**BIPS-FS
PRELIMINARY DESIGN
MISCELLANEOUS NOTES**

76-312000

June 15, 1976

MASTER

AIRESEARCH MANUFACTURING COMPANY OF ARIZONA
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PREFACE

This is a compendium of flight system preliminary design internal memos and progress report extracts for the Brayton Isotope Power System Preliminary Design Review to be held July 20, 21, and 22, 1975. The purpose is to bring together those published items which relate only to the preliminary design of the Flight System, Task 2 of Phase I. This preliminary design effort was required to ensure that the Ground Demonstration System will represent the Flight System as closely as possible.

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2.2 Task 2 -- IBFS Preliminary Design

2.2.1 Layout Drawings

Conceptual IBFS layouts are being generated for various engine and heat source orientations within both the Atlas/Centaur and Titan IIIC launch vehicles. More detailed work in this area will be accomplished upon receipt of further launch vehicle definition.

2.2.2 Control System

Controls effort is being directed toward defining the basic functional requirement with emphasis on review of the NASA LeRC controls for applicability. The alternator field configuration and voltage regulator section have been reviewed.

Alternator Field Configuration

A Failure Mode Effects Analysis (FMEA) was conducted to investigate several methods of connecting and energizing the field windings of the Isotope Brayton Flight System Alternator in order to determine the method that will provide the most reliable system. Four configurations were found to be suitable from a system reliability standpoint. Of these, two appeared easiest to implement electronically. These two configurations were then further compared with regard to impact on the total system.

The most reliable method of connecting and energizing the four field windings is as shown in Figure 2. If any single winding fails open, the other three provide parallel redundancy. Similarly, a shorted winding will blow the corresponding fuse with the same result. A failure of any one switching regulator (S1 through S4) will not cause a system failure because adequate excitation is provided by the other pair of windings. Field excitation can be provided by means of a simple, transformerless rectifier network using line-to-neutral voltage from the alternator.

Details of this analysis were documented for inclusion in the system reliability analysis.

ALTERNATOR FIELD WINDING CONNECTIONS

FIGURE 2

Voltage Regulator

The technical proposal for the Isotope Brayton Ground Demonstrator, 74-311015-2, Volume 2, proposed an on-off voltage regulator using a pulse-width-modulated system operating at a frequency that is fixed by the carrier generator (oscillator). That system employs redundant channels for reliability.

A study was conducted to investigate an on-off voltage regulator that eliminates the carrier generator and operates at the natural frequency of the closed loop. It is inherently more simple, with ultimate promise of higher reliability. This study was limited to investigating two schemes of "bang-bang" regulation from the standpoint of reliability such that no single component failure could result in a system failure. This study did not consider the adequacy of such a regulator from the standpoint of performance; this latter aspect will be the subject of a separate investigation.

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Two voltage regulator systems (one comprising 3 channels and 6 drivers and the other comprising 4 channels and 4 drivers) were considered. Each of these satisfied the criterion that no single component could cause system failure. All failure combinations that would cause a system failure were then tabulated for each scheme. Failure rates were assigned to various portions of the circuitry, based on similarity to previously known circuits. Calculations for each scheme yielded MTBF's of 0.33×10^{10} hours for one and 0.31×10^{10} hours for the other. For all practical purposes, these are equally reliable, but one appeared more desirable than the other since fewer drivers are used and lighter weight would be realized.

Details of this analysis were documented for inclusion in the system reliability analysis.

System Analysis

The initial work on this program has begun and consists of directing the recuperator analysis being performed by AiResearch-Los Angeles. The initial transient analysis of the system was presented in AiResearch Report No. 75-311279. This report is being analyzed in terms of start-up thermal stresses which will be imposed on the heat exchanger. Initial indication is that stresses may be severe, however, a detailed analysis is now being performed.

Data generated using an existing computer program will be compared with laboratory data for confirmation of accuracy. Also, the recuperator description for the transient program has been modified and will be run as a separate program for comparison. Further work in controls definition may then be accomplished using the transient computer program.

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2. TECHNICAL PROGRESS SUMMARY

2.2 Task 2 - IBFS Preliminary Design

2.2.1 Preliminary Design Trade-Off

A logic diagram has been completed acknowledging preliminary design trade-offs required to firm IBFS design specifications without including known design requirements of the various components. The diagram illustrates probable design logic paths for the IBFS with regard to launch and satellite environment, decisions to be made with each path, and alternatives associated with each design decision.

The preliminary design logic diagram and trade-off discussion are included in Appendix I.

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2.2.2 Recuperator Analysis

Analyses were continued to define recuperator behavior during the system startup transient. For this purpose, startup transient is the case where all components are initially at 450°R except the cooler (390°R) and heat source (2060°R). Speed acceleration will be from 0 to 100 percent in 4.5 seconds. Goals of these analyses are to: (1) predict steady-state and transient temperature distribution throughout the recuperator; (2) compute resultant distortions and thermal stress in critical areas; and (3) determine recuperator low-cycle fatigue life when subjected to the startup transient.

During this reporting period, steady-state and transient temperature distributions were computed for the assumed startup conditions. An analysis was initiated to compare predicted performance with experimental data obtained from a recuperator submodule test. Analyses are also being conducted to determine transient conditions required to utilize air to simulate Xe-He performance. The goal is to establish a transient air inlet condition that will produce approximately the same thermal response at the critical time as predicted for Xe-He.

An existing finite element thermal stress model (ANSYS program) will be utilized to simulate the recuperator. This program uses temperature data described above and computes distortions and thermal stresses in the heat exchanger manifold and core areas. Input was prepared to compute steady-state deflection of the mounting points and duct interfaces. This data is required for design of the component mounting scheme and interconnecting ducting.

2.2.3 Radiator Analysis

A conceptual design of the IBFS radiator was started. The baseline configuration was established to be a cylindrical radiator which is suitable for launch on the Titan IIIC or Space Shuttle. The Atlas Centaur was deleted as a possible launch vehicle.

2.2.4 Controls and Accessories

2.2.4.1 Alternator Field Fusing Schemes

During the reporting period, an investigation concerned with the MTBF of various alternator field fusing schemes was completed. Figure 2 shows the optimum fusing scheme considering the present alternator field configuration. Results of this study will be included in the system MTBF analysis.

FIGURE 2

2.2.4.2 Output Voltage Study

A qualitative study of various output voltages that are easily obtainable from the BIPS alternator/transformer/rectifier was conducted. These were then compared on the basis of reliability, weight, efficiency, and flexibility. Major points of this study are included in Appendix II.

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2.2.4.3 Speed Control

A linear description of the speed control loop for BIPS was created for programming on the 680 analog computer. Some preliminary data has been obtained and refinements of the control are in process. This effort will continue into the next reporting period.

2.2.4.4 Flight System Controls

Work is proceeding to define the conceptual flight system. Consideration has been given to a parasitic load consisting of 5 loads weighted to yield 16 coarse adjustments of load and 1 pulse-width modulated load of 150 watts. Such a system would minimize current transients caused by parasitic load switching and reduce maximum current in any one load to approximately 10 amps which is easily and reliably handled by current technology semiconductors. The configuration being considered is a high voltage (120 vdc) parasitic load controlled by switching transistors. The parasitic load and associated three-phase full wave rectifier would be connected directly to the alternator output.

Preliminary loop gain selections for both the major speed loop and minor power control loop have been made and will be evaluated on the 680 analog computer during the next reporting period.

2. TECHNICAL PROGRESS SUMMARY

2.2 Task 2 - Conceptual Flight System Design

2.2.1 Controls

During the reporting period, work continued toward defining a linear model to represent the Mini-BRU alternator in the computer analysis.

PROGRAM SCHEDULE AND MILESTONES

FIGURE 1

75-311457 (03)
Page 4

Work on Flight System definition was terminated and summaries of the completed work have been prepared. The summaries will be published during the next reporting period.

An engineering report entitled "Speed Control Loop for BIPS" was published on September 16, 1975. This report describes a candidate speed control system configuration for use on the BIPS GDS. The features of this speed control system include a power control minor loop and a parasitic load with digital control. The power control minor loop is intended to maintain total output power within five percent of a nominal setting as customer load varies between zero and full output. The digital parasitic load control is intended to minimize large current surges due to parasitic load switching. The parasitic load and load control will switch a maximum of 100 watts whenever necessary.

The performance capabilities of a simple "bang-bang" (i.e., either full on or off) voltage regulator and a "bang-bang" parasitic load speed regulator were analyzed by simulating these controls and the engine-alternator combination using the EAI Model PACE 680 Electronic Analog Computer.

This computer simulation showed voltage regulation was inadequate, yielding peak-to-peak excursions of approximately 10 volts on the 120 vdc line. The results of this study are being documented in an engineering report.

2.2.2 Preliminary Design Layouts

Principal activity during the reporting period was the production of the basic layouts of the Flight System for (a) spin stabilized and (b) 3-axis stable operation. A five-foot diameter radiator was chosen based on weight and compatibility with the FS components. Drawing L3621581 shows the basic component arrangement of the power conversion

system. Sheet 2 of Drawing L3621581 shows the relative thermal expansions and accumulative tolerances between the mount points and the ducting interfaces of the individual components.

Drawing L3621584 shows a proposed mounting scheme for the HSA subsystem which could meet the requirements of supporting the HSA and allowing for thermal growth at each mount. Mounts 1, 2, and 3 allow for radial growth of the HSA while fixing one end. Mount 4 allows axial growth of the HSA while supporting the cantilevered end of the HSA in the plane normal to the centerline of the HSA.

Drawing L3621585 shows the recuperator/Mini-BRU mount system. This consists of a bracket fabricated from 0.062 in. aluminum sheet, in the form of a box, for maximum strength and rigidity commensurate with light weight. The Mini-BRU interfaces with the ring on top of the bracket by means of a pilot surface and eight No. 10-32 bolts. The recuperator is mounted below the bracket by using a thermal isolation system to minimize heat loss from the recuperator. This system is shown in more detail on Sheet 2 of Drawing 3621585.

Mount 1 on the cool end of the recuperator takes loads in the X, Y and Z planes through a tubular cross-section strut which is also designed to reduce thermal loss by conduction (by having a small cross-sectional area). Mount 3 has the same design as Mount 1, but it sustains loads only in the Y and Z planes and allows for thermal expansion of the recuperator in the X direction.

Mount 2 on the hot end of the recuperator is designed to sustain loads in the Z and X planes while being sufficiently flexible to allow thermal growth of the recuperator in the Y direction.

The mount has low thermal conductivity due to the small cross-sectional area of the rods. Mount 4 is designed to sustain loads in the Z plane only, allowing for thermal expansion in the Y and X directions.

The recuperator/Mini-BRU mount bracket also incorporates the interface of Mount 4 of the HSAs.

The recuperator/Mini-BRU mount bracket will be supported by rods attached to the approximate mid-point of the radiator and the upper support structure. The upper support structure interfaces with the top of the radiator and supports the HSAs shown on Drawing L3621586.

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APPENDIX I

PRELIMINARY DESIGN TRADE-OFFS FOR BRAYTON ISOTOPE POWER SYSTEM

INTRODUCTION

The logic diagram (Figure 1) was constructed similarly to a computer-programming flow chart. The flow of charted events is from left to right, although the actual events need not be decided in that order. Information is given in rectangles, customer desires are noted in diamonds and have two possible decision paths, and octagons require a decision by the ERDA (and/or AIRPHX) to be made as to whether to proceed along a given path or not. The path of least resistance is that which does not require a major decision to be made (no octagons are encountered). The purpose of parallel paths (launch and satellite requirements) is to focus on the requirement of launch and satellite configuration compatibility.

DISCUSSION

Launch Requirements

Presuming that operation during launch is not a design requirement, battery power from the booster or the shuttle is required for startup in final orbit. It may be practical to start the Mini-BRU in the parking orbit prior to transfer from parking orbit to final orbit. Checkout on the launch pad is highly desirable in the event that the BIPS is not operated during launch. This requires that the HSA heat rejection be controllable; i.e., a method of changing from a standby mode to operational and back again prior to liftoff is required.

Checkout on the ground or operation during launch requires that the refractory-metal ducts from the HSAs and the turbine-plenum chamber and/or the foil insulation must be protected from reducing agents and oxidation (nitriding, sulphidation, etc). Since either ground checkout or operation during launch is deemed necessary to prove system integrity prior to liftoff, extra weight and system complication will probably be required. The extra weight involved would be minimal if the duct and turbine plenum were enclosed in a sheet-metal shroud over foil insulation that retained an inert atmosphere at a small positive pressure. To furnish the maximum anticipated on-pad (and presumably launch as well) power requirement of $320 W_e$, it may be necessary to place some conventional insulation between the foil insulation (effective only in a space/vacuum environment) and the sheet-metal shroud. Tradeoff studies are necessary to determine if this shroud must be partially evacuated. To complete the shroud-insulation system, a rupture diaphragm would be installed in the shroud to dissipate inert gas when a vacuum of approximately 5 in. Hg is reached during the launch. Further travel into the hard vacuum of space, plus heat from operation of the BIPS under these conditions, would evacuate the foil insulation which would enable fulfillment of design requirements.

Effective control of the HSHX insulation short-circuiting inert gas is required to control HSA-heat rejection during checkout or operation during launch. This control would consist of either replacing a high-thermal conductivity gas (such as helium) with a lower-thermal conductivity gas (such as argon) or a complete or partial evacuation of the short-circuiting inert gas. If ground checkout only is desired, high-thermal conductivity inert gas must be re-introduced.

Maximum stated preliminary design sustained g-loads for the Titan IIIC* occur at the Stage II shutdown; these are -10 g, +8 g longitudinal and $\pm 6 g$ ~~axial~~. If the Mini-BRU is to be mounted at some

Rationally

*GE-BIPS-20-001, "Mission Environments for the Isotope Brayton Flight System (Preliminary)," dated August 8, 1975, Paragraph 3.1.3 and Table 3.1-1.

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BIPS LOGIC AND TRADEOFF ANALYSIS DIAGRAM
FIGURE 1

distance away from the axis of a space vehicle that is spin stabilized, the centrifugal force thus imposed on the rotor (Figure 2) must be considered. To allow for this, a ± 7 g loading is shown in the Figure 1 logic chart (assuming transverse loads are taken by the journal bearings).

If satellite considerations indicate that the Mini-BRU is mounted horizontally (discussed below), the journal bearings must be designed for ± 10 g's.

Satellite Requirements

When decisions regarding launch mode are made, a parallel set of decisions remains to be resolved regarding operation of the satellite. Most of these decisions involve the effect of vehicle configuration on orbital-operation requirements. A salient requirement of any satellite is that, if any part of the vehicle system is rotating (such as the Mini-BRU or the entire satellite), the axis of rotation must be orientated perpendicular to the plane of orbit. If the BIPS is not operated during launch, there is no requirement for the Mini-BRU journal bearings to be redesigned to 7 or 10 g's; centrifugal force exerted by a spin-stabilized satellite (Figure 2) will generally be below 1 g, thereby not requiring bearing redesign.

For a spin-stabilized vehicle (launch or orbital) to avoid wobbling or a torroidal-flight path, the BIPS must be dynamically balanced. Dynamic balancing requires that components be placed so that moments are opposed by equal and opposite moments.

The present turbine-plenum design with opposing entrance ducts is critical in creep strength. Any change of duct direction will result in excess bending stresses being imposed on the plenum due to duct expansion, g-loads (launch conditions only), and gas pressure loads. Therefore, since dynamic balancing involves component placement in the

CENTRIFUGAL FORCES IN A SPINNING VEHICLE

FIGURE 2

manner previously described, the ducts may be required to enter the turbine plenum at an included angle of approximately 120 - 150 degrees, necessitating a turbine-plenum redesign. Every effort will be expended to place the components so that a redesign will not be necessary, however.

For the spin-stabilized vehicle, a possible compatibility problem exists with the antennas. By the time the BIPS is in orbit and the Mini-BRU axis (and the satellite axis as well in the case of a spin-stabilized satellite) is oriented normal to the plane of orbit, special design considerations will be required to avoid making antenna orientation a potential problem.

For the three axis stabilized system, care must be taken to avoid uncompensated reactive torques from the rotating group as a result of load changes. An isochronous control will minimize this problem; however, additional analysis will be required once the mission and vehicle are defined.

View factor of the radiator toward the earth is maximum when the radiator axis is horizontal, and minimum when vertical. Preliminary calculations indicate that the difference between effective-sink temperatures will be minimal, however.

Compatibility Requirements

The point emphasized under this heading is that a given configuration which satisfies launch requirements may not satisfy satellite requirements. The most difficult compatibility problems arise when spin stabilization of the satellite becomes a requirement. These will become more evident when Figure 1 is studied in detail; a more complete discussion may be found in the following Conclusions and Recommendations.

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CONCLUSIONS AND RECOMMENDATIONS

A general conclusion is that the path of least resistance (one that avoids all "stop signs" on Figure 1) provides a reliable system with the least redesign and compatibility complications. To ensure that the BIPS is in good working order, it seems appropriate to assume that the BIPS is operated on the launch pad for checkout purposes prior to liftoff. To accomplish this, the HSA-heat rejection must be controllable and heat loss from the hottest ducts must be minimized commensurate with proper protection of the duct and turbine-plenum material and the need to protect insulation from oxidation.

The journal bearing size need not be increased to accommodate the 7-g requirement imposed in the case of a vertical Mini-BRU but would require stiffer foil leaves. To accommodate the 10 g load in the case of a horizontal Mini-BRU, the length of the journal bearings would be increased approximately 30 percent. A summary of estimated bearing drag (expressed in watts) for the present design, 7 g design, and 10 g design is presented in Table I. The starting drag for a bearing system designed for operation through launch would increase by approximately 25 percent.

If the radiator axis is parallel rather than normal to the earth, the increased size should be minimal; therefore, this decision should not necessarily take precedence over other considerations. It is considered mandatory that the Mini-BRU axis be oriented normal to the plane of orbit since a reliability of single compensating gyro would not be acceptable. The double redundancy that would probably be required of a compensating gyro to ensure a seven-year operation would increase system weight and complication to that of a three-axis stabilization system (a preferable configuration). For these reasons, a gyro-compensated Mini-BRU system was not considered on Figure 1.

TABLE I
EFFECT OF MINI-BRU
JOURNAL BEARING REDESIGN

	Launch Designs		
	Present Design	Vertical Axis	Horizontal Axis
	1 g Journals	7 g Journals	10 g Journals
Journal	46 W_t	53 W_t	69 W_t
Thrust	82 W_t	109 W_t	109 W_t
	128 W_t	162 W_t	178 W_t
HSA Heat Required	448 W_t	567 W_t	623 W_t
HSA Heat Increase	0 W_t	119 W_t	175 W_t
% Thermal Capacity	9.33%	11.81%	12.98%
% Therm. Cap. Increase	0%	2.48%	3.65%

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With the design requirement that the Mini-BRU axis be normal to the plane of orbit to negate gyroscopic moment, the spin axis of a spin-stabilized satellite is also fixed since the two axes must be parallel. Since the spin axis of the satellite vehicle is normally thought to be coincident with the radiator axis, the vehicle spin axis would be parallel to the earth's surface, and the antennas would need to be designed accordingly.

Table I compares friction losses of the bearings using the present design and a vertical or horizontal Mini-BRU axis. The HSA heat required is estimated to be 3.5 times that of the bearing loss to compensate for system efficiency. Note that in the worst case (launch operation with Mini-BRU in a horizontal position), the required extra HSA heat is 175 watts, representing an increase of 3.65 percent (based on $4800 W_t$) from 9.33 percent to 12.98 percent. There is a tradeoff between the 3.65-percent HSA-heat requirement to operate the larger and stiffer bearings in orbit and the desirability of operation through launch. These bearing drag figures are based on laboratory tests which show the bearings had a larger safety factor than anticipated. Therefore, for a non-launch operated BIPS, redesign of the bearings to better fit the loads would lessen bearing drag.

Table II summarizes the conclusions and recommendations.

TABLE II
SUMMARY OF CONCLUSIONS AND RECOMMENDATIONS

LAUNCH REQUIREMENTS	SATELLITE REQUIREMENTS	DESIGN RECOMMENDATIONS	PENALTIES INCURRED FROM RECOMMENDED DESIGN
Mini-BRU not operated through launch	3-axis stabilized	Mini-BRU horizontal. Satellite vertical.	None
	spin stabilized	Mini-BRU vertical. Satellite and Mini-BRU axis normal to orbit plane.	No weight or BIPS penalties. Antennas must be compatible.
Mini-BRU operated through launch	3-axis stabilized	Mini-BRU horizontal. Satellite vertical.	Journal bearings redesigned. 3.65 percent more HSA heat used for bearing drag.
	spin stabilized	Mini-BRU vertical. Satellite and Mini-BRU axis normal to orbit plane.	Journal bearings redesigned. 2.48 percent more HSA heat used for bearing drag. Antennas must be compatible.

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APPENDIX A

FLIGHT SYSTEM PRELIMINARY DESIGN BRU ORIENTATION

The effort expended on Task 2 (FS Preliminary Design) has been confined to hardware layout. This effort has resulted in two candidate system layouts, (Mini-BRU both vertical and horizontal--Figure A). Note that for the vertical position, the HSA's remain in the same position relative to the structure (and radiator) as for the horizontal position, but the recuperator is shifted downward to allow for the extra ducting required (~ 5 lb); the Mini-BRU is simply rotated 90° about the turbine inlet centerline. The vertical position has been selected as the prime candidate for flight systems (and therefore for configuration selected for the GDS) because of the versatility of this configuration. The versatility comes from the vertical configurations compatibility with a spin stabilized vehicle, as well as with 3-axis stability vehicle (with certain control provisions as described below and in Figure B). Also, the thrust bearings on the Mini-BRU are better able to withstand the axial thrust of the booster rocket or takeoff (up to $10g$'s) without redesign.

The orientation of the Mini-BRU axis within a spacecraft can adversely affect the control of the spacecraft if all possible controls/Mini-BRU orientation combinations have not been recognized. A study was made which considered these combinations in order that the spacecraft customers and designers will be fully cognizant of the possible effects of the Mini-BRU orientation. The results are summarized in Figure B.

The study that resulted in Figure B was conducted using the following data:

COMPARISON OF HORIZONTAL AND VERTICAL
MINI-BRU CONFIGURATIONS

FIGURE A

75-311457(05)
Appendix A
Page A-3

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Mini-BRU polar moment of inertia = $0.00149 \text{ in-lb-sec}^2$
 Mini-BRU speed = 52,000 rpm
 Mini-BRU angular momentum = 0.6761 lb-ft-sec

Typical Mini-BRU precessional torques:

@ 1 rev/day (sync. orbit) = $4.917 \times 10^{-5} \text{ lb-ft}$
 @ 1 rev/1.5 hrs. (near earth orbit) = $7.867 \times 10^{-4} \text{ lb-ft}$
 @ 40 rpm (spin-stabilization) = 2.832 lb-ft
 Typical gas mass (hydrazine) required per revolution of a spacecraft with the Mini-BRU transverse to the spacecraft spin axis (b) = 0.002 lb
 Typical momentum wheel angular momentum range (a) = 5 - 20 ft-lb-sec
 Typical momentum wheel max speed (a) = 1000 rpm
 Typical momentum wheel weight range (a) = 5 - 15 lb
 Typical spacecraft spin rate (a) = 30 - 40 rpm
 Typical spacecraft polar moment of inertia(a) = 500 slug-ft^2

It has always been AIRPHX's position that a vertically oriented Mini-BRU was optimum for a spin-stabilized vehicle and a horizontally oriented Mini-BRU was optimum for a 3-axis stabilized satellite, assuming that the Mini-BRU axis is normal to the plane of the orbit. However, with the proper control system and Mini-BRU orientation, any precessional torque caused by the Mini-BRU may be controlled or counteracted (a).

(a) Memo from B. Raab (Fairchild Space and Electronics) to Dyer Kenney (ERDA-SNS). "Responses to AiResearch/Phoenix and Comments on BRU orientation", dated Oct. 27, 1975

(b) "Interim Report on Thermomechanical Power Systems for Planetary Applications", pg. 41, Jet Prop Lab CIT, Aug. 14, 1974, R. Caputo

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The Mini-BRU may be situated several feet from the axis of a spin-stabilized spacecraft, causing bearing loads similar to those encountered on the Titan IIIC (a). Assuming that the spacecraft is spinning at 40 rpm, the g-loading is given by:

$$g\text{-loading} = W^2 R/g_0 = 0.545R$$

where: R = the radius of the Mini-BRU from the spin axis, ft
 W = rate of spin, $4.19 \text{ rad sec}^{-1}$ (40 rpm)
 g_0 = standard gravity accel., ft sec^{-2}

From this it may be seen that, assuming 10g journal bearings, the limiting radius R is 18.3 ft.

The bearing losses have been re-evaluated using the latest laboratory test results and have been summarized on Table 1. Mini-BRU operation through launch or its operation at some radius on a spin-stabilized vehicle may require that the journal bearings absorb up to approximately 10g's. The thrust bearings will be able to absorb up to 10g's without a redesign and the journal bearings 7g's. If the journal bearings are required to operate at 10g's loading at any time during the mission, they must be stiffened, thus absorbing slightly more power, even at lower g-loadings.

From Table 1, it is evident that the additional bearing losses caused by launch g-loads are temporary and will be at a minimum when the vehicle is not accelerating or decelerating. In the case of a spin-stabilized vehicle, with the Mini-BRU situated 18.3 ft from the spin axis (assuming a spin rate of 40 rpm), the journal bearings would have to be stiffer (designed for the 10g load) and would exhibit the $79.4 W_t$ extra loss at all times, resulting in a 5.9 percent electrical output loss.

TABLE 1

MINI-BRU BEARING LOSS SUMMARY

Mini-BRU Orientation	Either	Either	Either	Vertical	Horizontal
Journal G-Load Design	7	10	7	7	10
Journal G-Load Actual	0	0	1	7	10
Thrust G-Load Actual	0	0	1	10	7
Journal Brg. Loss, W_t	84.4	120.8	120.8	130.4	135.0
Thrust Brg. Loss, W_t	<u>139.2</u>	<u>139.2</u>	<u>139.2</u>	<u>166.3</u>	<u>168.0</u>
Total Brg. Loss, W_t	<u>223.6</u>	<u>260.0</u>	<u>260.0</u>	<u>296.7</u>	<u>303.0</u>
ΔW_t Brg. Loss	0	36.4	36.4	73.1	79.4
$\Delta\%$ Brg. Loss	0	16.3%	16.3%	32.7%	35.5%
$\Delta W_{\text{Electrical Output Loss}}$	0	32.7	32.7	65.6	71.3
$\Delta\%$ Electrical Output Loss	0	2.7%	2.7%	5.4%	5.9%

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The conclusions reached are:

- (a) The weight and performance penalties of a vertically oriented Mini-BRU are minimal.
- (b) The vertical orientation of the Mini-BRU is chosen as the reference position for the GDS. A horizontal orientation is available with minimal ducting and structural changes.
- (c) Regardless of the Mini-BRU orientation, a properly designed space vehicle control system can compensate for the precessional torque caused by the Mini-BRU during orbital or interplanetary travel for a spin-stabilized or 3-axis stabilized vehicle.

APPENDIX I

BRAYTON ISOTOPE POWER SYSTEM FLIGHT SYSTEM (BIPS FS)

1.0 INTRODUCTION

This CCD is for a GDS designed to be representative of a flight system utilizing the Mini-BRU, MBR, and HSA designed for application in a space power system. This appendix briefly describes the BIPS FS design, operating parameters used in performance parametric studies, and design-point operating conditions. A conceptual system layout (L3621582) is included in Appendix III, which is based on 4800 W_t total from two radioisotope heat sources. Conceptual BIPS FS estimated performance is shown in Figure I-1, and the energy balance in Figure I-2.

BIPS FS design is such that optional features can easily be incorporated into delivered units. One such feature is one-g bearings that may be desirable for low power-level applications, or high-g bearings required for operation during launch.

The Mini-BRU is designed as an "all purpose" machine, capable of operating over a wide spectrum of power levels (0.35 to 2.1 kw_e) and orbital environments (near earth, synchronous, and deep space), with little compromise in performance. An example of this flexibility is that the "all purpose" Mini-BRU designed for 2.1 kw_e output adds only a six percent weight penalty to the BIPS FS when integrated into a 0.4 kw_e system.

GDS features, not in the BIPS FS, include a turbine bypass valve for overspeed protection, and a valved fill system. The BIPS FS will have no valves with external stems that could result in working fluid (gas) leakage.

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390°R EFFECTIVE
SINK TEMP
 $\epsilon = 0.80$
 $\alpha = 0.08$

$T_1 = 525$
 $P_1 = 44.4$
 $W_1 = 0.2304$

NET OUTPUT 1.3 KW
NET EFFICIENCY 0.270

1338 WATTS
OUTPUT (GROSS)

FLOW SCHEMATIC AND ESTIMATED PERFORMANCE
FOR THE 4800 W_t ERDA BIPS FS
FIGURE I-1

ENERGY BALANCE - 4800 W_t

FIGURE I-2

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The GDS HRHX (radiator simulator) will be replaced by a space radiator for the BIPS FS. The ducting system will necessarily be different, but will have a similar pressure drop.

All references noted in the paragraphs that follow are presented on the last page of this appendix.

2.0 BIPS FS OBJECTIVES AND DESIGN

The FS performance objectives are delineated on Table I-1. GDS design is based on these objectives except that an HRHX replaces the space radiator, and electrical resistance heat sources replaces radio-isotope heat sources. FS components and subcomponents are delineated in Table I-2.

The BIPS FS conceptual layout (L3621582) shows the Mini-BRU in the vertical position (same as GDS). This orientation was selected for two reasons:

- (a) For gyroscopic compatibility with a spin-stabilized spacecraft when mounted with the vehicle axis coincident with (or parallel to) the BIPS Mini-BRU axis.
- (b) Launch loads are usually greater in the vertical direction, and as presently designed, thrust bearings will withstand more load than journal bearings.

A horizontal orientation is equally feasible, but requires that some support structure and ducting be changed with the result that the MBR will be slightly lowered.

Because of potential future system requirements, some configuration changes may be required. These are:

- (a) Provision for Operation Through Launch - Since the BIPS FS is designed to operate in a space environment hard vacuum, a bell-jar approach must be taken to provide an inert gas atmosphere for refractory metals and Multi-Foil insulation protection. Bearings must also be checked for load adequacy (present journal and thrust bearing configurations are designed to accept 7 and 10 g acceleration loads, respectively).

TABLE I-1

FS PERFORMANCE OBJECTIVES

- (1) The FS will be capable of power output from 500 to 2000 W_e , EOM, with minimum redesign and requalification.
- (2) 28 ± 2 percent vdc is the reference output voltage.

NOTE

A capability for higher ac voltage output may be advantageous if the output can be achieved directly from the generator.

- (3) The control system will be capable of modulating power to the spacecraft from 0 to 100-percent in 20 milliseconds response times. This will be accomplished by a parasitic load.
- (4) FS life time requirements is $L = 7$ years and mean mission duration (MMD) = 5 years.

$$\text{where MMD} = \int_0^L R dt$$

and R = probability of producing design power at time t

(Notwithstanding this requirement, AIRPHX has established a more stringent objective of $R = 0.95$ after 5 years operation)

Prelaunch, launch, and operational phase failure modes will be identified.

- (5) Based on a GFE heat source cost of \$1000 per W_t , the recurring cost goal for the FS at 1300 W_e will be less than \$5000 per W_e .
- (6) A FS weight goal, at 1300 W_e , is 450 pounds.
- (7) The FS will be capable of launch on the space shuttle.
- (8) Due to a potential explosive vapor environment, the on-pad temperature of exposed surfaces will be maintained below 380°F.
- (9) The system will be designed with an assumed sink temperature equal to, or greater than, that of a geo-synchronous orbit (19,323 nm). The system will be capable of stable operation with unbalanced solar input.
- (10) Thermal input to the spacecraft must be of an amount that will not cause the spacecraft to rise above 100°F.
- (11) The magnetic field at one meter is not yet defined.
- (12) The on-pad biological dose at one meter will be ≤ 0.20 REM per hour.
- (13) The FS will be resistant to natural and man-made radiations.
- (14) FS components will be inherently flexible in mounting to facilitate integration with the required spacecraft configuration.
- (15) On pad and in-orbit fueling capabilities should be provided.

TABLE I-2
BIPS-FS COMPONENTS

Mini-BRU

- o 1600°F turbine inlet temperature (working fluid is Xe-He, MW = 83.8)
- o Single shaft turbine, Rice alternator and compressor supported on foil bearings with compressor bleed cooling
- o Alternator cooled by compressor discharge
- o Refractory metal turbine plenum (Columbium C-103)

Recuperator

- o Hastelloy X construction with gold alloy braze
- o Fin/plate counterflow design

Radiator

- o Material [Be, aluminum alloy or Lockalloy (alloy of Al and Be)]
- o Cylindrical design, offset fin/tube geometry
- o Sink temperature, 390°R
- o Micrometeoroid survival, 0.99 (assumed)
- o Used as part of a low gage pressure vessel to contain an inert gas for protection of the C-103 and multifoil insulation from launch to orbit.

Heat Source

- o ERDA Multi-Hundred Watt (MHW) heat source, (advanced ERDA HS concepts will also be investigated)
- o Electrically heated heat source simulator (EHS) for FS checkout

Heat Source Assembly (HSA)

- o Columbium Alloy C-103 Heat Source Heat Exchanger (HSHX)
- o Auxiliary Cooling System - Helium short circuit of HSHX Multi-Foil insulation
- o Emergency Cooling System - Melting of HSHX Multi-Foil insulation (alternatives have been identified)

Insulation

- o High temperature Multi-Foil insulation
- o Eight percent heat loss from HSA (Goal)
- o Two percent heat loss from turbine and hot interconnecting ducting (Goal)

Controls

- o Motor start inverter
- o Parasitic load
- o Speed control
- o Voltage regulator
- o Power supply
- o Rectifier/filter

Piping and Structure

- o Ducting
- o Bellows
- o Support structure (Beryllium or aluminum alloy)
- o Radiator is used as part of the support structure

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- (b) Operation on a Spin-Stabilized Platform - Bearing load limits must not be exceeded [see (a) above]. If the Mini-BRU axis is transverse to the vehicle spin axis, the resulting coning angle must not exceed that allowed by mission requirements. For a spinning vehicle and Mini-BRU combination, the coning half-angle (ϕ) is defined by:

$$\phi = M_m / M_v$$

where:

M_m = Mini-BRU angular momentum (0.68 lb-ft rad sec)

M_v = Vehicle angular momentum, lb-ft-sec

ϕ = Vehicle coning angle, rad

Presuming that M_v is 500, the coning half-angle would be approximately 0.08 degrees.

- (c) Spacecraft Integration - Launch vehicle constraints, heat input to the space vehicle, and/or BIPS placement on the space vehicle may require that the assumed BIPS FS be reconfigured. As presently configured, the BIPS can be mounted on any flat surface, and be designed to limit heat input to the spacecraft. The BIPS power supply furnishes the most power at 120 vdc, but by adding other power conversion components, any desired voltage and frequency combination may be obtained.

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The BIPS FS schematic, Figure I-3, shows all components within the working fluid loop, and inter-connecting ducting. The duct numbers correlate with Table I-3 data. Note that the radiator and radiator ducting materials have not yet been selected. Approximate duct sizes, and lengths of ducts 7, 8, 12, 13, and 14 (Table I-3) are based on the conceptual design, shown on Drawing L3621582 and will vary with different radiator configurations.

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Appendix I
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BIPS-FS DUCT SCHEMATIC

FIGURE I-3

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TABLE I-3

BIPS-FS DUCT SUMMARY

<u>Duct No.</u>	<u>Number of Joints</u>	<u>Material</u>	<u>Temperature °R</u>	<u>Pressure psia</u>	<u>OD Inch</u>	<u>Wall Inch</u>	<u>Duct Length</u>	<u>Number of Bellows</u>	<u>Bellows Length</u>
1	6	Hastelloy X	1760	66.9	2.50	0.035	43.3	3	3.00
2	3	Hastelloy X C-103	1760	66.9	1.60	0.035	10.0	1	2.00
3	2	Hastelloy X C-103	1760	66.9	1.60	0.035	6.0	1	2.00
4	1	C-103	2060	66.7	1.60	0.060	6.5	1	2.00
5	1	C-103	2060	66.7	1.60	0.060	6.5	1	2.00
6	9	C-103 Hastelloy X	1744	45.1	2.50	0.035	48.6	4	3.00
7	6	CRES/Be	721	44.9	2.50	0.035	37.0	3	3.00
8	9	Be/CRES	520	44.4	2.50	0.035	82.0	3	3.00
9	3	CRES	665	67.1	1.50	0.035	16.0	1	2.00
10	3	CRES	665	67.1	1.50	0.035	18.0	1	2.00
11	2	CRES	665	67.1	2.50	0.035	13.0	--	--
12	2/tube 32 tubes	TBD	721	44.8	TBD	TBD	2355	--	--
13	1	TBD	721	44.9	TBD	TBD	88.0	--	--
14	1	TBD	520	44.5	TBD	TBD	88.0	--	--
15	TBD	TBD

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3.0 PRELIMINARY SHOCK AND VIBRATION DESIGN CRITERIA

When considering structural dynamics, the BIPS consists of a cylindrical radiator within which four discrete masses are supported by means of interface structure consisting of beams and struts. The four discrete masses are; two heat source assemblies, recuperator, and Mini-BRU power package. As presently known, the BIPS will be mounted to the spacecraft, which will be mounted to the interim upper stage (IUS). The IUS will be mounted in a cradle or carrier in the space shuttle orbiter. This series of components comprises the orbiter payload. The Space Shuttle and IUS specifications (References 1 and 2, respectively) imply that the payload be cantilever-mounted, from the orbiter mount points when in the stowed configuration.

The principal source of structural vibration occurs during Space Shuttle powered ascent. Principal vibration levels the BIPS will experience depend on the payload dynamic response to orbiter vibration with the payload in the stowed configuration.

The principal source of shock occurs when using pyrotechnic devices on the IUS vehicle, such as that used for spacecraft separation. Principal shock levels the BIPS will experience depend on the payload dynamic response to pyrotechnic explosions, with the payload in the deployed configuration.

Shock and vibration levels at various interfaces, as estimated for conceptual design by the various referenced sources, are shown below.

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(a) Peak Random Vibration Levels

- o Shuttle Vehicle Payload Interface (Reference 1) - $0.1 \text{ g}^2/\text{Hz}$ (90 to 275 Hz) estimated response for an unloaded interface structure. Actual response should be lower due to mass of payload.
- o IUS-Spacecraft Interface (Reference 2) - $0.125 \text{ g}^2/\text{Hz}$ (100 to 1000 cps) recommended for conceptual design.
- o BIPS-Spacecraft Interface (Reference 3) - $0.1875 \text{ g}^2/\text{Hz}$ (900 to 1400 Hz) GE estimate based on LES 8/9 and Mariner J/S experience.

These vibration levels appear to represent a vibratory amplification through the payload. However, the levels are assumed to be estimates of the three reference sources that were determined independently. It will not be known whether the vibratory input to the payload will be amplified or attenuated until the total payload structure is defined and total payload dynamic analyses are conducted.

(b) Peak Shock Levels

- o IUS-Specification (Reference 2, Figure 6) - The shock response spectra gives a dynamic response of 115 g at the IUS-Spacecraft interface with a "Q" equal to ten for a 100 Hz system. Although

the BIPS response to shock at the IUS-Spacecraft interface is undefinable, it will be attenuated due to cantilever mounting and remoteness from the interface.

- o GE Specification Recommendations (Reference 3) -
50-g dynamic response for the Isotope Heat Source, and support structure, in the HSA housing and 25-g dynamic response for the overall BIPS structure, based on a shock response to a 775-g half sine pulse of 0.2 millisecond duration.

As noted above, actual shock and vibration levels the BIPS will experience depend on payload dynamic responses to the principal sources of excitation.

The Space Shuttle is well defined and the BIPS is conceptually defined. However, the spacecraft, IUS, and IUS support structure to the shuttle craft, are currently undefined except for the decision to utilize a solid rocket motor. It is anticipated that the solid rocket motor and structure will possess a high degree of stiffness. Because of many unknowns, no meaningful dynamic analyses can be made, at this time, for predicting BIPS, vibratory and shock responses. However, criteria does exist that system dynamics must satisfy to maintain BIPS responses within tolerable structural design limits. These are:

- (a) The cantilevered payload support structure must be designed to avoid coupling payload and Space Shuttle fundamental frequencies. To avoid this interaction the cantilevered payload fundamental frequencies should be at least two times greater than Space Shuttle fundamental frequencies. The shuttlecraft fundamental frequency range is estimated to be within 4 and 12 Hz. Therefore, the fundamental of the cantilevered payload should not be less than 24 Hz.

- (b) Support structures of BIPS discrete masses should be designed to avoid coupling the cantilevered payload fundamental frequencies with the rigid body fundamental frequencies of the BIPS masses. This means that BIPS components fundamental frequencies should be approximately 60 Hz.
- (c) Criteria (a) and (b) define cantilevered payload and BIPS components fundamental frequency limits. However, it is desirable to maintain the upper limits of natural frequencies no higher than necessary to satisfy (a) and (b), and to minimize BIPS components dynamic responses to shock and vibration. It is anticipated that the cascade of components, comprising the cantilevered payload, will respond to shock as a succession of simple systems, when each succeeding structure has a low mass relative to the next supporting structure.

3.1 Design Criteria

BIPS FS components (i.e., HSA, MBR, Mini-BRU) and mounting pads should be designed to a minimum limit loading of 50 g. BIPS interface structures (i.e., struts, beams, etc.) should be designed to a minimum limit loading of 25 g. The higher acceleration loading requirement provides increased structural integrity for BIPS components and mount pads. This is needed to compensate for the lower degree of mounting redundancy than the interface support structure (i.e., beams and struts). In the case of a statically determinate mounted component such as the MBR, failure of a single mount could result in separation of the component, whereas the interface structure is a many times redundant system capable of tolerating a number of member failures without total failure. The design criteria of providing sufficient stiffness to the BIPS interface structure, to maintain BIPS component

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natural frequencies above 60 cps, will inherently result in high strength. It is intended that this built-in strength will exceed 25 g. This limit loading should satisfy requirements for all other loadings such as pre-launch, launch, and operational environments, and in the case of an abort mission, crash landing requirements. This design criteria should be adequate for the BIPS preliminary design phase. After IUS and spacecraft definitions, dynamic analyses should be conducted on the orbiter cantilever-mounted payload, and the payload as deployed from the orbiter. The cantilevered payload and support design should minimize the BIPS dynamic response. Considerable latitude exists in designing the payload structure to minimize BIPS dynamic loading.

4.0 ESTIMATED PERFORMANCE

The estimated FS Mini-BRU performance, for 1, 2, and 3 HSA (2400 W_t each), is shown on Table I-4. The data was generated assuming the systems were optimized at minimum weight by changing the MBR and radiator sizes. The GFE recuperator was used for the 3-capsule design only and the weight of hypothetical recuperators of 1- and 2-capsule designs was lowered, mainly by shortening lengths. Note that for the 1-capsule case, the optimum system weight was achieved with a radiator larger than for the 2-capsule case. The same performance could be achieved with some weight penalty by using a smaller radiator and larger recuperator. Users of this data are referred to the objectives stated in Table I-1.

Cycle sensitivity to performance parameters that could change because of build technique, aging, intent, or usage, has been evaluated. Table I-5 enumerates those performance variables with linear sensitivity factors. For sink temperatures and turbine inlet temperature non-linear effects, refer to Figures I-4 and I-5. The partial differentials of cycle efficiency, thus obtained, can be added to estimate the effects of small changes on any of the three system sizes studied.

Parameters for a two-capsule BIPS are shown in Table I-6. To use the sensitivity factors, refer to Table I-4 for the original value, and Table I-5 for sensitivity factor.

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To illustrate sensitivity factors usage, consider the compressor efficiency calculation. A new 0.747 efficiency is assumed, which is 0.01 less than the original, and defines the delta as -0.01. This is also the 0.55 sensitivity factor multiplier shown on Table I-5. The resulting -0.0055 cycle efficiency change is noted on Table I-6. In some cases, the independent variable is a delta to base number ratio (i.e., $\Delta P/P$) and the multiplier is the delta of this parameter. As an example of pressure drop effect, if the pressure ratio parameter changes from 0.972 to 0.975, the increase is 0.003 $\Delta P/P$. The multiplier is the same number and $(0.003) \times (-0.8)$ results in a -0.0024 cycle efficiency change. As an example of bearing loss effect, if kw changes from 0.137 to 0.145, the delta increase is 0.008, but the multiplier is $0.008/0.137$, which is 0.0584. The cycle efficiency change is then $(0.0584) \times (-0.031)$, equalling -0.0018. To obtain $\Delta \eta_{cy}$ for other than design sink temperature (Figure I-4), or turbine inlet design temperature (Figure I-5), refer to the respective figures.

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TABLE I-4
CYCLE DATA OPTIMIZED FOR MINIMUM WEIGHT

	<u>Capsule 1</u>	<u>Capsule 2</u>	<u>Capsule 3</u>
Compressor Efficiency	0.746	0.757	0.760
Turbine Efficiency	0.791	0.818	0.831
Alternator Efficiency	0.905	0.927	0.921
Lost Pressure Ratio Parameter	0.959	0.972	0.9755
Recuperator Effectiveness	0.970	0.975	0.9758
Bearing Loss (kw)	0.124	0.137	0.151
Windage (kw)	0.030	0.053	0.073
Turbine Inlet Temperature (°R)	2060.0	2060.0	2060.0
Compressor Inlet Temperature (°R)	440.0	500.0	520.0
Compressor Pressure Ratio	1.615	1.537	1.514
Turbine Pressure Ratio	1.549	1.493	1.477
Compressor Inlet Pressure (psia)	18.2	42.3	66.2
Radiator Area (ft ²)	112.0	110.6	134.2
Recuperator Length (inch)	15.7	20.4	24.0
Fractional Heat Loss*	0.13	0.13	0.13
Net Output Power (kw)	0.619	1.356	2.067
Net Cycle Efficiency	0.2578	0.2825	0.2872
Radiator Emissivity	0.83	0.83	0.83
Radiator Sink Temperature (°R)	390.0	390.0	390.0
Compressor Flow Rate (lb/sec)	0.1064	0.2262	0.3442
Turbine Flow Rate (lb/sec)	0.1043	0.2217	0.3347
Heater Inlet Pressure (psia)	29.11	64.60	99.74
Heater Inlet Temperature (°R)	1738.9	1758.4	1762.5

*Thermal Loss/Gross Heat Release

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TABLE I-5

BIPS FS CYCLE EFFICIENCY SENSITIVITY FACTORS*

Independant Variable			Sensitivity Factors			
			Multiplier	Number of 2400 W _t Capsules		
Name	Calculation Range	Symbol		1	2	3
Compressor Efficiency	0.761 to 0.701	η_c	$\Delta \eta_c$	0.55	0.55	0.55
Turbine Efficiency	0.832 to 0.772	η_t	$\Delta \eta_t$	0.55	0.55	0.55
Alternator Efficiency	-0.02 to -0.04	$\Delta \eta_A$	$\Delta \eta_A$	0.32	0.32	0.32
Compressor Inlet Temperature	-40°F to 40°F	ΔT_1	ΔT_1	-0.0006	-0.0006	-0.0006
Emissivity	0.83 to 0.74	ϵ	$\Delta \epsilon$	0.05	0.077	0.075
Pressure Drop	0.005 to 0.010	$\frac{\Delta P}{P}$	$\Delta \left(\frac{\Delta P}{P} \right)$	-0.8	-0.8	-1.0
Bearing Loss	0.50 to 1.00	$\frac{\Delta W_B}{W_B}$	$\Delta \left(\frac{\Delta W_B}{W_B} \right)$	-0.056	-0.031	-0.02
Recuperator Effectiveness	0.00 to -0.03	ΔE_1	ΔE_1	1.25	1.20	1.15
Heat Loss	0.00 to -0.04	ΔW_t	ΔW_t	-0.37	-0.30	-0.25

* $\Delta \eta_{cy} = (\text{Sensitivity Factor}) \times \text{Multiplier}$

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SINK TEMPERATURE EFFECTS

FIGURE I-4

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EFFECT OF TURBINE INLET TEMPERATURE
ON OVERALL PERFORMANCE

FIGURE I-5

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TABLE I-6
OFF-DESIGN PERFORMANCE

<u>Parameter</u>	<u>New Value</u>	<u>Delta</u>	<u>Multiplier</u>	<u>$\Delta\eta_{cy}$</u>
Compressor Efficiency	0.747	-0.01	-0.01	-0.0055
Turbine Efficiency	0.808	-0.01	-0.01	-0.0055
Alternator Efficiency	0.917	-0.01	-0.01	-0.0032
Compressor Inlet Temperature	510	10	10.0	-0.0060
Emissivity	0.80	-0.03	-0.03	-0.0023
Pressure Drop	0.975	0.003	0.003	-0.0024
Bearing Loss	0.145	0.008	0.0584	-0.0018
Recuperator Effectiveness	0.950	-0.025	-0.02	-0.0300
Heat Loss	0.160	0.03	0.03	-0.0090
Sink Temperature	420	+30	--	-0.0088
Turbine Inlet Temperature	2010	-50	--	<u>-0.0080</u>
			Total	-0.0825
				(or -8.25%)

In more practical cases, only a few performance variables would generally be considered at one time. Assuming that compressor efficiency, radiator emissivity, and bearing loss have new values as indicated in Table I-6, cycle efficiency would then be reduced 0.96 percent.

To bring the cycle efficiency back to the original value, the compressor inlet temperature sensitivity factor is used to determine the required increased temperature:

$$T_{1_{\text{new}}} = 500 + 0.0096/(-0.0006) = 484^{\circ}\text{R}$$

From Figure I-7, $(\Delta \text{ area}) / (\Delta T_1) = -1.03$ and the required radiator area increase is therefore:

$$\Delta \text{ Area} = -16 \times (-1.03) = 16.48 \text{ ft}^2$$

for a total of 127.08 ft^2 , which represents a 14.5 percent increase.

This simplified approach to performance parameters effects on cycle efficiency, loses accuracy as parameters increase. Also, when the radiator size is changed, the performance of other components are changed slightly. Therefore, sensitivity factors, etc., should be used with caution.

To aid in assessing performance of a given system, with varying radioisotope quantities, off-design performance of the three systems is presented in Figure I-6. Since surface temperatures would be nearly constant, in each case it was assumed that the change in thermal loss would be insignificant.

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PERFORMANCE SPECTRUM OF 3 MINI-BRAYTON SYSTEMS

FIGURE I-6

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RADIATOR AREA EFFECTS

FIGURE I-7

5.0 LEAKAGE RATES

Estimated leakage rates of GDS components and joints are discussed in Section 3.3.5. FS hardware can be expected to meet or exceed existing estimates of leak tightness. However, assuming that all FS components and joints leak at the maximum rates estimated for GDS components, the total loss rate is calculated as:

<u>Component</u>	<u>Rate (Standard cc He/Sec)</u>
Mini-BRU	2.73×10^{-8}
MBR	1×10^{-7}
2 HSA at 1×10^{-9}	2×10^{-9}
Space Radiator	1×10^{-9}
Duct Welds (46)	4.6×10^{-9}
Bellows (19 with 2 welds each)	3.8×10^{-9}

Based on these rates the system total leakage rate is 1.387×10^{-7} standard cc He/sec. Since the allowable MBR leakage rate is 72 percent of the total, it may prove feasible to allow only a 1×10^{-9} standard cc/sec of He leakage rate in the FS MBR.

A constant maximum leakage rate for five years will result in 21.87 standard cc total loss. This loss equates to only 0.0146 percent of inventory, since the FS contains 1.16 pounds (150,079 standard cc) of Xe/He (MW = 83.8) gas at standard conditions. The resultant effects of this loss are negligible.

It is recognized that leakage rates do not normally remain constant, but rather, tend to grow with time. However, this is not considered a problem because the FS can tolerate leakage rates exceeding two orders of magnitude greater than the above noted maximum rate. For example assume the total working fluid loss percentage after 5 years will be as high as the thermal input reduction resulting from

radioisotope decay. This is the ideal and desired condition because the turbine inlet temperature and system efficiency remain constant.

It is estimated that thermal input will degrade to 96.13 percent after 5-years. An equivalent reduction in working fluid is $0.0387 \times 150,079$ cc, which equates to 5808 standard cc. This computes to a constant system loss of 3.68×10^{-5} standard cc He/sec, which is 265 times greater than the assumed 1.387×10^{-7} maximum rate. From this, it is obvious that the FS can tolerate an average maximum leakage rate exceeding 3×10^{-5} standard cc He/sec.

The FS control system will compensate for thermal input, and working fluid decreases. This is accomplished by altering engine speed to maintain a constant turbine inlet temperature (T_6).

AIRESEARCH MANUFACTURING COMPANY OF ARIZONA
A DIVISION OF THE GARRETT CORPORATION
PHOENIX, ARIZONA

LIST OF REFERENCES

1. JCS 07700 Volume XIV Revision D, "Space Shuttle System Payload Accommodations."
2. SS-ST5-100 Volume 3 Code Identification 07868 "Interim Upper Stage System Segment Basic Requirements for January 76."
3. GE-BIPS-20-001. "Mission Environments for The Isotope Brayton Flight System (Preliminary), August 8, 1975", Memo No. BIPS-GDS-G0017, 10-22-75.

BIPS SENSITIVITY CHARACTERISTICS

	PARAMETER VARIATION	RESULTING CHANGE $\Delta\eta_{cy}$
Compressor η	1%	.55%
Turbine η	1%	.55%
Alternator η	1%	.32%
E_R^* .97 to .96	1%	1.2 %
β where $\beta = 1 - \frac{\Delta P}{P}$	1%	.8 %
Thermal Loss	-160 Watts	1 %
Bearing Loss	-44.3 Watts	1 %
**Compressor Inlet Temp	-16°F	1 %
Turbine Inlet Temp	60°F	1 %
Emissivity	6.5%	.5 %
Sink Temp	-26°F	1 %

* E_R is the recuperator high pressure side effectiveness

**Can be offset by adding (12%) 14 ft² rad. area. Refer to Figure I-7, CCD.

OCT 27 1975

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 USERDA/SNS
 WASHINGTON, DC 20545
 AND
 BERNIE RAAB
 FAIRCHILD INDUSTRIES
 GERMANTOWN, MD 20767

BIPS-GDS

COORDINATION MEMO

TO Dyer Kenney FROM Howard Longee
93-24J AIRPHX

SUBJECT: ORIENTATION OF THE ROTATING GROUP

REFERENCE:

MEMO NO. BIPS-GDS- A0047

DATE: 10-27-75

REPLY BY: _____

☒ REQUEST ☐ INFORMATION

☐ REPLY TO: _____

COMMENT:

I had a brief conversation with Bernie Raab of Fairchild relative to our questions regarding orientation of the rotating group. Our understanding is that the communication satellite missions, as presently conceived, will be three-axis stabilized for transfer and orbit operation and that interplanetary missions will be spin stabilized. A qualitative analysis shows that mounting the rotating group spin axis perpendicular to the spacecraft major axis would be optimum for the three axis-stabilized craft and that mounting the unit coincident with the spacecraft major axis would be optimum for spin stabilized craft.

The present flight system conceptual design readily accommodates optional configurations for optimum orientation of the rotating group should the orientation have a significant impact on the spacecraft performance. Quantification of the effects of operating in a non-optimum orientation is needed as is a statement of the importance of a singular configuration.

In order to select a configuration for the GDS tests it is requested that ERDA have Fairchild either review the effects of orientation of the BIPS rotating group inertias

$$I_p = 0.00149 \text{ in}^2\text{lb}/\text{sec}^2$$

$$I_d = 0.0359 \text{ in}^2\text{lb}/\text{sec}^2$$

on the missions as presently conceived or provide the following information to AIRPHX to allow a quantitative evaluation of the effects of orientation:

CONTD

"Nothing contained herein shall be deemed to change the terms of any purchase order or contract between ERDA and AiResearch."

APPROVED BY: H.W. Longee

DATE: 10-27-75

DISTRIBUTION TO:

ERDA - G. Garborini, ERDA/SAN
 J. Lombardo, ERDA/SNS
 GE - W. Prickett; FCI - B. Rabb

AIRESEARCH

EDMONDS, ASHE, [REDACTED], MCCORMICK,
 STARK, PIETSCH/ROMANIN

BIPS-GDS

COORDINATION MEMO

MEMO NO:	BIPS-GDS-A0047
DATE:	10-27-75
REPLY BY:	
☒ REQUEST	☐ INFORMATION
☐ REPLY TO:	

1. What is the normal operating speed range for momentum wheels?
2. What is the typical moment of inertia and weight of a momentum wheel?
3. Is roll about the spacecraft center line SOP or allowable for 3-axis stable operation in synchronous orbit? If so, what range of roll speed would be typical (allowable)?
4. What values of spin speed and spacecraft moment of inertia are common for spin stabilized operation.
5. What are the power consumption characteristics of torque motors or other devices used to drive momentum wheels.
6. What coning angle can be accommodated by nutation dampers?
7. A slow rotation of the spacecraft would be desirable (orbital, 3-axis stab. or inter-planetary) to evenly spread the heat radiation of earth and planetary albedo and solar radiation (insolation).
 - a. Will a slow rotation of a 3-axis stabilized vehicle cause any control problem?
 - b. Will a spin-stabilized mission require that the spacecraft be precisely dynamically balanced?

The intent of this effort is to provide sufficient information to the ERDA to allow selection of the configuration for the KIPS and BIPS development programs. Bernie indicated that he would give immediate consideration to these questions. Your assistance in expediting this effort will be greatly appreciated as the GDS preliminary design effort is predicated on resolving this question.

Fairchild Space & Electronics Company Germantown, Maryland 20767 (301) 428-6000

November 6, 1975

Subject : Responses to AiResearch/Phoenix and Comments on BRU Orientation
Ref : BIPS-GDS Coordination Memo A0047 dated 10/27/75
To : Dyer Kenney (ERDA/SNS), A. Schock
cc : J. Lombardo, R. Brouns, R.J. Carpenter, B. Rock (ERDA/SNS)
H.W. Longee (AiR/Phx)
From : B. Raab

A. Responses to Memo

The overall concern of AiR/Phx in the reference memo regards the effects of the angular momentum of the BRU on spacecraft dynamics. Before proceeding to answer their questions specifically, I offer some general comments which may help to put things into perspective, and also to obviate several of the questions. [It is assumed herein that the values of BRU polar moment of inertia stated in the Memo is correctly stated in units of in-lb-sec², and furthermore that the BRU rotation speed is 52,000 RPM*]

AiR/Phx properly identifies two major classes of spacecraft: three-axis and spin-stabilized.

Three-axis -- Such spacecraft almost universally use internal momentum wheels or control-moment gyros (wheels on gimbals) to compensate for periodic torques. Such wheels are typically sized for maximum angular momenta in the range of 5-to-20 ft.-lb-sec and weigh some 5-to-15 pounds each. A set of three orthogonal wheels (plus one or more additional for redundancy) with their associated unloading devices, sensors and control circuits, typically constitute the spacecraft attitude control system (ACS). These wheels could be readily biased, in whatever combination is necessary to cancel out the angular momentum of the BRU (0.68 ft-lb-sec), whatever its orientation. In fact such compensation would generally be automatic, as the sensors detect unwanted

* verified by H. Longee

spacecraft motion and command the wheels to take it out. Clearly, with a control system of this type the weight, reliability, and power penalty due to the BRU are all negligible. Furthermore, from the viewpoint of on-orbit operations, no BRU orientation preference can be discerned.

It may be desirable ultimately to verify that the transient torques which may be introduced by action of the Brayton control system do not introduce transients which exceed the spacecraft limits and which may be too high in frequency for the nominal ACS. In view of the relative sizes of BRU and spacecraft moments, however, this seems hardly likely to be a problem.

Spin-stabilized -- Such systems have been preferred, where applicable, for simplicity and reliability of the attitude control, propulsion, and thermal control systems. Because of the large angular momentum available by spinning the entire spacecraft, typically of the order of $\sim 10^3$ ft-lb-sec, all wheels may be eliminated.

It is generally assumed that the BRU must be aligned with its rotational axis in the same direction as that of the spacecraft. Note, however, that if the BRU were aligned transverse to this axis, a coning motion would be produced with a half-angle given by $H_T/H_S \approx .7/1000 = 7 \times 10^{-4}$ radians = 0.04 degrees. The typical angular control requirement for a spinning communications spacecraft is of the order of 0.1-to-1 degree. Observatory-type spacecraft which may require closer angular control are invariably three-axis types.

It can be concluded, therefore, that with neither major spacecraft type will BRU orientation be dictated by on-orbit dynamic considerations, at least in the typical cases.* Therefore, orientation may be dictated by

* Note: This conclusion will not apply quite so generally to the KIPS rotating group, due to its considerably higher angular momentum.

D. Kenney
November 6, 1975
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thrust-load requirements of either boost or transfer-orbit operation, and/or by centrifugal loads imposed by spin-stabilized orbit-transfer or on-orbit service. Some thoughts relating to the latter loads are offered below, following responses to the specific questions raised by AiR/Phx.

The questions are repeated below, followed by the responses:

1. What is the normal operating speed range for momentum wheels?
A. 0-1000 RPM
2. What is the typical moment of inertia and weight of a momentum wheel?
A. (answered above)
3. Is roll about the spacecraft center line SOP or allowable for 3-axis stable operation in synchronous orbit? If so, what range of roll speed would be typical (allowable)?
A. No
4. What values of spin speed and spacecraft moment of inertia are common for spin stabilized operation?
A. 30-40 RPM, $I=500$ slug-ft²
5. What are the power consumption characteristics of torque motors or other devices used to drive momentum wheels?
A. typically ~10 watts/wheel
6. What coning angle can be accommodated by nutation dampers?
A. not relevant; "coning" due to non-coaxial internal angular momenta is not nutation and cannot be affected by nutation dampers.
7. A slow rotation of the spacecraft would be desirable (orbital, 3-axis stab. or inter-planetary) to evenly spread the heat radiation of earth and planetary albedo and solar radiation (insolation).
 - a. Will a slow rotation of a 3-axis stabilized vehicle cause any control problem?
A. In general, a three-axis spacecraft will be oriented by the requirements of the payload. A service function (e.g. electric power) cannot dictate orientation of

the spacecraft if it wants to succeed. Therefore, the power system must be capable of operation with fully unbalanced heat loads from external sources.

b. Will spin-stabilized mission require that the spacecraft be precisely dynamically balanced?

A. Yes; final balancing of the entire spacecraft on a spin table is generally done at the launch site prior to installation on the booster. A significant effort to balance the BIPS alone hardly seems warranted.

B. Launch and On-Orbit g-Loads

As AiR/Phx recognizes in their second monthly progress report (September 11, 1975), operation during launch can impose simultaneous axial and transverse loads (page 4). However, they also assume that starting in space will eliminate these requirements on the BRU bearings. I would like to introduce another scenario, however, which I believe to be highly likely for shuttle launches which now appears inevitable for all BIPS missions.

In this scenario, the shuttle is launched into parking orbit with the BIPS (or KIPS) quiescent. With the shuttle orbiter doors open and the spacecraft erected, but still physically attached to the orbiter, the payload specialist (4-th crew member) commands start-up of the machine and monitors progress on his payload console. If all satellite systems are satisfactory, the spacecraft is released. With adequate separation from the orbiter, the spacecraft spin rockets are fired in preparation for orbit transfer. The payload specialist verifies the spin axis prior to final approval of orbit transfer burn. A single or pair of solid rocket motors then completes orbit transfer to a final mission orbit, under ground control.

Although IUS or Tug upper stages will very probably be fully-stabilized vehicles, the above scenario assumes the dynamically more difficult spinning orbit transfer which is likely to be a very common approach for sub-synchronous or smaller satellites. Under these conditions, the rocket motors will of necessity occupy the vehicle axis. One end may be

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available for equipment on axis, but this location may well be required for payload equipment. Therefore, the BRU may of necessity be located several feet from the spin axis. Under these circumstances, and in this scenario, simultaneous axial and transverse loads will be experienced by the BRU bearings, very close in magnitude to those postulated for the Titan IIIC launch: 9-10 g's axially and 3-7 g's laterally would be a reasonable set.

A handwritten signature in dark ink, appearing to read "B. Raab", is written over a horizontal line.

B. Raab