157.1875 MHz are subject to adjustment to other frequencies in this band as long term U.S. maritime VHF planning develops, particularly that planning incident to support of the National VHF-FM Radiotelephone Safety and Distress System (See Doc. 15624/1-1.9.111/1.9.125).
- G110 Government ground-based stations in the aeronautical radionavigation service may be authorized between 3500 and 3700 MHz where accommodation in the 2700-2900 MHz band is not technically and/or economically feasible.
- G111 In the band 1215-1250 MHz, the frequency 1227.6 MHz with emissions limited to ± 12 MHz bandwidth, is also allocated to the Radionavigation Satellite Service, for satellite down link transmissions only. The power flux density at the earth's surface from such transmissions shall not exceed $-152\text{dBW/m}^2/4\text{kHz}$. The Radionavigation Satellite Service shall not cause harmful interference to the Amateur Service and shall accept any harmful interference that may be caused by the Amateur Service.
- G112 In the band 1558.5-1636.5 MHz, the frequency 1575.42 MHz with emission limited to ± 15 MHz bandwidth is also allocated to the Radionavigation Satellite Service for satellite down link transmissions only. The power flux density at the earth's surface from such transmissions shall not exceed $-152\text{dBW/m}^2/4\text{kHz}$.

Figure 2.8 Power distribution in $[(\sin x)/x]^2$ spectrum.

Figure 2.9 RF bandwidth restriction and its effect on direct sequence signals: (a) RF envelope of direct sequence signal for various RF bandwidths; (b) effect of bandwidth restriction on correlation function.

C. EFFECT OF SOLAR PRESSURE ON
GPS ORBIT PREDICTABILITY

C-1 Introduction

The GPS space vehicles are in nominal 12-hour orbits with 63° inclination. Perturbations to the orbits are caused by earth non-sphericity, lunar and solar gravitational effects, solar and vehicle emitted radiation pressures, impingement of particles, and vehicle mass expulsions. Reference 1 asserts that the total effect of these perturbing forces are sufficiently well-modelled that, after adequate ground-tracking, the actual vehicle position can be predicted within 1 to 2 meters (RMS). Unfortunately, there is no statement in Ref. 1 as to the length of time for which such accurate ephemerides may be predicted.

C-2 Analysis

Our interest here is the incremental effect of solar pressure on the vehicle. The analytical approach taken is to write the equations which describe the incremental accelerations of the satellite resulting from the incremental forces. These differential equations are non-linear and are therefore integrated numerically to determine actual incremental motions due to the incremental forces. The results of the computerized integrations are given in the attached figures.

C-3 Relative Motion of the Satellite with Respect to the Earth

Let: 0 be an inertial fixed point

$\bar{r}_1$ = the radius vector from 0 to the earth's center

$\bar{r}_2$ = the radius vector from 0 to the satellite

$\bar{r}_s$ = the radius vector from 0 to the sun

$\bar{r}_m$ = the radius vector from 0 to the moon

m_i = the mass of the body at $\bar{r}_i$

G = gravitational constant

$\bar{F}_s$ = force due to solar pressure

Then, if the earth's oblateness is neglected and there are no forces due to solar pressure, equation (1) gives the acceleration of the earth due to gravitational forces from the satellite, the moon, and the sun.

$$(1) \quad \frac{d^2}{dt^2} \bar{r}_1 = G m_2 \frac{(\bar{r}_2 - \bar{r}_1)}{|\bar{r}_2 - \bar{r}_1|^3} + G m_s \frac{(\bar{r}_s - \bar{r}_1)}{|\bar{r}_s - \bar{r}_1|^3} + G m_m \frac{(\bar{r}_m - \bar{r}_1)}{|\bar{r}_m - \bar{r}_1|^3}$$

Similarly, the acceleration of the satellite is given by (2).

$$(2) \quad \frac{d^2}{dt^2} \bar{r}_2 = G m_1 \frac{(\bar{r}_1 - \bar{r}_2)}{|\bar{r}_1 - \bar{r}_2|^3} + G m_s \frac{(\bar{r}_s - \bar{r}_2)}{|\bar{r}_s - \bar{r}_2|^3} + G m_m \frac{(\bar{r}_m - \bar{r}_2)}{|\bar{r}_m - \bar{r}_2|^3}$$

The relative acceleration of the satellite with respect to the earth is obtained by subtracting equation (1) from equation (2). Subtracting, and letting $\bar{r} = \bar{r}_2 - \bar{r}_1$, we have:

$$(3) \quad \frac{d^2}{dt^2} \bar{r} + \frac{G(m_1 + m_2)}{r^3} \bar{r} = G m_s \left[\frac{(\bar{r}_s - \bar{r}_2)}{|\bar{r}_s - \bar{r}_2|^3} - \frac{(\bar{r}_s - \bar{r}_1)}{|\bar{r}_s - \bar{r}_1|^3} \right] + G m_m \left[\frac{(\bar{r}_m - \bar{r}_2)}{|\bar{r}_m - \bar{r}_2|^3} - \frac{(\bar{r}_m - \bar{r}_1)}{|\bar{r}_m - \bar{r}_1|^3} \right]$$

Let the terms on the right be called $\bar{R}$ so that:

$$(4) \quad \frac{d^2}{dt^2} \bar{r} + \frac{G(m_1 + m_2)}{r^3} \bar{r} = \bar{R}$$

When solar pressure is seen by the satellite the equation of motion corresponding to (4) is:

$$(5) \quad \frac{d^2}{dt^2} \bar{r}' + \frac{G(m_1 + m_2)}{r'^3} \bar{r}' = \bar{R}' + \frac{\bar{F}_s}{m_2}$$

where primed values denote the values obtained with solar pressure. The acceleration of the earth due to solar pressure is negligible.

C-4 Relative Motion of the Solar-Forced-Satellite to the Unforced-Satellite (the incremental motion)

Subtracting equation (4) from (5) yields:

$$(6) \quad \frac{d^2}{dt^2} (\bar{r}' - \bar{r}) + G(m_1 + m_2) \left(\frac{\bar{r}'}{r'^3} - \frac{\bar{r}}{r^3} \right) = (\bar{R}' - \bar{R}) + \frac{\bar{F}_s}{m_2}$$

Let $\Delta \bar{r} = \bar{r}' - \bar{r}$

and $r' = r + \delta r$

Then
$$\frac{\bar{r}'}{r'^3} - \frac{\bar{r}}{r^3} = \frac{\bar{r} + \Delta \bar{r}}{(r + \delta r)^3} - \frac{\bar{r}}{r^3}$$

$$(7) \quad \approx \frac{\Delta \bar{r} - 3 \frac{\delta r}{r} \bar{r}}{r^3}$$

But δr is approximately equal to the projection of $\Delta \bar{r}$ on $\bar{r}$ when $|\Delta \bar{r}|$ is small compared to $|\bar{r}|$.

Therefore $\delta r \approx \Delta \bar{r} \cdot \frac{\bar{r}}{|\bar{r}|}$ and the equation of incremental motion (6) can be approximated by:

$$(8) \quad \frac{d^2}{dt^2} \Delta \bar{r} + \frac{G(m_1 + m_2)}{r^3} \left[\Delta \bar{r} - 3 \frac{\delta r}{r} \bar{r} \right] = (\bar{R}' - \bar{R}) + \frac{\bar{F}_s}{m_2}$$

C-5 Calculation of the Incremental Motion

In solving equation (8) we simplify by first assuming that the term $\bar{R}' - \bar{R}$ which is the sum of increments in solar and lunar gravitational forces, is negligible. This assumption is shown to be valid in Attachment C-1 for the condition in which the incremental motions are small.

For simplicity, the satellite orbit is assumed basically circular and the satellite mass is assumed small with respect to the earth. Then, the function $\frac{G(m_1 + m_2)}{r^3}$ equals the square of the satellite orbital rate ω .

Equation (8) then reduces to:

$$(9) \quad \frac{d^2}{dt^2} \Delta \bar{r} + \omega^2 \left[\Delta \bar{r} - 3 \frac{d\bar{r}}{dt} \bar{r} \right] = \frac{\bar{F}_s}{m_2}$$

The coordinate system used for the numerical evaluations is shown in the sketch below. The sun lies on the +X axis, the orbit is tilted through an angle i about the Z axis. θ is the vehicle orbit angle measured from the +Z axis. Unit vectors $\bar{i}$, $\bar{j}$, $\bar{k}$ lie along the X, Y, and Z axes, respectively.

Then, the radius vector $\bar{r}$ is given by:

$$(10) \quad \bar{r} = (\cos i)(\sin \theta) \bar{i} + (\sin i)(\sin \theta) \bar{j} + (\cos \theta) \bar{k}$$

The vector $\Delta \bar{r}$ is given by:

$$(11) \quad \Delta \bar{r} = \Delta x \bar{i} + \Delta y \bar{j} + \Delta z \bar{k}$$

Incorporating these relations in equation (9) gives the equations of incremental motion:

$$(12) \quad \begin{aligned} \frac{d^2 \Delta x}{dt^2} + \omega^2 \left[\Delta x - 3(\Delta x \cos i \sin \theta + \Delta y \sin i \sin \theta + \Delta z \cos \theta) \cos i \sin \theta \right] &= \frac{F_x}{m_2} \\ \frac{d^2 \Delta y}{dt^2} + \omega^2 \left[\Delta y - 3(\Delta x \cos i \sin \theta + \Delta y \sin i \sin \theta + \Delta z \cos \theta) \sin i \sin \theta \right] &= \frac{F_y}{m_2} \\ \frac{d^2 \Delta z}{dt^2} + \omega^2 \left[\Delta z - 3(\Delta x \cos i \sin \theta + \Delta y \sin i \sin \theta + \Delta z \cos \theta) \cos \theta \right] &= \frac{F_z}{m_2} \end{aligned}$$

It should be noted that equations (12) may also be regarded as differential equations which describe the motion of a satellite with a given solar pressure force relative to the satellite with a different solar pressure force, i.e., in equation (6) the term $\frac{F_{sp}}{m_2}$ can represent the increment in acceleration due to incremental solar pressure forces. With this interpretation, equation (12) can be used to determine the errors in predicted satellite location due to mis-estimates of the solar pressure forces.

C-6 Simulation Results

Numerical solutions to equation (12) for a 12-hour orbit were determined using the digital computer program, given in Attachment C-2, for the values for $i=0^\circ, 30^\circ, 60^\circ$ and 90° .

Solar force per unit mass (F_{sx}/m_2) was set at 1.4×10^{-7} ft/sec², based on a nominal total solar force of 70×10^{-7} lb and satellite mass of 50 slugs. The incremental effect of an incremental solar force of $\frac{1}{100}$ of the basic solar force was also simulated. The output plots give the components of the incremental displacement of the satellite along and normal to the radius vector.

Fig. C-1 thru C4 show the increase in amplitude of the displacement, resulting from full steady solar pressure force, as the satellite orbit plane approaches the sun-line. An orbit-to-orbit increase is shown when the inclination $i \neq 90^\circ$. This follows from equation (9) wherein ξr , the increment in radius vector length, is zero if the displacement $\Delta \vec{r}$ is normal to the radius vector $\vec{r}$. In this event ($i = 90^\circ$) the equations are linear in $\Delta \vec{r}$ and represent simple undamped harmonic oscillations with $\frac{\vec{r}_s}{m_2}$

as forcing function. As i approaches zero so that $\Delta \vec{r}$ and $\vec{r}$ are not normal there is an increase in ξr which allows "instability", e.g., if $\Delta \vec{r}$ is aligned with $\vec{r}$ then and equation (9) represents an unstable second-order system. The term

represents the increment in the gravitational force of the earth on the satellite due to the increment in satellite orbit radius vector. In effect there is an inherent "gain" in the solar pressure forces due to the change in gravitational effect.

Fig. C-5 depicts the time variation of the error in estimation of satellite position when an error of 1% is made in estimating constant solar pressure forces. Again, increasing errors occur as the inclination, i , approaches zero. After three orbits (1 1/2 days), errors up to 4 feet are obtained when $i = 0^\circ$. The errors oscillate during each orbit. A comparison of the results for full and partial solar pressure forces shows that the displacements are fairly linear to the solar pressure forces.

Fig. C-6 shows the effect of mis-estimating the solar pressure force on the satellite, when $i = 90^\circ$, due to an attitude error of the vehicle. The vehicle nominally points toward the center of the earth (along $\vec{r}$) but actually points 0.5 degree out of the YZ plane. A component of the solar pressure force is thereby exerted along the radius vector and is fixed relative to the vehicle. The assumed unknown disturbance acceleration is directed along $\vec{r}$ with magnitude $1.4 \times 10^{-7} \times \sin 0.5^\circ$. A gradual increase in satellite displacement, accumulating at the rate of about 1/2 foot per orbit, is noticed.

C-7 Conclusions

1. The effect of solar pressure forces, on a vehicle in nominal circular orbit, is to destabilize the basic vehicle motion so that increasing displacement increments occur with time.
2. Large variations in displacement occur during each orbit. (Just how long the destabilization remains is unknown.)
3. The destabilized motion increases with approach of the orbit plane to the sun-line.

4. The displacements increase relatively linearly with total solar pressure force.

5. Cyclic variation in solar pressure force at orbit frequency can cause increasing displacement amplitudes.

6. Small changes in solar pressure force can cause increasingly large displacement error oscillations.

Because facilitating ephemeris predictability is an important goal in GPS design, the GPS design should be as accurately modellable as possible. If a fixed percentage error is really necessary, one should minimize the basic solar pressure force by minimizing the surface area. Random changes in vehicle attitude, in external configuration, and in radiation from the vehicle should be minimized in order both to promote predictability before launch and to aid on-orbit evaluation of the forces on the vehicle. GPS design and/or operation should be such as to minimize force changes on the vehicle resulting from attitude perturbations or variations in radiation to or from the vehicle.

RELATIVE GPS DISPLACEMENT
 Due to BASIC SOLAR PRESSURE FORCE
 $i = 0^\circ$

FIGURE C-1

FIGURE C-2

FIGURE C-3

FIGURE C-4

Effect of 1% change in direct solar pressure force

T = component normal to radius vector

R = radial component

FIGURE C-5

Relative GPS displacement
 with radial force = basic solar pressure force $\times \sin 0.5^\circ$
 $i = 90^\circ$

FIGURE C-6

ATTACHMENT C-1

Evaluation of Approximation in Equation (8)

The incremental accelerations due to sun and moon gravity are:

$$\begin{aligned}\bar{R}' - \bar{R} &= G m_s \left[\frac{\bar{r}_s - \bar{r}_2'}{|\bar{r}_s - \bar{r}_2'|^3} - \frac{\bar{r}_s - \bar{r}_1}{|\bar{r}_s - \bar{r}_1|^3} \right] + G m_m \left[\frac{(\bar{r}_m - \bar{r}_2')}{|\bar{r}_m - \bar{r}_2'|^3} - \frac{(\bar{r}_m - \bar{r}_1)}{|\bar{r}_m - \bar{r}_1|^3} \right] \\ &\quad - G m_s \left[\frac{\bar{r}_s - \bar{r}_2}{|\bar{r}_s - \bar{r}_2|^3} - \frac{\bar{r}_s - \bar{r}_1}{|\bar{r}_s - \bar{r}_1|^3} \right] - G m_m \left[\frac{\bar{r}_m - \bar{r}_2}{|\bar{r}_m - \bar{r}_2|^3} - \frac{(\bar{r}_m - \bar{r}_1)}{|\bar{r}_m - \bar{r}_1|^3} \right] \\ &= G m_s \left[\frac{(\bar{r}_s - \bar{r}_2')}{|\bar{r}_s - \bar{r}_2'|^3} - \frac{(\bar{r}_s - \bar{r}_2)}{|\bar{r}_s - \bar{r}_2|^3} \right] + G m_m \left[\frac{(\bar{r}_m - \bar{r}_2')}{|\bar{r}_m - \bar{r}_2'|^3} - \frac{(\bar{r}_m - \bar{r}_2)}{|\bar{r}_m - \bar{r}_2|^3} \right]\end{aligned}$$

Consider first the term associated with the acceleration due to solar forces.

Let $\bar{R}_s = \bar{r}_s - \bar{r}_2$, $\bar{R}'_s = \bar{r}_s - \bar{r}_2'$, $|\bar{R}'_s| = R_s + \delta$

so that $\bar{R}'_s - \bar{R}_s = \bar{r}_2' - \bar{r}_2 = \Delta \bar{r}$.

$$(\bar{R}' - \bar{R})_{\text{sun}} = G m_s \left[\frac{(\bar{r}_s - \bar{r}_2')}{|\bar{r}_s - \bar{r}_2'|^3} - \frac{(\bar{r}_s - \bar{r}_2)}{|\bar{r}_s - \bar{r}_2|^3} \right]$$

$$\frac{(\bar{R}' - \bar{R})_{\text{sun}}}{G m_s} = \frac{\bar{R}'_s R_s^3 - \bar{R}_s R_s'^3}{R_s'^3 R_s^3}$$

$$= \frac{(\bar{R}_s + \Delta \bar{r}) R_s^3 - \bar{R}_s (R_s + \delta)^3}{R_s^3 (R_s + \delta)^3}$$

$$= \frac{\Delta \bar{r} - \bar{R}_s \left(3 \frac{\delta}{R_s} + 3 \left[\frac{\delta}{R_s} \right]^2 + \left[\frac{\delta}{R_s} \right]^3 \right)}{R_s^3 \left(1 + \frac{\delta}{R_s} \right)^3}$$

Attachment C-1

Evaluation of Approximation in Equation (8)

But $\delta \leq \Delta r$ and Δr is assumed small. So that:

$$\frac{(\bar{R}' - \bar{R})_{\text{sun}}}{G m_s} \approx \frac{\Delta r - \bar{R}_s^3 \frac{\delta}{R_s}}{R_s^3}$$

and

$$(\bar{R}' - \bar{R})_{\text{sun}} \approx G m_s \left(\frac{\Delta r - \bar{R}_s^3 \frac{\delta}{R_s}}{R_s^3} \right)$$

so that the amplitude of $(\bar{R}' - \bar{R})_{\text{sun}}$ is limited by:

$$|(\bar{R}' - \bar{R})_{\text{sun}}| \leq \frac{G m_s}{R_s^3} \cdot 4 |\Delta r|$$

But $\frac{G m_s}{R_s^3} \approx \omega_e^2$ the square of the earth's orbital rate about the sun.

$$\therefore |(\bar{R}' - \bar{R})_{\text{sun}}| \leq \omega_e^2 4 |\Delta r|$$

and for Δr in feet

$$\leq 1.58 \times 10^{-13} \Delta r \quad \text{ft/sec}^2$$

The equivalent relation for the incremental acceleration due to the moon is:

$$|(\bar{R}' - \bar{R})_{\text{moon}}| \leq \frac{G m_m}{R_m^3} \cdot 4 |\Delta r|$$

where $\bar{R}_m = \bar{r}_m - \bar{r}_2$. Thus the ratio of incremental acceleration due to the moon and sun is:

$$\frac{\frac{m_m}{R_m^3}}{\frac{m_s}{R_s^3}} = \frac{m_m}{m_s} \cdot \frac{R_s^3}{R_m^3} = \frac{\frac{m_m}{m_e}}{\frac{m_s}{m_e}} \cdot \left(\frac{R_s}{R_m} \right)^3$$

Attachment C-1

Evaluation of Approximation in Equation (8)

where m_e = earth mass.

$$\therefore \frac{|(\bar{R}' - \bar{R})_{\text{moon}}|}{|(\bar{R}' - \bar{R})_{\text{sun}}|} \approx \frac{\frac{1}{81.3}}{3.29 \times 10^{-5}} \cdot \left(\frac{1.496 \times 10^8 \text{ km}}{3.84 \times 10^5 \text{ km}} \right)^3$$

$$\approx 2.2$$

But the basic acceleration due to solar pressure force on the GPS is about $1.4 \times 10^{-7} \text{ ft/sec}^2$ which is large compared to $10^{-13} \text{ ft/sec}^2$. Thus, for small values of Δr (say $\approx 10^4 \text{ ft}$) with the basic solar pressure force, or for accelerations at least 10^3 greater than the incremental solar and lunar accelerations, the incremental accelerations due to sun and moon gravity may be neglected.

ATTACHMENT C-2

```
100 SET DEGREES
110 WINDOW 0,1080,-400,400
120 AXIS 90,20
130 MOVE 0,0
140 I=0
150 N=0
160 D=1
170 X0=0
180 Y0=0
190 X1=0
200 Y1=0
210 Z1=0
220 Z0=0
230 B=0
240 T2=0
250 C=-1.4E-7
260 O1=2*PI/(12*3600)
270 O2=O1*2
280 T1=2*PI/O1*D/360
290 T=0
300 A=0
310 PRINT
320 REM
330 J=COS(I)*SIN(A)
340 K=SIN(I)*SIN(A)
350 L=COS(A)
360 P2=0
370 B=X0*J+Y0*K+Z0*L
380 X2=C-O2*(X0-3*B*J)
390 Y2=-O2*(Y0-3*B*K)
400 Z2=-O2*(Z0-3*B*L)
410 R1=Y0*L-Z0*K
420 R2=Z0*J-X0*L
430 R3=X0*K-Y0*J
```

```
439 P1=T2
440 T2=SQRT(R1↑2+R2↑2+R3↑2)
450 U2=X2
460 V2=Y2
470 W2=Z2
480 U1=X1
490 V1=Y1
500 W1=Z1
510 X1=X1+(U2+X2)*T1/2
520 Y1=Y1+(V2+Y2)*T1/2
530 Z1=Z1+(W2+Z2)*T1/2
540 X0=X0+(U1+X1)*T1/2
550 Y0=Y0+(V1+Y1)*T1/2
560 Z0=Z0+(W1+Z1)*T1/2
570 N=N+1
580 T=T+T1
581 A1=A
582 MOVE A1,P1
590 A=D*N
600 DRAW A,T2
601 MOVE A1,P2
602 DRAW A,S
610 IF T=>T1*360*3 THEN 630
620 GO TO 320
630 I=I+30
640 IF I>90 THEN 670
650 MOVE 0,0
660 GO TO 150
670 END
```

D. AN EXAMPLE OF THE EFFECT OF EPHEMERIS
ERRORS ON USER POSITION DETERMINATION

INTRODUCTION

Several previous studies have determined GPS orbit perturbations as a function of external forces. Components of these perturbations along and normal to the satellite radius vector were determined. This study considers the relative effects of these components in causing user errors in position determination.

The specific situation examined assumes three satellites in 12 hour orbit available to the user. The satellites are assumed equally spaced in azimuth and at equal angles above the user's horizon. Ranges to the satellites are measured perfectly and two of the satellites' ephemerides are known perfectly. The ephemeris of the third satellite is in error in its radial distance or in distance normal to its radius vector.

ANALYSIS

Figure D1 illustrates the geometry of the situation:

the user is located at 0,0,0

Θ is the angle above the horizon for the satellites

R_e is the earth's radius

R_s is the satellites' radial distance to the center of the earth

a is the satellites' range to the user (measured perfectly)

ψ is the angle from the local horizontal plane XY to the intersection of spheres of radius a centered on the two perfectly located satellites A and B. A and B are symmetrically located with respect to the XZ plane.

ϵ is the user position determination error.

Φ is the angle between the plane containing satellites A, B and the user and the X axis.

β is the angle between the satellite C's radius vector and the vector from the user to the satellite

$\delta_{\perp}$ is satellite C's ephemeris error normal to the radius vector

δ_R is satellite C's ephemeris error along the radius vector.

The user calculates his position based on the intersection of spheres centered on the individual satellites, presumed at locations specified by the individual ephemerides. The radius of the spheres are the range measurements made by the user.

In the situation here, the user calculates his location as the intersection of

- 1) the line which passes through 0, 0, 0, lies in the XZ plane, and is inclined by angle ψ with respect to the X axis, and
- 2) a sphere of radius a centered on the ephemeris location for satellite C.

Figure D1. Satellite Geometry

The following equations therefore apply:

$$\phi = \tan^{-1} \left(\frac{1}{2 \tan \theta} \right)$$

$$\frac{a}{R_e} = -\sin \theta + \sqrt{\left(\frac{R_s}{R_e}\right)^2 - \cos^2 \theta}$$

$$\cos \beta = - \frac{1 - \left(\frac{R_s}{R_e}\right)^2 + \left(\frac{a}{R_e}\right) \sin \theta}{\frac{a}{R_e} \cdot \frac{R_s}{R_e}}$$

$$\frac{\epsilon_L}{\delta_L} = \frac{\sin \beta}{\cos(\theta - \psi)}$$

$$\frac{\epsilon_R}{\delta_R} = \frac{\cos \beta}{\cos(\theta - \psi)}$$

Values of $\frac{\epsilon}{\delta_L}$ and $\frac{\epsilon}{\delta_R}$ were computed as function of angle θ using the program listed in Figure 2. The results in that figure indicates that ephemeris errors in radial displacement are about 5 times as important as errors in normal-to-radius displacement -- for the conditions assumed.

It may be noted that the calculated errors due to δ_L occur when δ_L lies in the XZ plane. When δ_L is normal to that plane, no user position error exists. To emphasize this fact, the second column in Figure D2 is labelled "Maximum Position Error ...".

D- 3.0

CONCLUSIONS

Errors in GPS satellite radial position are several times as significant as equal errors in normal to radial vector position in user determination of position. When choice is possible between incurring radial or normal errors, normal errors should be accepted.

SATELLITE ANG FROM HORIZON θ	MAXIMUM POSITION ERROR PER UNIT LATERAL DISPLACEMENT ϵ/δ_L	POSITION ERROR PER UNIT RADIAL DISPLACEMENT ϵ/δ_R	ERROR DIRECTION ABOVE HORIZON ψ
THETA DEG	ERROR/DISPL	ERROR/DISPL	DEG
10	0.480719210664	1.9778771914	70.5797984826
20	0.271657625219	1.17446672097	53.9515851238
30	0.211497698276	0.996145451559	40.8964069135
40	0.186108989484	0.995819059085	30.7920010435
50	0.17337814534	1.1112882845	22.762152843
60	0.16640549912	1.37776454294	16.1032998561
70	0.16250312148	1.97454179305	10.3148645669
80	0.160497376369	3.85072873974	5.03873990656

LIST

```

110 R=(10898+3438)/3438
120 N=1
130 PRINT " ", "MAXIMUM", " ", " "
140 PRINT "SATELLITE ANG", "POSITION ERROR", "POSITION ERROR", "ERROR"
150 PRI "FROM HORIZON", "PER UNIT LATERAL", "PER UNIT RADIAL", "DIRECTION"
160 PRINT " ", "DISPLACEMENT", "DISPLACEMENT", "ABOVE HORIZON"
170 PRINT
180 PRINT "THETA DEG", "ERROR/DISPL", "ERROR/DISPL", "DEG"
190 T=PI/18*N
200 P=ATN(1/(2*TAN(T)))
210 A=-SIN(T)+SQR(R^2-COS(T)^2)
220 B=-(1-R^2+A*SIN(T))/(A*R)
230 E=SQR(1-B^2)/COS(T-P)
240 F=B/COS(T-P)
250 PRINT 10*N, E, F, P*57.3
260 N=N+1
270 IF N>8 THEN 290
280 GO TO 190
290 END

```

Figure D2

E. ORBIT PREDICTION UNCERTAINTY OF
DUAL-SPIN AND BIAS-MOMENTUM SPACECRAFT

E-1.0 Introduction

A dual-spin configuration has been proposed for a candidate GPS. This arrangement has the advantages of all dual-spin systems as well as the potential advantage of highly predictable forces due to solar pressure, especially when the despin section is enclosed in a spinning radome. The purpose of this study is to make an estimate of potential predictability of the GPS orbit as influenced by solar pressure on, and attitude control requirements of, such a vehicle.

Total vehicle angular momentum is a parameter in this study. When that angular momentum is sufficiently small, it can be stored in a reaction wheel (rather than the spinning motion of the vehicle itself) so that a bias-momentum system is equivalent, on a dynamic basis, to a low-momentum dual-spin system. It is considered here in that sense.

E- 2.0 Analyses

E- 2.1 Vehicle Angular Error Rate due to Solar Pressure Torques

The precession rate $\dot{\epsilon}$ equals the external torque (the solar pressure torque T) divided by the vehicle angular momentum H.

$$\dot{\epsilon} = \frac{T}{H}$$

$$= 1.2 \times 10^{-3} \text{ deg/day/inch offset}$$

when: vehicle sunlit area = 35 ft²
vehicle spin inertia = 200 slug ft²
vehicle spin rate = 6 rad/sec
axial offset between vehicle c. p. and c. m. = 1 inch

Reasonable offsets of 6 inches may be expected, depending upon vehicle attitude relative to the sun. So that $\dot{\epsilon}_{\text{max}} \cong .007^\circ/\text{day}$.

E-2.2

Precession Requirements Due to Orbit Nodal Regression

The orbit line-of-nodes rate due to earth oblateness (Reference 1) is:

$$\dot{\Omega} = \frac{-Jn \cos i}{a^2 (1-e^2)^2}$$

where: $J = 1.624 \times 10^{-3}$

$a = \text{semi-major axis/earth radius} = \frac{10898 + 3438}{3438}$

$n = \text{mean angular velocity} = 2\pi \text{ rad/12 hours}$

$i = \text{inclination} = 63^\circ$

$e = \text{eccentricity} = 0$

so that $\dot{\Omega} = -0.0305^\circ/\text{day}$

Therefore the orbit precession rate is:

$$\begin{aligned}\dot{\gamma} &= \dot{\Omega} \sin i \\ &= 0.0272^\circ/\text{day}\end{aligned}$$

If a 0.25° error is allowed for despin section earth pointing out of a total error budget of 0.5° , then attitude corrections must be made every

$$\frac{0.25}{0.0272} = 9.2 \text{ days}$$

- a. Angular momentum corrected every 9.2 days is

$$\begin{aligned}\Delta H &= \epsilon H \\ &= 5.23 \text{ ft-lb-sec. for a 1200 ft-lb sec. vehicle with} \\ \epsilon &= 0.25 \text{ deg.}\end{aligned}$$

b. Fuel use/ ΔH correction = $\frac{\Delta H}{I_{sp} d}$

$$= 0.0087 \text{ lb for } I_{sp} = 200 \text{ sec. with 3 ft. moment arm (d= A.)}$$

c. Fuel use/year = $0.0087 \times \frac{365}{9.2}$

$$= 0.34 \text{ lb. for } I_{sp} = 200 \text{ sec. with 3 ft. moment arm}$$

d. Jet pulses/day = $\frac{\Delta H}{f d t} \approx \frac{1}{9.2}$

$$= 20/\text{day for } 0.1\# \text{ thrusters, with 3 ft. moment arm with 0.1 sec. on-time}$$

E- 2.3 Estimate of Uncertainty in Predicting Solar Pressure Forces

The relative fixity of a spinning vehicle's attitude in inertial space permits an accurate evaluation of the solar pressure forces by ground observers. Observation noise and changes in vehicle attitude relative to the sun-line prevent perfect evaluation. Appendix E-A contains an evaluation of the rates of change of the attitude relative to the sun-line due to motion of the Earth about the sun. The maximum rate of change is low, only $1^\circ/\text{day}$, as shown in Appendix E-A, and therefore permits accurate estimation of future solar forces since the solar forces change at correspondingly low rates.

Estimation errors due to the changing sun aspect of the vehicle are primarily due to uncertainty as to the characteristics of the new areas which become sunlit, and the old areas which lose the sun, as time proceeds. It is assumed here that the daily increment in solar force is known within 5%, the persisting force is assumed known exactly. The maximum percentage change in sunlit area, over one day, for a cylindrical vehicle with height 1.5 x as great as diameter is:

$$\% \text{ change} = 100 \times \frac{(1.5) (D) \sin 1^\circ}{\pi D^2/4} = 5\%$$

when the circular face looks directly at the sun (orbit plane normal to sun-line).
Also,

$$\% \text{ change} = 100 \times \frac{\pi D^2/4 \sin 1^\circ}{(1.5 D) (D)} = 0.91\%$$

when the cylinder axis is normal to the sun line (orbit plane contains the sun-line).

Thus the total percentage force uncertainty varies between $5\% \times 5\% = 0.25\%$ and $5\% \times 0.9\% = 0.045\%$, depending upon the orbit axis inclination to the sun-line. It may be recalled (Reference 2) that the largest error in estimating a satellite's ephemeris is caused by mis-estimating solar forces which occur when the orbit plane contains the sun-line, and the smallest when the sun-line is normal to the orbit plane. Thus the largest uncertainty in estimating solar force occurs when the effect of those uncertainties is least and vice versa.

E-2.4 Displacement Uncertainty Due to Solar Force Uncertainty

The solar force uncertainty after 9.2 days (the required time between firings of the jets to force the vehicle spin axis to remain within 0.25° of the precessing orbit axis) causes a displacement error uncertainty which depends upon a) the mean force uncertainty and b) the sensitivity of displacement to incremental force. The latter is a function of the angle (i) between the orbit plane and the sun-line.

In Section 2.3, the force uncertainties of 0.25% per day and 0.045% per day were determined for angles $i = 90^\circ$ and 0° , respectively. After 9.2 days these uncertainties could accumulate to 2.3% and 0.415% , with mean values over that interval of 1.15% and 0.21% . From Reference 2 (which evaluates errors for a 50 slug SV), the maximum displacement errors for 9.2 days for a 30 slug SV and for 1% force error are:

Max errors after 9.2 days per 1% force error;

$$\begin{aligned} i = 90^\circ \quad \text{normal to radius vector} \quad \delta_{\perp} &= 0.2 \text{ ft} \times \frac{5}{3} \\ &= 0.33 \text{ ft} \\ \text{along radius vector} \quad \delta_R &= 0 \end{aligned}$$

$i = 0^\circ$	normal to radius vector	$\delta_{\perp} = 3 \text{ ft/day} \times \frac{5}{3} \times 9.2 \text{ day}$ = 46 ft
	along radius vector	$\delta_R = 1 \frac{3}{4} \text{ ft/day} \times \frac{5}{3} \times 9.2 \text{ day}$ = 27 ft

Using the estimated actual daily errors of 0.25% and 0.045%:

Maximum error after 9.2 days:

$i = 90^\circ$	normal to radius vector	$\delta_{\perp} = 0.33 \times 1.15$ = 0.38 ft
	along radius vector	$\delta_R = 0$
$i = 0^\circ$	normal to radius vector	$\delta_{\perp} = 46 \times 0.21$ = 9.6 ft
	along radius vector	$\delta_R = 27 \times 0.21$ = 5.7 ft

Values for intermediate angle i are estimated as:

Maximum error after 9.2 days:

$i = 60^\circ$	normal to radius vector	$\delta_{\perp} = 4 \text{ ft}$
	along radius vector	$\delta_R = 3 \text{ ft}$
$i = 30^\circ$	normal to radius vector	$\delta_{\perp} = 7 \text{ ft}$
	along radius vector	$\delta_R = 5 \text{ ft}$

All these values are maximum or 3σ values.

E-2.5 Estimation of Orbit Disturbances Due to Unmatched or Misaligned Thrusters

The GPS thrusters fired every 9.2 days move the vehicle spin axis 0.25° to account for vehicle orbit plane nodal regression. In order to minimize ΔV effects due to jet firing, the jets are fired in pairs so as nominally to apply couples to the vehicle. Due to mismatch between jets there will be a

ΔV along an axis normal to the nominal line-of-thrust. Evaluation of these orbital disturbances was accomplished using the computer program of Reference 1. That program, altered as required for the impulse disturbances, is documented in Appendix E-B. Appendix E-B also contains computer results for jet mismatches when line-of-thrust is normal to the orbit plane, along the radius vector, and along the vehicle's velocity vector. Jet misalignment effects are determined from jet mismatch effects by considering that the misalignment causes an error impulse along a normal axis. The effects of that error impulse are the same regardless of their source --mismatch or misalignment.

Computer runs were made for two sets of conditions. In the first, the required 0.25 degree reorientation was achieved by a single thrusting. In the second, two thrustings were made, one-half orbit apart. It was assumed here that the jet mismatches/misalignments remained unchanged during this time interval. A resultant compensating effect occurs yielding a decided improvement in orbit perturbation.* A comparison of the resultant orbit perturbations indicates that minimum perturbation, due to residual ΔV , occurs when the line-of-thrust of the jets is normal to the orbit plane; indeed there is a negligible effect due to thrust mismatch. A misorientation of the jets by 0.5 degrees can add an increment to the along-orbit velocity. The incremental impulse I equals $\sin 0.5^\circ$ times the sum of the individual jet impulses $2 F \Delta t$. Since the total precession angular momentum ΔH equals the impulse $2 F \Delta t$ times the moment arm d (taken as 3 feet) we have:

$$\Delta H = 2 F d \Delta t$$

and
$$I = \frac{\Delta H}{d} (\sin 0.5^\circ)$$

$$= \frac{5.23 \text{ ft lb sec}}{3 \text{ ft}} (\sin 0.5^\circ)$$

$$= .0152 \text{ lb sec}$$

* This is due to the approximate equality of orbit period before and after the two thrustings.

In Appendix E-B the orbit perturbation obtained with impulses of 0.029 lb sec, displacements, along and normal to the radius vector are 20 ft and 55 ft, respectively, for a vehicle with a 1200 ft lb sec angular momentum with delta V along the orbit. Displacements are much smaller for an equal delta V along the radius vector. Since orbit effects are roughly proportional to the perturbations, and the perturbations (jet impulses) are proportional to vehicle angular momentum H we can approximate * orbit effects by:

$$\text{Max. displacement along radius vector} = \frac{H}{1200} \times 20 \times \frac{0.0152}{0.029} \text{ ft}$$

$$\text{Max. displacement normal to radius vector} = \frac{H}{1200} \times 55 \times \frac{0.0152}{0.029} \text{ ft **}$$

E-2.6 Estimate of Uncertainty in Predicting Orbit Perturbations of a Bias Momentum System

Bias momentum stabilized vehicles, like dual spin vehicles, tend to respond slowly to external torques. Since the precession rate is proportional to vehicle angular momentum, and stored angular momentum is limited by allowable wheel weight to about 50 ft-lb-sec, they are far less fixed in space than reasonable dual-spinners.

Using the numerics in Section E-2.1, a 50-ft-lb-sec vehicle will have a precession rate, due to solar pressure torques, of $\dot{\epsilon}_{\text{max}} = \frac{1200}{50} \times 0.007^\circ/\text{day} = 0.168^\circ/\text{day}$. for a 6-inch offset between c. p. and c. m. If vehicle reorientation is required when the precession angle is 0.25° , then jet thrusting will be required every $\frac{.25}{.168} = 1.49$ days.

* When spin angular momentum goes below about 300 ft lb sec, the period at which jets must be fired to reorient the vehicle due to precession occasioned by solar pressure torques approaches 9 days, so that the total orbit disturbances are not closely approximated by jet firings to follow the precessing orbit.

** The along radius vector component increases slowly with time due to the change in orbit period from the nominal period. It may conservatively be estimated to double in 9.2 days. Doubled values are used in the Table 1 compilation.

Total jet impulses would be $\frac{1200}{50} = \frac{1}{24}$ as large as required for the 1200 ft lb sec dual-spinner, and all vehicle effects will be in the same proportion.

Over a 9.2 day interval $\frac{9.2}{1.49} = 6.16$ firings would be required so that the total orbital effects due to jet thrustings may be expected to be approximately

$6.16 \times \frac{1}{24} \times \frac{1}{2} = \frac{1}{78}$ as great as those obtained with a 1200 ft lb sec vehicle.

(The $\frac{1}{2}$ factor takes into account the spread-out over time effects of the thrustings.)

Uncertainties in solar pressure forces are more difficult to estimate : the attitude is not as constant, the tracking time between thrustings is less, different vehicle surfaces see the sun over the orbit, and the vehicle is not quite as thermally uniform as the dual-spinner. Assumptions of uncertainties are:

- a) Daily increment in solar force known within 10% (5% for spinner)
- b) Uncertainty in basic solar force due to attitude uncertainty of 0.1° is 0.17%.

Using the technique of Section E-2.3, these values lead to mean percentage errors of $0.17\% + \frac{10\% \times 5\% \times 9.2}{2} = 2.47\%$ when the vehicle orbit plane is normal to the sun-line and $0.17\% + \frac{10\% \times 0.9\% \times 9.2}{2} = 0.585\%$ when the vehicle axis is normal to the sun-line. Just as is true for the dual-spinner, the greatest uncertainty is due to the inability to know precisely the change in solar pressure force due to the changing solar aspect of the vehicle.

Thus, using the numerics of Section E-2.4, the maximum error after 9.2 days is:

$i = 90^\circ$	normal to radius vector	$\delta_{\perp} = 0.33 \times 2.47$ $= 0.82 \text{ ft}$
	along radius vector	$\delta_R = 0$
$i = 0^\circ$	normal to radius vector	$\delta_{\perp} = 46 \times 0.585$ $= 26.9 \text{ ft}$
	along radius vector	$\delta_R = 27 \times 0.585$ $= 15.8 \text{ ft}$

Estimates of maximum errors for intermediate angles i are:

$i = 60^\circ$	normal to radius vector	$\delta_1 = 12 \text{ ft}$
	along radius vector	$\delta_R = 8 \text{ ft}$
$i = 30^\circ$	normal to radius vector	$\delta_1 = 20 \text{ ft}$
	along radius vector	$\delta_R = 13 \text{ ft}$

E-2.7 Estimated Orbit Displacements Over 9.2 Days

This section determines the 9.2 day total estimates in orbit displacements based on mis-estimations of solar pressure forces and mismatched and mis-aligned thrusters. 9.2 days is chosen because it represents a complete thrusting and non-thrusting cycle. Errors determined by combining results from Sections 2.3, 2.4, 2.5, and 2.6 may be considered as those achievable in an operation in which after tracking to establish the true initial conditions a) a two-part jet thrusting cycle is instituted to re-orient the vehicle and b) a 9.2 day orbital drift occurs -- the ephemeris for the complete cycle being estimated before the jet thrusting. Tracking and curve-fitting is employed during the 9.2 day orbital drift to establish the "true" initial conditions of the orbit at the start of the next cycle and the "true" vehicle solar pressure force parameters.

Maximum errors in vehicle ephemeris over 9.2 days are taken as the sum of jet thrusting and solar pressure force mis-estimates. Values of error components are compiled in Table E-1. Since all component error values used in Table 1 are maximum, they may reasonably be considered as 3σ values so that $1/3$ the maximums are taken as the expected or 1σ values. These results are plotted in Figure E-1 as a function of orbit plane inclination i to the sunline.

ESTIMATED ORBIT UNCERTAINTIES FOR STORED MOMENTUM VEHICLES

FIGURE E.1

Table E-1

Ephemeris Error Compilation

H ft-lb-sec	50				300				600				1200			
i	0°	30°	60°	90°	0°	30°	60°	90°	0°	30°	60°	90°	0°	30°	60°	90°
$\delta_{\perp}$ Solar (ft)	52.8	24	16	1.6	9.6	7	4	.4	9.6	7	4	.4	9.6	7	4	.4
$\delta_{\perp}$ Jets (ft)	7.4	7.4	7.4	7.4	14.4	14.1	14.4	14.4	28.8	28.8	28.8	28.8	57.6	57.6	57.6	57.6
Totals $\approx 3\sigma$	60.2	31.4	23.4	9.0	24.0	21.1	18.4	14.8	38.4	35.8	32.2	29.2	67.2	64.6	61.6	58.0
1σ (ft)	20.1	10.5	7.8	3.0	8.0	7.0	6.1	4.9	12.8	11.9	10.7	9.7	22.4	21.5	20.5	19.7
δ_R Solar (ft)	15.8	13	8	0	5.7	5	3	0	5.7	5	3	0	5.7	5	3	0
δ_R Jets (ft)	1.3	1.3	1.3	1.3	2.6	2.6	2.6	2.6	5.3	5.3	5.3	5.3	10.5	10.5	10.5	10.5
Totals $\approx 3\sigma$	17.1	14.3	9.3	1.3	8.3	7.6	5.6	2.6	11.0	10.3	8.3	5.3	16.2	15.5	13.5	10.5
1σ (ft)	5.7	4.8	3.1	0.4	2.8	2.5	1.9	0.9	3.7	3.4	2.8	1.8	5.4	5.2	4.5	3.5

Conclusions

A major contributor to GPS ephemeris uncertainty for a dual-spin satellite is due to the imperfection in the required jet thrustings. These thrustings force the satellite spin axis to remain normal to the orbit plane. The best direction for the thrustings is normal to the orbit plane. Because jet misorientation is the prime imperfection it is worthwhile to investigate means of increasing the precision of control and determination of actual jet thrust directions.

Estimated 1σ ephemeris errors as low as 3 feet in radial distance and 8 feet in distance normal to the radius vector are attainable with dual-spin vehicles. Bias momentum system estimates are about twice these values but suffer from greater uncertainty in the estimates themselves.

Persons with actual experience in estimating on-orbit vehicle characteristics should be consulted to (hopefully) confirm or (hopefully not) reject the computational assumptions of this study.

References

1. Basic Physics of the Solar System by V. M. Blanco and S. W. McCuskey. (Addison-Wesley Publishing Company - 1961).
2. Effect of Solar Pressure on GPS Orbit Predictability, Appendix C.

APPENDIX E-A

Determination of rate of change of angle between the sun-line orbit axis, as a function of time of year.

The geometry of the situation is shown in figure A-1. Axis systems are denoted by subscripts:

- I is the inertially fixed system with Z_I pointing to the north ecliptic pole and Y_I to the first point of Aries.
- e is an earth centered inertial system with Z_e pointing to the north pole and Y_e along Y_I . Z_e is displaced i_e ($=23.5^\circ$) from Z_I .
- OA is the orbit axis system with Z_{OA} pointing along the orbital rate vector. Z_{OA} is displaced by angle i ($=63^\circ$) from Z_e . Y_{OA} lies in the $X_e Y_e$ plane and is displaced by angle Ω_e from Y_e .
- Φ is the earth centered system with Z_Φ along Z_I and Y_Φ pointing from the earth to the sun so that Y_Φ is displaced by angle Φ from Y_I .

With these definitions the angle α between the orbit axis and the sun-line is given by:

$$\alpha = \cos^{-1} V_{\Phi Y}$$

where:

$$\begin{bmatrix} V_{\Phi X} \\ V_{\Phi Y} \\ V_{\Phi Z} \end{bmatrix} = \begin{bmatrix} c_\Phi & s_\Phi & 0 \\ -s_\Phi & c_\Phi & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} c_{i_e} & 0 & -s_{i_e} \\ 0 & 1 & 0 \\ +s_{i_e} & 0 & c_{i_e} \end{bmatrix} \begin{bmatrix} c_{\Omega_e} & -s_{\Omega_e} & 0 \\ s_{\Omega_e} & c_{\Omega_e} & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} c_i & 0 & s_i \\ 0 & 1 & 0 \\ -s_i & 0 & c_i \end{bmatrix} \begin{bmatrix} 0 \\ 0 \\ 1 \end{bmatrix}$$

and $c = \cos$, $s = \sin$.

Thus:

$$\alpha = \cos^{-1} (-s\Phi c i_e c\Omega si + c\Phi s\Omega si + ci s\Phi si_e)$$

And:

$$-s\alpha \frac{\partial \alpha}{\partial \Phi} = -ci_e c\Omega si c\Phi - s\Omega si s\Phi + ci si_e c\Phi$$

These functions were calculated using the program of Figure E-A-2 with the results plotted in Figure E-A-1. The maximum rate of change

$\frac{\partial \alpha}{\partial \Phi}$ is seen to be unity.

```

100 INIT
110 SET DEGREES
120 WINDOW 0,360,-1,1
130 AXIS 30,0.2
151 PRINT
160 E=23.5
170 I=63
180 O=0
190 F=-90
191 K1=COS(E)*SIN(I)
192 K2=SIN(I)
193 K3=COS(I)*SIN(E)
200 A=ACS(+(-SIN(F)*COS(O)*K1+COS(F)*SIN(O)*K2+SIN(F)*K3))
210 D=+(-COS(O)*COS(F)*K1-SIN(O)*SIN(F)*K2+COS(F)*K3)/-SIN(A)
220 IF O=0 THEN 270
221 DRAW A,D
230 O=O+15
240 IF O>360 THEN 290
250 GO TO 200
270 MOVE A,D
271 GO TO 230
290 O=0
300 F=F+15
310 IF F>90 THEN 330
320 GO TO 200
330 END

```

***** THIS PROGRAM IS ON FILE 9 *****

$\phi = -90^\circ$ = winter solstice
 $\phi = 0^\circ$ = vernal equinox
 $\phi = 90^\circ$ = summer solstice

Rate of change of angle between
 orbit normal and sun line with
 change in earth position in its
 orbit about sun ($d\alpha/d\phi$)
 as a function of
 Angle to sun-line (α)
 and
 Earth's orbital position measured
 from vernal equinox (ϕ)

```

LIST
100 SET DEGREES
110 WINDOW 0,1000,-200,200
120 AXIS 90,20
130 MOVE 0,0
140 I=0
150 N=0
160 D=1/2
170 X0=0
180 Y0=0
190 X1=0
200 Y1=0
210 Z1=0
220 Z0=0
230 B=0
240 T2=0
250 C=8.05E-6/2
251 C1=C
260 O1=2*PI/(12*3600)
270 O2=O1/2
280 T1=2*PI/O1*D/360
290 T=0
300 A=0
310 PRINT
320 REM
330 J=COS(I)*SIN(A)
340 K=SIN(I)*SIN(A)
350 L=COS(A)
360 P2=B
370 B=X0*J+Y0*K+Z0*L
380 X2=-O2*(X0-3*B*J)+C
390 Y2=-O2*(Y0-3*B*K)
400 Z2=-O2*(Z0-3*B*L)
410 R1=Y0*L-Z0*K
420 R2=Z0*J-X0*L

```

```

430 R3=X0*K-Y0*J
439 P1=T2
440 T2=SQR(R1↑2+R2↑2+R3↑2)
450 U2=X2
460 V2=Y2
470 W2=Z2
480 U1=X1
490 V1=Y1
500 W1=Z1
510 X1=X1+(U2+X2)*T1/2
520 Y1=Y1+(V2+Y2)*T1/2
530 Z1=Z1+(W2+Z2)*T1/2
540 X0=X0+(U1+X1)*T1/2
550 Y0=Y0+(V1+Y1)*T1/2
560 Z0=Z0+(W1+Z1)*T1/2
570 N=N+1
580 T=T+T1
581 A1=A
582 MOVE A1,P1
590 A=D*N
600 DRAW A,T2
601 MOVE A1,P2
602 DRAW A,B
610 IF A=>1080 THEN 630
611 C=0
612 IF A=180 THEN 680
613 IF A=180.5 THEN 680
614 IF A=0.5 THEN 680
620 GO TO 320
630 I=I+30
640 IF I>90 THEN 670
650 MOVE 0,0
660 GO TO 150
670 END
680 C=C1
681 GO TO 620

```

Figure E-B-1. Relative Displacement of 30 slug, 1200 ft-lb-sec GPS Due to Procession Jet Mismatch of $3 \frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluse Normal to Orbit Plane Single Firing at Orbit = 0.

Figure E-B-2. Relative Displacement of 30 slug, 1200 ft-lb-sec GPS Due to Precession Jet Mis-match of $3 \frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluse Along Orbit Single Firing at Orbit = 0.

Figure E-B-3. Relative Displacement of 30 slug, 1200 ft-lb-sec GPS Due to Precession Jet Mis-match of $3\frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluse Along Radius Vector Single Firing at Orbit = 0.

Figure E-B-4. Relative Displacement of 30 slug, 1200 ft-lb-sec GPS Due to Precession Jet Mis-match of $3 \frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluses Normal to Orbit, Two Equal Firings: One at Orbit = 0, One at Orbit = $1/2$.

Figure E-B-5. Relative Displacement of 30 slug, 1200 ft-lb-sec GPS Due to Precession Jet Mis-match of $3\frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluses Along Orbit, Two Equal Firings: One at Orbit = 0, One at Orbit = $1/2$.

Figure E-B-6. Relative Displacement of 30 slug, 1200 ft-lb-sec. GPS Due to Precession Jet Mismatch of $3 \frac{1}{3}\%$. Total Impluse = 0.029 lb-sec Impluse Along Radius Vector, Two Equal Firings: One at Orbit = 0, One at Orbit = $1/2$.

F. ACS ELEMENTS FOR BIAS-MOMENTUM-
CONTROLLED GPS SPACECRAFT

F1 Introduction

This appendix documents the ACS requirements for a GPS/SV with biased angular momentum control. Two vehicle orientations are considered: 1) the vehicle axis points to the Earth and 2) the vehicle axis is normal to the orbit plane. In both, the bias momentum is normal to the orbit plane. The orientations impose similar sensor requirements, but due to different solar aspects require different bias momentum requirements. These have been evaluated for the two configurations assuming the vehicle as cylindrical, 8 ft. long and 6 ft. in diameter about the yaw axis, and covered with white paint. Center-of-mass displacement from the vehicle geometric center is assumed as 3 inches maximum. As shown below, the actuator sizing depends upon the allowable time between jet thrustings for vehicle unload of stored momentum.

F2 Sensor Complement

Since the bias momentum GPS control requirements differ from the NDS requirements only in needing an earth pointing signal about the single continuously controlled axis rather than the two axis requirement of NDS, the sensor complement of NDS will be suitable for the bias momentum system. While probably not optimum, it will be proposed here for simplicity.

Table E-1 AVCS Physical Characteristics [†]

Component	Qty Per Ship	Estimated Unit Size (cm)			Weight Per Ship [*] Lbs
		Length	Width	Height	
Control Electronics Assembly	1*	38.1	22.9	15.3	18.5
Reaction Wheel	1	N/A	N/A	N/A	N/A
Combined Earth Sensor	1*	31.0	18.9 dia	N/A	13
Spin Sun Sensor	2	5.1	3.9	3.1	2.5
Nutation Damper	1	61.0	6.0 dia	N/A	2.5 [‡]
Electromagnets	2*	43.0	2.5 dia	N/A	2

*Internally redundant

[†] from Rockwell GPS design CID-SV-101A Vol 5, Part 1 1975 June 06

[‡] from Rockwell GPS design 16 GP 6296

[‡] estimated

F3 Z Axis to Earth Orientation

An evaluation was made of the vehicle's angular momentum accumulation in the orbit plane (i.e., normal to the bias momentum). The program of Fig. F1 was utilized and graphical results are given in Fig. F2. Over the set of conditions shown, the maximum accumulation ΔH of 0.035 ft lb sec per orbit is noted for the condition $i = 45^\circ$ with zero center-of-mass offset.

I= ORBIT INCLINATION TO SUN-LINE (DEG)
H= CHANGE OVER ONE ORBIT OF ANGULAR MOMENTUM (FT-LB-SEC)
DUE TO SOLAR TORQUES
R0= OIS ENGINE OPENING RADIUS (FT)
D= CENTER OF MASS DISPLACEMENT (IN)

I	H	R0	D
0	5.453134394E-15	0	0
15	0.0333748776284	0	0
30	0.0578069837488	0	0
45	0.0667497552569	0	0
60	0.0578069837488	0	0
75	0.0333748776284	0	0
90	6.943146353E-15	0	0
0, 3			
0	0.0721975203953	0	3
15	0.109040180952	0	3
30	0.127539710934	0	3
45	0.122332836273	0	3
60	0.0941683413117	0	3
75	0.0497504832999	0	3
90	7.831520866E-15	0	3
0, -3			
0	0.0721975203953	0	-3
15	0.0422904256949	0	-3
30	0.0119257434368	0	-3
45	0.0111666742408	0	-3
60	0.0214456261859	0	-3
75	0.0169992719569	0	-3
90	6.05477184E-15	0	-3

Figure 1

Figure 2. (Sheet 1 of 3)

Figure 2. (Sheet 2 of 3)

Accumulation of Angular Momentum
in Orbit Plane
FT-LB-sec

Figure 2. (Sheet 3 of 3)

If $\Delta\theta$ is the permitted angular error in pointing to the nadir, then the relation between bias momentum H and the time interval Δt between reorientations (achieved by jet firing) is:

$$\Delta t = \frac{H \Delta\theta}{(\Delta H/\text{orbit})}$$

If $\Delta\theta = 0.25$ deg

$\Delta H/\text{orbit} = 0.035$ ft lb sec

then $\Delta t = 0.125$ H orbits

or $\Delta t_d = 0.062$ H days

Table F-2 below puts numerics in this equation and also shows the incremental H required by a GPS which contains an Organic Rankine cycle engine with angular momentum of 17.6 ft lb sec. The weight of wheel with that incremental H is included.

Table F-2

Δt days between reorientations	H ft lb sec	Incremental H with O.R. engine ft lb sec	Wheel Weight lbs (Estimated)
1	16	1.6	10
2	32	14.4	21
3	48	30.4	24
5	80	62.4	35
7	112	94.6	45

It should be noted that, in configuration layouts for this on-orbit orientation, the wheels and/or power supply rotating machine are oriented with their spin axes normal to the booster thrust axis.

The above calculations are based on the assumption that the nuclear engine's coolant flow contributes no net angular momentum to the system other than about the bias momentum axis. Angular momentum due to coolant flow either causes vehicle angular errors or is removed by jet action, nulled by auxiliary wheels or nulled by suitable slight inclination of the reaction wheel or rotating machine. Preset inclination of the rotating machine appears to be the most satisfactory alternative if the angular errors are unacceptable.

F4 Z Axis Normal to Orbit Plane Orientation

An evaluation was made of the vehicle's angular momentum accumulation in the orbit plane. A modification of the program of Ref. 1 was utilized and tabular results are given in Fig. 2. Over the set of conditions shown, the maximum accumulation ΔH of 0.127 ft lb sec per orbit is noted for the condition $i = 30^\circ$ with center-of-mass offset of 3 in.

Table F-3 below gives the data for this orientation equivalent to that given in Table F-2 for its orientation.

$$\Delta t_d = 0.0172 H \text{ days}$$

Table F-3

Δt days Between reorientations	H ft lb sec	Incremental H with O.R. engine ft lb sec	Wheel Weight lbs (Estimated)
1	58.4	40.8	25
2	116.8	99.2	45
3	175.2	167.6	65

Basic spinner ACS component complement, used on Hughes HS-333 Anik-A.

	Wt. #
(2) Earth Sensors	1.20
(2) Sun Sensors	0.24
On-board dynamic balance mechanism	19.99
Motor bearing ass'y.	9.82
Nutation damper SN6	<u>0.98</u>
Total weight	32.23 lbs.

As shown, practically all the required weight is due to mechanical components not ordinarily considered as ACS elements. More important, these weights are not reviewable for validity or improvement without detailed explicit design; they therefore, must be accepted at face value.

"Standard" spinners and dual-spinners use only two jets, one radial for velocity control, the other axial for attitude control. It has already been shown that axially oriented jets, firing in pairs as couples, are most effective in minimizing orbit perturbations due to required jet thrusting for attitude control. Thus, at least two jets axially oriented and oppositely firing are required as part of the jet complement.

G. ORGANIC RANKINE CYCLE INPUT DATA *

* From Sunstrand Energy Systems, Rockford, Ohio

G-1 GPS System Performance

System performance was analysed using the computer programs developed for the KIPS program. These model the interactions of the components and allow individual components to be sized to meet given mission requirements. Alternatively, the system performance in an off-design mode can be determined with no change in component configuration. The design program is controlled by an optimization procedure which allows component designs to be varied to meet preselected constraints on the system or to maximize efficiency, or minimize weight or some compromise between the two.

The optimization procedure consists of a computerized mathematical model of the system incorporating both cycle analysis and component design. The logic is arranged in the form of a mainline program which reads input data, performs a cycle analysis, and prints output. The mainline is the vehicle used to call subroutine elements which contain mathematical relationships describing working fluid properties, component design logic, and a multivariable optimization routine.

The operation of the program is illustrated by the flow chart shown in Figure G-1. For the salient inputs such as condenser temperature, regenerator effectiveness, heat transfer component pressure losses, prime mover design variables, and radiator component probabilities, a cycle analysis is undertaken and cycle efficiency and system weight calculated. A criterion function (PAOFF) defined as follows is then computed:

Figure G-1 Basic Flow Chart of Rankine Cycle System Design Optimization Program

$$PAOFF = \frac{(1-C)}{\eta_c} + (C)W$$

Where: C = weighting factor

η_c = system efficiency

W = system weight

The inputs to the program are then varied by the optimization subroutine and the cycle calculation and component design procedure repeated until PAOFF is minimized. By conducting the optimization with values of the weighting factors between zero and unity a locus of optimum systems can be generated as a function of system weight and efficiency as shown by the general curve in Figure G-2. Each point on the curve corresponds to a different optimum system design, the relative importance of efficiency and weight being different.

The modular construction of the program coupled with the generalized approach to optimization enables Sundstrand to optimize the design of Rankine cycle engines required for various applications in a very rapid manner. The basic main-line program is tailored to call the subroutines containing the logic for each component included in the particular system. Sundstrand has developed a comprehensive inventory of computer subroutines to support the overall system design optimization program. The optimizations have been performed about a representative system comprising a turbo-alternator-feed pump combined rotating unit (CRU), a vapor-liquid regenerator, a jet condenser, and liquid radiator.

Typical Cycle Efficiency - Weight Relationship

Figure G-2

By interconnecting detail design and evaluation subroutines for the various system components with the correct logical criteria, the mathematical model can duplicate the operating characteristics of a real system. This technique offers a utility much more powerful than single parameter trade-offs. The information gained correctly reflects not just the first order variations, but because the complete system is being studied, it also shows effects not available in single component parameter trade-offs. For the GPS study the only components whose design was allowed to change from the KIPS system were the jet condenser, turbine nozzles, pump impeller, radiator and radiator bypass valve. The radiator outlet temperature was fixed by the mission requirements at 86°F, and the liquid/vapour flow ratio of the jet condenser was preserved at the same value (9:1) as for the KIPS system. The system schematic, figure G-3 was modified to include a heat addition between the radiator and bypass valve equal to the total electrical output less 200 watts. Sample printouts from the design and off-design programs are presented in figures 4 and 5 respectively showing relevant operating conditions. Based on the current KIPS Heat Source Assembly heat loss of 200 Btu/hr per module, the end-of-mission heat input, heat rejected and power output is slightly higher than shown in figures G-4 and G-5.

FIGURE G-3 GPS SYSTEM SCHEMATIC

FAIRCHILD GLOBAL POSITIONING SYSTEM ORC KIPS/MODIFIED

OUTPUT

STATE POINT	PRESS(PSIA)	TEMP(F)	ENTHALPY(B/Lb)
TURBINE INLET	57.0187	650.000	396.265
REGEN VAPOR IN	0.962139E-01	521.204	340.012
JET CONDENSOR VAP IN	0.638046E-01	225.542	224.820
JET COND MIXED OUTFLOW	30.3801	196.065	56.7087
PUMP INLET	29.8706	197.479	57.2875
RADIATOR AND ALT IN	98.0577	197.990	57.6671
RADIATOR OUTLET FLOW	78.0577	85.2692	12.1169
JET CONDENSOR LIQ IN	75.9504	150.614	37.8951
REGEN LIQ IN	67.5482	211.605	63.4907
BOILER INLET	65.9137	453.311	179.179
BOILER OUTLET	60.2347	653.286	397.662
TURBINE NOZZLE IN	57.0187	652.024	397.248

CYCLE CONDITIONS

ALT EFF	CONDITION EFF	GEAR LOSS(KW)	NET PWR(KW)	DPL BOILER
0.928996E 00	0.9217815E 00	0.7999998E-01	0.7500000E 00	0.5679031E 01

RFT(LB/LH.R)	REGEN FACT	DP THROTTLE	PARA PWR(KW)
0.9315000E 01	0.1000000E 01	0.3000000E 02	0.2500000E-01
TB EFF	FEED PUMP EFF	REGEN EFFEC	UPV REGEN
0.6556544E 00	0.5643690E 00	0.9550000E 00	0.3209000E-01
			DPL REGEN
			0.4009999E 01

PUMP NS = 0.8782556E 03

CRU EFF	FLOW(LH/SEC)	Q ADD(H/HR)	Q REJ(H/HR)	CYC EFF
0.5182277E 00	0.1863309E-01	0.1472271E 05	0.1403964E 05	0.1738641E 00

TURBINE

TURBINE DIA = 5.9500	TURBINE RPM = 33683.	INLET TEMP = 1110.	INLET PRES = 57.0
TURBINE HP = 1.414	EXH PRES = 0.066	GAMMA = 1.0221	GAS CONST = 9.31
SPEC HEAT = 0.4239	MOLEC WGT = 0.0	BLADE HIGHT = 0.5207	BLADE CHORD = 0.2992
TOTAL PRES R = 861.129	TOTAL MDOT = 67.1	DIFFUSER REC = 0.0	TARGE LAP = 1.0000
IN BLADE ANG = 27.3630	OUT BLDE ANG = 27.3630	VISCOSITY = 0.0284	REF TEMP = 1110.
NOZ EDGE = 0.0100	TURB EDGE = 0.0100	SCAV LOSS = 0.0075	DISC FRIC LO = 0.0574
VIS EXP = 1.0000	PUMP LOSS = 0.0005	BLADE AREA = 0.01864	CLP RADIAL = 0.0100
BLADE DEN R = 0.5500	CLR AXIAL = 0.0200	PUMP POWER = 0.0	TIP VEL = 951.
NUM BLADES = 115.	TOTAL ADMIS = 0.9726	SPEC SPEED = 35.3493	SPEC DIA = 1.0721
TURBINE EFF = 0.6556	TOTAL AD MD = 63688.		
TURBINE TORQ = 0.2205	SPEC PROP CO = 47.4254		

*** STAGE 1

NUMBER NOZ = 15.00	NOZ SHAPE IS CIRC	AREA RATIO = 121.4817	TOTAL NOZ AH = 0.0094
BLD EX AREA = 9.0240	NOZ ANGLE = 15.000	VELOCITY HTO = 0.4320	ADMISSION = 0.9726
ROT IN REL M = 2.0870	STAGE MDOT = 0.0186	ROT IN AH V = 1943.	ROT EX AH V = 0.433.
ROT IN RE V = 1122.	ROT EX REL M = 1.1213	ROT VEL COEF = 0.5479	NOZ COEF = 0.9000
EXH LS COEF = 1.0000	ROT EX RE V = 615.	BL MDI/NOZ D = 1.6728	REC'D BH = 0.5269
NOZ INT DIA = 0.0282	NOZ FLOW COF = 0.9600	NOZ FLOW RAT = 0.9996	SIG ADIA MD = 43940.
HYD EFF = 0.7732	NOZ EXIT DIA = 0.3113	JOYN LEAK = 0.0055	SCAV FRACTION = 0.0075
NOZ EX M N = 0.6156	STATIC LEAK = 0.0	ROT FLOW RAT = 1.0000	ROT EX VEL A = 40.0800
STAGE IN P = 57.0199	NOZ ACT M N = 3.7963	ROT EX I PHS = 0.691	ROT EX S PHS = 0.066
ROT RT P IN = 0.582	ROT T P IN = 31.295	ROT EX MACH = 0.7914	ROT EX RT T = 989.
ROT IN TEMP = 1115.	ROT EX RT P = 0.126	ROT OUT TEMP = 980.	ROT OI S TEM = 971.
ROT REL TEMP = 992.	ROT IN S TEM = 937.	PRES RATIO = 801.1269	STAGE EFF = 0.6899
GAS HP = 1.47	REYN N M DIA = 1809.		
LXHAUST T T = 985.			

FIGURE G-4 DESIGN PROGRAM, END-OF-MISSION

FAIRCHILD GLOBAL POSITIONING SYSTEM ORC OFF DESIGN				KIPS/MODIFIED	02/11/77	650.0	7.000	377.0	0.7465
STATE POINT	PRESS (PSIA)	TEMP (F)	ENTHALPY (B/LB)						
TURBINE INLET	56.8745	650.000	396.274	0.875801					
REGEN VAPOR IN	0.891699E-01	521.489	340.139	0.140311E-02					
JET CONDENSOR VAP IN	0.667138E-01	228.186	225.695						
RADIATOR AND ALT IN	97.2414	199.801	56.2353	62.4642					
RADIATOR OUTLET FLOW	77.2414	87.3047	11.9413						
JET CONDENSOR LIQ IN	73.4378	151.556	38.1186	63.8616					
JET COND MIXED OUTFLOW	29.3751	197.577	57.2891						
PUMP INLET	28.8985	198.958	57.8766	62.4990					
REGEN LIQ IN	67.0720	212.632	63.7318	62.1189					
REGEN LIQ OUT	65.5439	452.322	177.666						
BOILER INLET	65.5439	452.322	177.666	54.6853					

TB EFF 0.6444522E 00	ALT EFF 0.9298318E 00	H+S LOSS 0.7999998E-01	CONDITION EFF 0.9218003E 00	REGEN EFFEC 0.9501891E 00
P TB IN 0.5687448E 02	T TB IN 0.6500000E 03	P COND 0.6671470E-01	R (FT.LB/LB.R) 0.9315000E 01	COND REC FACT 0.4000000E 00
DPV REG 0.3200000E-01	DPL REG 0.4009999E 01	DPL BOIL 0.5679999E 01	DP THROI 0.2777751E 02	DP HEAT REJ 0.2000000E 02
NET POWER 0.7464821E 00	CRU RPM 0.3368300E 05			
J COM MACH NO 0.5536596E 00	CUL 0.9600000E 00	AREA LIQ JETS 0.2605699E-04		
TB CP 0.3326416E 00	TB GAMMA 0.1037337E 01	PUMP NO 0.1575522E 03	PUMP GPM 0.1318236E 01	PUMP EFF 0.5646465E 00
M.F. (LB/SEC) 0.1858279E-01	J COND EFFEC 0.9275655E 00	CRU EFF 0.4975965E 00	CYCLE EFF 0.1731111E 00	
HEAT AD (B/HR) 0.1471845E 05	HEAT RJ (HEAT) 0.1195438E 05	HEAT RJ (COND LOSSES) 0.2161331E 03		
FEED PUMP FR =	9.87745	RAU BYPASS FRAC=	0.4894	
REGEN BYPASS =	0.0			
FEED PUMP KW =	0.0694	NET KW =	0.7465	
ALT POWER INPUT =	0.9031	WINDAGE =	0.0089	
HEATING KW =	0.0420	SEAL KW =	0.0400	
ALT EL LOSS KW =	0.0543	COND LOSS KW =	0.0654	
TURBINE KW =	1.0496	PANA POWER =	0.0250	

0.4042E-02 0.7465 0.5917E-01 199.8 56.87 650.0

FIGURE-5 OFF-DESIGN PROGRAM, END-OF-MISSION

G-2 Radiator Design Procedure

The radiator design procedure is illustrated by the basic flow chart shown in figureG -6. The logic is computerized and integrated with the Rankine cycle system design optimization program. Radiator variables under control of the multi variable optimization routine include the following:

- 1) header probability of non puncture
- 2) inside tube probability of non puncture
- 3) outside (spaceside) tube probability of non puncture
- 4) liquid pressure drop
- 5) fin efficiency

Other quantities such as overall probability of non puncture, radiator diameter, material and fluid properties and armor correlation coefficients are input to, or calculated by, the systems program.

The design procedure consists of satisfying the heat transfer, armor thickness and friction pressure drop equations for a specified heat load, fluid condition, heat sink temperature and probability of no puncture in a prescribed time period.

Heat transfer coefficients for the inside of the radiator tubes are determined using heat transfer data from reference 1 shown in figureG-7for tubes with dimpled surfaces. The purpose of the dimpled surface is to extend the turbulent heat transfer into lower Reynolds numbers then is possible with a plain tube. An equivalent heat transfer coefficient on the outside surface is defined in terms of the Stefan radiant. heat transfer equation. The overall heat transfer coefficient

* CALCULATED BY DESIGN OPTIMIZATION PROGRAM
** UNDER CONTROL OF OPTIMIZATION ROUTINE

Figure G-6 Radiator Design Procedure

Fig. 10-3. Flow inside dimpled flattened tubes, surface FTD-1.

Tube ID before flattening = 0.234 in.

Tube inside dimension perpendicular to flats = 0.114 in.

Tube inside dimension parallel to flats = 0.302 in.

Length of flat along tube = 3.60 in.

Length of section from flat-to-circular-to-flat cross section = 0.345 in.

Dimple depth = 0.03 in.

Minimum distance between dimples = 0.5 in.

Flow passage hydraulic diameter, $4r_h = 0.01116$ ft

Minimum free-flow area per tube = 0.000169 ft²

FIGURE G-7 HEAT TRANSFER AND PRESSURE DROP DATA

is computed using the inside and outside film coefficients together with the conduction term and an overall efficiency term for the extended surface. This latter quantity is determined from the input fin efficiency and a fin length calculation per reference 2. A fin thickness of 0.025 inches is used to match the KIPS design.

Tube pressure drop is determined from the friction factor data given in figure G-7. Armour thickness on the header and tube backside surface is derived from data in reference 3, while the armour requirements of the tube space side came from reference 4.

In order to determine the optimum configuration, various radiator diameters and number of tubes were selected. Sample computer printouts are presented in figures G-8 through G-10 for one, two and three tubes.

VECTOR LOC.		VARIABLE VALUE					
641.3	0.8335	18.00	0.2500E-01	0.9823	2	164.6	56.67
HEAT REJ	T LIO IN	DELTA T	RAD FLOW	VISC LIO			
0.140400E 05	0.1980000E 03	0.1127300E 03	0.8561999E-01	0.4494691E 01			
RHO LIO	T CON LIO	DP TUBE	DP HEADER	PPOR HEADER			
0.6409724E 02	0.8176351E-01	0.1800000E 02	0.2000000E 01	0.9455100E 00			
ETA FIN	S/D	DBW	TUBE I D	HEADER AV I D			
0.8331500E 00	0.5000000E 01	0.2520400E 00	0.2717667E 00	0.1276109E 00			
NO TUBES	AV T LIO	AV T FIN HASE	TUBE THK	CALC TUBE THK			
0.1000000E 01	0.5954646E 03	0.5876628E 03	0.1337096E 00	0.1337096E 00			
HEADER THK	CALC HDR THK	TUBE BSDE THK	FIN THK	CONNECT. LENG			
0.2000000E-01	0.1422115E-01	0.2000000E-01	0.2499680E-01	0.446413E-01			
HEADER LENG	TUBE LENG	PROB IN	PROB OUT	PROB TUBE			
0.7682272E 00	0.1456317E 03	0.9568117E 00	0.9874100E 00	0.9442619E 00			
PARTICLE DIA	TUBE AREA	HEADER AREA	TOT AREA	FIN WT			
0.5534638E-01	0.6543787E 01	0.0	0.1118401E 03	0.3684898E 02			
TUBE WT	TUBE LINER WT	INVENT WT	HEADER WT	HEAD LINER WT			
0.1603265E 02	0.0	0.3773842E 01	0.1663040E-01	0.0			
TOT WT							
0.5667224E 02							
0.9823	2399.	164.6	4.900	0.7162E-03	0.2000E-02		

INC9001 EXECUTION TERMINATING DUE TO ERROR COUNT FOR ERROR NUMBER 217

INC2171 FIOCS - END OF DATA SET ON UNIT 1

TRACEBACK ROUTINE CALLED FROM ISN REG. 14 REG. 15 REG. 0 REG. 1

IRCOM 00360EF4 0036E140 00000000 00360860

MAIN 0000EEF2 0136046A 0004EED0 0036EFFF

ENTRY POINT= 0136046A

FIGURE G-8 RADIATOR DESIGN, ONE TUBE

OLD BASE POINT PAOFF		NEW BASE POINT PAOFF			
0.6560781E 03		0.6560781E 03			
VECTOR LOC.		VARIABLE VALUE			
2		0.8323156E 00			
9		0.2437558E 00			
656.1	0.8323	12.00	0.2500E-01	0.7229	1766. 198.3 51.41
HEAT REJ	T LIO IN	DLTA P	RAD FLOW	VISC LIO	
0.1404000E 05	0.1980000E 03	0.1127300E 03	0.8561994E-01	0.4499691E 01	
RMO LIO	T CON LIO	DP TURE	DP HEADER	PROR HEADER	
0.6405724E 02	0.8176351E-01	0.1800000E 02	0.2000000E 01	0.9855100E 00	
ETA FIN	S/O	DRW	TUBE I D	HEADER AV. I D	
0.8323156E 00	0.5000000E 01	0.2520400E 00	0.1846468E 00	0.1473470E 00	
NO TUBES	AV T LIO	AV T FIN BASE	TURE THK	CALC TURE THK	
0.2000000E 01	0.5958494E 03	0.4459941E 03	0.1268985E 00	0.1268985E 00	
HEADER THK	CALC HDR THK	TURE RSOE THK	FIN THK	CONNECT. LENG	
0.2000000E-01	0.1526874E-01	0.2000000E-01	0.2499992E-01	0.5535096E-01	
HEADER LENG	TURE LENG	PROR IN	PROR OUT	PROR TURE	
0.1537044E 01	0.7396761E 02	0.9968117E 00	0.9874100E 00	0.9842619E 00	
PARTICLE DIA	TURE AREA	HEADER AREA	TOT AREA	FIN WT	
0.5087251E-01	0.5405106E 01	0.0	0.1132422E 03	0.3774284E 02	
TURE WT	TURE LINER WT	INVENT WT	HEADER WT	HEAD LINER WT	
0.1183141E 02	0.0	0.1795081E 01	0.3754976E-01	0.0	
TOT WT					
0.5140720E 02					
0.7229	1766.	198.3	4.900	0.7162E-03	0.2006E-02

IHC9001 EXECUTION TERMINATING DUE TO ERROR COUNT FOR ERROR NUMBER 217

IHC2171 FIOCS - END OF DATA SET ON UNIT 1

TRACERACK	ROUTINE	CALLD FROM	TSN	REG. 14	RFG. 15	REG. 0	REG. 1
	IBCOM			00330EE4	0033E140	00000000	00330860
	MAIN			0000FEF2	01330488	0006EFD0	0035EFF8

ENTRY POINT= 01330488

FIGURE G-9 RADIATOR DESIGN, TWO TUBES

0.0001500E 03 0.0001500E 03
 VECTOR LOC. VARIABLE VALUE
 0.0300013E 00
 0.2454515E 00
 688.2 0.8300 18.00 0.2500E-01 0.6057 1480. 218.1 49.67
 HEAT REJ T LIQ IN DELTA T HAD FLOW VISC LIQ
 0.1404000E 05 0.1900000E 03 0.1127300E 03 0.6561999E-01 0.4899691E 01

RND LIQ T CON LIQ DP TUBE DP HEADER PROB HEADER
 0.6469724E 02 0.8176351E-01 0.1800000E 02 0.2000000E 01 0.9855100E 00
 ETA FIN S/D DBW TUBE I D HEADER AV I D
 0.8300013E 00 0.5000000E 01 0.2520400E 00 0.1469182E 00 0.1505660E 00
 NO TUBES AV T LIQ AV T FIN BASE TUBE THK CALC TUBE THK
 0.3000000E 01 0.5458896E 03 0.5846340E 03 0.1233807E 00 0.1233807E 00
 HEADER THK CALC HDER THK TUBE SSDE THK FIN THK CONNECT. LENG
 0.2000000E-01 0.1599760E-01 0.2000000E-01 0.2500039E-01 0.6311756E-01
 HEADER LENG TUBE LENG PROB IN PROB OUT PROB TUBE
 0.2314081E 01 0.4955823E 02 0.4968117E 00 0.9874100E 00 0.9842619E 00
 PARTICLE DIA TUBE AREA HEADER AREA TOT AREA FIN WT
 0.4839806E-01 0.4877453E 01 0.0 0.1148882E 03 0.3843437E 02
 TUBE WT TUBE LINER WT INVENT WT HEADER WT HEAD LINER WT
 0.9999016E 01 0.0 0.1178058E 01 0.6127809E-01 0.0
 TOT WT
 0.4467308E 02
 0.6057 1480. 218.1 4.980 0.7162E-03 0.2006E-02

IMC9001 EXECUTION TERMINATING DUE TO ERROR COUNT FOR ERROR NUMBER 217

IMC2171 FLOCS - END OF DATA SET ON UNIT 1

TRACEBACK ROUTINE	CALLER FROM ISN	REG. 14	REG. 15	REG. 0	REG. 1
INCOM		00370EE4	0037E140	00000000	00370860
MAIN		0000EEF2	01370488	00D6F720	0039EFF8

ENTRY POINT= 01370488

FIGURE-10 RADIATOR DESIGN, THREE TUBES

G-3. MULTIVARIABLE OPTIMIZATION TECHNIQUE

The general form for all optimization techniques is:

$$\underline{X} (n+1) = \underline{X} (n) - A(\underline{X})^{-1} f(\underline{X})$$

Where $\underline{X}$ is a vector of model parameters

$f(\underline{X})$ is the payoff function

$A(\underline{X})$ is a matrix determined by the system under study which carries the information the optimization technique requires to find a minima of the payoff.

There are many optimization methods available with the basic difference being the logic used to form the $A(\underline{X})$ matrix. The multivariable optimization technique used for most problems studied by Sundstrand uses a slope-curvature technique to search out a minimum by two modes. The first is a numerical evaluation of the first and second partial derivatives of the payoff function with respect to each of the parameters under study. The second search mode then uses the gradient vector defined by the first mode to vary the system parameters toward an optimum point. This process continues until no worthwhile improvement is possible.

References

- 1) Kays and London, Compact Heat Exchangers AliGraw Hill
- 2) "Analysis of Temperature Distribution and Radiant Heat Transfer Along a Rectangular fin of Constant Thickness" S. Lieblein, NASA TN-D-196
- 3) "A Penetration Criterion for Double Walled Structures Subject to Meteoroid Impact" J.P.D. Wilkinson Paper 68-1058, AIAA 5th Annual Meeting
- 4) "Results of Hypervelocity Impacts Into Space Radiator Materials " N. Clough & J. Diedrich AIAA Specialists Conference on Ranking Space Power Systems. NASA Lewis 1965

H. CONTROL SYSTEM DESIGN FOR A YAW
ROTA TOR GPS SPACECRAFT

INTRODUCTION

A candidate nuclear-powered GPS Space Vehicle spins about the earth pointing axis. This arrangement aids in making the effective solar pressure forces quite predictable because the vehicle responds to the average, over time, of the instantaneous solar pressure forces. These instantaneous solar pressure forces vary with vehicle solar aspect and vehicle surface characteristics. The effect of these variations is made more predictable with spin because the effective solar force on the vehicle due to the changing solar aspect becomes a function only of the angle between the sun-line and spin axis even for a vehicle whose surface is not a surface of revolution. In addition, those surface characteristics which are a function of temperature, are maintained nearly constant because the surface is approximately isothermal over many spins. Thus, the surface characteristics vary slowly over each orbit.

The nuclear power systems of interest contain (speed controlled) rotating machines which are assumed aligned with the vehicle spin axis. When the vehicle's and rotating machine's spin angular momenta are equal and oppositely directed, the vehicle responds to external torque as if it were non-rotating. Without the compensating effect of the rotating machine's angular momentum, there would be either a need for continual momentum unload as the vehicle moves in orbit, or an alternate compensation through on-board control rotating devices. Thus, the inherent angular momentum of the nuclear power system may be employed advantageously by eliminating or reducing the need for counter-rotating devices in the proposed operating mode.

The purpose of this study is to verify that relatively simple, lightweight control systems can be designed suitable for the required use--0.5° earth pointing accuracy with spin rate nominally fixed with a basic value in the range of 0.2 to 2 rpm.

REQUIRED MOMENTUM STORAGE

Solar pressure torques cause angular momentum accumulation in the vehicle. The torques are functions of vehicle solar aspect and surface properties. Appendix H-B contains an analysis of the effect of these torques in causing the angular momentum buildup which requires storage in the vehicle control torques.

The satellite is assumed to be 6 ft in diameter and 5 ft long. A 3-ft black "hole" in the skin is made available for the orbit insertion engine. The vehicle cg is taken as displaced 3 inches (an estimated maximum value) along the axis of symmetry from the geometric center. The skin is painted white. Under these conditions Fig. H-B-4 indicates that the required storage capability is approximately 0.025 ft lb sec times the number of orbits desired between momentum unloads. Fig. H-B-4 also indicates that the hole in the skin causes only a negligible portion of the total angular momentum storage requirement. The maximum storage requirement occurs when the orbit plane is inclined about 45 degrees from the sun-line.

In this study a 10-day interval between unloads will be posited so that the required storage is 0.5 ft lb sec. The actual implementation would include at least a 50% contingency resulting in a 0.75 ft lb sec on-board capability.

(In the event that the nuclear engine combined rotating unit (CRU) is not along the vehicle axis, as postulated in 2.0, an increment must be made to the required storage of magnitude $2r\&Q$.)

H-3 ON-ORBIT CONTROL SYSTEM DESCRIPTION

The approach taken in the normal operation control of the satellite, which spins about the satellite-earth line, is to control the vehicle attitude about axes normal to that line and to allow the vehicle spin rate to be uncontrolled. Variations in spin rate will occur primarily as the vehicle accumulates angular momentum due to external torques and secondarily during vehicle attitude transients. Periodic unloading of the accumulated angular momentum to reestablish the desired spin rate is accomplished by jet thrusting; spin rate information indicating the unload requirement is available from body-mounted sun sensors when the vehicle spin axis is normal to the vehicle-sun-line, i.e., twice each orbit.

Thus, continuous control is only required about the two axes normal to the spin axis. Body-fixed horizon sensors are used to measure the angle "components" about the vehicle X and Y axes, Z, and the local vertical. Derivatives of the horizon sensor outputs may be used to stabilize the control loops. Four different non-redundant torquer configurations may be considered (see Fig. H-1).

- a. Two reaction wheels may be used. One wheel torques the vehicle about its X axis; the other about its Y axis.
- b. Two single-gimballed control moment gyros may be used. With one gimbal axis along the vehicle X axis, the other along the vehicle Y axis and spin of both wheels nominally along the vehicle Z axis, the units act very much like high torque reaction wheels.
- c. One double-gimballed reaction wheel may be used. One gimbal axis is along the vehicle X axis, the other along the vehicle Y axis.
- d. One single-gimballed reaction wheel may be used. The wheel spin axis is normal to the vehicle spin axis, and the gimbal axis parallel to the vehicle spin axis. Along the wheel axis the unit acts as a reaction wheel; normal to the wheel axis the unit acts as a control moment gyro.

Variable wheel speeds in 2 and 3 above could also be considered.

In all configurations the addition of a single like unit, reaction wheel or CMG, will provide redundancy without performance degradation.

H-3.1 Control Loop

The vehicle-fixed earth sensors scan in the vehicle XZ and vehicle YZ planes to provide vehicle angular positions (θ , ϕ) about Y (pitch) and X (roll) axes. Because

the vehicle is rotating about its Z axis sequential θ or ϕ measurements are not about the same inertial axes. Thus, derivatives of ϕ and θ are not really inertial rates but are only rates of change of earth sensor outputs.

Over small fractions of a vehicle spin cycle the outputs θ , $\dot{\theta}$, ϕ , $\dot{\phi}$ are approximately the same as would be obtained if the vehicle were non-rotating. This suggests the use of the "standard" and simple control law:

$$\begin{aligned} T_{cx} &= \text{X axis control torque} = -K_1 \dot{\phi} - K_2 \phi \\ T_{cy} &= \text{Y axis control torque} = -K_1 \dot{\theta} - K_2 \theta \end{aligned}$$

where K_1 , K_2 are constant.

Because the vehicle is rotating in yaw, there is an $\bar{\omega}_s \times \bar{H}$ torque generated due to the stored angular momentum in the torquer configuration, i.e., stored X (or Y) angular momentum H_x (or H_y) results in Y (or X) axis torques. These torques are predictable within the uncertainty in the yaw spin rate ω_s , and compensating terms can be added into the basic control laws to reduce the torque effects:

$$\begin{aligned} T_{cx} &= \text{X axis control torque} = -K_1 \dot{\phi} - K_2 \phi - \omega_s H_y \\ T_{cy} &= \text{Y axis control torque} = -K_1 \dot{\theta} - K_2 \theta + \omega_s H_x \end{aligned}$$

Whether the compensating terms are included or not in the control laws, they define a lower bound on the required torquer output capability as a function of spin speed and desired momentum storage capability.

To retain the similarity of the control loop for this rotating vehicle to the control loop for a non-rotating vehicle, the bandwidth of the loop should be made high compared to the spin rate, i.e.

$$\sqrt{\frac{K_2}{I}} \gg \omega_s$$

H-3.2 Commands for Singly Redundant Torquer Configurations

H-3.2.1 Reaction Wheels

For wheels mounted as shown in Figure H-1a, the wheel motor torque commanded on the individual wheels are:

$$\begin{aligned} \text{Motor torque on Wheel 1} &= -T_{ex} \\ \text{Motor torque on Wheel 2} &= -T_{cy} \\ \text{Motor torque on Wheel 3} &= 0 \text{ (wheel power OFF)} \end{aligned}$$

In the event of failure of Wheel 1, it is turned off and the wheel torque commands are:

$$\text{Motor Torque on Wheel 2} = -T_{cy} + T_{cx} \frac{.707}{.707}$$

$$\text{Motor torque on Wheel 3} = \frac{-T_{cx}}{0.707}$$

In the event of failure of Wheel 2, it is turned off and the wheel torque commands are:

$$\text{Motor torque on Wheel 1} = -T_{cx} + T_{cy} \frac{.707}{.707}$$

$$\text{Motor torque on Wheel 3} = \frac{-T_{cy}}{.707}$$

When two wheels fail, the system has failed.

H-3.2.2 Single Gimballed Control Moment Gyros

For gyros mounted as shown in Fig. H -1b, the commanded gimbal rates are:

$$\text{Gimbal rate of gyro 1} = \frac{T_{cy} / \cos \gamma_1}{H}$$

$$\text{Gimbal rate of gyro 2} = \frac{-T_{cx} / \cos \gamma_2}{H}$$

$$\text{Gimbal rate of gyro 3} = 0 \text{ (gyro power OFF)}$$

where γ_i is gimbal displacement of unit i .

In the event of failure of gyro 1, it is turned off and the commanded gimbal rates are:

$$\text{Gimbal rate of gyro 2} = \frac{(T_{cx} - T_{cy} / \cos \gamma_3) / \cos \gamma_2}{.707H}$$

$$\text{Gimbal rate of gyro 3} = \frac{T_{cy}}{.707H} / \cos \gamma_3$$

In the event of failure of gyro 2, it is turned off and the commanded gimbal rates are:

$$\text{Gimbal rate of gyro 1} = \left(\frac{T_{cy}}{H} + \frac{T_{cx}}{.707H} / \cos \gamma_3 \right) / \cos \gamma_1$$

$$\text{Gimbal rate of gyro 3} = \frac{-T_{cx}}{.707H} / \cos \gamma_3$$

When two gyros have failed, the system has failed.

H-3.2.3 Gimballed Reaction Wheel

For the unit orientation shown in Fig. H-1c the commanded gimbal rates and motor torques are:

$$\begin{aligned} \text{Motor torque} &= \begin{bmatrix} T_{cx} \cos \gamma + T_{cy} \sin \gamma \end{bmatrix} \\ \text{Gimbal rate} &= \frac{-1}{H} \begin{bmatrix} -T_{cx} \sin \gamma + T_{cy} \cos \gamma \end{bmatrix} \end{aligned}$$

In the event of failure of the operating unit its non-operating redundant equal is activated. It uses the same commands.

H-3.2.4 Double-Gimballed CMG

For the unit orientation shown in Fig. H-1d, the commanded gimbal rates are:

$$\text{Outer gimbal rate command} = \frac{T_{cy} - T_{cx} \sin \gamma_1 \tan \gamma_1}{H \cos \gamma_i \cos \gamma_o}$$

$$\text{Inner gimbal rate command} = \frac{-T_{cx}}{H \cos \gamma_i}$$

In the event of failure of the operating unit, its non-operating redundant equal is activated. It uses the same commands.

H-3.3 System Performance

With the intent of verifying the basic control concept, a series of tests of loop stability and performance were made, by digital simulation, for two of the torquer configurations--the reaction wheel set and the gimballed reaction wheel. (The two CMG sets act dynamically very much like reaction wheels, of either high or low torque capability, and their performance can be estimated from the performance of the reaction wheels.)

The simulation is documented in Appendix H-C.

Parameters used in the simulation were:

Vehicle spin inertia I_z	100 slug-ft ²
Vehicle lateral inertia I_x, I_y	90 slug-ft ²
Vehicle spin rate W_z	0.1 rad/sec
Vehicle orbital period	12 hours
Control System natural freq. ω_n	1 rad/sec
$K_2 = \omega_n^2 I$	90 ft lb/rad
Control System damping ratio ζ	1
$K_1 = 2\zeta\omega_n I$	180 ft-lb/rad/sec
Reaction Wheel torque capability	20 in-oz
Reaction Wheel Maximum H	0.75 ft-lb-sec
Gimballed wheel torque capability	20 in-oz
Gimballed wheel maximum H	0.75 ft-lb-sec
Gimballed wheel gimbal rate (max)	1 rad/sec

Loop stability and error closure toward zero was verified with both torquer configurations using initial attitude errors. However, no attempt was made at optimizing nor has a wide variety of possible situations been treated. Typical results are given in Fig. H-2.

H-4.0 TORQUER DISCUSSION

Provided that the associated size, weight, power costs are acceptable, reaction wheels would undoubtedly be the preferred storage devices/torquers. This results from the success experienced in their on-orbit operations. In most of the on-orbit experience the wheels have been operated in quasi-steady speed, low torque (hence low power) modes. Application of this experience to the present use implies only low power operation of the wheels. This may be the determining factor in selection or non-selection of wheels as control system torquers.

As noted in H-3.1 the wheel torque must be at least $\omega_s H$. If a maximum 2 rpm vehicle spin rate is assumed, and H is taken as 0.75 ft lb sec, the wheel torque requirement T_w is given by:

$$\begin{aligned}
 T_w &= \omega_s H = 2 \left(\frac{2\pi}{60} \right) 0.75 (192) \text{ in-oz} \\
 &= 30 \text{ in-oz}
 \end{aligned}$$

H-2
(1 of 3)

Fig. H-2
(2 of 3)

The resulting wheel motor input power P_w requirements are given by the product of torque T_w , speed ω_w , and reciprocal of efficiency η_w . An efficiency of 50% may be assumed. Also, the wheel H is the product of wheel speed and spin moment of inertia I_w .

$$P_w = T_w \omega_w / \eta_w$$

Maximum power is therefore given by:

$$\begin{aligned} P_w \text{ max} &= 2 (\omega_s H) \omega_w \\ &= 2 (\omega_s H^2 / I_w) \end{aligned}$$

Spin rate ω_s is primarily a function of the angular momentum of the nuclear engine's rotating machine, the wheel angular momentum H depends upon the storage requirement, and the wheel speed ω_w (or equivalently its moment of inertia I_w) depend upon its physical size. The latter is the only element at the designer's option unless the spin rate is adjusted by the use of auxiliary Z axis wheels).

If an Organic Rankine engine with $H_{or} = 17.6 \text{ ft lb sec}$ is assumed for a 100 slug ft^2 vehicle, then:

$$\omega_w = 49 (P \text{ watts}) \text{ rpm}$$

With 10 watts estimated as allowable power, the maximum allowable wheel speed is 490 rpm and the approximate wheel weight is 11 pounds.

The gimballed reaction wheel does not have the torquing requirements of the reaction wheel system because momentum transfer from axis to axis during spin is accomplished by gimbal rotation relative to the vehicle. Thus, higher wheel speeds may be used. The wheel assembly weight would be about 6 pounds and an additional 4 lbs might be expected for the gimbal, its torquer, position sensor, its cover, etc. A singly redundant reaction wheel configuration would therefore weigh about ten pounds more and require about 10 watts more than a singly redundant gimballed reaction wheel system.

There appears to be small advantage in using CMGs as actuators when the Organic Rankine engine, with its large angular momentum, is used. Some improvement in transient response would be achieved due to the large torque output of these units. If a Brayton engine, with its rotating machine's angular momentum of 0.68 ft lb sec is used, the angular momentum of the CMGs can be profitably used to set the vehicle spin rate. Thus, with CMG $H = 1 \text{ ft lb sec}$, one can achieve a spin rate of 0.25 rpm when 2 single gimballed CMGs are used and a spin rate of 0.16 rpm when one double gimballed CMG is used.

H-5.0 CONCLUSION

A simple, lightweight control system for an earth-pointing rotating GPS satellite has been defined and its stability and error closure toward zero verified. Of all the control loop elements, only the torquer differs from that in the existing GPS satellites. Fixed or gimballed reaction wheel configurations are suitable, the former being preferred if the costs due to its higher weight and power are acceptable.

APPENDIX H-B

CALCULATION OF ANGULAR MOMENTUM ACCUMULATION DUE TO SOLAR PRESSURE

H-B.1.0 SOLAR PRESSURE TORQUES - NOMINAL

A simplified model of the vehicle configuration was developed for use in assessing the level of cyclic and bias solar torques and resulting angular momentum accumulation during earth orbit. The model consists of flat and cylindrical surfaces equivalent in area to those of the proposed configuration, detailed in Fig. H-B-1, where surface properties are described.

The equations used have been derived based on the usual photon momentum transfer theory and the appropriate geometric treatment of surfaces exposed to solar flux. Normal (F_n) and tangential (F_t) components of force are defined for flat and cylindrical surfaces as follows:

Flat Plate

where:

$$F_n = \left[(1 + s\rho) \cos^2 \psi + \frac{2}{3} \rho (1 - s) \cos \psi \right] P_f A_p$$

$$F_t = \left[(1 - s\rho) \cos \psi \sin \psi \right] P_f A_p$$

$$P_f = \text{solar pressure, } 1 \times 10^{-7} \text{ lb/ft}^2$$

$$A_p = \text{plate area, ft}^2$$

$$L = 5 \text{ ft}$$

$$D = 6 \text{ ft}$$

$$D_c = 0 \text{ or } 4 \text{ ft}$$

$$\delta = 3 \text{ inches}$$

$$\text{vehicle painted white all over} \quad s = 0, \rho = 0.75$$

$$\text{cutout for OIS} \quad s = 0, \rho = 0.1$$

(s = specularity, ρ = reflectivity)

Fig. H-B-1 Simplified Vehicle Model

Cylinder

where
$$F_n = \left[(1 + s\rho/3) \cos^2 \phi + \rho(1-s) \frac{\pi}{6} \cos \phi \right] P_{\text{f}} A_c$$
$$F_t = \left[(1 - s\rho) \cos \phi \sin \phi \right] P_{\text{f}} A_c$$
$$A_c = 2 r l$$

and where

ρ = fraction of incident radiation reflected

s = fraction of reflect radiation which is specular

H-B.2 CALCULATION OF TORQUES DUE TO SOLAR PRESSURE

The vehicle moves in an orbit inclined angle i to the sun-line. It is located angle Ω from the ascending node with respect to an inertial reference system X_s, Y_s, Z_s , where Z_s looks at the sun, and Y_s points along the ascending line-of-nodes. The vehicle, s with coordinate system X_v, Y_v, Z_v , has its $-Y_v$ axis pointing to the earth and X_v pointing along the orbit axis X_{OA} . The orbit axis system has its y_{OA} axis along the inertial Y_s axis. The geometry is illustrated in Fig. B-2.

Transformation relating vectors V_v, V_s, V_{OA} in the vehicle, sun inertial, and orbit axis system are given below:

Unit vectors along coordinate axes
in space i : $\bar{e}_i, \bar{f}_i, \bar{k}_i$

Vehicle in-orbit geometry

fig H-B-2

$$(V_s) = \begin{bmatrix} c i & 0 & s i \\ 0 & 1 & 0 \\ -s i & 0 & c i \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & c \Omega & -s \Omega \\ 0 & s \Omega & c \Omega \end{bmatrix} (V_v)$$

$$(V_{0A}) = \begin{bmatrix} 1 & 0 & 0 \\ 0 & c \Omega & -s \Omega \\ 0 & s \Omega & c \Omega \end{bmatrix} (V_v)$$

where $c = \cos$ $s = \sin$

Thus:

$$(V_s) = \begin{bmatrix} c i & s i s \Omega & s i c \Omega \\ 0 & c \Omega & -s \Omega \\ -s i & c i s \Omega & c i c \Omega \end{bmatrix} (V_v)$$

$$(V_v) = \begin{bmatrix} c i & 0 & -s i \\ s i s \Omega & c \Omega & c i s \Omega \\ s i c \Omega & -s \Omega & c i c \Omega \end{bmatrix} (V_s)$$

The normal component of solar force on the cylindrical portion of the vehicle ($\bar{F}_{nc}$) lies normal to the vehicle Y_v axis and in the $Y_v Z_v$ plane. It has a component away from the sun. Thus, the unit vector $\bar{e}_{nc}$ along $\bar{F}_{nc}$ is given by:

$$\begin{aligned}\bar{e}_{nc} &= \frac{\bar{Y}_v \times (\bar{Y}_v \times \bar{E}_s)}{|\bar{Y}_v \times (\bar{Y}_v \times \bar{E}_s)|} \\ &= \frac{si \bar{Y}_v - ci c \Omega \bar{K}_v}{(si^2 + (ci c \Omega)^2)^{1/2}}\end{aligned}$$

The tangential component of solar force on the cylindrical portion of the vehicle ($\bar{F}_{tc}$) is directed along the $-Y_v$ axis when $0 \leq \Omega < \pi$ and along the $+Y_v$ axis when $\pi \leq \Omega \leq 2\pi$

$$\bar{e}_{nt} = \begin{cases} -\bar{Y}_v & \text{for } 0 \leq \Omega < \pi \\ +\bar{Y}_v & \text{for } \pi \leq \Omega < 2\pi \end{cases}$$

The normal component of solar force on the flat faces of the vehicle ($\bar{F}_{nf}$) lies along the vehicle Y_v axis. (It causes no vehicle torques because the vehicle c.g. is assumed on the Y_v axis.) The tangential component of solar force on the flat faces of the vehicle is directed parallel to $\bar{e}_{nc}$.

The moment arm vector for $\bar{F}_{tc}$ is directed opposite the $\bar{e}_{nc}$ vector and has half the diameter of the vehicle as magnitude. Due to the vehicle center-of-mass displacement (δ) along the $+Y_v$ axis, from the vehicle geometric center, the moment arm vector for $\bar{F}_{nc}$ is $-\delta \bar{Y}_v$. The moment arm vector for $\bar{F}_{tf}$ on the $+Y_v$ face is $(L/2 - \delta) \bar{Y}_v$. For $\bar{F}_{tf}$ on the $-Y_v$ face, the moment arm vector is $-(L/2 + \delta) \bar{Y}_v$.

Total torque on the vehicle is given as the sum of individual moment arm vectors crossed with their force vectors:

$$\begin{aligned}\bar{T}_{total} &= \frac{D}{2} (-\bar{e}_{nc}) \times |\bar{F}_{tc}| \bar{Y}_v M \\ &\quad - \delta \bar{Y}_v \times |\bar{F}_{nc}| \bar{e}_{nc} \\ &\quad + \left(\frac{L}{2} + \delta M\right) \bar{Y}_v \times |\bar{F}_{tf}| \bar{e}_{nc} (-M)\end{aligned}$$

where: $M = -1$ for $0 \leq \Omega < \pi$
 $M = 1$ for $\pi \leq \Omega < 2\pi$

Evaluation of the force magnitudes requires evaluation of the acute angles between the sun-line and the flat surface normal: $\phi' = \cos^{-1} |\bar{r}_v \cdot \bar{e}_s|$

and the acute angle between the sun line and the cylinder normal: $\phi = \frac{\pi}{2} - \phi'$

Accumulation of angular momentum in inertial space is calculated by taking the components of torques about vehicle axes, resolving them into the orbital axis system and integrating. The total accumulation is the sum of squares of the individual accumulations.

The above equations have been programmed for digital computer simulation, with the program documented in Figure H-B-3, and results for various orbit inclinations i are given in Figure H-B-4.

```

100 REM THIS PROGRAM IS ON FILE 13
110 INIT
120 DIM F(3,3)
130 D0=0
150 H6=0
160 H7=0
170 H8=0
180 PRINT "      CHANGE OVER ONE ORBIT OF ANGULAR MOMENTUM (FT-LB-SEC)"
190 PRINT "      DUE TO SOLAR TORQUES "
200 PRINT
210 PRINT "      OIS ENGINE OPENING RADIUS (FT)= ";
211 INPUT R0
212 PRINT "      CENTER OF MASS DISPLACEMENT (IN)= ";
213 INPUT D
214 D=D/12
225 I=0
226 C=0.05
230 WINDOW 0,2*PI,-C,C
240 AXIS P1/4,C/10
245 MOVE 0,C
246 PRINT C
247 MOVE 0,C/2
248 PRINT C/2
249 MOVE 0,0
250 O1=2*PI/72
260 H1=0
270 H2=0
280 H3=0
300 O=1.0E-3
310 P=1.0E-7
340 T=123600*O1/(2*PI)
350 L=5
360 R=3
370 S1=0

```

```

700 R2=R3
710 H=-1
720 GO TO 500
730 A2=A4
740 S2=S4
750 R2=R4
760 GO TO 500
770 D1=(L/2-D*M)/D0
780 X1=R1*COS(I)*COS(O)/D0*F(1,2)*H
800 X2=D1*COS(I)*COS(O)/D0*F(1,1)
830 X3=D1*COS(I)*COS(O)*F(2,2)*H
835 Z1=R1*SIN(I)*H*F(1,2)/D0
840 Z2=D1*SIN(I)/D0*F(1,1)
850 Z3=D1*SIN(I)*F(2,2)*H
860 X=X1+X2-X3
870 Z=Z1+Z2-Z3
880 T1=X
890 T2=-SIN(O)*Z
900 T3=COS(O)*Z
920 H1=H1+T1*T
930 H2=H2+T2*T
940 H3=H3+T3*T
950 H=SQR(H1*2+H2*2+H3*2)
960 O=O+O1
1030 DRAW 0,H
1110 IF O>2*PI THEN 1130
1120 GO TO 310
1130 T=1+PI/12
1140 IF T>PI/2 THEN 1160
1150 GO TO 249
1160 END

```

```

380 R1=0.75
390 A1=2*R*L
400 S2=0
410 R2=0.75
420 A2=P1*R12-P1*R012
430 S3=0
440 R3=0.75
450 A3=P1*R12
460 S4=0
470 R4=0.1
480 A4=P1*R012
490 D0=SQR(SIN(I)*2+(COS(I)*COS(O))*2)
491 IF D0<>0 THEN 500
492 D0=1.0E-10
500 P1=PI/2-ACS(ABS(COS(I)*SIN(O)))
510 F(1,1)=F*A1*((1+S1*R1/3)*COS(P1)*2+R1*(1-S1)*PI/6*COS(P1))
520 F(1,2)=P*A1*(1-S1*R1)*COS(P1)*SIN(P1)
530 F1=0
540 F2=0
550 H=0
560 M=1
570 IF O>PI THEN 600
580 F2=ACS(ABS(COS(I)*SIN(O)))
590 F(2,1)=P*A2*((1+S2*R2)*COS(P2)*2+2/3*R2*(1-S2)*COS(P2))
600 F(2,2)=P*A2*(1-S2*R2)*COS(P2)*SIN(P2)
610 IF O>PI THEN 770
620 F(2,1)=F1+F(2,1)
630 F(2,2)=F2+F(2,2)
640 F1=F(2,1)
650 F2=F(2,2)
660 H=H+1
670 IF H>2 THEN 770
675 GO TO 730
680 A2=A3
690 S2=S3

```


Figure H-4

Figure H-B-4

APPENDIX H-C

CONTROLLED VEHICLE DYNAMIC SIMULATION

The computer program is documented in Figure H-1.

H-C.1.0

COORDINATE FRAME DEFINITIONS AND TRANSFORMATIONS

The earth centered inertial system (I) has its X_I axis pointing toward the first point of Aries, γ , and its Z_I axis along the vehicle orbit axis.

The reference system (R) with origin at the spacecraft center of mass has its Z_R axis point to the earth center below, and its X_R axis in the direction of motion. The vehicle is located angle γ from γ measured about the orbit axis.

The rotated reference system (RR) has its Z_{RR} axis along the Z_R axis and is rotated γ about the Z_{RR} axis.

The vehicle body axes (B) are nominally along the rotated reference axes but are displaced by rotation ϕ about X_{RR} , then angle θ about the rotated Y_{RR} axis.

Based on these definitions, the following transformations apply:

$$(V_R) = \begin{bmatrix} -s\gamma & c\gamma & 0 \\ 0 & 0 & -1 \\ -c\gamma & -s\gamma & 0 \end{bmatrix} (V_I) = E(V_I) \quad (C-1)$$

where: (V_I) = vector V components in coordinate system I

s = sin

c = cos

$$(V_{RR}) = \begin{bmatrix} c\gamma & s\gamma & 0 \\ -s\gamma & c\gamma & 0 \\ 0 & 0 & 1 \end{bmatrix} (V_R) = F(V_R) \quad (C-2)$$

$$\text{Let: } (V_B) = A(V_I) \quad (C-3)$$

$$\text{Then: } (V_B) = AE^T F^T (V_{RR}) \quad (C-4)$$

$$\text{also: } (V_{RR}) = \begin{bmatrix} c\theta & 0 & s\theta \\ s\phi s\theta & c\theta & -s\phi c\theta \\ -c\phi s\theta & s\phi & c\phi c\theta \end{bmatrix} (V_B) = FEA^T(V_B) \quad (C-5)$$

H-C.2.0 DYNAMICS

The total spacecraft momentum vector, $\bar{H}$, is comprised of on-board wheel momentum and momentum due to spacecraft rates:

$$\bar{H} = [I] \bar{\omega} + \bar{h}$$

where $[I]$ is spacecraft inertia tensor expressed in body coordinates
 $\bar{\omega}$ is spacecraft angular rate expressed in body coordinates
 $\bar{h}$ is total on-board wheel momentum expressed in body coordinates

$$[I] = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{xz} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix}$$

$$\bar{\omega} = \begin{bmatrix} \omega_x & \omega_y & \omega_z \end{bmatrix}^T$$

$$\bar{h} = \begin{bmatrix} h_x & h_y & h_z \end{bmatrix}^T$$

From Euler's equation of motion, the external torque on the spacecraft causes a rate of change of total momentum:

$$\begin{aligned} \bar{T} &= \frac{d}{dt} (\bar{H}) \\ &= [I] \dot{\bar{\omega}} + \dot{\bar{h}} + \bar{\omega} \times ([I] \bar{\omega}) + \bar{\omega} \times \bar{h} \end{aligned} \quad (C-6)$$

Solving for the derivative of inertial rate:

$$\dot{\bar{\omega}} = [I]^{-1} \left[\bar{T} - \dot{\bar{h}} - \bar{\omega} \times ([I] \bar{\omega}) - \bar{\omega} \times \bar{h} \right] \quad (C-7)$$

H-C-3.0 KINEMATICS

Euler symmetric parameters are used as the Kinematic variables.

Thus:

$$\begin{bmatrix} \dot{\xi} \\ \dot{\eta} \\ \dot{\zeta} \\ \dot{\chi} \end{bmatrix} = \frac{1}{2} \begin{bmatrix} 0 & \omega_3 & -\omega_2 \\ -\omega_3 & 0 & \omega_1 \\ \omega_2 & -\omega_1 & 0 \\ -\omega_1 & -\omega_2 & -\omega_3 \end{bmatrix} \begin{bmatrix} \xi \\ \eta \\ \zeta \\ \chi \end{bmatrix} \quad (C-8)$$

The direction cosine values a_{ij} from equations C-1, C-4 are used for initialization with $\gamma = 0$.

$$\begin{aligned}\Phi &= \cos^{-1} \frac{1}{2} (a_{11} + a_{22} + a_{33} - 1) \\ \xi &= (a_{23} - a_{32}) \frac{\sin(\Phi/2)}{2 \sin \Phi} \\ \eta &= (a_{31} - a_{13}) \frac{\sin(\Phi/2)}{2 \sin \Phi} \\ \zeta &= (a_{12} - a_{21}) \frac{\sin(\Phi/2)}{2 \sin \Phi} \\ \chi &= \cos(\Phi/2)\end{aligned}\tag{C-9}$$

During the run the direction cosines are obtained from the symmetric parameters by:

$$\begin{aligned}a_{11} &= \xi^2 - \eta^2 - \zeta^2 + \chi^2 \\ a_{21} &= 2(\xi\eta - \zeta\chi) \\ a_{31} &= 2(\xi\zeta + \eta\chi) \\ a_{12} &= 2(\xi\eta + \zeta\chi) \\ a_{12} &= -\xi^2 + \eta^2 - \zeta^2 + \chi^2 \\ a_{32} &= 2(\eta\zeta - \xi\chi) \\ a_{13} &= 2(\xi\zeta - \eta\chi) \\ a_{23} &= 2(\eta\zeta + \xi\chi) \\ a_{33} &= -\xi^2 - \eta^2 + \zeta^2 + \chi^2\end{aligned}\tag{C-10}$$

H-C.4.0 CONTROL LOOP

H-C.4.1 Sensor

The earth sensor is assumed perfect. Assuming small attitude errors its outputs are angles ϕ , θ relative to the RR system. Thus, from C-5, C-2:

$$\begin{aligned}\phi &= \sin^{-1} FEA^T(3,2) = \sin^{-1} EA^T(3,2) \\ \theta &= -\sin^{-1} FEA^T(3,1) = -\sin^{-1} EA^T(3,1)\end{aligned}\tag{C-11}$$

Attitude output rates $\dot{\phi}, \dot{\theta}$ are due to motion of the body relative to the RR system. They are given by:

$$\begin{aligned}\dot{\phi} &= \omega_1 + EA^T(2,1) \omega_o - EA^T(3,1) \dot{\theta} \\ \dot{\theta} &= \omega_2 + EA^T(2,2) \omega_o - EA^T(3,2) \dot{\phi}\end{aligned}\quad (C-12)$$

where ω_o is orbital rate.

This follows from the equation:

$$\bar{\omega}_{B/RR} + \bar{\omega}_{RR/R} + \bar{\omega}_{R/I} = \bar{\omega}_{B/I} \quad (C-13)$$

where $\bar{\omega}_{i/j}$ is angular rate of i with respect to j .

$$\therefore \bar{\omega}_{B/RR} = \bar{\omega}_{B/I} - \bar{\omega}_{R/I} - \bar{\omega}_{RR/R} \quad (C-14)$$

The components of these vectors in the base space are given by (C-12).

H-C.4.2 Actuators

Only non-redundant configurations are modelled. The wheels are assumed to achieve their commanded torque perfectly (see H-3.3) with limits of torque and total angular momentum.

The gimballed reaction wheel achieves its commanded motor torque and gimbal rate (see H-3.2.3) perfectly within the wheel torque, wheel angular momentum and gimbal rate limits.

H-C.4.3 Control Law

The control law simulated is:

$$\begin{aligned}T_{cx} &= X_{axis} \text{ control torque} = -K_1 \phi - K_2 \dot{\phi} \\ T_{cy} &= Y_{axis} \text{ control torque} = -K_1 \theta - K_2 \dot{\theta}\end{aligned}\quad (C-15)$$

where: K_1, K_2 are constant

and $\sqrt{\frac{K_1}{I}} = 10 \times \text{spin rate.}$

Fig C-2

```

100 REM THIS FILE 11 CALLS ON FILE 7
110 INIT
120 REM W9(1) IS DERIVATIVE OF U9(1)
130 REM U9(1), U9(2), U9(3), U9(4) ARE EULER PARAMETERS
140 REM U9(5), U9(6), U9(7) ARE VEHICLE INERTIAL RATES ABOUT X,Y,Z, AXES
150 REM U9(8) IS VEHICLE ORBITAL RATE
160 REM W9(9), W9(10), W9(11) ARE X,Y,Z CONTROL TORQUES DUE TO WHEELS
170 REM A1, A2 ARE PHI, THETA VEHICLE DISPLACEMENTS
180 REM J1, J2, J3 ARE VEHICLE X,Y,Z MOM OF INERTIA
210 REM H = INTEGRATION STEP SIZE
220 REM H8 = MAXIMUM INTEGRATION STEP SIZE FOR CURRENT INTEGRATION STEP
230 REM H2 = INTEGRATION STEP CONTROL VARIABLE. WHEN H2=0 STEP
240 REM SIZE FIXED, WHEN H2 = .04 STEP SIZE AUTOMATICALLY ADJUSTED
250 REM P0 = INTERVAL BETWEEN OUTPUTS
260 REM MATRIX A: (A) (USUB INERTIAL) = (U SUB BASE)
270 REM MATRIX E: (E) (USUB INERTIAL) = (U SUB REFERENCE AXES)
280 REM MATRIX E1: (E1)=(E)(A TRANSPOSE)
290 REM MATRIX E1: (E1)(U SUB BASE) = (U SUB REFERENCE AXES)
300 REM P,Q, ARE VEHICLE MEASURED RATES ABOUT X,Y AXES USING ESA
310 REM THE NEGATIVE OF THE INTEGRAL OF CONTROL TORQUES IS EQUAL TO
320 REM WHEEL ANGULAR MOMENTUM INCREMENT
330 REM T9 IS END TIME
340 REM .....
350 REM .....
360 DIM W1(11), W9(11), U(11), U9(11), W2(11)
370 DIM A(3,3), E(3,3), E1(3,3)
380 READ T,H,H2,J1,J2,J3,T7
390 DATA 0,0.1,0.04,90,90,100,0.75
400 H0=H
401 K1=1
410 H0=H
420 E=0
430 U9=0
440 W9=0

```

```

716 U9(9)=-U9(9)
717 U9(10)=U9(10)/57.3
718 GO TO 740
720 FIND 7
721 PRINT " INITIAL X WHEEL H ( FT-LB-SEC) = ";
722 INPUT U9(9)
723 U9(9)=-U9(9)
724 PRINT " INITIAL Y WHEEL H ( FT-LB-SEC) = ";
725 INPUT U9(10)
726 U9(10)=-U9(10)
730 GO TO 750
740 FIND 12
750 WINDOW T,T9,-S,S
760 AXIS T9/10,S/2
770 MOVE T,A1
775 DELETE A$
780 DELETE 100,780
790 APPEND 800
800 REM

```

```

460 U9(7)=0.1
480 U9(11)=+J3*U9(7)
490 W9(8)=2*PI/(12*3600)
500 PRINT "WHEELS (W) OR CMG (C) ?"
510 INPUT A$
520 PAGE
521 IF A$="C" THEN 524
522 PRINT " FIXED REACTION WHEEL SYSTEM"
523 GO TO 525
524 PRINT " GIMBALLED REACTION WHEEL SYSTEM"
525 PRINT
530 PRINT " FULL SCALE TIME (SEC) = ";
540 INPUT T9
550 O9=10*U9(7)
560 Z=1
570 T9=20
580 PRINT " INITIAL PHI (RAD) = ";
590 INPUT A1
600 PRINT " INITIAL THETA (RAD) = ";
610 INPUT A2
620 T4=T
630 W0=A2
640 S=ACS(0.5*(SIN(A1+A2)-1))
650 A3=SIN(S/2)/(2*SIN(S))
660 U9(1)=(-COS(A1)-SIN(A2))*A3
670 U9(2)=-COS(A1+A2)*A3
680 U9(3)=(COS(A2)+SIN(A1))*A3
690 U9(4)=COS(S/2)
700 S=(ABS(A1) MAX ABS(A2))*2
710 IF A$="C" THEN 712
711 GO TO 720
712 PRINT " INITIAL GIMBAL ANGLE (DEG) = ";
713 INPUT U9(10)
714 PRINT " INITIAL WHEEL H (FT-LB-SEC) = ";
715 INPUT U9(9)

```

Figure H-C-1
(2 of 3)

```

100 REM THIS IS ON FILE 12 AND IS USED WITH FILE 11
110 T0=T
120 H=H8
130 W1=W9
140 U=U9
150 REM
180 T=T0+0.5*H
190 FOR I=1 TO 11
200 U9(I)=U(I)+0.5*H*W1(I)
210 NEXT I
220 GOSUB 700
230 FOR I=1 TO 11
240 W2(I)=W9(I)
250 U9(I)=U(I)+0.5*H*W2(I)
260 NEXT I
270 GOSUB 700
280 H9=0
281 H=0
290 FOR J=1 TO 11
300 A1=W2(J)-W1(J)
310 IF A1=0 THEN 332
320 S=W9(J)-W2(J)
330 IF H9<(S/A1)*2 THEN 350
331 GO TO 340
332 H=H+1
333 GO TO 350
340 H9=(S/A1)*2
350 NEXT J
351 IF H=11 THEN 460
360 IF H9<H2*2 THEN 420
370 IF H2=0 THEN 460
380 IF H<0.2 THEN 460
390 H=0.5*H8
400 H8=H

```

```

410 GO TO 150
420 IF H9<(0.4*H2)*2 THEN 460
430 IF H<H8 THEN 460
440 H8=2*H8
460 T=T0+H
480 FOR I=1 TO 11
490 S=(W1(I)+W9(I))*H/6
500 U9(I)=U(I)+H*W9(I)
510 NEXT I
520 GOSUB 700
530 FOR I=1 TO 11
540 S=(W1(I)+W9(I))*H/6
550 U9(I)=U(I)+S
570 NEXT I
580 GOSUB 700
590 GO TO 620
610 END
620 IF T>T9 THEN 610
630 DRAW T,A1
640 MOVE T4,W0
650 DRAW T,A2
660 MOVE T,A1
670 W0=A2
680 T4=T
690 GO TO 110
700 REM
710 Q1=0.9*2*J1
720 Q2=2*2*0.9*J1
730 S=1-0.5*(U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11))
740 U9(1)=U9(1)*S
750 U9(2)=U9(2)*S
760 U9(3)=U9(3)*S
770 U9(4)=U9(4)*S
780 W9(1)=0.5*(U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20))
790 W9(2)=0.5*(-U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20))

```

```

800 W9(3)=0.5*(U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20))
810 W9(4)=0.5*(-U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20))
820 A(1,1)=U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
830 A(2,1)=2*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
840 A(3,1)=2*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
850 A(1,2)=2*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
860 A(2,2)=-1*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
870 A(3,2)=2*U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
880 A(1,3)=2*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
890 A(2,3)=2*U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
900 A(3,3)=-1*U9(1)+U9(2)+U9(3)+U9(4)+U9(5)+U9(6)+U9(7)+U9(8)+U9(9)+U9(10)+U9(11)+U9(12)+U9(13)+U9(14)+U9(15)+U9(16)+U9(17)+U9(18)+U9(19)+U9(20)
910 E(1,1)=-SIN(U9(8))
920 E(1,2)=COS(U9(8))
930 E(2,3)=-1
940 E(3,1)=-E(1,2)
950 E(3,2)=+E(1,1)
960 FOR I=1 TO 3
970 FOR J=1 TO 3
980 E1(I,J)=0
990 FOR K=1 TO 3
1000 S=E(I,K)*A(J,K)
1010 E1(I,J)=E1(I,J)+S
1020 NEXT K
1030 NEXT J
1040 NEXT I
1050 A1=ASH(E1(3,2))
1060 A2=ASH(E1(3,1))
1070 P=U9(5)+E1(2,1)*W9(8)-E1(3,1)*U9(7)
1080 Q=U9(6)+E1(2,2)*W9(8)-E1(3,2)*U9(7)
1090 C1=-(Q1*A1+Q2*P)
1100 C2=-(Q1*A2+Q2*Q)
1101 U9(9)=T7 MIN U9(9)
1102 U9(9)=-T7 MAX U9(9)
1106 IF U9(9)=0 THEN 1108
1107 GO TO 1110

```

```

1108 W9(10)=0
1109 GO TO 1140
1110 W9(10)=1/U9(9)*(-C1*SIN(U9(10))+C2*COS(U9(10)))
1120 W9(10)=R1 MIN W9(10)
1130 W9(10)=-R1 MAX W9(10)
1140 W9(9)=+(C1+COS(U9(10))+C2*SIN(U9(10)))
1150 W9(9)=T8/192 MIN W9(9)
1160 W9(9)=-T8/192 MAX W9(9)
1162 IF U9(9)=T7 THEN 1166
1164 GO TO 1167
1166 W9(9)=0 MIN W9(9)
1167 IF U9(9)=-T7 THEN 1169
1169 GO TO 1170
1169 W9(9)=0 MAX W9(9)
1170 C1=-U9(9)*COS(U9(10))
1180 C2=-U9(9)*SIN(U9(10))
1190 Q1=J1*U9(5)+C1
1200 Q2=J2*U9(6)+C2
1210 Q3=J3*U9(7)+U9(11)
1220 W9(5)=(U9(6)+Q3-U9(7)+Q2+W9(9)*COS(U9(10))+W9(10)*C2)/J1
1230 W9(6)=(U9(7)+Q1-U9(5)+Q3+W9(9)*SIN(U9(10))-W9(10)*C1)/J2
1240 W9(7)=(U9(5)+Q2-U9(6)+Q1+W9(11))/J3
1250 RETURN

```

Figure H-C-1
(3 of 3)

99 REM THIS IS ON FILE 7 AND IS USED WITH FILE 11

```
100 T8=T
101 H=H8
110 W1=W9
130 U=U9
150 REM
170 T=T0+0.5*H
180 FOR I=1 TO 11
190 U9(I)=U(I)+0.5*H*W1(I)
200 NEXT I
220 GOSUB 3000
230 FOR I=1 TO 11
240 W2(I)=W9(I)
250 U9(I)=U(I)+0.5*H*W2(I)
260 NEXT I
270 GOSUB 3000
280 H9=0
281 H=0
290 FOR J=1 TO 11
320 A1=W2(J)-W1(J)
330 IF A1=0 THEN 352
340 S=W9(J)-W2(J)
350 IF H9>(S/A1)*T2 THEN 390
351 GO TO 360
352 H=H+1
353 GO TO 390
360 H9=(S/A1)*T2
390 NEXT J
391 IF H=1 THEN 510
410 IF H9<H2*T2 THEN 470
420 IF H2=0 THEN 510
430 IF H<0.2 THEN 510
440 H=0.5*H8
450 H0=H
```

```
3260 W9(3)=0.5*(U9(6)+U9(1)-U9(5)+U9(2)+U9(7)+U9(4))
3270 W9(4)=0.5*(-U9(5)+U9(1)-U9(6)+U9(2)-U9(7)+U9(3))
3290 A(1,1)=U9(1)*T2-U9(2)*T2-U9(3)*T2+U9(4)*T2
3300 A(2,1)=2*(U9(1)+U9(2)-U9(3)+U9(4))
3310 A(3,1)=2*(U9(1)+U9(3)+U9(2)+U9(4))
3320 A(1,2)=2*(U9(1)+U9(2)+U9(3)+U9(4))
3330 A(2,2)=-1*(U9(1)+T2+U9(2)+T2-U9(3)+T2+U9(4)+T2
3340 A(3,2)=2*(U9(2)+U9(3)-U9(1)+U9(4))
3350 A(1,3)=2*(U9(1)+U9(3)-U9(2)+U9(4))
3360 A(2,3)=2*(U9(2)+U9(3)+U9(1)+U9(4))
3370 A(3,3)=-1*(U9(1)+T2-U9(2)+T2+U9(3)+T2+U9(4)+T2
3400 E(1,1)=-SIN(U9(8))
3410 E(1,2)=COS(U9(8))
3420 E(2,3)=-1
3430 E(3,1)=-E(1,2)
3440 E(3,2)=+E(1,1)
3450 FOR I=1 TO 3
3460 FOR J=1 TO 3
3470 E(I,J)=0
3480 FOR K=1 TO 3
3490 S=E(I,K)*A(J,K)
3500 E(I,J)=E(I,J)+S
3510 NEXT K
3520 NEXT J
3530 NEXT I
3550 A1=ASN(E(3,2))
3560 A2=-ASN(E(3,1))
3565 P=U9(5)+E(1,2)*W9(8)-E(3,1)*U9(7)
3566 Q=U9(6)+E(1,2)*W9(8)-E(3,2)*U9(7)
3580 W9(9)=-Q*(A1+Q2*P)
3590 W9(10)=-Q*(A2+Q2*Q)
3600 W9(11)=0
3601 W9(9)=T8/192 MIN W9(9)
3602 W9(9)=-T8/192 MAX W9(9)
3603 W9(10)=T8/192 MIN W9(10)
```

```
3604 W9(10)=-T8/192 MAX W9(10)
3605 U9(9)=T7 MIN U9(9)
3606 U9(9)=-T7 MAX U9(9)
3607 U9(10)=T7 MIN U9(10)
3608 U9(10)=-T7 MAX U9(10)
3610 IF U9(9)=T7 THEN 3614
3612 GO TO 3616
3614 W9(9)=0 MIN W9(9)
3616 IF U9(9)=-T7 THEN 3620
3618 GO TO 3622
3620 W9(9)=0 MAX W9(9)
3622 IF U9(10)=T7 THEN 3626
3624 GO TO 3628
3626 W9(10)=0 MIN W9(10)
3628 IF U9(10)=-T7 THEN 3632
3630 GO TO 3640
3632 W9(10)=0 MAX W9(10)
3640 Q1=J1+U9(5)-U9(9)
3650 Q2=J2+U9(6)-U9(10)
3660 Q3=J3+U9(7)-U9(11)
3680 W9(5)=(U9(6)+Q3-U9(7)+Q2+W9(9))/J1
3690 W9(6)=(U9(7)+Q1-U9(5)+Q3+W9(10))/J2
3700 W9(7)=(U9(5)+Q2-U9(6)+Q1+W9(11))/J3
3800 RETURN
```

```
460 GO TO 150
470 IF H9>(0.4*H2)*T2 THEN 510
480 IF H<H8 THEN 510
490 H8=2*H8
510 T=T0+H
530 FOR I=1 TO 11
540 S=(W1(I)+W9(I))*H/6
550 U9(I)=U(I)+H*W9(I)
560 NEXT I
570 GOSUB 3000
580 FOR I=1 TO 11
590 S=(W1(I)+W9(I))*H/6
600 U9(I)=U(I)+S
620 NEXT I
640 GOSUB 3000
655 GO TO 2000
670 END
2000 IF T>T9 THEN 670
2003 DRAW T,A1
2004 MOVE T4,W0
2005 DRAW T,A2
2006 MOVE T,A1
2007 W0=A2
2110 T4=T
2111 GO TO 100
3000 REM
3150 Q1=0912*J1
3160 Q2=2*2*09*J1
3180 S=1-0.5*(U9(1)+T2+U9(2)+T2+U9(3)+T2+U9(4)+T2-1)
3190 U9(1)=U9(1)*S
3200 U9(2)=U9(2)*S
3210 U9(3)=U9(3)*S
3220 U9(4)=U9(4)*S
3240 W9(1)=0.5*(U9(7)+U9(2)-U9(6)+U9(3)+U9(5)+U9(4))
3250 W9(2)=0.5*(-U9(7)+U9(1)+U9(5)+U9(3)+U9(6)+U9(4))
```


J. ESTIMATE OF POSITION RATE UNCERTAINTY
FOR THE REFERENCE NUCLEAR GPS SPACECRAFT

J-1.0 INTRODUCTION

The candidate nuclear-powered GPS Space Vehicle (SV) as shown in Fig. J-1, is a large squat cylinder, 6 ft in diameter by 6 ft long, with a set of antennae protruding from one cylinder base and a small vehicle launch support structure protruding from the other. Nominally, the antenna farm aligned with the cylinder axis looks down at the Earth and the vehicle spins at relatively fixed rate at about 0.5 to 1 rpm.

Differences between predicted and actual vehicle orbits result from imperfect estimation of the forces on the SV. Bias-type differences between expected and actual parameters, such as size and reflectivity, are expected to be determined by the filtering procedures employed, using measured orbital data over extended periods of time. But changes in vehicle forces which are random or with time variations more rapid than the filter time constants, will impede accurate prediction. The purpose of this memo is to consider such forces due to SV error attitude, temperature variation, and error unload impulse, and to evaluate the resultant orbit perturbations.

J-2.0 TEMPERATURE EFFECTS ON BODY FORCE

J-2.1 Evaluation of Normal Body Force due to Heat Transfer by Diffuse Radiation

A small area element of a black body dS_1 at temperature T_1 radiates at the rate of $(\sigma/\pi) T_1^4 \cos \Theta dS_1 dW$ into an element of solid angle dW . The solid angle of an element with area dS_2 normal to the direction Θ is $dW = \frac{dS_2}{r^2}$. Therefore, the rate at which radiation passes from dS_1 to dS_2 is

$$dQ = \frac{\sigma}{\pi} T_1^4 \frac{\cos \Theta}{r^2} dS_1 dS_2$$

Integrating over area 1 and area 2 gives total radiation rate

$$Q = \int_{S_1} \int_{S_2} dQ$$

The ΔQ into a conical shell zone with thickness dQ is given by

$$\Delta Q = \frac{\sigma}{\pi} T_1^4 \frac{\cos \Theta}{r^2} (r \sin \Theta) 2\pi (r \cos \Theta) d\Theta$$

FIGURE J-1

N/GPS ORGANIC RANKINE CYCLE SPACECRAFT PROFILE

J-2

Since the reaction force on the body is proportional to the emitted radiation, the axial force component due to ΔQ is proportional to the "axial component" of ΔQ

$$\text{i.e.} \quad \Delta Q_{\text{axial}} = \Delta Q \cos \theta$$

Then, the total axial force due to radiation from dS_1 is proportional to

$$\begin{aligned} Q_{\text{axial}} &= \int_0^{\pi/2} \frac{\sigma}{\pi} T_1^4 \frac{\cos \theta}{r^2} (r \sin \theta) 2\pi (r \cos \theta) d\theta \\ &= \frac{1}{3} \sigma T_1^4 \end{aligned}$$

i.e., the effective axial force is 1/3 that which would be expected in a simple calculation, based on total heat radiated.

Because emitted radiant heat transfer is given by $\sigma T_1^4 \Delta S$

we have:
$$Q = \frac{1}{3} \sigma T_1^4 \Delta S$$

and
$$\frac{dQ}{dT} = \frac{1}{3} \sigma (4T^3) \Delta S$$

$$\frac{\Delta Q}{Q} = 4 \frac{\Delta T}{T}$$

The tabulation below indicates numerics when the nominal emitting area temperature is 100° C (i.e., 373° K). A one-degree centigrade temperature change is shown to cause about a one percent change in radiated heat.

TABLE J-2

$\frac{\Delta T}{T}$	$\frac{\Delta Q}{Q}$	$\Delta T \text{ (}^\circ\text{C)} \text{ } [T = 373^\circ\text{K}]$
0.03	0.12	11.28
.02	0.08	7.45
.01	0.04	3.73
.005	0.02	1.86
.001	0.004	0.37

J-2.2 Sensitivity of SV Displacements to Surface Temperature Change

J-2.2.1 Flat Surfaces

If Q_I is the rate of total internally generated heat, Q_S is the rate of heat input from the sun, $\dot{E}_R$ is the radiated power from the antenna, then the steady-state radiation rate Q_R from the SV is:

$$Q_R = Q_I + Q_S - \dot{E}_T$$

To approximate the heat radiated from the SV's earth-viewing face, the emitted radiation is assumed uniformly distributed over the external surface. Then the rate at which heat is radiated from that earth-viewing face is Q_{EVF} :

$$Q_{EVF} = \left[\frac{\pi R^2}{2(\pi R^2) + 2\pi RL} \right] Q_R$$

where: R is vehicle radius

and L is vehicle length

The heat rate Q_{EVF} may be normalized by comparing it to the rate of solar energy radiation to the surface when the sun line is normal to the surface.

$$\text{i.e.} \quad F_{evn} = \frac{Q_{EVF}}{E (\pi R^2)}$$

where: E is the solar constant (1353 watts/m²).

Therefore, the radiation force due to diffuse emission of heat at rate Q_{EVF} is given by:

$$\begin{aligned} F_{ev} &= K Q_{EVF} / 3 \\ &= K F_{evn} (E) (\pi R^2) / 3 \\ &= K^1 F_{evn} (\pi R^2) / 3 \end{aligned}$$

where K is the constant which relates power emitted to radiation force = 7.42×10^{-10} lb/watt and K^1 is solar pressure = 1×10^{-7} lb/ft²

The GPS SV has the parameters: mass = 1105 lbs, Q_I = 4800 watts, Q_S = 500 watts, and $\dot{E}_T$ = 200 watts, so that Q_R = 510 watts.

With $R = 3$ ft and $L = 6$ ft, $Q_{EVF} = 850$ watts and $F_{evn} = 0.25$. But a 4% change in Q_{EVF} , which corresponds to about a 4-degree centigrade change in surface temperature, causes a change in F_{evn} of 0.01.

By use of the orbit perturbation equations given in App. C, we have calculated the effect of constant radial forces due to diffuse radiant heat transfer from the earth-viewing face of the SV. The computer simulation, given in Appendix J-A, indicates a buildup in perturbation displacements, linear with radial force. With radiated heat increment given in normalized form, the perturbation displacement rates are:

Radial displacement (ft/ day when $F_{evn} = 0.01$) $\cong 0.01$

Normal-to-radius-vector displacement (ft/ day when $F_{evn} = 0.01$) $\cong 0.1$

Another way of stating these results is: as a result of a 4° C mis-estimation of the temperature of the SV's earth-viewing side, there will be an error in predicting the radial displacement of 0.01 ft/ day and in normal-to-radius-vector displacement of 0.1 ft/ day.

Assuming a 10° C error in temperature prediction, the resultant error radial displacement rate is 0.025 ft/ day and the resultant error normal-to-radius vector displacement is 0.25 ft/ day.

Similarly, the expected 2° C prediction error for the SV's space-viewing flat face will yield predictions with errors of 0.005 ft/ day and 0.05 ft/ day, respectively.

J-2.2.2 Cylindrical Surfaces

The temperatures of the surface elements on the cylindrical portion of the SV vary slowly with time so that during the course of single vehicle revolution all area elements on the cylinder may be considered isothermal. Therefore, as a result of the vehicle spin all radiation forces, due to cylindrical surface radiation, tend to integrate to zero over a full spin when the spin is about the cylinder axis.

When the spin does not rotate about its cylinder axis there will be non-zero orbit radial forces if there are differences between temperatures of symmetric cylinder areas.

To determine the orbit radial forces on the vehicle we first evaluate the total force in a direction normal to a plane containing the cylinder axis. In accordance with eq 6, each small area has an "axial" Q equal to $1/3$ the total Q emitted from that area. Integrating the components of these Q 's, over half the cylinder, to find their total resultant in a direction normal to the plane one obtains:

$$Q = Q_x = \int_0^L \int_0^{\pi/2} 2 \left(\frac{1}{3} \sigma T_s^4 \right) (R d\theta dL) \cos \theta$$

$$= \frac{2}{3} \sigma T_s^4 R L$$

If the temperature of the opposite side of the cylinder is T_2 then its total normal Q is:

$$Q_{-x} = \frac{2}{3} \sigma T_2 RL$$

The difference $\Delta Q_X = Q_X - Q_{-X}$ is proportional to the net radiation force normal to the plane which divide the two halves of the cylinder.

$$\text{Thus } \Delta Q_X = \frac{2}{3} \sigma RL (T_1^4 - T_2^4)$$

$$\text{if } T_2 = T_1 + \Delta T$$

$$\Delta Q_X \cong \frac{2}{3} \sigma RL (4T^3 \Delta T)$$

$$\text{so that } \frac{\Delta Q_X}{Q_X} = 4 \frac{\Delta T}{T}$$

With total radiation Q_R from the satellite, the radiation from the cylindrical section Q_{CF} is:

$$Q_{CF} = \left[\frac{2\pi RL}{2\pi RL + 2\pi R^2} \right] Q_R$$

Also, for uniform heat radiation over the cylinder:

$$Q_{CF} \cong 2\pi RL \sigma T_1^4$$

Thus:

$$\frac{Q_X}{Q_{CF}} = \frac{\frac{2}{3} \sigma RL T_1^4}{2\pi RL \sigma T_1^4} = \frac{1}{3\pi}$$

and

$$\Delta Q_X = \frac{4\Delta T}{T} \left(\frac{Q_{CF}}{3\pi} \right)$$

Define the normalization as before:

$$F_{cfn} = \frac{Q_{CF}}{\epsilon(\pi R^2)}$$

Then

$$\Delta Q_X = \frac{4\Delta T}{T} \left(\frac{\epsilon(\pi R^2)}{3\pi} \right) F_{cfn}$$

If the SV does not rotate about its cylinder axis but about an axis displaced by an amount α then there will be a net radiation in force over a spin, along the actual spin axis, due to the net axial radiation ΔQ_X . This force F_{SA} will be given by:

$$F_{SA} = K \Delta Q_X \sin \alpha$$

and

$$F_{SA} = K' \left(\frac{\pi R^2}{3\pi} \right) F_{cfm} \left(\frac{4 \Delta T}{T} \right) \sin \alpha$$

With the SV parameters in Sec. J-2.1, $F_{CFM} = 0.955$.

With $\alpha = 0.5^\circ$ and $\Delta T = 10^\circ \text{ C}$, $T = 373^\circ \text{ K}$ one has

$$F_{SA} = K' \frac{\pi R^2}{3} (2.82 \times 10^{-4})$$

The computer results mentioned in Sec. 2.1 show that radial forces of this magnitude have negligible effect on the SV's orbit since the radial errors accumulate at 2.8×10^{-4} ft/day and normal-to-radial errors are 10 times greater.

J-3.0 UNLOAD PERTURBATIONS

Periodically, unload is required for the momentum which has accumulated on the SV. Approximately 0.5 ft lb sec will be unloaded during a single unload by firing, for 0.42 sec, as a couple, a pair of 0.2-lb jets spaced 6 ft apart (Section 7). When these jets are misoriented (or mismatched), their firing causes a net impulse on the vehicle perpendicular (or aligned with) their nominal thrust direction. The orbital effects of such jet errors have been considered in App. E for a somewhat larger vehicle and for larger error impulses than obtained by the vehicle here considered. But the errors do change linearly with mass and impulse so that the results of that reference can be used to obtain the perturbations of interest here.

For the present vehicle, a jet mismatch of 5% can be assumed and a jet misorientation of 0.5 degrees can be assumed. Then the error impulses are:

Jet mismatch impulse	= (0.2 #) (0.42 sec) (.05) lb sec
	= 0.0042 lb sec

Jet misalignment impulse	= (2 jets) (.2#/jet) (.42 sec) (sin 0.5 deg) lb sec
	= .00146 lb sec

As indicated in Table J-3, and in App. E, error impulses along the velocity vector cause the greatest orbit perturbations. The AVCS system design, therefore, achieves in-orbit-plane momentum unload by firing jets normal to the orbit plane, and achieves normal-to-orbit-plane unload by firing orbit-radius pointing jets. A vehicle attitude error, α , during firing can cause a component of the error impulse, ΔI , to be seen along the velocity vector. The along-velocity-vector impulse is $\Delta I \sin \alpha$. Resultant orbit error rates are shown in the table.

J-4.0 ORBIT PERTURBATIONS DUE TO SV MISALIGNMENTS

The purpose of this section is to estimate the incremental (error) solar forces on the SV due to misorientation of the vehicle. As a result of the control loop and the spin about the nominal earth pointing axis, the actual motion will be one in which the vehicle axis of symmetry will describe a cone in space, the cone half-angle is the attitude error.

Two error sources are considered below. With respect to the first, intuitively, it may be predicted the average error forces on the cylindrical body of the SV over a spin cycle, will be near zero, since the forces do not change greatly for small attitude changes with respect to the sun and there is an inherent cancelling of errors between attitudes obtained one-half spin cycle apart. With regard to the second - changes in vehicle self-shadowing - it may be observed that the errors are small.

J-4.1 SOLAR FORCES

A simplified model of the vehicle configuration was developed for use in assessing the level of solar forces during earth orbit. The model consists of flat and cylindrical surfaces equivalent in area to those of the proposed configuration, detailed in Fig. J-4 where surface properties are described.

The equations used have been derived based on the usual photon momentum transfer theory and the appropriate geometric treatment of surfaces exposed to solar flux. Normal (F_n) and tangential (F_t) components of force are defined for flat and cylindrical surfaces as follows:

Flat Plate

TABLE J-3

	App. E Vehicle (Spinner)	Reference-Design N/GPS			
		0.5° jet misalignment		5% jet mismatch	
		no att. error	1 degree att error	no att. error	1 degree att error
vehicle mass slugs	30	34.4	34.4	34.4	34.4
error impulse lb sec	0.029	0.0015	0.0015	0.0042	0.0042
impulse along velocity vector					
radial component ft/day	2	0.09	0.09	0.29	0.29
⊥ to radial component ft/day	200	9	9	29	29
impulse normal to orbit plane					
radial component ft/day	0	0	0.0016	0	0.005
⊥ to radial component ft/day	0	0	0.155	0	0.5
impulse along radius vector					
radial component ft/day	0	0	0.0016	0	0.005
⊥ to radial component ft/day	~0	0	0.155	~0	0.5

where:

$$F_{nf} = \left[(1+sp) \cos^2 \psi + \frac{2}{3} p(1+s) \cos \psi \right] P_f A_p$$

$$F_{\perp f} = \left[(1-sp) \cos \psi \sin \psi \right] P_f A_p$$

$$P_f = \text{solar pressure, } 1 \times 10^{-7} \text{ lb/ft}^2$$

$$A_p = \text{plate area, ft}^2$$

where

$$F_{nc} = \left[(1+sp/3) \cos^2 \phi + p(1-s) \frac{\pi}{6} \cos \phi \right] P_f A_c$$

$$F_{tc} = \left[(1-sp) \cos \phi \sin \phi \right] P_f A_c$$

$$A_c = 2rl$$

and where

p = fraction of incident radiation reflected

s = fraction of reflected radiation which is specular

The vehicle in-orbit geometry is illustrated in Figure J-5. In that figure, i is the orbit plane inclination to the earth-sun line, the axis Y_{OA} , also denoted Y_S , is normal to the orbit axis X_{OA} and the earth-to-sun vector Z_S . The coordinate axis X_S completes the right-handed triad X_S, Y_S, Z_S . The SV is located at angle Ω from the Y_{OA} axis taken positively with right-hand sense about the X_{OA} axis. The reference axis Y_R is radially outward from the earth thru the SV's radius vector, the reference axis X_R is parallel to the orbit axis X_{OA} , and the reference axis Z_R completes the right-handed triad.

The vehicle axes X_V, Y_V, Z_V are located with respect to the reference axis by rotating thru angle γ about the Y_R axis and then rotating through angle δ about the rotated X_R axis, also called X_V . Angle δ is the SV attitude error angle.

Our goal now is to evaluate the net force over one spin cycle in three directions: radial, along the velocity vector, and normal to the orbit plane.

The normal component of solar force on the cylindrical portion of the SV ($\bar{F}_{nc}$) lies normal to the vehicle Y_V axis and in the $Y_V X_S$ plane. It has a component away from the sun. Thus, the unit vector $\bar{e}_{nc}$ along $\bar{F}_{nc}$ is given by

$$\bar{e}_{nc} = \frac{\bar{Y}_V \times (\bar{Y}_V \times \bar{Z}_S)}{|\bar{Y}_V \times (\bar{Y}_V \times \bar{Z}_S)|}$$

The tangential component of solar force on the cylindrical portion of the vehicle ($\bar{F}_{tc}$) is directed along the $-Y_V$ axis when $Y_V \cdot Z_S$ is positive and along the $-Y_V$ axis when $Y_V \cdot Z_S$ is negative.

$$\bar{e}_{tc} = \begin{cases} -\bar{Y}_V & \text{when } \bar{Y}_V \cdot \bar{Z}_S \geq 0 \\ +\bar{Y}_V & \text{when } \bar{Y}_V \cdot \bar{Z}_S < 0 \end{cases}$$

The normal component of solar force on the flat faces of the SV ($\bar{F}_{nf}$) lies along the vehicle Y_V axis and is positive when $Y_V \cdot Z_S < 0$ i.e., along $\bar{e}_{tc}$. The tangential component of solar force on the flat faces is directed along $\bar{e}_{nc}$.

Then, the force on the SV's surface is

$$\bar{F} = (|\bar{F}_{nc}| + |\bar{F}_{tf}|) \bar{e}_{nc} + (|\bar{F}_{tc}| + |\bar{F}_{nf}|) \bar{e}_{tc}$$

With the aid of the computer programs listed in Appendix J-C, the net error forces have been calculated as the difference between the average of the force F over a spin revolution and the value of force F when the misalignment angle δ is zero. Figures in the App. J-C are graphs of these errors over a full turn. Then the net error forces over the turns are employed in a set of orbit perturbation equations, from App. C to evaluate the orbital effects. Figures in Appendix J-D show how SV orbital positions are perturbed for several values of orbit inclination with respect to the sun line. Maximum displacements occur when the sun lies in the orbit plane. When attitude errors are 0.5 degrees, maximum radial error rates are 0.025 ft/day and maximum normal-to-radius-vector error rates are 0.05 ft/day.

J-4.3 SELF-SHADOWING

When on-orbit, there will be predictable shadowing of the antennae and the launch support cone by the SV's main cylindrical body. In addition, there will be unpredictable shadowing due to vehicle attitude uncertainties. We consider here the effect of such shadowing on the SV's ephemeris. For the present purpose, the vehicle is assumed to have the external configuration shown below.

When the sun-line is normal to the cylinder axis within angle Θ part of one or the other of the small cylindrical projections will be shadowed. Small changes $\Delta\Theta$ in the sun angle Θ cause small changes in shadowed area by amount $\Delta A = (18 \sin \Delta\Theta) 36$ in projected area normal to sun-line.

Angular changes of the SV with respect to the sun-line are caused by spinning of the spacecraft about an axis skewed with respect to the SV's cylinder axis. When the angle of skew is $\Delta\Theta$, there is an average deviation, over one vehicle spin, of about $\Delta\Theta/2$ between the nominal sun angle and actual sun angle. Thus, the effective change in vehicle projected area is

$$\Delta A_{AVG} = (18 \sin \Theta/2) 36$$

Since the total area normal to sun-line is approximately $A = 72 \times 72$ sq in, the fractional area change is

$$\frac{\Delta A_{AVG}}{A} = \frac{(18 \sin(\theta/2)) 76}{72 \times 72}$$

When $\Delta\theta = 0.5^\circ$, the fractional area change is

$$\frac{\Delta A_{AVG}}{A} \approx 0.001$$

The effect on the vehicle ephemeris, of the force variation due to this fractional area change during times when shadowing occurs (i.e., when $|\tan \theta| \leq \frac{8.5}{15}$) has been computed with the program documented in Appendix J-E. The results indicate that with a 0.5° skew angle, the self-shadowing causes an orbit perturbation rate of 0.04 ft/day change in radial distance and a 0.08 ft/day change in normal-to-radius distance.

J-5.0

COMPILATION OF RESULTS

The tabulation of errors on table J-5-1 shows a maximum rate in radial position unpredictability of about 0.1 ft/day and in normal-to-radius position of 1.75 ft/day when all bias-type force perturbations are correctly estimated by ground filtering of orbit data.

References

1. Appendix C
2. Appendix E
3. Appendix H

Table J-5-1

Error Sources		Error Rates	
		Radial Displacements	Normal to Radial Displacements
		ft/day	ft/day
Thermal	1. 10°C error prediction in earth-viewing face temperature.	0.025	0.25
	2. 2° C error prediction in space-viewing face temperature.	0.005	0.05
	3. 10°C temperature difference on opposite sides of cylinder plus 0.5° attitude error.	0.0003	0.003
Unload Perturbations	1. 0.5° jet misalignment with 1° attitude error during firing		
	a. impulse nominally $\perp$ to orbit plane	0.0016	0.155
	b. impulse nominally along radius vector	0.0016	0.155
	2. 5% jet mismatch with 1° attitude error during firing		
	a. impulse nominally $\perp$ to orbit plane	0.005	0.5
	b. impulse nominally along radius vector	0.005	0.5
Spacecraft Attitude Error (Conical Motion of $\hat{L}$)	1. 0.5° error-uniform reflectivity	0.025	0.05
	2. 0.5° error-self-shadowing	0.04	0.08

```

1 REM THIS PROGRAM IS ON FILE 16
100 SET DEGREES
110 WINDOW 0,1080,-0.5,0.5
120 AXIS 90,0.05
130 MOVE 0,0
140 I=0
150 N=0
160 D=1
170 X0=0
180 Y0=0
190 X1=0
200 Y1=0
210 Z1=0
220 Z0=0
240 B=0
250 T2=0
260 C=PI*3*2/(1105/32.17)*1.0E-7/3
265 F=0.01
270 O1=2*PI/(12*3600)
280 O2=O1*2
290 T1=2*PI/O1*0/360
300 T=0
310 A=0
320 PRINT
330 REM
340 J=COS(I)*SIN(A)
350 K=SIN(I)*SIN(A)
360 L=COS(A)
370 P2=B
380 B=X0*J+Y0*K+Z0*L
390 X2=-O2*(X0-3*B*J)+C*F*J
400 Y2=-O2*(Y0-3*B*K)+C*F*K
410 Z2=-O2*(Z0-3*B*L)+C*F*L
420 R1=Y0*L-Z0*K

```

```

430 R2=Z0*J-X0*L
440 R3=X0*K-Y0*J
450 F1=T2
460 T2=SQRT(R1*2+R2*2+R3*2)
470 U2=X2
480 V2=Y2
490 W2=Z2
500 U1=X1
510 V1=Y1
520 W1=Z1
530 X1=X1+(U2+X2)*T1/2
540 Y1=Y1+(V2+Y2)*T1/2
550 Z1=Z1+(W2+Z2)*T1/2
560 X0=X0+(U1+X1)*T1/2
570 Y0=Y0+(V1+Y1)*T1/2
580 Z0=Z0+(W1+Z1)*T1/2
590 N=N+1
600 T=T+1
610 A1=A
620 MOVE A1,P1
630 A=D*PI
640 DRAW A,T2
650 MOVE A1,P2
660 DRAW A,B
670 IF T>T1*360*3 THEN 700
680 GO TO 320
700 MOVE 0,0

```

Appendix J-B
Orbit Perturbations Resulting
From Unload Impulse Errors
(reproduced from Appendix E)

Relative displacement of
30 slug, 1200 ft-lb-sec GPS
due to
precession jet mismatch of $3\frac{1}{3}\%$. Total impulse = 0.029 lb-sec
impulse normal to orbit plane
single firing at orbit = 0

Relative displacement of
30 slug, 1200 ft-lb-sec GPS
due to
precession jet mismatch of $3\frac{1}{3}\%$. Total impulse = 0.029 lb-sec
impulse along radius vector
single firing at orbit = 0

Relative displacement of
30 slug, 1200 ft-lb-sec GPS
due to
precession jet mis-match of $3\frac{1}{3}\%$. Total impulse = 0.029 lb-sec
impulse along orbit
single firing at orbit = 0

Evaluation of Perturbation of Forces Due to Misalignment of the Rotating N/GPS.

```

100 REM THIS PROGRAM IS ON FILE 14
110 REM THIS PROGRAM CALCULATES THE ERROR FORCES ON A GPS SATELLITE
120 REM DUE TO MISALIGNMENT WHICH RESULTS IN A CONICAL MOTION BY THE
130 REM VEHICLE'S AXIS OF SYMMETRY
140 SET DEGREES
150 J=1
151 DIM M(100)
200 R=3
210 L=6
220 R1=0.88
230 S1=1
240 S2=1
250 R2=0.88
260 I=45
270 G=0
280 D=0.5
290 O=0
300 C=1.0E-7
310 PRINT "      CHANGE OVER ONE SPIN TURN OF BODY FORCES DUE TO"
320 PRINT "      SOLAR PRESSURE"
330 PRINT "      MISALIGNMENT = 0.5 DEGREES"
340 WINDOW 0,360,-C,C
350 AXIS 45,C/10
360 MOVE 0,C
370 PRINT C
380 MOVE 0,C/2
390 PRINT C/2
391 MOVE 0,-C/10
392 PRINT "0"
393 MOVE 0,0
400 K1=1.0E-7
410 A1=PI*R1
420 A2=2*PI*L
430 N=0

```

160 SPIN ANGLE(DEG)*

```

440 Q4=0
450 Q5=0
460 Q6=0
500 REM
530 O1=SIN(O)
540 O2=COS(O)
550 G1=SIN(G)
560 G2=COS(G)
570 I1=SIN(I)
580 I2=COS(I)
590 O1=SIN(O)
600 O2=COS(O)
610 X1=D2*G1*D1*I2*O1
620 X2=D2*D2*I1
630 X3=G2*D1*G1*D1*I2*O2
640 X4=G2*D1*G2*D1*I1
650 X=X1+X2+X3+X4
660 Y1=-G1*D1*G1*D1*I2*O1
670 Y2=-G1*D1*D2*I1
680 Y3=G2*D1*D2*I2*O2
690 Y4=-G2*D1*G2*D1*I2*O1
700 Y=Y1+Y2+Y3+Y4
710 Z1=-G1*D1*G1*D1*I2*O2
720 Z2=-G1*D1*G2*D1*I1
730 Z3=-D2*D2*I2*O2
740 Z4=D2*G2*D1*I2*O1
750 Z=Z1+Z2+Z3+Z4
760 A=SQR(X*2+Y*2+Z*2)
765 A=A*MAX 1.0E-60
770 X=X/A
780 Y=Y/A
790 Z=Z/A
800 P2=-G1*D1*I1+D2*I2*O1+G2*D1*I2*O2
802 M=1
803 IF P2>0 THEN 805

```

```

804 M=-1
805 P2=ABS(P2)
810 Q=ACS(P2)
820 P1=SIN(Q)
830 F=90-Q
840 F1=SIN(F)
850 F2=COS(F)
860 REM
870 REM
880 REM
890 REM H1 AND T1 RELATE TO FLAT SECTION
900 H1=((1+S1*R1)*P2+2+2/3*R1*(1-S1)*P2)*K1*A1
910 T1=((1-S1*R1)*P1+P2*K1*A1
920 REM N2 AND T2 RELATE TO CYLINDRICAL SECTION
940 N2=((1+S2*R2/3)*F2+2+2/3*R2*(1-S2)*P1/6)*K1*A2
950 T2=((1-S2*R2)*F1+F2*K1*A2
1000 U1=(ABS(N2)+ABS(T1))*X+(ABS(T2)+ABS(N1))*G1*D1*-1*M
1010 U2=(ABS(N2)+ABS(T1))*Y+(ABS(T2)+ABS(N1))*D2*-1*M
1020 U3=(ABS(N2)+ABS(T1))*Z+(ABS(T2)+ABS(N1))*G2*D1*-1*M
1026 J=J+1
1050 IF N=1 THEN 1100
1060 U1=U1
1061 U2=U2
1063 U3=U3
1080 D=0
1090 N=1
1095 GO TO 500
1100 N=0
1105 D=0.5
1190 Q1=U1-U1
1191 Q2=U2-U2
1192 Q3=U3-U3
1200 Q4=Q4+Q1/24
1210 Q5=Q5+Q2/24
1220 Q6=Q6+Q3/24

```

when reflectivity lower with
angle to sun:

$$R1 = 0.88 \left(1 + \left[\frac{0.1}{.12} \right] \frac{Q}{90} \right)$$

$$R2 = 0.88 \left(1 + \left[\frac{0.1}{.12} \right] \frac{E}{90} \right)$$

```

2000 DRAW G,Q3
2200 G=G+15
2210 IF G=360 THEN 3000
2220 GO TO 500
3000 G=0
3002 MOVE 0,-C/2-0/30*C/20
3003 PRINT 0,Q4,Q5,Q6
3004 MOVE 0,0
3010 O=O+30
3015 IF O>180 THEN 4000
3020 GO TO 430
4000 END

```

Evaluation of Effects of Perturbation Forces Due to Misalignment of the Rotating N/GPS.

```

LIST
1 REM THIS PROGRAM IS ON FILE 5
100 SET DEGREES
101 K9=0.1
110 WINDOW 0,1000,-K9,K9
120 AXIS 90,K9/10
121 MOVE 0,-K9/10
122 PRINT
123 MOVE 0,K9
124 PRINT K9
130 MOVE 0,0
140 I=45
150 N=0
160 D=1
170 X0=0
180 Y0=0
190 X1=0
200 Y1=0
210 Z1=0
220 Z0=0
240 B=0
250 T2=0
260 M=1105/32.17
270 O1=2*PI/(12*3600)
280 O2=O1/2
290 T1=2*PI/O1/360
300 T=0
310 A=0
320 PRINT
325 A8=A
330 REM
340 J=COS(I)*SIN(A)
350 K=SIN(I)*SIN(A)
360 L=COS(A)
365 P2=B

```

```

370 B=X0*J+Y0*K+Z0*L
371 K1=1
372 K2=1
373 K3=1
375 IF A8<90 THEN 1000
376 IF A8=>90 AND A8<=180 THEN 1700
377 IF A8=>180 AND A8<=270 THEN 1800
378 IF A8=>270 AND A8<=360 THEN 1900
379 A8=A8-360
380 GO TO 330
381 F2=COS(I)*X+SIN(I)*SIN(A)*Y+SIN(I)*COS(A)*Z
382 F3=COS(A)*Y-SIN(A)*Z
383 F1=-SIN(I)*X+COS(I)*SIN(A)*Y+COS(I)*COS(A)*Z
390 X2=-O2*(X0-3*B*J)+F1/M
400 Y2=-O2*(Y0-3*B*K)+F2/M
410 Z2=-O2*(Z0-3*B*L)+F3/M
420 R1=Y0*L-Z0*K
430 R2=Z0*J-X0*L
440 R3=X0*K-Y0*J
450 P1=T2
460 T2=SQR(R1^2+R2^2+R3^2)
470 U2=X2
480 U2=Y2
490 W2=Z2
500 U1=X1
510 V1=Y1
520 W1=Z1
530 X1=X1+(U2+X2)*T1/2
540 Y1=Y1+(U2+Y2)*T1/2
550 Z1=Z1+(W2+Z2)*T1/2
560 X0=X0+(U1+X1)*T1/2
570 Y0=Y0+(V1+Y1)*T1/2
580 Z0=Z0+(W1+Z1)*T1/2
590 N=N+1
600 T=T+1

```

2 ORBITS *

```

610 A1=A
620 MOVE A1,P1
630 A=D*I
640 DRAW A,T2
650 MOVE A1,P2
660 DRAW A,B
670 IF T=>T1*360*3 THEN 700
680 GO TO 320
700 MOVE 0,0
800 END
1000 A9=A8
1010 IF A9=>0 AND A9<15 THEN 1100
1020 IF A9=>15 AND A9<45 THEN 1200
1030 IF A9=>45 AND A9<75 THEN 1300
1040 IF A9=>75 AND A9<90 THEN 1400
1100 X=-1.15E-9
1110 Y=0
1120 Z=-1.15E-9*K1*K2
1130 GO TO 381
1200 X=-4.99E-11
1210 Y=-2.5E-10*K2*K3
1220 Z=4.32E-11*K1*K2
1230 GO TO 381
1300 X=1.03E-10
1310 Y=-1.32E-10*K2*K3
1320 Z=-5.14E-11*K1*K2
1330 GO TO 381
1400 X=1.03E-10
1410 Y=-4.53E-11*K2*K3
1420 Z=0
1430 GO TO 381
1700 A9=180-A8
1710 K1=-1
1720 GO TO 1010
1800 A9=A8-180

```

```

1810 K2=-1
1820 GO TO 1010
1900 A9=360-A8
1910 K3=-1
1920 GO TO 1010

```

Appendix J-E

Displacement of N/GPS Due to Self-Shadowing Error

```

1 REM THIS PROGRAM IS ON FILE 16
100 SET DEGREES
101 I=0.2
110 WINDOW 0,1000,-1,I
120 AXIS 90,1/10
130 MOVE 0,I
131 PRINT I
140 I=0
150 N=0
160 D=1
170 X0=0
180 Y0=0
190 X1=0
200 Y1=0
210 Z1=0
220 Z0=0
240 B=0
250 T2=0
260 C=-1.0E-7*36/((1105/32.17)*1.0E-3
265 F=ATN(8.5/18)
270 O1=2*PI/(12*3600)
280 O2=O1*T2
290 T1=2*PI/O1*D/360
300 T=0
310 A=0
320 PRINT
330 REM
340 J=COS(I)*SIN(A)
350 K=SIN(I)*SIN(A)
360 L=COS(A)
370 P2=B
380 B=X0*J+Y0*K+Z0*L
385 C=0
386 IF A<F OR A>180-F AND A<180+F OR A>360-F AND A<360+F THEN 1000

```

```

387 IF A>540-F AND A<540+F OR A>720-F AND A<720+F THEN 1000
388 IF A>900-F AND A<900+F OR A>1080-F AND A<1080+F THEN 1000
390 X2=-02*(X0-3*B*K)+C
400 Y2=-02*(Y0-3*B*K)
410 Z2=-02*(Z0-3*B*L)
420 R1=Y0*L-Z0*K
430 R2=Z0*J-X0*L
440 R3=X0*K-Y0*J
450 P1=T2
460 T2=SQR(R1*2+R2*2+R3*2)
470 U2=X2
480 U2=Y2
490 W2=Z2
500 U1=X1
510 U1=Y1
520 W1=Z1
530 X1=X1+(U2+X2)*T1/2
540 Y1=Y1+(U2+Y2)*T1/2
550 Z1=Z1+(W2+Z2)*T1/2
560 X0=X0+(U1+X1)*T1/2
570 Y0=Y0+(U1+Y1)*T1/2
580 Z0=Z0+(W1+Z1)*T1/2
590 N=N+1
600 T=T+1
610 A1=A
620 MOVE A1,P1
630 A=D*N
640 DRAW A,T2
650 MOVE A1,P2
660 DRAW A,B
670 IF T>T1*360*3 THEN 700
680 GO TO 320
700 MOVE 0,0
1000 C=-1.0E-7*36/((1105/32.17)*1.0E-3
1010 GO TO 390

```

K. RETROFIT DESIGNS FOR A NUCLEAR GPS

K-1. INTRODUCTION AND SUMMARY

During the first quarter of the study, retrofits to the current Phase I GPS satellite design were considered. Replacements of the solar-array/battery system by radioisotope thermoelectric generators (RTG), organic Rankine cycle (ORC), and Brayton isotope power systems (BIPS) were evaluated. In addition, an ORC with a thermoelectric (TE) " topping " cycle was considered. Each of the systems could be feasibly retrofitted using the space-qualified MHW heat source. Within the weight constraints of the study groundrules, (to 1200 lbs exclusive of apogee motor (AKM) weight), any of the candidate nuclear power systems could be employed. Within the retrofit limitations, operational benefits could be achieved in the form of:

- availability of the high power mode throughout the design life of the satellite
 - elimination of the risk of a cold spacecraft during transfer orbit
 - reduction in the clock baseplate temperature variation
 - possibility of extended satellite design life.
 - improvement in predictability of satellite ephemerides
- By using an enlarged version of the MHW heat source, which would

provide 25% additional thermal power, the following added benefits would result on a retrofit basis:

- simultaneous transmission of the L-2 link P and C/A signals with the C/A link power 3 dB above its current level
- increased user power to the following levels:

L1 (P)	L1(C/A)	L2(P)	L2(C/A)
-160 dBW	-157 dBW	-165 dBW	-162 dBW

K-2 CANDIDATE RADIOISOTOPE POWER SYSTEMS

K-2.1 GENERATOR OPTIONS

Three advanced radioisotope power systems now in development under ERDA sponsorship form the bases for the power systems considered in this study. These are the: 1.) selenide RTG 2.) mini-Brayton cycle (BIPS) and 3.) organic-Rankine cycle (ORC) power systems. These are shown schematically in Figures K-1, K-2 and K-3.

FIGURE K-1. RTG SCHEMATIC

Current program plans call for carrying both dynamic generator systems through the ground demonstration phase in 1978 (Figure K-4). Following a period of ground testing, one of the two systems will be selected for flight development and testing. The selenide RTG will be developed independently, to the schedule shown in Figure K-5. Both static and dynamic systems are now scheduled for flight test readiness in FY 81.

K-2.2 HEAT SOURCE OPTIONS

An ERDA management decision of December 1975 has specified that the Multi-Hundred-Watt heat source (MHW/HS) shall be used for the 1981 test flight program, and for applications through 1983. A new-design heat source will be phased into programs thereafter. The MHW/HS has been qualified for launch and is currently in use aboard the LES 8 and 9 satellites, powering the MHW/RTG's. It is also scheduled for use on the Mariner-Jupiter/Saturn mission (MJS-77) due to be launched in mid 1977.

Because this heat source is already space -qualified, it imposes minimal schedule risk for missions in the 1981-83 time frame. Consequently, it was used as the baseline heat source in the retrofit phase of the study herein reported.

COMPRESSOR + ALTERNATOR + TURBINE = MINI - BRU
(Brayton Rotating Unit)

ALTERNATOR : 52000 RPM

1.16 lb of Xe/He gas mixture

FIGURE K-2 MINI BRAYTON CYCLE SCHEMATIC

PUMP + ALTERNATOR + TURBINE = CRU (Combined Rotating Unit)

Working Fluid: Dowtherm A

FIGURE K-3 ORGANIC RANKINE CYCLE SCHEMATIC

MILESTONES	FY75	FY76	FY77	FY78	FY79	FY80	FY81	
DEV. FLT. CONCEPT		■						
SYS. GROUND DEMO.		■	■	■				
SYS. SELECTION				▲				
FLT. SYS. DESIGN AND QUAL				■	■	■		
FLT. SYS. PRODUCTION						■	■	
DELIVERY FOR FLT. TEST							▲	

Figure K-4 Dynamic Generators Development Schedule

MILESTONES	FY77	FY78	FY79	FY80	FY81	
TECHNOLOGY READY		▲				
FLT. SYSTEM DESIGN		■	■			
RTG PRODUCTION AND DELIVERY			■	■		
FLT. SYSTEM PRODUCTION				■	■	

Figure K-5 Selenide RTG Development Schedule

The MHW/HS as qualified produces 2400 thermal watt (w_t) at BOL, built up of an assembly of 24 fuel sphere assemblies (FSA's) (Figure K-6). Each FSA contains a plutonium-dioxide (PuO_2) sphere which produces 100 w_t of power. These are arranged in four trays of eight spheres each, bolted together and enclosed by an aeroshell reentry structure.

Initial consideration of the retrofit designs was based on the 2400 w_t unit. Subsequently, the addition of another tray of eight FSA's was postulated to bring the thermal power of each heat source to 3200 w_t , thus enabling the determination of the effects of increased electrical power, where warranted by weight margin.

Based on the current GPS spacecraft power budget of 373 w_e on orbit (reference App. A), it was anticipated that both of the dynamic power conversion systems would meet the basic requirement with a single heat source, while two heat sources, and two RTG's, would be required for use with the thermoelectric converters.

The parametric and physical characteristics of each of the power systems configured for use with both 2400 and 3200 w_t heat sources is described in section K-2.4.

K-2.3 BASIC INTERFACE LIMITATIONS

In the retrofit design approach, the spacecraft design is largely determined, without regard for the desires or limitations of nuclear power systems. Therefore, a certain amount of fitting and tailoring is called for, which may be relieved in the new spacecraft designs to be considered in the next phase of the study. Other interface limitations, however, are less likely to be relieved, and are basic to spacecraft operation as presently conceived. The fundamental interface characteristics which were applied to the power systems in the retrofit phase are:

- Volume -- Potential power system mounting locations are limited to two regions, of which only one ultimately proved to be preferable for all systems. This region is the annular volume surrounding the AKM nozzle and thrust cone (Figure K-7).

MHW HEAT SOURCE

Figure K-6

Figure K-7, Retrofit Volume Limitations

- Launch Power -- Approximately 80 w_e of power is required from T-5 minutes through injection, a maximum of T + 45 hours
- Spinning Orbit Transfer -- A spin-stabilized orbit transfer of 45 hours maximum at 95 RPM is used. This requires both lateral g-loading tolerances and dynamic balance of the power system about the spin axis.

K-2.4 POWER SYSTEM DESIGN PARAMETERS

The generator design characteristics to be described were derived by the respective power system development prime contractors with Fairchild guidance and modifications as necessary to meet the various interface characteristics. These contractors are:

Selenide RTG	-	Teledyne Energy Systems, Timonium, Maryland
Organic Rankine	-	Sundstrand Energy Systems, Rockford, Illinois
Mini-Brayton	-	AiResearch Mfg. Co. of Arizona, Phoenix, Arizona

K-2.4.1 Selenide RTG

Because of the particular annular volume limitation of the retrofit design approach, the RTG's must be designed with only two fins which can wrap around the annulus. This suggests a design featuring two columns of selenide thermoelectric converters at opposite sides of the heat source, with heat-pipe-finned radiators extending from these columns, as shown in Figure K-8. The remainder of the heat source surface is thermally insulated, and the entire generator is enclosed by an aluminum housing.

Within the given design limitations two options were considered: fixed generators and deployable generators. Because the latter can radiate from both sides of the radiator, it was believed that these could offer an additional weight and/or efficiency advantage.

The fixed generators were finally preferred because:

- 1.) The weight advantage of the deployed system vanished when the deployment mechanism was included, and

TELEDYNE RTG DESIGN

Figure K-8

- 2.) The long spinning transfer orbit with RTG's in the stowed position would result in overheating of the thermoelectric modules, since only one side of the radiators would be effective during this period.

Table K-9 is a summary of the major performance parameters of the RTG unit. The radiator temperature was selected so that the radiator would fit within a semi-space of the available annulus without protruding beneath the $Z = 0$ station. (The retrofit spacecraft design is shown in section 6.)

Table K-10 shows the detailed weight breakdown for the 2400 w_t case, and table K-11 lists the materials and configurations of the individual components in the 2400 w_t version.

K-2.4.2 Organic Rankine Cycle System (ORC)

The ORC system currently under development for the ground demonstration phase of the program is optimized for a 1300 w_e output. This is the KIPS (Kilowatt Isotope Power System) unit, which requires three MHW heat sources. The minimum power requirements of the GPS can be met with a slightly modified KIPS powered by a single MHW heat source.

Added power can be achieved by using the 3200 w_t heat source. The only major component parts of the 1300 w_e KIPS system which are reconfigured for these cases are the regenerator and the turbine admission ports, which are specially adjusted for each system. In addition, the components have been repackaged in order to fit within the specified volume, and to provide the necessary dynamic balancing. Of course, connecting plumbing, accumulator, valves, wiring harness, and parasitic load radiator are all re-sized from the KIPS baseline case.

The basic system layout is shown in figures K-12 and K-13. The summary of performance parameters is given in Table K-14 for both 2400 w_t and 3200 w_t cases.

TABLE K-9
PERFORMANCE SUMMARY
SELENIDE RTG* FOR GPS MISSION

Output power (watts)	
BOL	228
7 years	
Load voltage, volts	12.3
Specific power (watts/pound), BOL	2.3** (Non-deployed RTG)
	2.4** (Deployed RTG)
Steady state temperatures, BOL (°C)	
Sink	-69
Radiator	
Cold junction	185
Hot junction	900
Isolation hot frame	930
Efficiencies, BOL (%)	
Overall	9.5
Thermoelectric	12.0
Thermal	80.5
Shoes/Straps	98.5
Fuel inventory (watts (t))	2400

* Two RTG's required

** Support structure not included

TABLE K-10
SIZE/WEIGHT SUMMARY
SELENIDE RTG FOR GPS

<u>Component</u>	<u>2400 w_t System Weight (lbs)</u>
Isolation Hot Frame & Wedges	6.3
Isolation Hot Frame Liner & Seal	5.5
Thermoelectric Elements & Hot Shoes	4.0
Straps & Cold End Hardware	4.0
Insulation	10.4
Heat Pipes	7.1
Heat Pipe Saddles	1.9
Fin (Web)	3.8
Fin Ribs	2.1
Housing	5.0
End Covers	2.8
Housing to Fin and Cold End Hardware Transitions	4.3
Housing Ribs	1.5
Emissive Coating	2.3
Connectors and Wiring	0.3
Heat Source	<hr/>
Total Weight	100.7
RTG Height (inches)	22.0
Fin Tip-to-Tip, along surface (inches)	102.0
Housing diameter (inches)	11.0

TABLE K-11
CONFIGURATION/MATERIAL SUMMARY
LCHPG/GPS II SELENIDE GENERATOR

<u>Item</u>	<u>Material</u>	<u>Configuration</u>
Heat Source	MJS Helipak	Right circular cylinder 7.4" dia. x 16.5" high with 24 balls (2400 W(t))
Isolation Hot Frame Support Structure	POCO AXF-5Q	Modified right circular cylinder 7.45 " I. D. x 8.25 " O. D. x 16.5 " high
Isolation Hot Frame Liner	Pt - 3008	Right circular cylinder with closed bottom, 7.42 " I. D. x .015 " wall x 17.0 " high
Bellows	Pt - 3008	7.42 " I. D. x .010" thick x 3.9 " developed length
Transition Bulkhead	Inconel 600	.010 " thick x 7.62 " I. D. x 10.37 " O. D. ; connects bellows to seal ring
Bi-metallic Seal Ring	Aluminum/Stainless	10.37 " I. D. x 11.10" O. D. , Al/ss halves are explosively bonded. Al part welded to housing, ss part welded to transition bulkhead
Hot Side Dielectric	Al_2O_3	Either sprayed on straps or separate sheet
Hot Straps	Molydenum	0.060 " thick
Thermoelectric	Copper-Silver Selenide	Partitioned, length = 0.3 dia. = .392 " , number = 128 per generator
Thermoelectric N element	Current Standard Gasolinium Selenide	Length = 0.3 " Diameter = .472 " Number = 128 per generator

TABLE K-11 (Cont'd)

A. Non-Deployable System (Continued)

<u>Item</u>	<u>Material</u>	<u>Configuration</u>
Cold Straps	Copper	0.010" thick, 2 string parallel circuit
Cold End Hardware	Aluminum rail copper follower BeO electrical insulation	Rail with spring loaded follower
Converter Side and End	Min-K TE 1400, Cotronics Ceramic Board	Cotronics on hot side Min-K on cold side Cotronics/Min-K interface held at 1600°F
T/E Module Insulation	Fiberfrax Paper	0.030" thick sheets
Housing	Aluminum	Right circular cylinder 0.060" wall x 11.0" O.D. x 20" high
Housing Ribs	Aluminum	.060" thick x 1" high circumferential ribs number = 7
Radiator Fin	Aluminum	Taper from 2" high at RTG to 5/32" high at fin tip. Thickness is 0.020" with a 1/4" wide flange on free edge. Number = 7 per fin
Heat Pipes	Copper/Water	Internal axial grooves 5/16" O.D., 14 pipes per RTG (7 on each side)
Electrical Connectors	TBD	Hermetically sealed to housing
Emissive Coating	ZnO + K ₂ SiO ₄ Binder (IITRI Z-93)	0.005" thick
Converter Gas Fill	TBD (Xenon or Vacuum)	Sealed between housing, ends, and isolation hot frame liner

ORC POWER SYSTEM FROM BELOW
(PLUMBING INCOMPLETE FOR CLARITY)

FIGURE K-12

ROLL OUT OF ORC POWER SYSTEM (PLUMBING NOT SHOWN FOR CLARITY)

COMPONENT SCALE 1/10

FIGURE K-13

TABLE K-14
PERFORMANCE AND MASS SUMMARY OF ORC SYSTEM

Heat Released, w_t	2400	3200
Heat Absorbed by Boiler	2311	3081
Power Output, w_e @ 28 VDC)	350	484
Heat Source Heat Loss, w_t	89	119
System Efficiency (350 W @ 28 VDC/2400 W)	14.58%	
MHW Heat Source/Boiler and Multi-Foil Insulation	80 lb	90 lb
Combined Rotating Unit (CRU)	41	41
Regenerator and Duct (Wet)	12	16.5
Jet Condenser	3.5	3.5
Accumulator (Wet)	11	15
Electronic Control Box	10	10
Parasitic Load Radiator (PLR)	2.5	3.5
Valves and Plumbing	7.5	9
Wiring Harness, Support Structure, Insulation, Misc.	13.5	17.5
Power Conversion Unit, Weight	101 lb	116 lb
Radiator Assembly (wet) (72" dia x 1/4" radial thickness)	<u>30.5 lb</u>	<u>38.5 lb</u>
Total Wet Weight	211.5 lb	244.5 lb
Radiator Length, inches	23.2	30.7

The combined rotating unit (CRU) contains the single-stage turbine wheel, the homopolar inductor rotor, and the rotor of the boiler feed pump. The angular momentum of the CRU at full design speed of 33,700 RPM is 17.6 ft-lb-sec. In many satellite systems this rather substantial angular momentum might be used for attitude stabilization, by aligning the momentum vector normal to the orbit plane. In the GPS retrofit case as in others, this would involve orienting the CRU laterally, i.e., normal to the Z-axis, in the spacecraft. This can be accommodated in the ORC system by stiffening the lateral bearings at a slight cost in system efficiency.

If the vehicle is spin-stabilized for orbit transfer and orbit injection, as is the GPS spacecraft and many others, a lateral orientation of the CRU would produce a slight coning motion of the vehicle and, therefore, a potential misalignment of the thrust vector and increased propellant consumption. These factors were investigated for the GPS case and found to be negligible (Appendix K-A). Although the coning angle is relatively large during PKM-2 burn (~ 5 degrees), the maximum increment in 3σ orbit trajectory error is less than 5% (Appendix K-A) which can be easily accommodated. The velocity loss appears during AKM burn for orbit injection and is less than 1% (Appendix K-A). This can be easily corrected by a corresponding increase in fuel loading of the AKM, now some 10% off-loaded. Therefore, no impediment appears to the use of the CRU as a bias-momentum wheel from a launch dynamics viewpoint.

Unfortunately, this use of the CRU for attitude stabilization in the retrofit case had to be rejected for reasons relating to the need for continuous spacecraft yaw rotation. In the solar-powered design this yaw steering is needed for solar-array orientation. However, it also produces a favorable thermal control condition for the frequency standards. Therefore, even though solar array orientation is no longer required in the nuclear powered case, the thermal control requirement still dictates that yaw steering be retained, at least in the retrofit cases. Consequently, the use of the ORC in this application will necessitate the inclusion of a counter-rotating momentum wheel to cancel out the angular momentum of the CRU. This will involve a weight penalty of approximately 15 pounds, but negligible power consumption by the wheel, which is to be used in a basically fixed-speed mode.

K-2.4.2.1. Thermoelectric - ORC Cascaded System

A conceptually simple yet effective modification of the ORC system is to use a thermoelectric topping cycle to increase system efficiency and specific power. This consists of the same MHW heat source with a selenide (or silicon-germanium) thermoelectric converter layer between the heat source and the ORC boiler tubes. The physical details of this package have not been worked out, but the performance numbers have been derived, as shown in Table K-15.

The efficiency of the thermoelectric segment is only 5.5% because the overall system optimizes at a relatively high cold-strap temperature for the selenides. Nevertheless, the turbine inlet temperature requirement of the ORC system is low enough ($\sim 340^{\circ}\text{C}$) so that significant performance may still be obtained from the thermoelectrics with a 900°C hot junction temperature.

Table K-15

Performance and Mass Summary of ORC Topped by 5.5% T.E. Generator

Heat Released, w_t	2400	3200
Heat Absorbed by ORC, w_t	2179	
Power Output, w_e @ 28 VDC	<div> <div>ORC 330</div> <div>TE 132</div> <div>}</div> <div>462</div> </div>	<div> <div>456</div> <div>176</div> <div>}</div> <div>632</div> </div>
Heat Source Heat Loss, w_t	89	119
System Efficiency (462 W @ 28 VDC/2400 W), %	19.25	19.7
MHW Heat Source/Boiler/T.E. G. and Insulations	87	102
ORC/PCU	101	116
Radiator Assembly (72" dia. x 1/4" radial thickness)	28.5	36.5
Total	216.5 lb	254.5 lb.
Radiator Length, inches	21.7	29

As with the organic Rankine cycle system, the BIPS is being developed for ground demonstration at 1300 w_e , although with its higher projected efficiency this is expected to be achieved with two MHW heat sources. The Brayton system differs from the ORC system mainly in two respects: it is a single phase (all gas) cycle, thereby eliminating the boiling/condensing aspects of the Rankine cycle; and it operates at a higher turbine inlet temperature, thereby requiring refractory metals in the hot side of the loop, including the heat source, heat source heat exchanger, turbine housing and connecting ducting.

The system design for the retrofit application was based on the following assumptions:

- single heat source of either 2400 or 3200 w_t
- Brayton rotating unit (BRU) is fixed, same as for 1300 w_t BIPS
- recuperator is tailored in size for each case, but similar in geometry and design throughout
- ducting similar to BIPS
- Columbium (C-103) hot loop alloy with 1600^o F turbine inlet temperature, as for BIPS
- radiator design for 0.99 reliability in 7 years
- sink temperature of 390^o R with radiator emissivity of 0.8

With these assumptions, a computer analysis was performed by AiResearch/Phoenix in which several design cases were investigated. The compressor inlet temperature (CIT), recuperator effectiveness (E_R), and the pressure drop ($\Delta P/P$) of the radiator and recuperator were all varied.

Because the radiator area is known to be a critical parameter, this was first used to generate a typical radiator area vs. output power curve (Figure K-16). Within the available area for a non-deployable radiator, approximately 360 w_e would be produced. If a deployable radiator is postulated, the effective radiator area is doubled because radiation can now emit from both faces of the radiator surface, and the output power is increased to approximately 460 w_e . Therefore, a deployable radiator, requiring a flexible joint for the working fluid (at $T_{max} = 212^{\circ}F$), was considered as an option.

It must be noted in this connection in particular that the launch power requirement dictates a ground-start up condition for the system, and hence a system in which the hot columbium components are not exposed to oxygen. This requires that the entire system be sealed and either evacuated or flooded in a inert noble gas atmosphere. The radiator is usually assumed to form a part of the enclosing envelope, a much more difficult proposition if it is to be deployable as well. This design problem will be examined in detail in the forthcoming stage of the study within the context of the special nuclear-powered spacecraft. For the retrofit scoping phase no further consideration was given to this aspect of the Brayton system.

From the computer analysis, three candidate systems were chosen at each thermal input power, a total of six candidates. The first two in each group were for non-deployed radiators, in which weight and efficiency are traded off. The third case in each group is for the deployed radiator, approximating maximum specific power within the assumptions.

Tables K-17 and K-18 show the weight and power breakdowns for the $2400 w_t$ and $3200 w_t$ cases respectively.

Table K-19 shows the computer summary of the critical cycle parameters for the $413 w_e$ case.

Figures K-20 and K-21 show the component layouts within the prescribed envelope for the $413 w_e$ and $453 w_e$ cases respectively. The smaller recuperator heat exchanger in the latter case necessitates a rearrangement of the components in order to maintain system balance. Fortunately, the electronic control box can be located at any position to assist in the balancing.

The angular momentum of the BRU is 0.68 ft-sec at design speed of 52,000 RPM. At this value, a factor of 25 less than that of the ORC/CRU, any effects on spacecraft attitude are well within the capability of the normal attitude control system to compensate for, and launch effects would be negligible.

1. Typical Characteristics
2. $E_R = .96$
3. $\Delta P/P$ Radiator & Recuperator = .024
4. 10% Weight Variation
5. $T_{\text{sink}} = 390^\circ\text{R}$
6. $\epsilon = .8$

FIGURE K-16 EFFECT OF RADIATOR AREA ON SYSTEM PERFORMANCE

TABLE K-17

POWER AND WEIGHT SUMMARY OF BIPS WITH 2400 w_t HEAT SOURCE

Output Power, w_{e2}	359	413	453
Radiator Area, ft^2	38	38	*65
Weight, lb			
Recuperator	75	140	60
Radiator	35	38	65
Mini-BRU	36	36	36
HSA	81	81	81
Ducting	20	20	20
Insulation	28	30	28
Structure	20	20	20
Controls	10	10	10
Total, lb.	305	375	320

TABLE K-18

POWER AND WEIGHT SUMMARY OF BIPS WITH 3200 w_t HEAT SOURCE

Output Power, W Electric	522	563	700
Radiator Area, Ft Squared	38	38	*65
Weight, lb			
Recuperator	95	140	110
Radiator	38	38	65
Mini Bru	36	36	36
HSA	101	101	101
Ducting	20	20	20
Insulation	28	30	28
Structure	20	20	20
Controls	10	10	10
Total, Lb.	348	395	390

* Deployed Radiator

TABLE K-19

CYCLE OUTPUT DATA, 413 w_e CASE

COMP INLET TEMP (DEG R) .570633E+03	TURB INLET TEMP (DEG R) .206000E+04	GROSS PWR OUTPUT (KW) .428005E+00	CRU ROTAT.SPEED (RPM) .520000E+05
COMP INLET PRESS (PSIA) .248017E+02	TURB INLET PRESS (PSIA) .358622E+02	NET POWER OUTPUT (KW) .413377E+00	RECUPERATOR EFF (----) .980000E+00
COMP PRESS RATIO (----) .146790E+01	TURB PRESS RATIO (----) .142524E+01	ALT.PERIF.WINDAGE (KW) .358144E-01	BETA (----) .970938E+00
COMP EFFICIENCY (----) .764010E+00	TURB EFFICIENCY (----) .796975E+00	ALT.RADIAL WINDG. (KW) 0.	ALTERNATOR EFF (----) .877779E+00
COMP WORK FACTOR (----) .787356E+00	GG TURB VEL. FACTOR (----) .101440E+01	BEARING POWER (KW) .126705E+00	BLEED FLOW (LBS/SEC) .234938E-02
COMP WEIGHT FLOW (LBS/SEC) .117469E+00	TURB WEIGHT FLOW (LBS/SEC) .115119E+00	PUMP POWER (KW) 0.	ACCESSORY POWER (KW) .146281E-01
COMP POWER (KW) .908716E+00	TURBINE POWER (KW) .155883E+01	GROSS CYCLE EFF (----) .178336E+00	NET CYCLE EFF (----) .172240E+00

FIGURE K-20

BIPS RETROFIT LAYOUT (453 W_E CASE)

FIGURE K-21

K-3

RETROFIT DESIGN APPROACHES

K-3.1

SOLAR POWER SUBSYSTEM ELEMENTS ELIMINATED

Figures K-22 through K-24 show the major elements of the solar (electrical) power system (EPS) used on the Phase I space vehicle (Φ1 SV). It was assumed that all of the EPS system elements shown in the figures would be removed except the Load Control Unit, DC/DC converter, sun sensors and current sensor. While being retained, these latter boxes would require some modification for use with any of the isotope systems. The following weight breakdown of the EPS total was derived from the 21 July 1976 weight status report of the Φ1 SV.

<u>EPS Item</u>	<u>Weight (lb)</u>
Solar Array (with deployment mechanism)	93.0 lb.
Shunt Dissipator	9.0
Orientation Drives	18.4
Solar Array Control Electronics	6.5
Power Conditioning Electronics	25.0
Batteries (3 @ 31.5)	<u>96.9</u>
Total Removed	248.8
Sun Sensor	.8
Load Control Unit	15.0
Current Sensor	1.0
DC/DC Converter	<u>10.0</u>
21 July 76 Status	275.6

The weight of the fore and aft stow latches, panel edge restraint hooks and other array related hardware which is also removed was assumed to be counterbalanced by cover panels, doubler plates and other secondary structure added to accommodate the retrofitted subsystems.

GPS SOLAR POWER ELEMENTS

FIGURE K-22

GPS SOLAR POWER INSTALLATION

FIGURE K-23

FIGURE K-24 SOLAR/BATTERY POWERED GPS

It was evident (reference current GPS Spacecraft layout shown by Figure K-25) that the required volume for any of the proposed isotope systems precluded their integration within the main body of the satellite. Of the solar power elements being removed only two, the power conditioning unit and the solar array drive electronics, are located in the main body. In addition because the same structural configuration and louver panels were being retained and the fact that the RCS thruster modules would offer mechanical and jet plume interference, it was clear that the radioisotope system radiators could not be integrated around the main body of the vehicle. A primary desire for the retrofit exercise was to keep structural changes to a minimum, to minimize interference and/or changes to other subsystems, to provide attachment points that tie in with the established structural design and to maximize accessibility to the power subsystem, particularly the heat source.

Accessibility is considered important, particularly if a viable method can be developed for installing the heat source in the radioisotope power subsystem that has already been integrated in the spacecraft, thereby reducing the prelaunch time during which cooling must be provided and access to the spacecraft is limited.

These considerations led to radioisotope power system designs that would occupy the volume between the thrust cone and the dynamic envelope at the aft end of the satellite, with mounting points forward on the aft bulkhead and laterally on the thrust cone.

Figure K-25 GPS Space Vehicle Configuration (1 of 2)

Figure K-25 GPS Space Vehicle Configuration (2 of 2)

As indicated in section K-2, four nuclear powered subsystems were considered for the GPS Phase I design retrofit. Layouts were made to verify that the components could be fitted into the annular volume surrounding the Orbit Insertion Motor nozzle and that required radiator areas could be accommodated without interfering with either the ATLAS-F shroud or the Stage Vehicle System (SVS) attachments. Thermal analyses were also conducted to verify that the radiator area predictions presented by the various nuclear power subsystem contractors were valid and based on consistent assumptions. The radiators for all of the retrofitted nuclear power subsystems form a new, cylindrical surface at the aft end of the spacecraft.

A single electrical interface cable incorporating power, control and telemetry leads runs from the control/monitor unit in each of the subsystems to the Load Control Unit inside the main body of the spacecraft. The power subsystem components are, in general, thermally isolated from the primary structure through the use of insulating washers, pads, etc. that are part of the attachment structure.

Both 2400 and 3200 watt (thermal) sources were considered. System layouts for both approaches are essentially the same.

Descriptions of mechanical interfaces and quantitative data on subsystem power, weight and performance are provided in the following sections.

K-3.2.1

Radioisotope Thermoelectric Generator (RTG) Approach

The RTG's were designed with integral radiator wings that are fastened together at their far ends by a splice joint. Figure K-26 shows the tapered stiffeners on the outer surface that parallel the heat pipes around the radiator surface. The generators are supported at their aft ends by tubular struts attached to the thrust cone. A single clevis fitting and a shear panel bracket on the forward end of the generator are fastened to the corner of the aft bulkhead and the GPS shear panel. Side and end views of the RTG

FIGURE K-26, RTG POWERED GPS

installation are shown by figure K-27. An analysis was made to determine attachment hardware required to mate the RTG components to the spacecraft and, as indicated by Table K-22, it was found that attachment hardware weights totalled less than 6 lbs. Provision is made for heat source accessibility through the end of the spacecraft and thermal insulation is provided to isolate both the GPS spacecraft body and the transfer orbit Stage Vehicle from the Radioisotope heat source thermal load.

6.2.2 Brayton Isotope Power Subsystem (BIPS) Approach

In both the BIPS and ORC units the MHW heat source and associated heat exchanger are attached to the aft bulkhead using a cylindrical bracket incorporating thermal isolators attached to the thrust cone at their aft ends with tubular struts. The loads of other dynamic system components are supported by brackets fastened to the aft bulkhead, the thrust cone or both. Figure K-29 shows a pictorial view of the BIPS retrofit installation with the Brayton Rotating Unit (BRU) oriented with its momentum vector parallel to the spacecraft Z axis (spin axis during transfer orbit and earth pointing axis during operation). Figure K-30 shows an alternate installation with the BRU momentum vector oriented at right angles to the spacecraft Z axis. Since the momentum vector generated by the BRU is small, either orientation is feasible. Table K-31 lists the mounting hardware required to mate the BIPS components with the GPS Phase I Spacecraft. As indicated by this table, the mounting hardware weight totals less than 13 lbs.

In the deployable version, the BIPS radiators would utilize bellows joints and spring/damper deployment mechanisms. The cylindrical radiators shown would require some local envelope modification to avoid jet impingement from the 5 lb orbit-correction hydrazine thrusters. Jet plume shielding may also be required but would add negligible weight since this can be accomplished with thin nickel foil. Figure K-32 shows a side and end view of the BIPS installation.

FIGURE K-27 RTG SYSTEM INSTALLATION

TABLE K-28

WEIGHT BREAKDOWN - RTG ATTACHMENT HARDWARE

ITEM	MTL	UNIT WEIGHT	QTY/ VEH	TOTAL WEIGHT	LOC.
Ftg, Upr Spt, X-Y Plane	Machined Al	.14	4	.56	Rft Blkd
Shear Web, X-Y Plane	Al Sht.	.13	2	.26	Aft Blkd
Thrust Ftg. Z Loading	Machined Al	.29	2	.50	Shear Panel
Struts, Lwr Spt, X-Y Plane	Al Tubing	.06	4	.24	Thrust Cone
Strut End Fittings	Machined Al	.15	8	1.20	Thrust Cone
Ftg, Strut To Thrust Cone	Machined Al	.15	4	.60	Thrust Cone
Doubler, Thrust Cone	Al Sht	.09	4	.20	Thrust Cone
Radiator Supt On X-Axis	Sht Al Intercostal	.60	2	1.20	Aft Blkd and Thrust Cone
Bushings, Thermal Isolation	Glass Reinf. Epoxy	—	A/R	.10	In Ftg Lugs
Attach Hdwre	Cres Blts, Nuts	.02	24	.48	In Ftg Lugs
Wt. of Structure Provisions				5.34 lb	
Contingency			10%	.54 lb	
Total Structure w _t				5.88 lb

FIGURE K-29, BIPS RETROFIT INSTALLATION

FIGURE K-30, BIPS RETROFIT INSTALLATION
(ALTERNATE APPROACH)

TABLE K-31

WEIGHT BREAKDOWN - BIPS MOUNTING HARDWARE

ITEM	MTL	UNIT WEIGHT	QTY/ VEH	TOTAL WEIGHT	LOC.
HSA Support Cradle	Cres, Saddles Thermal Isolators, Al Bkts	2.0	1	2.0	Thrust Cone
Stiffeners, Doublers to Beef Up Thrust Cone		.5	1	.5	Thrust Cone
Mini-BRU	Cres, Ring, Thermal Isolation, Al Pedestal	1.0	1	1.0	Aft Blkd
Doubler For Aft Blhd H/C Face Sht	Al Sht	.1	1	.1	Aft Blkd
Recuperator Supt Bkts	Cres, Isolation Al	.8	1	.8	Thrust Cone
Thrust Cone Reinforcement	Al	.5	1	.5	Thrust Cone
Control Box Mounting	Bolts, Inserts	—	—	.1	Aft Blkd
Radiator Support	Alum Intercostals and Bkts	1.40	4	5.6	Aft Blkd, Longerons
Mounting Hdw, Tube Clamps			A/R	1.0	
Wt. of Structure Provisions				11.6 lb	
Contingency			10%	1.2 lb	
Total Structure Wt.				12.8 lb

FIGURE K-32

BIPS INSTALLATION (Z AXIS BRU, DEPLOYED RADIATOR)

K-3.2.3 Organic Rankine Cycle (ORC) Approach

Installation interfaces for the ORC approach are similar to those previously described for BIPS. Figure K-33 provides a pictorial view of the ORC retrofit assembly and figure 6.2-7 shows side and end views. In Figure K-34 the axis of the ORC rotating turbine/generator/pump combined relating unit (CRU) is shown parallel to the Z axis. As shown in Appendix K-A, a lateral orientation of the CRU is feasible. However, because of thermal control considerations, all of these retrofit designs maintain the yaw steering requirement. Therefore, the spacecraft must rotate in both yaw and pitch, making it necessary to cancel out the angular momentum of the CRU (17.6 ft-lb-sec) in either orientation. A momentum wheel with very low torquing requirement can be provided for this function at a weight penalty of approximately 15 pounds. This is included in the spacecraft weight estimates for both ORC and TE-topped ORC systems.

K-3.2.4 Structure/Mechanical Interfaces

Layouts of the radioisotope subsystems options based on the 2400 watt heat source show that they fit well within the aft end annular volume bounded by the aft bulkhead, the separation plane, the thrust cone and their respective cylindrical radiators. When the power level is increased to 3200 w_t , however, the heat source assembly with its associated thermoelectric generator, boiler, or gas heat exchanges protrudes below the separation plane by several inches. The Rankine cycle radiators were equally long. Additional studies, based on detailed drawings of the separation plane area would be required to determine whether or not the thrust cone would need to be modified for the 3200 w_t systems. The AKM plume problem in this area is treated below.

Mounting of the radioisotope systems would require new attachment inserts in the aft bulkhead and a number of stiffeners on the thrust cone. Layouts of the support structure for the heat source assemblies, which are located somewhat outboard of the shear panels, suggested attachment at the edge formed by the aft bulkhead and the shear panels. This corner is a strong point and lends itself to the use of brackets with clevis joints at the location of the (removed) aft solar stow latch (Figure K-34). A combination of brackets (with and without shear webs) and struts is then used to support the

FIGURE K-33 ORGANIC RANKINE (ORC) POWERED GPS

FIGURE K-34 ORC INSTALLATION

lateral, thrust axis and rotational loads of other components through the bulkhead and/or thrust cone. The support structure weights are based on preliminary stress analysis of selected members.

The arrangement of power system components in the annular volume was constrained by the requirement of a balanced configuration during spin. As the weight data for components becomes more accurate, the locations shown previously will need to be adjusted, but in any case, provision for balance weights will be included. Estimates have been made of the new CG position along the Z axis, but the effect of such a change on the moments of inertia ratio which affects the vehicle during PKM burn or on the AVCS throughout the mission have not been made. The July 21 mass properties indicated CG shifts for the solar powered Φ I SV between the 36 and 38 inch Z-axis station during the mission. The estimated change resulting from retrofitted isotope power systems were as follows assuming the solar powered GPS at Z = 38.0:

	<u>Z Station</u>	<u>Δ</u>
RTG	33.1 "	4.8 "
BIPS	31.5	6.5
ORC	34.2	3.8

These are sizable shifts and their effect will need to be analyzed.

Access to subsystems and components inside the main body of the satellite utilizes four curved panels on the +X and -X sides of the spacecraft. The thrust cone is removed for installation of the AKM in the center of the main body. Access to the radio-isotope power systems will be from below between the thrust cone and cylindrical radiators. Orientation of the various heat sources with their cylindrical axis parallel to the SC centerline was generally considered to be desirable. Orientation of other dynamic system components was made to minimize plumbing complexity. Line diameters are as specified by the power system subcontractors but, in some cases, line lengths and routing have been modified, leading to small changes in system pressure drops and flow rates.

The BIPS deployable radiators pose a difficult design problem because of the large size of their gas flow passages. A preliminary layout indicates that the deployable cylindrical sections utilizing flexible bellows joints is a possible solution, although the use of such a radiator with its added mechanical complexity is a clear disadvantage for the BIPS.

K-3.2.5 Power Subsystem Interfaces

Each of the radioisotope power subsystems has a power conditioning unit that interfaces with the LCU retained from the solar power system. This unit would also act as the command and telemetry interface and would contain any necessary active monitor/control circuits in the case of the dynamic isotope systems. In the retrofit case it has been assumed that all power matching would be accomplished on the power system side of the interface leaving the LCU with minimum design changes. An example of these latter changes would be removal of the solar array power leads and the added provision for ordinance power.

Internal to the isotope systems, the start up sequence, provisions for prelaunch cooling, launch and ascent temperature profiles and launch phase available power profiles will be studied as part of the redesign effort beyond the retrofit phase.

The effects of internal G forces on the operation of the RTG heat pipes during the spin mode was minimized by a redesign of the heat pipe attachment and radiator curvature as indicated below. In the BIPS gas system, these effects are negligible. In the case of the organic Rankine cycle that has both vapor and liquid loops, the effect requires that the jet condenser be mounted at an angle to the axis of rotation in order to correct for the coriolis acceleration of the focussing jet stream.

K-3.2.6 Thermal Control Subsystem Interfaces

An independent assessment of the tradeoff between isotope system radiator size and operating temperature was made (see Appendix K-B). The size of these radiators is limited by aft end dynamic envelope, while the operating temperature has a direct affect on RTG and dynamic cycle efficiency. Parametric data on radiators sized to handle both the 2400 and 3200 watt heat sources are given and confirm contractor designs.

In the case of the RTG's the (water/copper) heat pipes were redesigned to account for G forces and the effective radiating area was increased by moving structural ribs from the inside to the outside of the cylindrical radiator surface. These radiators would operate at a pipe wall temperature of about 304° F at the evaporator end with a fairly constant vapor to wall ΔT along the pipe length of about 1 to 2° F. The following parametric data were calculated for these radiators:

Heat Pipe Vapor Temperature, °F	Radiator Diameter *	
	62"	72"
Full Sun	271°	226°
Full Shadow	255°	209°

*L = 19" @ 2400 w_t, = 25 @ 3200 w_t

The shape and orientation of the pipes were also modified to overcome any "puddles" of water between the evaporator and condensor due to the approximate 10g centrifugal force at the 100 rpm spin rate. The ΔP required to overcome 10g is 131 lbf/ft² whereas the available capillary pressure within the heat pipes is 2.8 lbf/ft². The bends where puddles could collect and block heat pipe action were eliminated.

Similar parametric radiator temperature data were calculated for the Brayton and Rankine cycle systems that included additional parameters such as stowed vs. deployed configurations and variable deployed areas (see Appendix K-B).

The heat soakback from the retrofit isotope heat source assemblies into the GPS structure will be a small fraction of the total GPS heat rejection load. A detailed analysis of the conductive paths from the RTG source assemblies through their support structure is given in the Appendix K-B. The total conducted heat load from two RTG's is 26 watts at a GPS structural temperature of 40° F, decreasing linearly to 16 watts at 100° F. The conducted heat from the BIPS and ORC systems will be less than half of the RTG value because of their higher source efficiency and lower radiator temperature. In addition, with the solar arrays removed the louver heat rejection capability is increased by 60 watts (at 70° F). The total vehicle heat rejection load is about 300 watts at normal power and 385 watts at high power with 100% TT&C. The worst case RTG load would then be 6% of the total with the structure at 70° F.

In retrofitting the radioisotope systems, the thermal subsystem interface would change to reflect the removal of battery heaters, the modification or addition of thermal doublers on the aft bulkhead and changes and/or removal of the external multi-layer insulation blankets on the thrust cone. Thermal analysis of the aft end of the spacecraft would include the heat exchange interface between the isotope systems and the AKM during coast and burning phases and AKM plume heating.

This latter interface is an important liability of the retrofit designs. In the Φ I SV the AKM plume is free to expand. Except for the thrust cone and aft-facing TT&C antenna no equipment is encountered. In the retrofit designs, particularly the 3200 w_t cases where the radiator as well as the heat source assemblies protrude below the separation plane, the plume expansion could result in circulation of hot gases within the annular volume occupied by the power system. The addition of an annular plume deflector causes complications, in that it would upset the radiative heat balance within the power system cavity since about 10 to 20% of heat rejected is from the back face of the radiators. Analysis however, might show that the relatively short burn time of the AKM (35 seconds) does not produce unacceptable temperature or heating levels.

A thermal interface also exists between the radiators and the RCS hydrazine thruster plumes. A 5-lbf precession and Δ V thruster fires aftward along the -X face, while two 0.1-lbf thrusters fire aft along both the + and -X faces*. The primary problem with the larger plume would be radiator heating during steady state firings (maximum 13 minutes), while the added area of the isotope radiators also offers a surface for the development of side forces during on-orbit firing of the 0.1-lbf thrusters. These impingement forces and heating are acceptably small in the Φ I SV design because of the outboard modular mounting of the jets that provides separation from adjacent surfaces. The outside envelope of the RTG radiators has been made flush with the + and -X panel surfaces to avoid direct plume impingement (see Figure K-27). An iteration on the dynamic systems radiator design would be required to modify their shape in the area of the 5-lbf thruster, or to provide deflection/shielding of this plume.

* The other 5-lbf thruster fires forward.

K-3.2.7

Tracking Telemetry and Command (TT&C) Interfaces

It is assumed that conditioning circuitry for power system telemetry would be part of the omnibus power conditioning unit included in each of the power subsystems and that sufficient channels would be available in the TT&C after removal of the solar power yaw and pitch sun sensor and drive position telemetry.

It is possible that the aft-facing TT&C antenna would either have to be relocated or have its length modified because of possible interference with isotope power components, particularly those in the dynamic systems. The integration layouts shown in previous figures maintain the antenna in its present location and do not violate its field of view, but power system components are in close proximity to the antenna and any changes could lead to mechanical or field of view interference.

K-3.2.8

Attitude/Velocity Control System (AVCS) Interfaces

As already discussed, the thermal control requirements dictate that yaw steering and pitch control be maintained, which permitted the axis of the rotating unit to be retained along the Z-direction.

K-4 Conclusions

K-4.1

Weight and Power Comparisons

The weight and power capabilities of the candidate nuclear power subsystems were previously defined in section K-2. In summary, these characteristics are:

<u>For 2400 w_t Sources</u>	<u>Current Solar</u>	<u>RTG</u>	<u>BIPS</u>	<u>ORC</u>	<u>ORC/RTG</u>
Gross EOL power (watts)	410	420	428	331	465
Net EOL power (watts)	326	412	420	323	457
Subsystem weight (lbs)	276	249	340	253	258

For 3200 w_t Sources

Gross EOL power (watts)	559	662	457	635
Net EOL power (watts)	548	651	446	624
Subsystem weight	285	412	286	296

As may be seen from the above summary, the 2400 w_t sources are capable of maintaining the current GPS spacecraft high power mode (373 watts net power required) through end of life and three of the four systems have sufficient power margin to provide some small improvements to the "user power " critical parameter.

The 3200 wt sources are capable of providing significant improvements to the "user power " critical parameter. Power-to-user options vs DC loads are listed in Appendix B, which encompasses spacecraft power levels up to 654 watts. Estimated incremental Navigation subsystem weights corresponding to these options are:

<u>Option</u>	<u>Δ Power</u>	<u>NAV Subsystem Weight Increment</u>
(1)	Baseline	Baseline
(2)	+18 W	+ 2.1 lbs
(3)	+54	9.9
(4)	+86	19.7
(5)	+43	24.6
(6)	+50	31.8
(7)	+30	36.1

As indicated in previous sections, the spacecraft housekeeping subsystems required 67 watts D.C. and the basic spacecraft initial on-orbit weight (after AKM firing) is 889 lbs.

Table K-35 and K-36 show the impact on overall spacecraft weight of the 2400 watt and 3200 watt sources respectively.

Compared to an initial on-orbit weight budget of 897 lbs. (GPS spacecraft launch weight estimate as of July 21, 1976, was 1665 lbs. which included 8 lbs. of ballast) only marginal improvements are possible and the BIPS approval is not feasible. However, if the initial on orbit weight budget is increased to 1200 lbs. (as stated in the study guidelines), any of the nuclear power source approaches is feasible and most of the user power options can be accommodated.

K-4.2 Weight/Reliability Tradeoffs

GPS reliability and failure rates at the subsystem level are as follows:
(reference: Rockwell 7/76 Design Review).

	<u>R 5 year</u>	<u>$\lambda \times 10^{-9}$/hr.</u>
TT&C	.8927	2591.4
AVCS	.9847	352.0
EPS	.9628	865.5
RCS	.9945	125.9
TCS	.9999	2.2
STR	.9992	18.3
NAV	.9407	1395.7
<u>OIS</u>	<u>.9963</u>	<u>73.2</u>
SV	.7885	5424.2

The TT&C subsystem has the lowest reliability and of its components and sub-assemblies, the following two are the major contributors:

	<u>R_{5year}</u>	<u>$\lambda \times 10^{-9}$/hr.</u>	<u>Weight</u>
Transponder	.96105	907.1	16.0
KIR-23	.95841	909.9	8.6

The KIR-23 function is handled in a switched parallel redundant arrangement. As an example

TABLE K-35, SPACE VEHICLE: POWER/WEIGHT SUMMARY (2400 w_t SOURCE)

GPS POWER SYSTEM	SOLAR	RTG (FIXED)	BIPS DEPLOYABLE RADIATOR	ORC	ORC (TE TOPPING)
POWER (WATTS)					
NAV Subsystem (HiPower)	305.5	305.5	305.5	305.5	305.5
EPS (Losses + Battery Chrg)	84.0	8.1	8.1	8.1	8.1
Housekeeping	61.4	61.4	61.4	61.4	61.4
Power Conversion Loss	NA	<u>41.7</u>	<u>0</u>	<u>0</u>	<u>0</u>
TOTAL REQUIRED	450.9	416.7	375.0	375.0	375.0
EOL AVAILABLE POWER, WATTS	410	420	428	331	465
MARGIN	-40.9	+3.3	+53	(-44)	+90
SV WEIGHTS:					
EPS Retained/Added	275.6	249.1	339.8	253.3	257.8
Other Subsystems	<u>613.2</u>	<u>596.5</u>	<u>601.5</u>	<u>601.5</u>	<u>601.5</u>
TOTAL (Less AKM)	888.8	845.6	941.3	854.8	859.3
MARGIN	-	+43.2	(-52.5)	+34.0	+29.5
EPS EOL POWER/WEIGHT	1.49	1.69	1.26	1.31	1.80

TABLE K-36. SPACE VEHICLE: POWER/WEIGHT SUMMARY (3200 w_t SOURCE)

GPS POWER SYSTEM	SOLAR	RTG (FIXED)	BIPS DEPLOYABLE RADIATOR	ORC	ORC (TE TOPPING)
POWER (WATTS)					
NAV Subsystem (Hi Power)	305.5	305.5	305.5	305.5	305.5
EPS (Losses and Battery Chrg)	84.0	11.1	11.1	11.1	11.1
Housekeeping	61.4	61.4	61.4	61.4	61.4
Power Conversion Loss	NA	0	0	0	0
TOTAL REQUIRED	450.9	378.0	378.0	378.0	378.0
EOL AVAILABLE POWER, WATTS	410.	559.	662.	457.	635.
MARGIN	(-40.9)	+181	+284	+79	+257
SV WEIGHTS					
EPS Retained/Added	275.6	284.7	411.8	286.3 *	296.3 *
Other Subsystems	613.2	596.5	601.5	601.5	601.5
TOTAL	888.8	881.2	1013.3	887.8	897.8
MARGIN (WITH 1200 LB DRY WT.)	311.2	318.9	186.7	312.2	302.2
EPS EOL POWER/WEIGHT	1.49	1.96	1.61	1.60	2.14

*Includ. 15 lb. wheel

of weight/reliability tradeoff, the effect on MMD of adding a third parallel KIR-23 was calculated. The submodel at that point was:

The reliability increased about 3% from .9584 to about .9900 which in turn increased the TT&C subsystem from .8927 to .9222. This improved the overall SV reliability from .7885 to .8146 ($\lambda = 4682.6 \times 10^{-9}/\text{hr}$), resulting in an increase in MMD from 4.45 to 4.74 years or 3-1/2 months, at a cost of 8.6 lb. weight increase. A similar improvement in MMD brought about by an additional TT&C transponder would cost 16 lb. weight increase.

K-4.3 POTENTIAL ADVANTAGES OF RETROFIT DESIGNS

In addition to the user power improvements documented in section K-4.1, two significant advantages accrue from the use of retrofitted isotope power on the GPS. Both occur in the navigation area, as a result of the removal of the solar panels:

- 1) Improved Frequency Standard temperature control and,
- 2) Reduced navigation error over a given time or, conversely, increased time between required updates.

The solar panels on the current GPS spacecraft radiate both reflected solar and infrared energy into the louvered radiators. It is the orbital fluctuation of this heat load on the louver that thermally controls the frequency standard that is the major cause of variation in its interface temperature. Removal of the solar panels reduces the temperature variation and in turn improves stability of the standard.

Details are given in Appendix K-B of simplified math models for the thermal input/output to the louver and of the frequency standard/louver panel composite structure. The model assumes a constant frequency standard dissipation of 18.6 watts and utilizes experimental performance data for the GPS Fairchild-supplied louver panels. The model calculates a maximum daily variation of 2.6°C for the worse case noon orbit compared to a measured

variation of about 2.2°C reported at the 21 July 1976 GPS review. The specified stability tolerance is 4°C per day.

The residual thermal load after removal of the solar panels consists of direct solar radiation at small angles in pitch and roll. Following an eclipse the worse case sun angle could approach 8° over a period of about 8 minutes. The net input under these conditions is a complex function of both sun and louver blade angle due to secondary reflections within the louver mechanism. The residual load however would be a small fraction of that from the solar array. If, for example, the residual were as high as 13% the interface temperature variation could be reduced to about 0.3°C.

The second effect of removing the GPS solar arrays is that the area seeing the sun and therefore the effect of solar radiation pressure is reduced by a factor of 2/3. The advantage in this arises from the fact that navigation accuracy depends on the degree of uncertainty in orbit predictability. Where a fixed percentage error exists in the prediction of solar forces, a reduction in the absolute level of the solar force reduces proportionally this contribution to the total error. Perturbations to the orbit are caused by earth non-sphericity, lunar and solar gravitational effects, solar and vehicle emitted radiation pressures, particle impingement and vehicle mass expulsions. The total effect of these perturbing forces are sufficiently well-modelled that, after adequate ground-tracking, the actual space vehicle position is currently predicted to within 1 to 2 meters.

The analysis of the predictable effect of solar pressure in Appendix K-A leads to the following conclusions:

- 1) In a nominal circular orbit, solar pressure destabilizes the basic vehicle motion so that increasing displacement increments occur with time,
- 2) Large variations in displacement occur during each orbit,
- 3) The destabilized motion increases as the orbit plane approaches the sunline,
- 4) The displacements increase relatively linearly with total solar pressure force,
- 5) Cyclic variation in solar pressure force at orbit frequency can cause increasing displacement amplitudes,

6) Small changes in solar pressure force can cause increasingly large displacement errors.

The following table indicates the maximum magnitude of the relative displacements of the GPS solar configuration from a circular orbit path due to basic solar pressure force only. The displacements are in terms of an increase in the length of the radius vector (i. e., altitude) and a sideward displacement perpendicular to the vector.

Orbit Inclination from Sun Line	Maximum Relative Orbit Displacements Due to Solar Pressure, ft.			
	Single Orbit		Σ^3 Orbits	
	"Altitude" Error (@3/4 orbit)	Normal* Error (@3/4 orbit)	"Altitude" Error (@3/4 orbit)	Normal* Error (@3/4 orbit)
0° (Ecliptic)	104	192	352	656
30°	88	168	304	568
60°	53	99	176	320
90°	0	24	0	24

An uncertainty of 1% in the solar pressure force would produce an approximately proportional uncertainty in displacement.

The nuclear powered spacecraft however would experience 1/3 the predictable displacements shown as well as a 1/3 reduction in uncertainty of position prediction.

APPENDIX K-A
DYNAMIC ANALYSES

K-A.1 Rotating Machinery Orientation Tradeoffs

K-A.1.1 Introduction

The candidate nuclear power supplies for the GPS space vehicle contain rotating machines. These machines, after start on the ground, remain operating throughout launch and on-orbit life. They can be mounted with their spin axes either along the launch direction or normal to it.

Insertion into the nominal final orbit follows the sequence:

1. Launch by Atlas burn.
2. Separation, spin-up, and launch into drift orbit by two successive PKM burns.
3. Insertion of the spinning payload into nominal final orbit using the orbital insertion system (OIS)

Up to 48 hours of spin may pass between the end of event 2 and the start of event 3. By the start of event 3, the vehicle normally is rotating stably about its axis (X) of maximum moment of inertia, nutation having been damped out by a passive damper responding to lateral axis accelerations. Stable rotation about the X axis, or close to it, is necessary for efficient OIS operation.

In this memo, we consider the vehicle motion at start of event 3 as determined by the orientation of the spin axis of the nuclear supply's rotating machine.

K-A.1.2 Analysis

While in drift orbit, the vehicle is in essentially external torque-free motion which is described by the Euler equations:

$$(1) \quad \dot{\omega}_x = -\omega_y \frac{H_z}{I_x}$$

$$(2) \quad \dot{\omega}_y = \omega_x \frac{H_z}{I_z} - \omega_z \frac{H}{I_z} + \omega_x \omega_z \left(1 - \frac{I_x}{I_z}\right) - \frac{T_{dy}}{I_z}$$

$$(3) \quad \omega_z = \omega_y \frac{H_x}{I_{\mathcal{L}}} - \omega_x \omega_y \left(1 - \frac{I_x}{I_{\mathcal{L}}} \right) - \frac{T_{dz}}{I_{\mathcal{L}}}$$

Where: ω_i = Angular rate about vehicle axis i

$\dot{\omega}_i$ = Angular acceleration about vehicle axis i

H_1 = Angular momentum of laterally oriented rotating machine spin axis, assumed along Z

H_x = Angular momentum of rotating machine spin axis along nominal spin axis X

I_i = Vehicle moment of inertia about axis i

T_{di} = Damper torque about axis i

By assumption:

$$I_x > I_y = I_z = I_{\mathcal{L}}$$

K-A.1.2.1

Rotating Machine Spin Axis Along Vehicle X Axis

By definition for this configuration: $I - I_x > 0$; $H_{\mathcal{L}} = 0$

Because the vehicle is symmetrical about the X axis, the axes can be redefined so that the y axis will be assumed to feel the damping torques and $T_{dz} = 0$.

In the steady state condition $\dot{\omega}_y = 0$ because the damper torque T_{dy} removes rotational kinetic energy from the system whenever accelerated axis motion exists. Thus, ω_y will be constant in steady state.

Since $\dot{\omega}_y$ is zero then from equation (2) $\omega_z = 0$ and/or $\omega_x = \frac{H_x}{I_{\mathcal{L}} - I_x}$

And if $\omega_z = 0$, from equation (3) $\omega_y = 0$ and/or $\omega_x = \frac{H_x}{I_{\mathcal{L}} - I_x}$.

Two possibilities for the steady state condition therefore exist:

either $\omega_x = \frac{H_x}{I_{\mathcal{L}} - I_x}$ and ω_z, ω_y are constant (because $\dot{\omega}_y = \dot{\omega}_z = 0$) or $\omega_y = \omega_z = 0$.

Now, it is shown that the steady state solution $\omega_x = \frac{H_x}{I_z - I_x}$ is impossible for arbitrary initial conditions if any damping has occurred.

Total angular momentum of the system is conserved so that with initial X axis rate ω_{x0} and initial lateral axis rate ω_{y0} (where $\omega_z^2 = \omega_x^2 + \omega_y^2$):

$$(4) \quad (I_x \omega_{x0} + H_x)^2 + (I_z \omega_{y0})^2 = (I_x \omega_x + H_x)^2 + (I_z \omega_y)^2$$

Also, total vehicle rotational kinetic energy cannot increase:

$$(5) \quad \frac{I_x \omega_{x0}^2}{2} + \frac{I_z \omega_{y0}^2}{2} \geq \frac{I_x \omega_x^2}{2} + \frac{I_z \omega_y^2}{2}$$

Combining equation (4) and (5) with the assumed steady state value $\omega_x = \frac{H_x}{I_z - I_x}$ leads to the condition: $(\omega_{x0} - \omega_x)^2 \leq 0$.

This is impossible unless $\omega_x = \omega_{x0}$ in which instance no damping has taken place because the equality sign applies. Thus, when damping has occurred $\omega_x \neq \frac{H_x}{I_z - I_x}$ and the true steady state condition has $\omega_y = \omega_z = 0$ with the angular momentum of the system concentrated along the X axis.

K-A.1.1.2.2 Rotating Machine Spin Axis Aligned Near Vehicle X Axis

In this configuration, $H_x \neq 0$ and $H_z \neq 0$ but $H_x \gg H_z$. As in 2.1, because the vehicle is symmetrical about the X axis the axes can be redefined, so that the Y axis will be assumed to feel the damping torques, and $T_{dz} = 0$.

In the steady state condition $\dot{\omega}_y = 0$ due to the damping action, so that ω_y is a constant. But ω_y must be zero since a non-zero ω_y would change ω_x at constant rate in accordance with equation (1). Therefore, from equation (3) $\dot{\omega}_z = 0$, $\omega_z =$ constant and:

$$(6) \quad 0 = \omega_x H_z - \omega_z H_x + \omega_x \omega_z (I_z - I_x)$$

Solutions to equation (6) can be understood if H_z and H_x are assumed components of a single vector H in the X Z plane making an angle θ with respect to the X axis.

ω_x and ω_z are components of a total rate vector ω in the ZX plane making an angle α with respect to the X axis.

$$\text{Then: } H_x = H \cos \theta ; H_z = H \sin \theta$$

$$\omega_x = \omega \cos \alpha ; \omega_z = \omega \sin \alpha$$

Combining these equations with equation (6) leads to:

$$(7) \quad \sin 2\alpha = \frac{-2H}{\omega(I_x - I_z)} \sin (\theta = \alpha)$$

for $\theta = \alpha$ and α small:

$$\alpha = \frac{K}{2 + K} \theta$$

$$\text{where } K = \frac{2H}{\omega(I_x - I_z)}$$

Thus, when damping occurs about the Y axis, α is at most equal to θ so that the vehicle's steady state axis of rotation is near the X axis when there is a small deviation of the H vector from the X axis.

Note that because all vehicle rates are constant the same steady condition also satisfies the dynamic equation when damping occurs along the Z axis.

K-A.1.2.3

Rotating Machine Spin Axis Along Vehicle Z Axis

For this configuration, the analyses of section 1.2.2 applies with $\theta = 90^\circ$. The resultant steady state motion then has $\omega_y = 0$, ω_x and ω_z are constant, and from equation (7) $\alpha = \sin^{-1} \frac{H}{(I_x - I_z) \omega}$. It should be noted that the steady state axis of rotation can be greatly offset^x from the X principal axis - the intuitively "obvious" axis.

The table below shows x , ω_x , $\frac{\Delta V}{V}$ as a function of H for the Rockwell GPS wherein $I_x - I_z = 14 \text{ slug ft}^2$, $\omega \cong 9.19 \text{ r/s}$. The function $\frac{\Delta V}{V}$ ($= \cos \alpha$) shows the fractional loss in thrusting, at the end of drift orbit, incurred as a result of the rotation of the vehicle X axis about the vehicle axis of rotation. Thrusting is of such long duration that it covers several spin cycles.

H ft. lb. sec.	α deg.	ω_x rad/sec	$\frac{\Delta V}{V}$
0.5	0.22	9.19	0.0000
1.0	0.45	9.19	0.0000
5.0	2.23	9.18	0.0008
10.0	4.46	9.16	0.0030
15.0	6.70	9.13	0.0068
20.0	8.94	9.08	0.0120
25.0	11.2	9.01	0.0190

K-A.1.3

Conclusions

The steady state effects on drift orbit vehicle attitude due to rotating machine orientation have been shown to be small with possible exceptions due to large laterally oriented machines. These are discussed below.

K-A.2

Errors Introduced by Laterally-Oriented Machines

The launch sequence for the GPS includes separation from the Atlas launch stage, rapid spin-up, ignition of the first PKM, separation from the spent PKM, short coast, ignition of the second PKM, and separation from the spent second PKM. Attitude errors and vehicle nutations which develop during these phases propagate into transfer orbit inclination, and apogee and perigee attitude errors.

This study evaluates the effects on the drift orbit trajectory at first apogee, due to an operating, laterally oriented rotating machine in the GPS.

The analysis technique in Reference (1) determines the 3 angular deviation, from the desired direction, which occurs during each launch phase. RSS'ing of these increments yields the total 3 misdirection of the vehicle's velocity vector from the desired direction. In addition, the nutation cone angle increment, during each phase, is found. Table D.2-1 contains a summary of the angular disturbances documented in Reference (1).

Here, we are concerned with the effects of operating, laterally oriented, rotating machines in the payload. The approach taken is based on using the techniques and results of Ref. (1). The approach assumes two reasonable propositions:

1. Equal lateral axis angular momenta, due to lateral axis torques, are added to the basic GPS vehicle (i. e., with no lateral machine) and to the vehicle with a lateral rotating machine. The causative torques are due to imperfections in the separation spring systems, the spin-up torque system, and PKM thruster misalignments.
2. Angular offsets during PKM burns are the same for the basic GPS and the vehicle with a lateral rotating machine. The averaging effect of the thrusting PKMs on the spinning vehicle is imperfect only due to the change in thrust levels during PKM burns. These changes are due to start-up transients, imperfect "constant" thrust levels, and burn-out transients. The net torque resultant of the non-uniform thrusting over a single spin cycle precesses the system angular momentum vector and causes angular offsets between the system's angular momentum vectors existing before and after that cycle. Thus, for equal spin rates and similar PKM thrust histories, one can expect incremental vehicle offsets to vary inversely with the total vehicle angular momentum.

BASIC VEHICLE *

K-66

$\Delta \theta$ - Increment during phase; β - Cumulative total error; η - nutation $\frac{1}{2}$ cone angle.

* Basic Vehicle values from Reference 1

A third proposition, based on digital computer simulation results, is also assumed:

3. The total nutation angle of the vehicle, when a laterally-oriented machine exists, is obtained by taking the sum of the nutation half-cone angle of the "basic" vehicle and half the maximum nutation angle of an untorqued vehicle containing the rotating machine.

Using these three propositions and the techniques and results of Reference (1), one can estimate the offset increments $\Delta\theta_i^1$ and nutation angles n_i^1 at each stage i as a function of lateral axis machine angular momentum:

- a. for phases 3, 4, 5, 8, 9* (from proposition 2)

$$\Delta\theta_i^1 = \Delta\theta \frac{I_{si} \omega_{si}}{\left[(I_{si} \omega_s)^2 + H_i^2 \right]^{1/2}}$$

where $\Delta\theta$ = increment during phase for basic vehicle

(from Table 2.1 Reference (1) - deg)

I_{S_i} = moment of inertia about nominal spin axis at end of
phase i - slug ft²

ω_s = spin speed - rad/sec

H_i = angular momentum of lateral axis machine - ft-lb-sec

* For phase description see Table D. 2-1.

- b. for phase 2 $\Delta \theta \frac{1}{2} = \Delta \theta$ at Atlas/stage vehicle separation

$$\eta_1^1 = 0$$

This number is negligible due to the small time that the rates resulting from the small disturbing torques exist.

- c. for phase 7 $\Delta \theta \frac{1}{7} = \eta_6^1 - \eta_7^1$ (from analysis in Ref. 1)

where η_6^1, η_7^1 are half of the nutation angles at end of phases 6, 7 as calculated below, for vehicle with lateral angular momentum,

- d. for phases 3-9 (from proposition 3)

$$\eta_i^1 = \left[\frac{\frac{I_{li} \omega_{li}}{I_{si} \omega_s} \cdot \frac{I_{si} \omega_s}{\left[(I_{si} \omega_s)^2 + H_{\ell}^2 \right]^{1/2}} \cdot 57.3 \right] + \eta_{ih}$$

where: ω_{li} = lateral vehicle rate - rad/sec

η_i = nutation half-cone angle of untorqued vehicle - deg
(from Reference 1)

η_{ih} = half maximum nutation angle of untorqued vehicle with
lateral angular momentum H -deg (from computer
simulation)

I_{li} = moment of inertia about lateral axes at end of phase i
-slug ft²

and for phases 3, 4, 5, 6

$$\eta_i = \frac{\frac{I_{li} \omega_{\ell}}{I_{si} \omega_s} \cdot 57.3}{1}$$

and for phases 7, 8

$$\omega_{\ell 8} \cong \omega_{\ell 7} \cong \frac{\eta_6^1}{57.3} \cdot \frac{I_{S6} \omega_s}{I_{\ell 6}}$$

and for phase 9

$$\omega_{\ell 9} = \left[\left(\frac{I_{\ell 8} \omega_{\ell 8}}{I_{\ell 9}} \right)^2 + \left(\frac{\Delta H}{I_{\ell 9}} \right)^2 \right]^{1/2}$$

where ΔH = incremental lateral angular momentum added during phase 9 ft-lb-sec
and since

$$(I_{\ell 8} \omega_{\ell 8})^2 + (\Delta H)^2 = (I_{\ell 9} \omega_{\ell 9})^2$$

$$\frac{\Delta H}{I_s} = \left[\eta_9^2 - \eta_8^2 \left(\frac{I_{s8}}{I_{s9}} \right)^2 \right]^{1/2} \frac{\omega_s}{57.3}$$

The results of using these equations for lateral angular momenta (H_{ℓ}) of 5, 15, and 26.1 ft/lb/sec are given in Table D.2-1. The values of η_{ih} were determined by digital computer simulation.

D.2.2 Trajectory Errors Due To Angular Disturbances

The analysis technique in Reference (2) determines the transfer orbit errors due to:

1. Atlas hand-off errors
2. Atlas/Stage vehicle separation
3. Spin-up
4. Coast

In addition to error sources associated with PKM burns:

- 1) PKM Total Impulse Variations
 - a) 3σ variations due to propellant load, propellant performance, manufacturing tolerances, etc., of 0.75%.
 - b) Variations due to PKM grain temperature variations from the nominal rating temperature of 70°F.
- 2) PKM Ignition Time Variations
- 3) 3σ pitch attitude error during burns

4) 3σ yaw attitude error during burns

5) 3σ nutation during burns

The results of the analysis is repeated in Table K-A.2.2. Nominal three errors are calculated by separately RSS'ing the 3σ positive and negative errors.

In this study, the techniques and results of Reference (2) are utilized, together with angular disturbance data from Table D.2-1. Results of this re-application are given in Tables D.2-3, D.2-4, and D.2-5.

Since nutation during PKM burn always shows itself as a reduction in the velocity increment for that burn, and since nutation is an inherent characteristic of the vehicle with lateral angular momentum, it may be concluded that the true 3σ apogee error, for apogee high, is less than that indicated in the tables.

K-A.2.3 Conclusions

A comparison of the transfer orbit trajectory errors for the basic GPS vehicle and for vehicles with lateral angular momenta up to at least 26 ft/lb/sec shows no major differences.

The maximum difference occurs for $H_{\phi} = 26.1$ ft/lb/sec in the 3σ apogee radial distance. It is only 36 km less than the basic vehicle's total error of 530 km (apogee low). The nominal specification value is 525 km. The incremental error is primarily due to inherent loss in along-track velocity increment due to the relatively large nutation angle (4.97 deg) which exists during the PKM 2 burn. Thus, by altering the nominal drift orbit to have a somewhat higher apogee, one can somewhat reduce the true apogee error (apogee low) while still keeping the apogee high error within specification.

It is useful to note, as Reference (2) does, that "Considering only Atlas errors and PKM (which are GFE) impulse variations of $\pm 0.75\%$, the resulting apogee dispersion is ± 514.4 km. . . " If the "true" 3σ impulse variation is 0.9 of the nominal then the total apogee altitude error would be only 525 km (apogee low).

TABLE K-A.2-2

PREDICTED 3σ TRANSFER ORBIT ERRORS

PHASE	ERROR SOURCE	ERROR MAGNITUDE	ERROR EFFECTS		
			APOGEE RADIUS (km)	PERIGEE RADIUS (km)	INCLINATION (DEG)
ATLAS	ALTITUDE	(± 2000 FT)	± 15.1	± 0.6	0
	VELOCITY MAGNITUDE	(± 15 FT/SEC)	± 125.3	0	0
	FLIGHT PATH ANGLE	± 0.1 DEG	± 18.0	± 2.2	0
	ORBIT INCLINATION	± 0.2 DEG	± 3.4	0	± 0.189
PKM 1 BURN	IMPULSE	$\pm 0.75\%$	± 226.7	0	0
	GRAIN TEMP VARIATIONS	$+10, -15^\circ\text{F}$	$+14.5, -21.8$	0	0
	IGNITION TIME*	± 1 SEC	± 1.4	± 0.2	0
	PITCH ATTITUDE*	± 1.84 DEG	$- 90.0$ $+ 29.5$	$+ 4.9$ $- 6.8$	0
	YAW ATTITUDE*	± 1.84 DEG	$- 30.0$	0	± 0.546
	NUTATION	0.78 DEG	$- 2.8$	0	0
PKM 2 BURN	IMPULSE	$\pm 0.75\%$	± 443.8	0	0
	GRAIN TEMP. VARIATIONS	$+10, -15^\circ\text{F}$	$+28.4, -42.6$	0	0
	IGNITION TIME	± 4 SEC	± 7.7	± 0.6	0
	PITCH ATTITUDE	± 1.07 DEG	$- 65.0$ $+ 49.0$	$+ 0.7$ $- 1.0$	0
	YAW ATTITUDE	± 1.07 DEG	$- 8.0$	0	± 0.216
	NUTATION	1.266 DEG	$- 14.4$	0	0
			$+518.6/529.6$	$+5.5/-7.3$	± 0.617

TABLE K-A.2-3

PREDICTED 3σ TRANSFER ORBIT ERRORS H_p = FT-LB-SEC

PHASE	ERROR SOURCE	ERROR MAGNITUDE	ERROR EFFECTS		
			APOGEE RADIUS (KM)	PERIGEE RADIUS (KM)	INCLINATION (DEG)
ATLAS	ALTITUDE	(± 2000 FT)	± 15.1	± 0.6	0
	VELOCITY MAGNITUDE	(± 15 FT/SEC)	± 125.3	0	0
	FLIGHT PATH ANGLE	± 0.1 DEG	± 18.0	± 2.2	0
	ORBIT INCLINATION	± 0.2 DEG	± 3.4	0	± 0.189
PKM 1 BURN	IMPULSE	$\pm 0.75\%$	± 226.7	0	0
	GRAIN TEMP VARIATIONS	+10, -15°F	+14.5, -21.8	0	0
	IGNITION TIME*	± 1 SEC	± 1.4	± 0.2	0
	PITCH ATTITUDE*	± 1.84 DEG	- 90.0 + 29.5	+ 4.9 - 6.8	0
	YAW ATTITUDE*	± 1.84 DEG	- 30.0	0	± 0.546
	NUTATION	1.2 DEG	- 6.6	0	0
PKM 2 BURN	IMPULSE	$\pm 0.75\%$	± 443.8	0	0
	GRAIN TEMP. VARIATIONS	+10, -15°F	+28.4, -42.6	0	0
	IGNITION TIME	± 4 SEC	± 7.7	± 0.6	0
	PITCH ATTITUDE	± 78 DEG	-46. +38.	+0.8 -0.8	0
	YAW ATTITUDE	± 78 DEG	-4	0	± 0.158
	NUTATION	2.53 DEG	- 57.7	0	0
RSS TOTAL			+517 / -530	+5.5 / -7.3	$\pm .60$

*Errors propagated through PKM 2 Burn

TABLE K-A.2-4
PREDICTED 3rd TRANSFER ORBIT ERRORS $H_e = 15 \text{ ft-lb-sec}$

PHASE	ERROR SOURCE	ERROR MAGNITUDE	ERROR EFFECTS		
			APOGEE RADIUS (km)	PERIGEE RADIUS (km)	INCLINATION (DEG)
ATLAS	ALTITUDE	($\pm 2000 \text{ FT}$)	± 15.1	± 0.6	0
	VELOCITY MAGNITUDE	($\pm 15 \text{ FT/SEC}$)	± 125.3	0	0
	FLIGHT PATH ANGLE	$\pm 0.1 \text{ DEG}$	± 18.0	± 2.2	0
	ORBIT INCLINATION	$\pm 0.2 \text{ DEG}$	± 3.4	0	± 0.189
PKM 1 BURN	IMPULSE	$\pm 0.75\%$	± 226.7	0	0
	GRAIN TEMP VARIATIONS	+10, -15°F	+14.5, -21.8	0	0
	IGNITION TIME*	$\pm 1 \text{ SEC}$	± 1.4	± 0.2	0
	PITCH ATTITUDE*	$\pm 1.84 \text{ DEG}$	- 90.0 + 29.5	+ 4.9 - 6.8	0
	YAW ATTITUDE*	$\pm 1.84 \text{ DEG}$	- 30.0	0	± 0.546
	NUTATION	1.65 DEG	- 12.5	0	0
PKM 2 BURN	IMPULSE	$\pm 0.75\%$	± 443.8	0	0
	GRAIN TEMP. VARIATIONS	+10, -15°F	+28.4, -42.6	0	0
	IGNITION TIME	$\pm 4 \text{ SEC}$	± 7.7	± 0.6	0
	PITCH ATTITUDE	$\pm .83 \text{ DEG}$	-50. +39.	+ 0.6 -0.8	0
	YAW ATTITUDE	$\pm .83 \text{ DEG}$	-5	0	± 0.168
	NUTATION	3.81 DEG	-130.8	0	0
RSS TOTAL			+516/-544	+5.5/-7.3	± 0.60

Error propagated through PKM 2 burn.

TABLE K-A.2-5
PREDICTED 3 σ TRANSFER ORBIT ERRORS

$H_{\ell} = 26.1$ ft-lb-sec

PHASE	ERROR SOURCE	ERROR MAGNITUDE	ERROR EFFECTS		
			APOGEE RADIUS (KM)	PERIGEE RADIUS (KM)	INCLINATION (DEG)
ATLAS	ALTITUDE	(± 2000 FT)	± 15.1	± 0.6	0
	VELOCITY MAGNITUDE	(± 15 FT/SEC)	± 125.3	0	0
	FLIGHT PATH ANGLE	± 0.1 DEG	± 18.0	± 2.2	0
	ORBIT INCLINATION	± 0.2 DEG	± 3.4	0	± 0.189
PKM 1 BURN	IMPULSE	$\pm 0.75\%$	± 226.7	0	0
	GRAIN TEMP VARIATIONS	+10, -15 $^{\circ}$ F	+14.5, -21.8	0	0
	IGNITION TIME*	± 1 SEC	± 1.4	± 0.2	0
	PITCH ATTITUDE*	± 1.84 DEG	- 90.0 + 29.5	+ 4.9 - 6.8	0
	YAW ATTITUDE*	± 1.84 DEG	- 30.0	0	± 0.546
	NUTATION	2.27 DEG	- 23.7	0	0
PKM 2 BURN	IMPULSE	$\pm 0.75\%$	± 443.8	0	0
	GRAIN TEMP. VARIATIONS	+10, -15 $^{\circ}$ F	+28.4, -42.6	0	0
	IGNITION TIME	± 4 SEC	± 7.7	± 0.6	0
	PITCH ATTITUDE	± 1.25 DEG	-78. +58.	± 0.8 -1.2	0
	YAW ATTITUDE	± 1.25 DEG	-12.	0	± 0.252
	NUTATION	4.97 DEG	-252.5	0	0
RSS TOTAL			+517/-566	+55/-7.3	+0.63

* For propagation through PKM 2 burn.

It therefore appears that lateral oriented machines in the GPS do not cause unacceptable orbit trajectory errors.

References

- Reference (1) Report No. 309-CD-0501 " Stage Vehicle System, Subsystem Design Analysis Report, Stability and Control Analysis CDRL No. A005, dated 28 May 1975, and prepared for SAMSO under Contract Number F04701-75-C-0028, by Fairchild Space and Electronics Company.
- Reference (2) Report No. 309 - CD-0601 " Stage Vehicle System, Subsystem Design Analysis Report Error Analysis " CDRL No. A006, dated 30 May 1975, and prepared for SAMSO under Contract Number F04701-75-C-0028, by Fairchild Space and Electronics Company.

K-A.3 Reaction Wheel/Propellant Weight Tradeoffs

K-A.3.1 Introduction

A candidate revision of the GPS space vehicle may contain a laterally - oriented rotating machine.

At launch into its drift orbit, the vehicle is spun up about its longitudinal axis in an attempt to ameliorate the misalignment effects of the drift orbit insertion engines. It is shown above that at drift orbit apogee the resultant motion of the vehicle, after nutation has damped out, is a rotation at constant speed about a vehicle-fixed axis displaced (by angle α) from the vehicle longitudinal axis. This displacement causes a loss in velocity increment from the orbit insertion system engine which thrusts along the longitudinal axis.

A trade-off now exists between the propellant weight increment necessary to make-up for the velocity loss and the weight of a reaction wheel, aligned with rotating machine spin, and operating so as to decrease the angle offset α .

This study considers this trade-off relation.

K-A.3.2 Analysis

K-A.3.2.1 Propellant Requirements

Reference (1) evaluates the velocity loss ΔV , in the normalized ratio $\Delta V/V$, (V = total velocity increment), as a function of lateral axis angular momentum

H_e. These results are shown in Table K-A.3-1. Here, the value of the total velocity increment is first calculated. Then, the fuel weight necessary to increment the velocity is calculated by using the Rockwell estimate of propellant capability at 3 lb. per 23 ft/sec.

The vis-viva integral for elliptical orbits about the earth is:

$$v^2 = \mu \left(\frac{2}{r} - \frac{1}{a} \right)$$

where v = velocity in orbit

μ = gravitational constant for the earth

r = orbit radius

a = orbit semi-major axis

Drift orbit perigee = earth radius + 77 nm

$$\begin{aligned} r_p &= 3438 + 77 \text{ nm} \\ &= 3515 \text{ nm} \end{aligned}$$

Drift orbit apogee = earth radius + 10818 nm

$$\begin{aligned} r_a &= 3438 + 10818 \text{ nm} \\ &= 14,256 \text{ nm} \end{aligned}$$

$$\text{Semi-major axis} = \frac{3515 + 14256}{2} \text{ nm}$$

$$a = 8885 \text{ nm}$$

$$\frac{\text{transfer orbit apogee velocity}}{\text{circular orbit velocity}} = \frac{V_{xa}}{V_c}$$

$$\begin{aligned} &= \left[\mu \left(\frac{2}{r_a} - \frac{1}{a} \right) \right]^{1/2} \\ &\quad \left[\mu \left(\frac{2}{r_a} - \frac{1}{r_a} \right) \right] \\ &= 0.63 \end{aligned}$$

$$\text{Therefore, } \Delta V = 0.37 V_c$$

$$\text{Also } V_c = 12752 \text{ ft/sec.}$$

So that the increment for circularization ΔV is 4718 ft/sec.

Table K-A.3-1 contains the numerical values of interest.

Table K-A.3-1

H ℓ ft. lb. sec.	$\Delta V/V_c$	ΔV ft./sec.	Fuel Weight lb.
1.0		0.7	0.09
5.0	.0008	3.78	0.49
10.0	.0030	14.15	1.85
15.0	.0068	32.08	4.18
20.0	.0120	56.62	7.39
25.0	.0190	89.65	11.69

Values of propellant weight versus net angular momentum are plotted in the graph of Figure K-A.3-1

K-A.3.2.2 Reaction Wheel

Figure D.3-2 contains an accumulation of current reaction wheel weight versus angular momentum capability. A curve matching these points is indicated.

Numerical values from Figure K-A.3-1 are replotted in Figure K-A.3-2 so as to show the reduced propellant requirements when a reaction wheel cancels some of the lateral rotating machine's angular momentum (R.M.). For example, if R.M. = 20 ft. lb. sec. then 7.4 lb. of propellant are required. If a 4 ft. lb. sec. reaction wheel weighing 16.2 lbs, is added the propellant requirement is reduced to 4.7 lbs.

K-A.3 Results

Over the range of net angular momenta shown in Figure K-A.3-2 there is no net weight saving when a reaction wheel is used to cancel some of the rotating machine's angular momentum. An uncompensated rotating machine should be used. If a reaction wheel is required, perhaps for attitude control reasons, it should have the smallest size suitable to meet those needs. Any larger size is unprofitable from a total system weight requirement view-point.

Figure K-A.3-1

RELATION BETWEEN
REACTION WHEEL WEIGHT
AND
ANGULAR MOMENTUM CAPABILITY

CIRCLED POINTS ARE
UNITS ALREADY IN EXISTENCE

Figure D.3-2

VARIATION IN
INCREMENTAL ΔV FUEL REQUIRED DUE TO
OFFSET SPIN AXIS RESULTING FROM
LATERALLY ORIENTED ROTATING MACHINE (RM)
AS A FUNCTION OF
COMPENSATING REACTION WHEEL
WEIGHT AND ANGULAR MOMENTUM

WEIGHTS - FUEL OR REACTION WHEEL - LBS

WHEEL ANGULAR MOM. CAPABILITY FT-LB-SEC

$$H_{\text{ROT MACH.}} - H_{\text{REAC WHL}} = \text{NET ANGULAR MOMENTUM}$$

APPENDIX K-B
THERMAL ANALYSES

Analyses pertaining to RTG, ORC and BIPS radiator sizings were performed and Teledyne, Airesearch, and Sunstrand's sizing of RTG, ORC, and BIPS radiators to handle 2400 watts heat source have been verified. Parametric data on radiator designs to handle both 2400 and 3200 watt heat sources are given by figures K-B-1 through K-B-5. Data given is preliminary and is to be consider as "first cut estimates " for radiator sizing.

Teledyne's concept of using heat pipes in the radiators to cool the RTG system were evaluated first. Teledyne proposed using copper/water, grooved heat pipes. The pipes are spaced approximately 2.70 inches apart on a 0.02 aluminum panel. In order to stiffen the radiator panel, tapered ribs (0.02 Alum) are used. The pipes are located at the junction between the ribs and panel. The evaporator length of pipe was designed to eliminate the possibility of nucleate boiling of the water. The fundamental heat flux limit of the evaporator is a maximum of 476,000 BTU/°R-ft² (Reference Dynatherm Heat Pipe Design Handbook). The evaporator heat flux in the proposed design is:

$$(q) \text{ evap.} = 170.0 \times 3.414 / \frac{(3/16) \Delta T \ 3.25}{144} = 44,644 \text{ BTU/}^{\circ}\text{R ft}^2$$

The original configuration had a number of bends in the heat pipe and the evaporator section turned in toward the spacecraft's spin axis. During transfer orbit (45 hours), the GPS spins at 100 RPM in order to maintain stabilization. At this spin rate, approximately 10g centrifugal acceleration is acting on the water inside the pipes. This high centrifugul acceleration forces the water into the pipe bends and also prevents the liquid from reaching the evaporator.

Typical Pipe Layout

The phenomenon shown above can be demonstrated mathematically. The amount of pressure head needed to pump the water against 10g can be computed.

$$(\Delta P)_{needed} = \rho \omega^2 / 2g (l_1^2 - l_2^2) = 60(10.47)^2 / 2 \times 32.2 (2.67^2 - 2.42^2) = 1310 \text{ ft}$$

However, the approximate capillary pressure is:

$$(\Delta P)_{available} = 2\sigma \cos \theta / w$$

$$\theta = \text{contact angle} = 0$$

$$w = \text{is assumed groove width (ft)} = 0.0025$$

$$\sigma = \text{surface tension of } H_2O = 3.4 \times 10^{-3} \text{ lbs/ft.}$$

$$\therefore (\Delta P)_{available} = 2 \times 3.4 \times 10^{-3} / 0.0025 = 2.8 \text{ lb/ft}^2$$

Obviously, the fluid cannot move against such a high acceleration.

Fairchild proposes that the sharp bends be removed and a constant large radius bend be put into the pipes, and that the evaporator end of the pipes be bent outward. Also, in order to improve the radiator efficiency, the ribs should be moved to the outside of the panel. A thermal math model was constructed that represents the proposed configuration. (See Figure K-B-1. Also, given in Figure K-B-1 are the detailed steady state temperatures ($^{\circ}\text{F}$) along a typical pipe, rib, and fin composite. Figure K-B-2 shows the temperature gradient along the pipe wall in relation with the vapor temperature. As the assumed radiator and heat pipe radius is increased, the vapor temperature in the pipes decrease. This general relation is shown in Figure E-3 for no sun and full sun conditions and for 2400 and 3200 watt heat sources. In order to minimize weight, the radiator should be sized as to give a steady state vapor temperature of approximately 300°F . The heat transport factor of water reaches its maximum value at a temperature of 300°F . (Reference Dynatherm Handbook, Part 1, Page F-33).

Airesearch's concept of a deployable radiator could actually be used for both the BIPS and KIPS. Figures K-B-4 and K-B-5 are general working curves for a 72 in. diameter stationary radiator plus deployable radiators of various lengths. The height of stationary and deployable radiators is assumed to be 23.9 inches. Both the stationary and deployable radiators are assumed to be completely covered with Z-93 thermal control paint. In both Figures K-B-4 and K-B-5, the radiator temperatures are given as a function of Gross Cycle Efficiency. The Gross Cycle Efficiency for both the BIPS and KIPS varies between 15 and 20%.

Depth of Model = 2.7 m.

FIGUREK-B-1.PROPOSED HEAT PIPE LAYOUT FOR RTG SYSTEM

FIGURE K-B-2. TYPICAL TEMPERATURE GRADIENT ALONG RADIATOR HEAT PIPE

LCHPGI INSTALLATION ON GPS

HEAT PIPE VAPOR TEMPERATURE vs RADIATOR RADIUS

FIGURE K-B-3

KIPS & BIPS STOWED RADIATOR TEMPERATURES

----- 2400 Watt Heat Source Used
_____ 3200 Watt Heat Source Used

Note:

- All radiator surfaces covered with E-93
- Radius of radiators = 36 inches
- Height of radiators = 23.9 inches

FIGURE K-B-4

KIPS & BIPS DEPLOYED RADIATOR TEMPERATURES

FIGURE K-B-5
 K-88

This section presents preliminary findings from investigations which were conducted to evaluate various potential changes to the thermal control subsystem of the GPS. These changes are being studied as a part of the present study to substitute a nuclear power source for the existing solar powered subsystem. Significant to thermal control of a nuclear powered spacecraft will be the absence of the stowed arrays during transfer orbit, absence of reflected and I. R. inputs from the arrays to the radiator surfaces when the arrays are deployed, and the effect of a relatively constant and predictable heat load in the equipment housing.

Although final design parameters and many details of the GPS configuration are unknown, it is possible to make comparisons and discuss the relative merits of the various alternatives.

Current Design

The GPS louvers are furnished to Rockwell by Fairchild Industries. There are seven louvers per ship; each measures 15.6 x 16.54 inches for a total area of 12.53 ft². Effective emittance in the open position is specified at 0.70; in the closed position it is 0.14. Since these values are with respect to a black baseplate ($\epsilon = 0.9$) and Rockwell is using a silver teflon radiator ($\epsilon = 0.76$); the corrected values become

$$F_{\epsilon} = 0.7 \frac{.76}{.9} = 0.591 \quad \text{Open}$$

$$F_{\epsilon} = 0.14 \frac{.76}{.9} = 0.12 \quad \text{Closed}$$

The louvers are mounted to the plus and minus Y facing sides which, because of solar syn. yaw control, receive no solar input during orbit. Neglecting inputs from the extended arrays, the maximum heat rejection through the louvers at 90°F is

$$Q = 0.591 \frac{12.53}{3.413} \sigma T / ^4 = 340 \text{ watts} \\ 550^\circ \text{ R}$$

(In the closed position, 55 watts will suffice to maintain 60° F.)

Elimination of Yaw Control

Yaw control is necessary to maximize power output from the solar arrays. It conceivably could be eliminated with the substitution of a nuclear source and an analysis was conducted to evaluate the effect on thermal control of the equipment items inside the spacecraft.

The effective solar absorptance of a typical louver assembly as a function of blade and solar angle may be taken from Figure K-B-6 attached. Assuming a rotisserie mode (full rotation) and hot case with the blades open, the 90° blade angle curve may be used to obtain an average solar absorptance of 0.169 during a half cycle when the louvers receive sun. Integrated over a full revolution, the absorbed solar flux on the radiator baseplates is 136 watts and the maximum heat rejection through the louvers is reduced to 204 watts. It is apparent, then, that the present louvers in a rotisserie mode will have their capacity reduced about 40% and additional louvered and/or passive radiating area must be incorporated to accommodate a rotisserie mode. If, for example, the additional area is all passive, 8.9 ft² would have to be added.

Removal of Louvers

If the louvers are removed, the present radiators will provide adequate area to reject the component waste heat during a rotisserie mode. Either of two coating systems, the silver teflon presently being employed behind the louvers or the more stable OSR's (Optical Solar Reflectors), might be considered. At 90°F, the use of the silver teflon will provide a rejection capability of 365.4 watts ($\alpha = 0.14$, $\epsilon = 0.76$) whereas the OSRs will reject 386 watts ($\alpha = 0.2$, $\epsilon = .85$).

FIGURE K-B-6

There are several additional considerations which should be carefully evaluated if the louvers are to be removed. The first concerns the average baseplate temperature change as a function of heat dissipation. Since the louvers provide a variable radiation resistance as a function of temperature, it is obvious their removal will result in a larger heater power requirement during a given "cold case" or, alternatively, a larger temperature drop per unit change in heat generation as the heat generation varies from the design point. This may be shown graphically by considering two radiators, one covered with louvers and one without, but each designed to maintain 90°F at a given dissipation. When the dissipation decreases below the design point, the decrease in temperature per unit change is much greater for the unlouvered radiator. The two curves are shown in Figure K-B-7.

A second concern is related to the transient behavior of a given radiator (and components thermally coupled thereto) during a given rotation about the yaw axis. Figure K-B-8 shows the results of an analysis in which a large, 300-lb equivalent mass radiator was assumed to be rotating at one revolution per 12 hours about an axis normal to the sun. The minimum to maximum temperature variation is about 19°F and this is probably unacceptable for temperature control of the frequency standard clock. (The clock is thermally coupled to a louvered radiator by means of 0.1 inch doubler strap and the orbital temperature variation of the radiator which, in this case is the heat sink, is very significant to the clock's temperature control). Of course, the curve shown in the figure represents a particular case and additional runs should be made in which the two significant parameters, effective mass and rotational velocity, are varied.

A third consideration relates to the thermal coupling between the components and various radiators on the Y facing sides. As presently designed, any imbalance in heat distribution will be partially compensated for by the thermal louvers which, by virtue of their independent actuation, can attain different blade positions and vary heat rejection to space in accordance with the temperature of a particular radiator. In the absence of louvers, temperature gradients should be evaluated along and between radiators on both sides.

FIGURE K-B-7

Transfer Orbit

During the transfer orbit, the arrays are stowed so as to nearly surround the louvered radiators and provide an additional radiation resistance to space. This, of course, is necessary for thermal control of the electronics since the power input from the arrays is small and the associated low dissipation mode tends to result in low component temperatures.

Using the current configuration, it is computed that in a cold extreme (spin axis aligned with the sun and louvers closed) 42 watts will be required to maintain a mean radiator temperature above 40°F . If the louvers and arrays are removed, the same environment will dictate the necessity for providing 334 watts to maintain 40°F on the radiators. However, because of the relatively constant output of most nuclear energy sources, it is not anticipated that this will impose a problem.

K-B-3 Heat Soakback Into GPS Structure

Heat soak-back into the GPS basic structure from the RTGs was estimated. Teledyne's design concept was used as a baseline configuration for this study. The assumed 3200 watt RTG attachment arrangement is shown in Figure K-B-9. The upper fitting and lower truss supports the RTG laterally while the top cap fitting carries the vertical loads. In order to minimize the conducted heat input to the structure from the RTGs, epoxy glass isolators should be used between the RTG and attachment fittings. It appears that the most feasible mounting configuration is to attach the upperfittings to the RTG's upper end cap and the lower truss to a support ring. The louver truss cannot readily be attached to the RTG lower end cap due to the RTG's overall length. Both Fairchild and Teledyne agree that approximately 1.6% of the RTG's total heat is lost through the end caps. Therefore, a total of 25.6 watts/RTG ($0.016 \times 3200/Z$) was assumed to be conducted through the two upper fittings and radiated from the cap to space.

The support ring was assumed to have the same temperature as a typical radial fin (300°F). The total amount of heat being conducted into the structure was calculated parametrically as a function of structural temperature. Twenty-six watts is conducted into the structure from the RTGs when the structure is approximately sixteen watts as shown by figure K-B-10.

FIGURE K-B-8

Heat soak back into GPS structure from RTGs

General data of mounting fittings

- Lower trusses are made of 5/8" O.D. tubing, 2024-T6 Al, length = 10.0 in.
- Upper fitting has a cross section area of 0.42 in², length = 4.0 in, Mat'l: 2024-T6 Al
- Upper Cap fitting has an effective cross section area of 0.64 in², length = 1.0 in, Mat'l: 2024-T6 Al.

Conductance calculations

Upper cap:

$$KA/\Delta x = 109.2 \times 0.64 / 12 \times 1 = 5.82 \text{ BTU/hr}$$

Upper fitting

$$KA/\Delta x = 109.2 \times 0.42 / 12 \times 4 = 0.91$$

Lower truss

$$KA/\Delta x = 109.2 \times \pi (5/8)^2 \times 0.035 / 12 \times 1 = 0.125$$

FIGURE K-B-9 Assumed Attachment Conf.

Thermal Isolator Insert

Let thickness of epoxy glass = 1/16 inches

$$d = 3/16$$

$$l (\text{in}) = 0.64$$

Epoxy Glass Insert

Epoxy Glass Spacers or Washers = 1/2 O.D.; 3/16 in

Total conductance through one interface

Total possible contact area between (1) & (2)

$$= \frac{2\pi (1.5^2 - 0.127^2)}{4} = 0.34 \text{ in}^2$$

Total possible contact area between (1) & (2)

FIGURE K-B-10

Total Heat Conducted into GPS structure
from the Two RTGs

The soak-back heat to the structure from the heat sources used in the BIPs and KIPs concepts is less than determined in this analysis. With the solar arrays removed, the present GPS louvered areas (14.2 ft^2) will be able to dump an additional 60.0 watts at 70°F . Therefore, heat conducted from the RTGs, KIPs heat source, and/or BIPs heat source to the GPS structure is not expected to be a problem.

K-B-4 Frequency Standard Temperature Variation

The Rockwell International GPS spacecraft design has the Frequency Standard mounted directly to a louvered honeycomb panel. The temperature of panel/Frequency Standard interface varies by approximately 3°C during max eclipse orbit. This variation of 3°C is within the thermal specification. However, the performance of the GPS Navigation equipment could be improved if this interface temperature variation could be reduced.

The purpose of this memo is to show that the frequency standard mounting surface temperature variation during a max eclipse orbit is caused mainly by radiation coming from the solar array. This variable source of external heat is both IR and reflected solar energy coming from the deployed solar array. Fluctuations in the IR and reflected solar energy levels to the louvered panel is caused by rotation of the solar arrays and by the earth eclipse.

Figure K-B-11 shows the total solar array radiant heat absorbed by all seven GPS louvered panels over the full range of the louver operating conditions. The absorbed heat is given as a function of panel temperature and time. After the neat heat loss was determined for the louvered panel in which the frequency standards are mounted, the net effective emissivity of the panel was determined. (See Figure K-B-12.) A thermal math model was constructed to represent the frequency standard/louvered panel composite. Assuming a constant thermal dissipation in the Frequency Standard of 18.6 watts and varying the effective thermal mass of the composite parametrically, the orbital temperature of the frequency/standard/panel interface was estimated. As Figure K-B-13 shows, the effective thermal mass (mC_p), has to be approximately $1.8 \text{ BTU}/^\circ\text{F}$ in order to obtain a mounting face temperature variation of 3.0°C . When considering the $mC_p = 1.8$, a variation of $\pm 1^\circ\text{C}$ can be attributed to the fact that the arrays are rotating and a $\pm 2^\circ\text{C}$ variation can be attri-

FIGURE K-B-11 Total Absorbed Heat* to GPS Louvered Areas

During
Noon Orbit

* IR ϵ Reflected Solar Heat from Deployed Solar Arrays

FIGURE K-B-12

Effective Emissivity of 4reg. Std. Louver
During Moon Orbit

buted to earth shadow. An effective thermal mass of $1.8 \text{ BTU}/^{\circ}\text{F}$ is equivalent to 9 lbs. of aluminum. The frequency standard unit is designed thermally like three concentric ovens (one inside the other). The weight of the outer oven plus the louvered panel is approximately nine pounds.

This study has shown that the frequency standard mounting base temperature variation ($\pm 3^{\circ}\text{C}$) that the Rockwell GPS design experiences during max eclipse orbit is caused mainly by the solar arrays. The variable external heat input to the louvered panels and therefore the orbital variation of the frequency standard temperature could be dramatically reduced by eliminating the solar arrays and substituting a fixed nuclear power system.

21.7

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Estimate of Frequency Standard Temp. Variation During Noon Orbit

FIGURE K-B-13

18.9

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