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25 kW POWER MODULE UPDATED BASELINE SYSTEM

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NASA

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

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LIST OF ABBREVIATIONS

Symbol	Definition
A	Analog
ac	Alternating Current
ACS	Attitude Control System
AMP	Amplifier
ATM	Apollo Telescope Mount
A/D	Analog to Digital (Analog/Digital)
A-h	Ampere-Hour
BATT	Battery
BI-ϕ-L	Bi-Phase-Level
BOL	Beginning-Of-Life
BRPC	Battery Reconditioning and Protection Circuit
BUS 1	Regulated Bus No. 1 (Nominal 28 Vdc)
BUS 2	Regulated Bus No. 2 (Nominal 28 Vdc)
BUS 3	Regulated Bus No. 3 (Nominal 28 Vdc)
CHG	Charger
cm	Centimeter
CMG	Control Moment Gyro
CMGEA	Control Moment Gyro Electronics Assembly
CMGIA	Control Moment Gyro Inverter Assembly
cm^2	Square Centimeter
COMM	Communications
CONV	Converter
CONV/REG	Converter/Regulator
CRT	Cathode Ray Tube
CU	Central Unit
C&DH	Communications and Data Handling
C&W	Caution and Warning
D/A	Digital to Analog (Digital/Analog)
dB	Decibel
dBW	Decibel Relative to 1 Watt
deg/s	Degrees per Second
DISS	Dissipated
DOD	Depth of Discharge

LIST OF ABBREVIATIONS (Continued)

Symbol	Definition
EA	Electronics Assembly
EA.	Each
EF	MSFC Symbol for Data Systems Laboratory
EOL	End-Of-Life
EPS	Electrical Power System
EVA	Extra-Vehicular Activity
F	Fuse
FF	Free-Flyer
FMDM	Flexible Multiplexer/ Demultiplexer
ft	Feet
ft-lb	Foot Pounds
ft^2	Square Feet
GPC	General Purpose Computer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HEAO	High Energy Astronomical Observatory
HI V BUS	High Voltage Power Bus
H_x	Heat Exchanger
Hz	Hertz or Cycles per Second
IA	Inverter Assembly
IBM	International Business Machines, Inc.
IG	Inner Gimbal
IOM	Input/ Output Module
I/O	Input/ Output
k	Kilo (Thousand)
K	Relay Contact
kbps	Kilobits Per Second
kHz	Kilohertz or Kilocycles Per Second
kW	Kilowatt
kW_t	Kilowatt (Thermal)
KYBD	Keyboard

LIST OF ABBREVIATIONS (Continued)

Symbol	Definition
lb	Pound
lb/hr	Pounds per Hour
lb_m/hr	Pounds (Mass) per Hour
m	Meter
mA	Milliamperes
MAX.	Maximum
Mbps	Megabits Per Second
MCU	Module Control Unit
MDM	Multiplexer/Demultiplexer
MGT	Management
MIA	Multiplexer Interface Adapter
MIN.	Minimum
MIU	Modular Interface Unit
mm	Millimeter
MMS	Multimission Modular Spacecraft
Module	25 kW Power Module
MON	Monitor
MSFC	Marshall Space Flight Center
MTBF	Mean Time Between Failure
mV	Millivolt
m^2	Square Meters
N/A	Not Applicable
NiCd	Nickel-Cadmium
nm	Nanometer
Nm	Newton-Meter
Nms	Newton-Meter-Second
NRZ-L	Non-Return to Zero-Level
NRZ-M	Non-Return to Zero-Mark
NSSC-1	NASA Standard Spacecraft Computer — I
NSSC-2	NASA Standard Spacecraft Computer — II
N-tuple	A Multidimensional Representation, N Signifying N Dimensions

LIST OF ABBREVIATIONS (Continued)

Symbol	Definition
OA	Optical Assembly
OFT	Operational Flight Test
OP.	Operating
ORB	Orbiter
OSC	Oscillator
PCU	Power Control Unit
PK-to-PK	Peak-to-Peak
PM	25 kW Power Module
PMP	Premodulation Processor
Power Module	25 kW Power Module
PPG	Power Processing Group
PREMOD.	Premodulation
PROM	Programmable Read Only Memory
P/L	Payload
P/P_o	Power/Original Power
P/S	Power Supply
P^3	Programmable Power Processor
p-p	Peak-to-Peak
RCCB	Remote Controlled Circuit Breaker
REG	Regulator
RF	Radio Frequency
RFP	Request For Proposal
RGP	Rate Gyroscope Processor
RIU	Remote Interface Unit
RMS	Remote Manipulator System
rms	Root Mean Square
RPC	Remote Power Controller
rpm	Revolutions Per Minute
R_C	Line Resistance from Charger
R_B	Line Resistance from Battery
R_O	Line Resistance (Output)

LIST OF ABBREVIATIONS (Continued)

Symbol	Definition
R_R	Line Resistance of Bypass Circuit
R.A. MGT.	Retrieval Assurance Management
R_1	Line Resistance from Solar Array to Charger
R_2	Line Resistance from High V Bus to Converter
R_3	Line Resistance from Converter to Remote Controlled Circuit Breaker
SA	Solar Array
SCIU	Signal Conditioning Interface Unit
SCU	Sequence Control Unit
SEPS	Solar Electric Propulsion Stage
SEQ. MEMORY	Sequence Memory
SLCC	Standard Load Center Converter
SMA	S-Band Multiple Access
SN	Serial Number
SPS	Samples Per Second
SSA	S-Band Single Access
STACC	Standard Telemetry and Command Component
STA.	Station
STDN	Spaceflight Tracking and Data Network
STINT-2	Standard Interface — 2
STS	Space Transportation System
S/F	Scale Factor
TBD	To Be Determined
TCS	Thermal Control System
TDRSS	Tracking and Data Relay Satellite System
TEMP.	Temperature
TN	Technical Note
T_a	Actuator Stall Torque
T_w	Wire Torque
UV	Ultraviolet

LIST OF ABBREVIATIONS (Concluded)

Symbol	Definition
V	Volts
Vdc	Volts Direct Current
VRCS	Vernier Reaction Control System
V_{mp}	Voltage at Maximum Power
V_{oc}	Voltage Open Circuit
V/deg/s	Volts per Degree per Second
V/V_o	Voltage/Original Voltage
W	Watts
W_a	Actuator Weight
X	X-Axis
X-LV	X-Axis — Local Vertical
X-POP	X-Axis — Perpendicular to Orbital Plane
Y	Y-Axis
Y-LV	Y-Axis — Local Vertical
Z	Z-Axis
Z-LV	Z-Axis — Local Vertical
Z-POP	Z-Axis — Perpendicular to Orbital Plane
°	Degrees
°C	Degrees Celsius
°F	Degrees Fahrenheit
°/min	Degrees per Minute
ϕ	Phase
ϕ	Gimbal Angle ±180°
$\dot{\phi}_c$	Normal Control Rate
$\dot{\phi}_{max.}$	Maximum Slew Rate

TECHNICAL MEMORANDUM

25 kW POWER MODULE UPDATED BASELINE SYSTEM

1.0 INTRODUCTION

Most future payloads that will utilize the Space Transportation System (STS) will require power, attitude control, heat rejection, and a communications system other than that contained within the payloads themselves. Some potential payloads may also have orbit stay-time and power requirements beyond that which the Orbiter can furnish. In addition, interest has been shown in a power source for free-flying payloads. The 25 kW Power Module is proposed to meet the basic power requirements of the payloads both as a supplement to the Orbiter and as a prime source for free-flyers. In September 1977, MSFC issued a preliminary definition report on the 25 kW Power Module in which a baseline system was proposed. Since that report was issued, studies, trades, and additional preliminary designs have been conducted to further refine the original baseline system proposal. This report defines a refined baseline system that is proposed to supersede the September 1977 depiction. Where significant departures occur between the system proposed in this report and the September 1977 system, the rationale for incorporating these changes is discussed. However, where significant changes are not proposed, for the sake of brevity, the rationale for proposing the original system or subsystem is not covered. Similarly, although ground support equipment, operational requirements, etc., were considered in the systems trades and analyses, these items are not discussed in this report.

2.0 REQUIREMENTS

The 25 kW Power Module requirements are dedicated to the support of STS payloads that have requirements beyond those that can be supplied by the Orbiter. These requirements are summarized in the Power Module System Design Requirements Document [1] and are based upon a desired low-cost approach utilizing existing hardware and current technology where practical. These requirements, which have driven the studies, trades, and development of concepts and options, were developed to meet the more general requirements expressed in the 25 kW Power Module Project Requirements Document [2]. The system requirements were developed in an evolutionary manner as the various system concepts, options, and trades were conducted.

3.0 SYSTEM DESCRIPTION

The 25 kW Power Module baseline design as refined is shown in Figures 1 through 3. The design allows the 25 kW Power Module to extend forward from the Orbiter payload bay and be cantilevered directly above the forward end of the Orbiter. This position will allow for extension, retraction, and articulation of the solar arrays. Thermal radiators are provided that can be extended for heat rejection or retracted to a stowed position. Two docking ports are available for docking with the Orbiter in a sortie mode, with a free-flyer, or simultaneously with both. Alongside these docking ports are umbilicals to interface with the Orbiter/payload or free-flyer. These interfaces will allow transfer of electrical power, heat rejection, communications and data, and caution and warning information.

Within the in-flight portion of the 25 kW Power Module, there are five basic systems: structures, electrical power, attitude control, thermal control, and communications and data handling. During this study, work has been performed in each of the system areas; in some cases, refining earlier concepts and in other cases, developing better approaches than earlier envisioned. The remainder of this section is a summary depicting the five systems being proposed as the new baseline Power Module system. Detailed information concerning each system is contained in sections 3.1 through 3.5.

Although the Apollo Telescope Mount (ATM) structure was found not to be cost effective to use for the 25 kW Power Module, the ATM structure did influence the structure proposed in this report. Such things as the ATM structure safety factor being too low (and thus requiring expensive redesign and testing to overcome) and having to move the tie-down points combined to cause analysts to decide not to use the ATM structure per se. However, the octagonal ATM rack was found to be a good compromise, in size and shape, towards fully utilizing the 180-in. diameter of the payload bay of the Orbiter, while also providing flat panels of a practical size for component mounting. In the envisioned structure, the forward end supports the solar array and the aft end

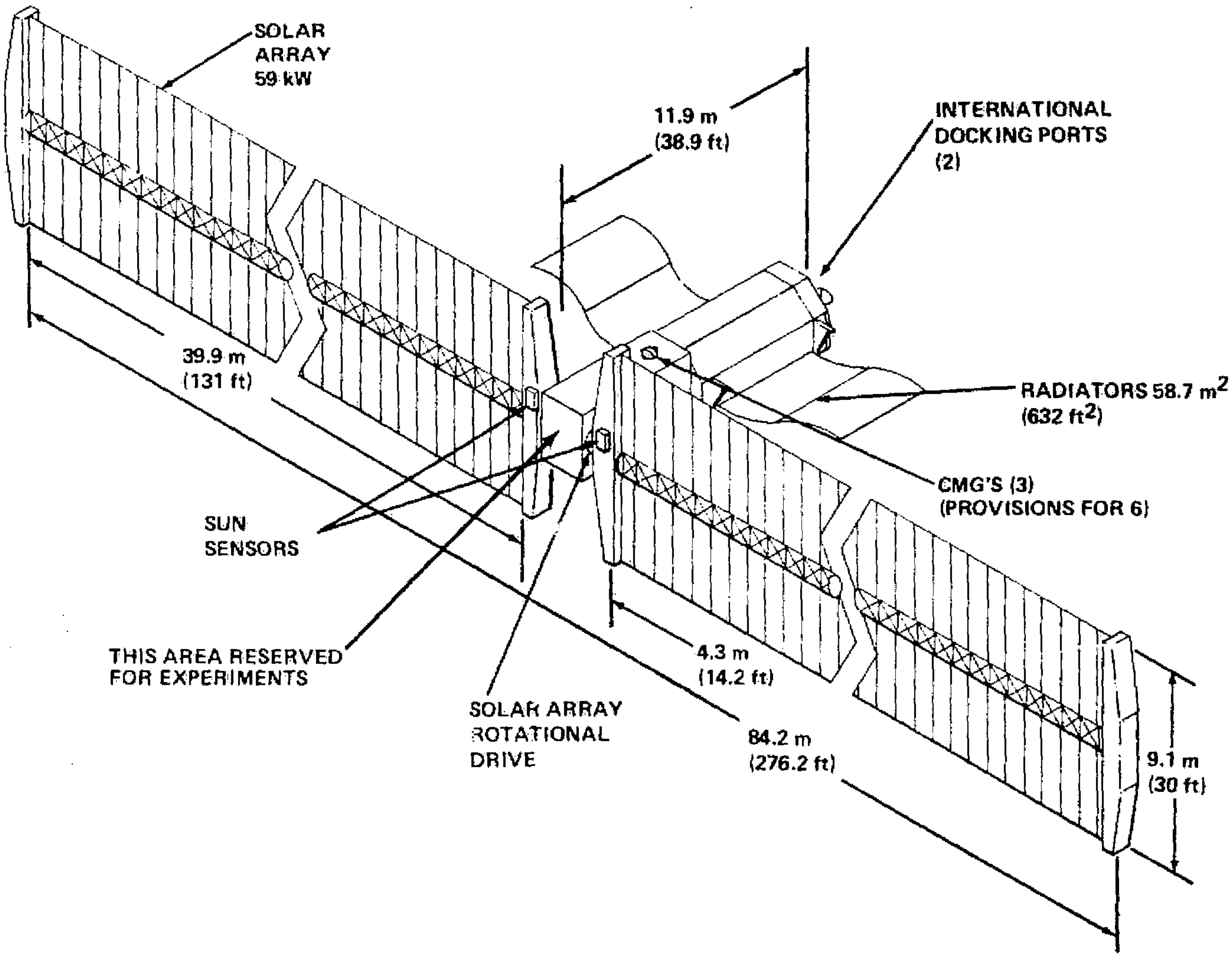


Figure 1. 25 kW Power Module baseline design — December 1978.

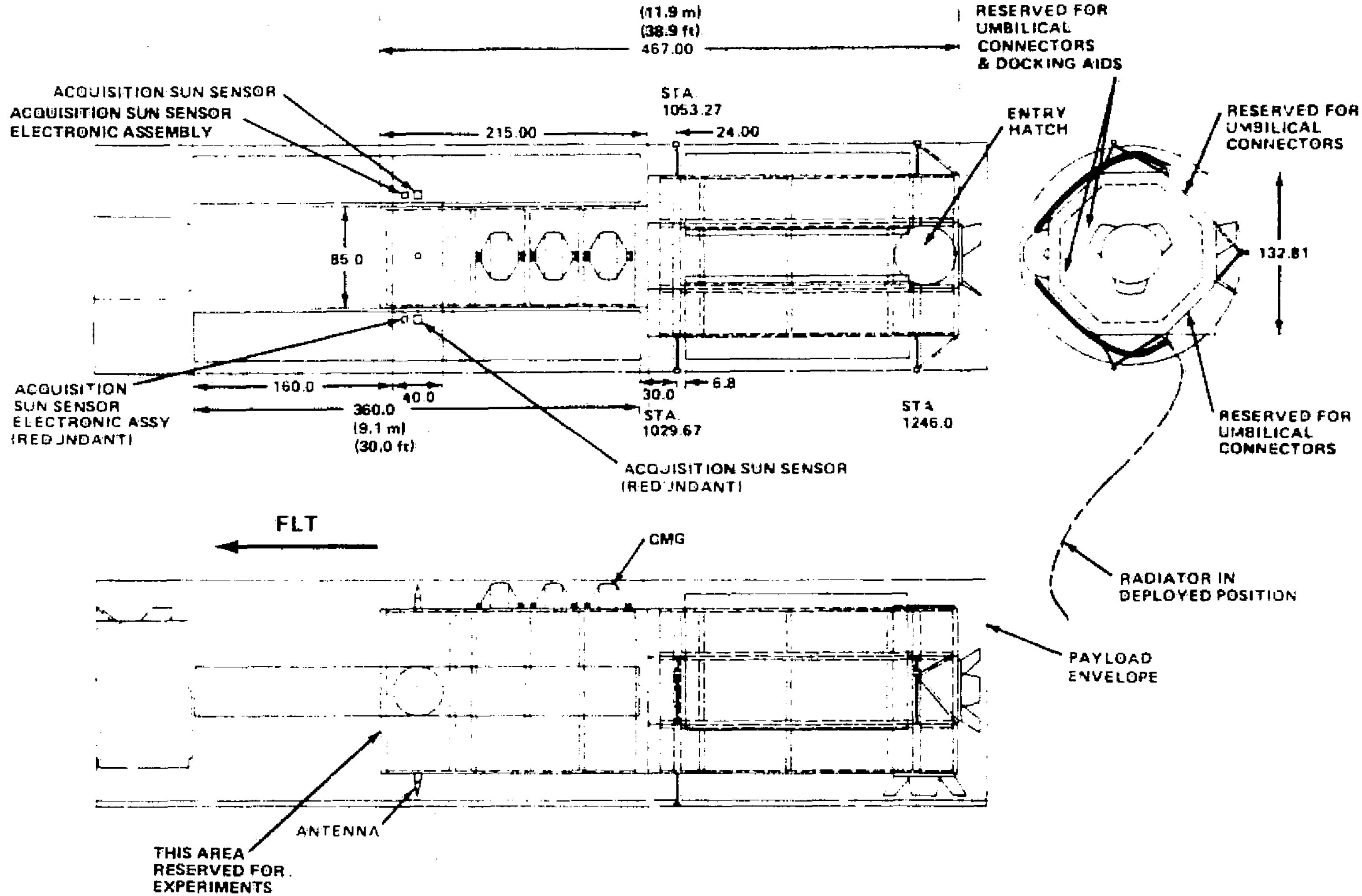


Figure 2. 25 kW Power Module structure — December 1978.

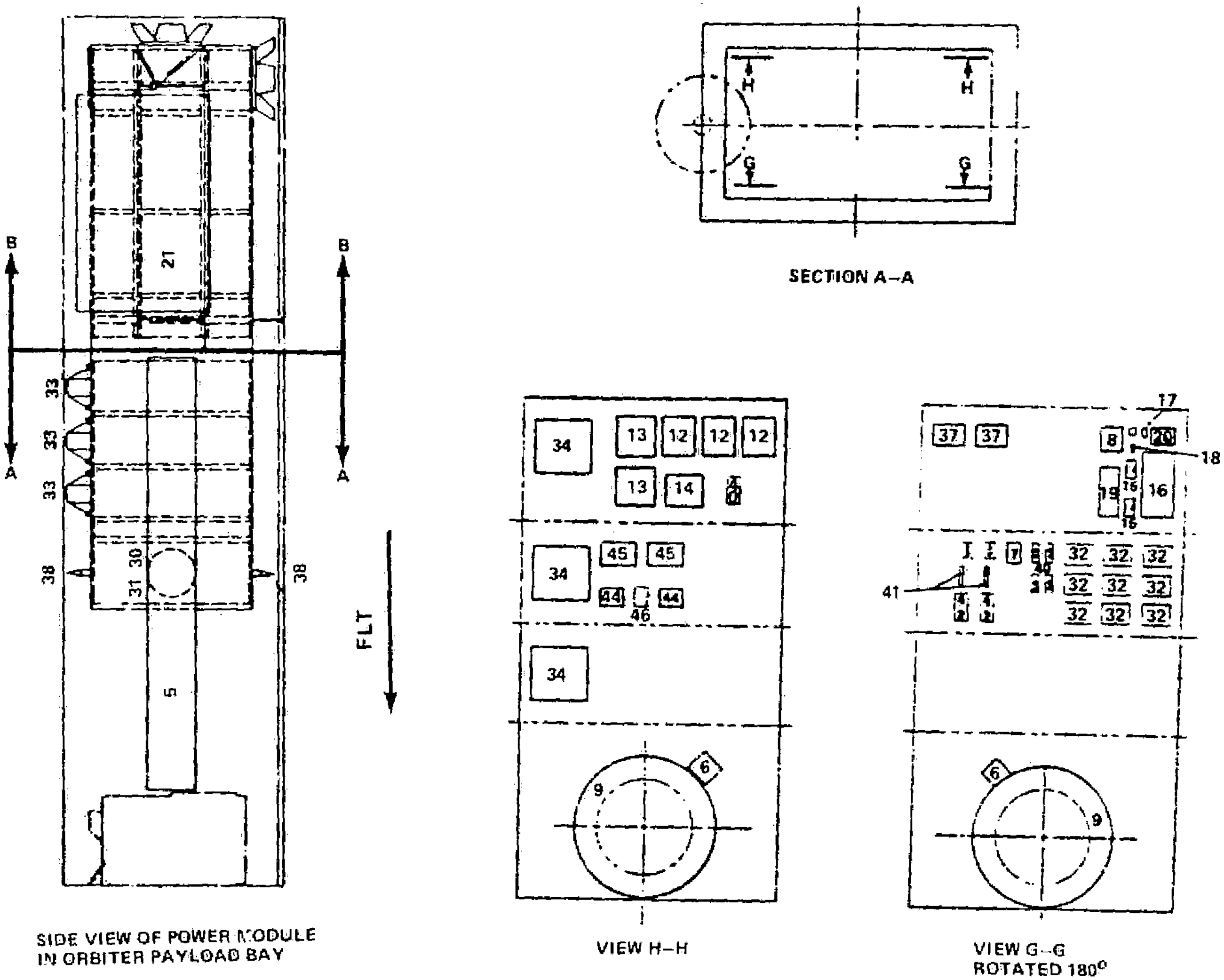
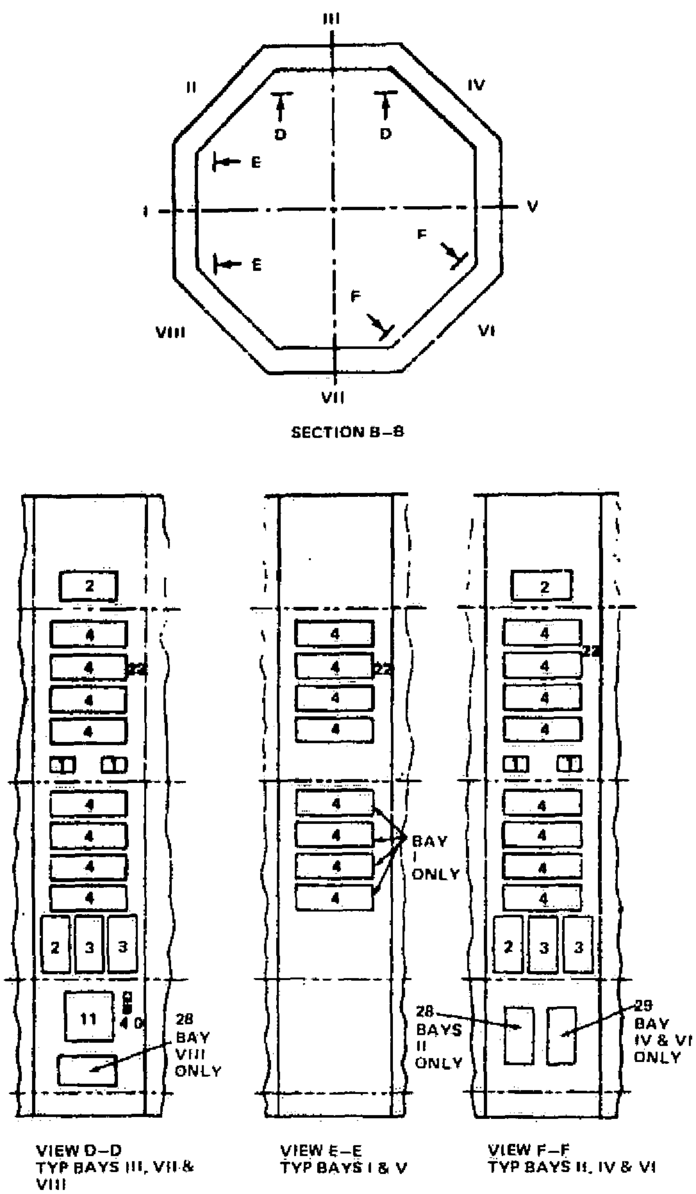


Figure 3. 25 kW Power Module racks — internal view — December 1978.



ITEM	COMPONENT
1	BATTERY RECONDITIONING MODULE
2	BATTERY CHARGER
3	BUS REGULATOR
4	BATTERY MODULE
5	SOLAR ARRAY WING
6	SOLAR ARRAY ARTICULATION CONTROL ASSEMBLY
7	SLCC
8	PUMP INVERTER
9	SOLAR ARRAY ARTICULATION ASSY POWER BRIDGING UNIT
10	DELETED
11	POWER TRANSFER DISTRIBUTOR
12	CONTROL DISTRIBUTOR
13	AUXILIARY POWER DISTRIBUTOR
14	MEASURING DISTRIBUTOR
15	FREON PUMP
16	FREON ACCUMULATOR
17	FILTER
18	CHECK VALVE
19	PAYLOAD HEAT EXCHANGER
20	FLOW PROPORTIONING MODULE
21	ORBITER RADIATOR PANELS
22	COLD PLATES
23	DELETED
24	THERMAL SYSTEM CONTROLLER (NS)
25	SHUT-OFF VALVES (NS)
26	FLOW BALANCE VALVES (NS)
27	TEMPERATURE CONTROL VALVES (NS)
28	RADIATOR PANEL DEPLOYMENT MECHANISM
29	RADIATOR PANEL LATCHES
30	ACQUISITION SUN SENSOR
31	ACQUISITION SUN SENSOR ELECTRONIC ASSEMBLY
32	ATM RATE GYRO
33	CMG
34	CMG INVERTER ASSEMBLY
35	DELETED
36	DELETED
37	SIGNAL CONDITIONER & INTERFACE UNIT
38	OMNI ANTENNA
39	STACC CENTRAL UNIT
40	REMOTE INTERFACE UNIT
41	STACC STANDARD INTERFACE (STINT)
42	NSSC II COMPUTER
43	DELETED
44	POWER AMPLIFIER
45	NASA STANDARD TRANSPONDER
46	RF TRANSFER SWITCH
47	RETRIEVAL ASSURANCE UNIT (NS)
48	POWER CONTROL UNIT (NS)
49	PREMODULATION PROCESSOR (NS)

(NS) = NOT SHOWN

Figure 3. (Concluded).

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supports a free-flyer. Thermal radiators are supported on each side of the module, and can be extended for operation or retracted for berthing operations or retrieval by the Orbiter remote manipulator system (RMS). A grappling fixture is available near or at the module center of gravity for RMS use in removing the module from the payload bay and for docking and berthing operations. This structure design is based upon a module weight of approximately 29 000 lb. Section 3.1 depicts some alternate structural concepts recently considered and describes the approach proposed for use on the new baseline structure in more detail.

The technology development leading to the large electrical power systems such as that used in Skylab has been continued and emphasized at MSFC. As with the 1977 baseline system proposal, this proposal centers largely around the facts that the electrical power will be furnished by a solar array-battery system and will use the technology developed under both the Skylab and Solar Electric Propulsion Stage (SEPS) activities. The 25 kW Power Module solar array will produce approximately 60 kW of power and furnish an average of 25 kW of regulated electrical power to the module interface. Part of the solar array power will be used to charge 12 high voltage batteries using MSFC developed programmable power processors (P^3's). Battery reconditioning and self-protection circuits directly developed from information gained during Skylab experience will allow longer, higher efficiency usage of the electrical power subsystem. The solar array will have two wings with each wing approximately 131 ft (39.6 m) long and 30 ft (9 m) wide, and the envisioned solar array will be able to articulate approximately ±180° about an axis perpendicular to the module centerline. Sun sensors will be used to search for and point the solar array to the Sun. Section 3.2 outlines the activities relating to the electrical power system in more detail.

The proposed 25 kW Power Module baseline system provides its own attitude control system (ACS) and will utilize rate gyros and control moment gyros (CMG's) available from the Skylab program. CMG's were selected for the ACS not only because they are on hand at MSFC, but they provide a clean environment for experiments. The Skylab CMG's are the only fully developed actuators of this type with the required capability. Experience in Skylab operation and subsequent technology development and analyses provide the knowledge necessary to modify the existing hardware and configuration designs. A particularly favorable feature of the proposed ACS is that it will not only provide attitude control for the 25 kW Power Module when in a free-flying mode, but it will also provide attitude control for the docked Orbiter/25 kW Power Module combination. However, in the docked configuration, the Orbiter vernier reaction control system (VRCS) may be required for some orbit orientations and dumping of the CMG momentum. A more detailed description of the 25 kW Power Module attitude control system is provided in section 3. 3.

The proposed 25 kW Power Module baseline thermal control system will provide heat rejection for the Power Module as well as some heat rejection for the Orbiter and Orbiter payloads when docked, or for other payloads when the Power Module is functioning as a free-flyer. The heat rejection for payloads will be accomplished through fluid lines into a heat exchanger within the Power Module. From there two sets of radiators on each side will reject the heat into outer space. A dual loop will be used in this system to provide 12 kW of heat rejection to the Orbiter or other payloads. Section 3. 4 provides more details of the proposed thermal control system.

The baseline on-board communications and data handling (C& DH) system utilizes a data bus concept to enable telemetry, commands, and attitude control. Two C& DH options are proposed in this report. One is oriented toward the Multimission Modular Spacecraft (MMS) command and data handling system. Another option utilizes a modified flexible multiplexer/demultiplexer (FMDM) subsystem and is seen as equally viable. The preliminary studies of the communications portion of the C& DH system revealed that an approximate limit of 4 kbps for the return link (from the spacecraft to ground) and 1 kbps for the forward link could be accommodated without incurring the significant step-function increase in cost that would be necessary to go to high gain or steerable antennas. Consequently, this 4 kbps/1 kbps requirement was used as a self-imposed constraint for the proposed C& DH system. The 25 kW Power Module will normally communicate with Earth by two means. When docked with the Orbiter, it will use the Orbiter's communications system. When in orbital stowage or in a free-flying mode, it will communicate with its own C& DH system via the Tracking and Data Relay Satellite System (TDRSS). In addition to handling the Power Module data, the C& DH of the Power Module will also accommodate limited amounts of payload housekeeping and command data. Section 3. 5 describes the two system options proposed for the new baseline and discusses the rationale for the variations from the September 1977 baseline.

The subject of redundancy management cuts across several systems. The original assumption in the earlier baseline work was that all redundancy management for the various flight systems would be done from the ground.

However, to ensure proper redundancy management under the condition of non-continuous RF coverage (a more stringent condition than originally provided for), the original assumption was modified. As a rule, redundancy management will be accomplished by ground evaluation and command except for automatic fail-operational switching needed to preserve the safety of the Power Module following a failure.

3.1 Structures System

3.1.1 Introduction. The selected overall structural arrangement of the 25 kW Power Module is indicated in Figure 2, which shows the Power Module in relation to the outline of the payload bay of the Orbiter. The stowed Power Module occupies all of the usable volume of the payload bay aft of Orbiter station 660, thus reserving adequate space for the structural struts of the docking adapter since this envelope is not currently well defined.

The incorporation of existing ATM racks in the Power Module structure was investigated. Consideration was given to using the ATM racks "as is" and to dismantling them and using parts, but this was found to be more expensive than designing and building an all new structure. This was largely because the ATM was designed for a different carrier vehicle (Saturn V) and used a lower factor of safety than that required for the Space Shuttle system. However, the octagonal ATM rack was found to be a good configuration in shape and dimension toward fully utilizing the 180-in. diameter of the payload bay of the Orbiter while also providing flat panels of a practical size for component mounting.

3.1.2 Assumptions. A summary of the assumptions used in the structural design is as follows:

a) Volume, weight, and load limitations as defined in JSC 07700, Volume XIV, Space Shuttle System Payload Accommodations.

b) The 25 kW Power Module when stowed in the payload bay of the Orbiter shall not extend forward of Orbiter station 660. Rationale — As noted in section 3.1.1.

c) Minimum ultimate factor of safety for design of 1.4. Rationale — Imposed by Shuttle.

d) All Power Module system components are accessible to a suited astronaut for purposes of orbital maintenance. Rationale — Systems requirements document.

e) The docking adapters are Apollo-Soyuz units scaled up to a 1-m diameter passageway. Rationale — Preliminary discussions with Johnson Space Center (JSC) revealed that it would likely be some time before the Shuttle Office was able to rigorously define the characteristics of the docking adapter. Consequently, this assumed definition, which has been informally used before, was used in the absence of the final definition.

3.1.3 Structural Description. The proposed new baseline structure of the Power Module is divided on the basis of geometry into two major subassemblies. The aft subassembly is an octagon in cross section, 132.8 in. wide by 252 in. long. This octagon structure carries the trunnion trusses for interfacing with the Orbiter payload retention fittings at stations 1053.27 and 1246.00. A five-point, indeterminate support system is used in the payload bay with the Orbiter X- and Z-loads reacted by the two sill trunnions at station 1246, and the Z- and Y-loads reacted by the two sill trunnions and the one keel trunnion at station 1053.27. Studies showed that with the Power Module configuration, and the current weight estimate of approximately 29 000 lb, loads would be too high with the four-point support system. The exterior space on the Power Module structure between the forward and aft trunnion trusses is required for the four radiator panels, grouped in two sets of two panels each with one set on each side of the Power Module. The radiator panels, which are modified Orbiter payload bay door radiator panels, curve about the octagon in the stowed position.

There are two identical docking adapters at the aft end of the octagon structure. One docking adapter, located on the same side of the octagon as the keel trunnion truss, is for berthing the Power Module to a docking structure in the payload bay of the Orbiter. The second docking adapter is located on the aft bulkhead of the octagon and is to be used for berthing of the experiment loaded pallets of the free-flyer.

The forward structural subassembly supporting the solar arrays is rectangular in cross section, 85 in. wide by 132.8 in. deep, and 215 in. long. Near the forward end of this box structure are two bearings, one in each lateral surface of the box, for the solar array articulation system. The stowage canisters for the solar array masts are located end-to-end within this box structure along the centerline connecting these two bearings. The CMG's are mounted in a parallel arrangement on one of the 85 in. by 215 in. sides of the box structure, and the CMG's are installed and removed from the outside of the structure. All other black-box components are mounted inside the structure.

Cold plates are located within the octagon structure for all those components requiring active thermal control. This arrangement places the cold plates adjacent to the radiators. Components are also mounted to the interior of the forward rectangular box structure. The final arrangement of components will be influenced to a large degree by the center-of-gravity requirements for payloads carried in the Orbiter.

Both the rectangular box and the octagon structures have main beam members located longitudinally along each corner of the subassembly, i.e., four main beams in the box and eight in the octagon. Primary frames are located at stations where major external loads are introduced; for example, the trunnion trusses; radiator hinge, stowage latch, and deployment/ retraction actuator anchor points; solar array articulation bearings and stowage latches; CMG supports; and docking adapters. A major frame is also required at the junction of the rectangular box to the octagon structure. The lateral sides of both structural subassemblies, as well as the forward and aft bulkheads, are honeycomb panels.

To minimize problems such as distortions, heating at welds, and difficult inspections, the entire structure is mechanically fastened together, as opposed to welding, and the material, except for the trunnions, is aluminum. The trunnions will be either steel or titanium, since the Shuttle requirements on the trunnion diameters preclude use of aluminum for the loads anticipated.

The 25 kW Power Module structure is not a pressurized structure. Orbital access to all internally mounted components will be by a suited astronaut through the keel-side docking adapter. Ground access will be by way of a hatch on the anti-keel side. This hatch is also usable when the Power Module is in the payload bay of the Orbiter and the payload bay doors are open.

The weight of the structure as described is estimated at 7700 lb and is based on an ultimate factor of safety of 1.4 for the design.

3.1.4 On-Orbit Maintenance. No on-orbit maintenance of the structures system is planned for this baseline system.

3.2 Electrical Power System (EPS)

3.2.1 Introduction. The requirement for 25 kW of regulated electrical power is considerably larger than that of any previous flight power system. In recent MSFC studies, four major alternate concepts were investigated in addition to the configuration (Fig. 4) proposed in the September 1977 baseline. The alternative concepts were:

a) Alternate No. 1 — 12 power processing groups (Fig. 5)

b) Alternate No. 2 — Based on Skylab Airlock Module power conditioning group usage (Fig. 6)

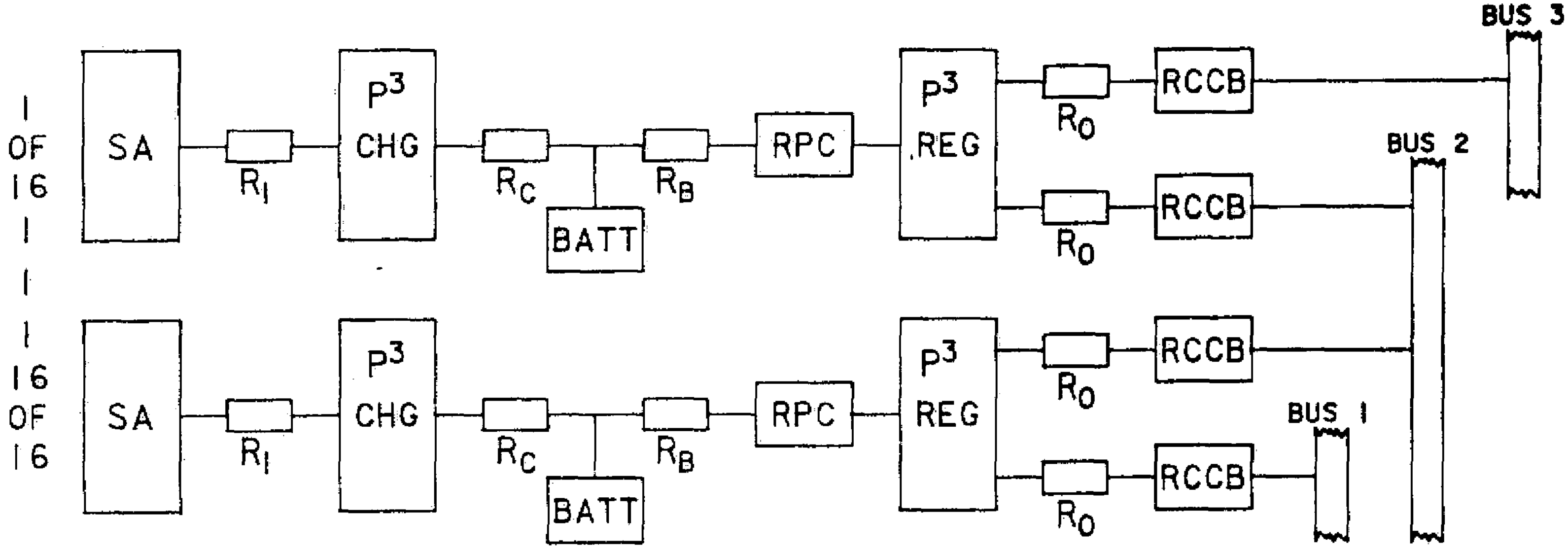
c) Alternate No. 3 — Based on six power processing groups and four converters (Fig. 7)

d) Alternate No. 4 — Based on six programmable power processors and four converters (Fig. 8).

Past experience in design and operation of the Skylab power system as well as data from continuing technology programs at MSFC were used in the configuration evaluations and selection of the baseline EPS depicted in this report. In addition, actual test data and accurate cost estimates were available for the trade studies and final configuration selection.

The configuration of Alternate No. 1 (Fig. 5) was selected over the original 1977 baseline EPS because additional information became available from ongoing technology work which allowed, for example, additional advantages such as a larger depth of discharge of the batteries — ultimately allowing fewer power processing groups (PPG's) to be used (12 as opposed to 16) with a proportional savings in cost and weight.

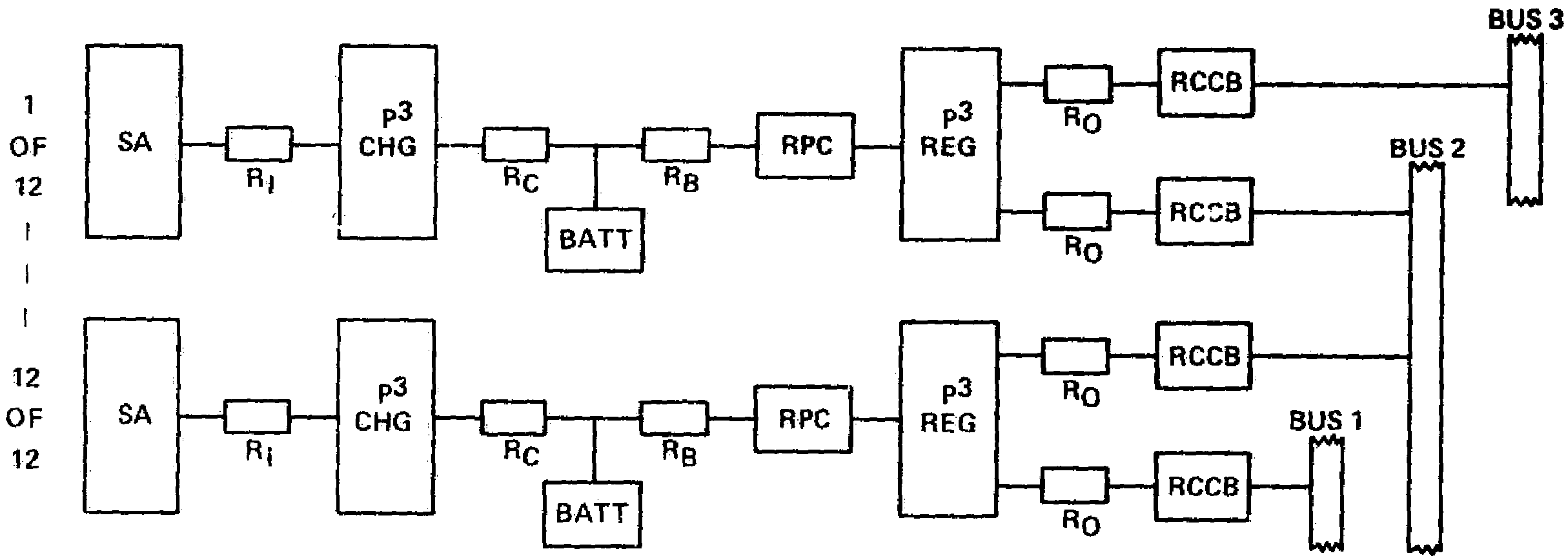
A functional block diagram of the new baseline EPS is shown in Figure 5. There are 12 primary PPG's consisting of solar array sources, programmable power processor chargers, 110 cell NiCd batteries with their reconditioning and protection circuits (BRPC), programmable power processor regulators, and their associated switches providing three nominal 30 Vdc buses to supply power to the Shuttle Orbiter or a free-flying payload and/or to the 25 kW Power Module (PM) internal loads. The system is sized to provide an orbital average power of 25 kW to the payload (Orbiter or free-flyer) and 2 kW to internal systems. With this configuration, a peak power of 35 kW is available within thermal and orbital average power limitations. These figures are based on a 57.8 kW solar array source at a beta angle of zero and with the solar panels perpendicular to the solar vector during sunlight portions of the orbit. The accompanying power



LEGEND:

SA	– SOLAR ARRAY
R_I-R_O	– CABLE RESISTANCES
P_3	– PROGRAMMABLE POWER PROCESSOR
CHG	– CHARGER
BATT	– BATTERY
RPC	– RECONDITIONING & PROTECTION CIRCUITS
REG	– REGULATOR
RCCB	– REMOTE CONTROLLED CIRCUIT BREAKERS

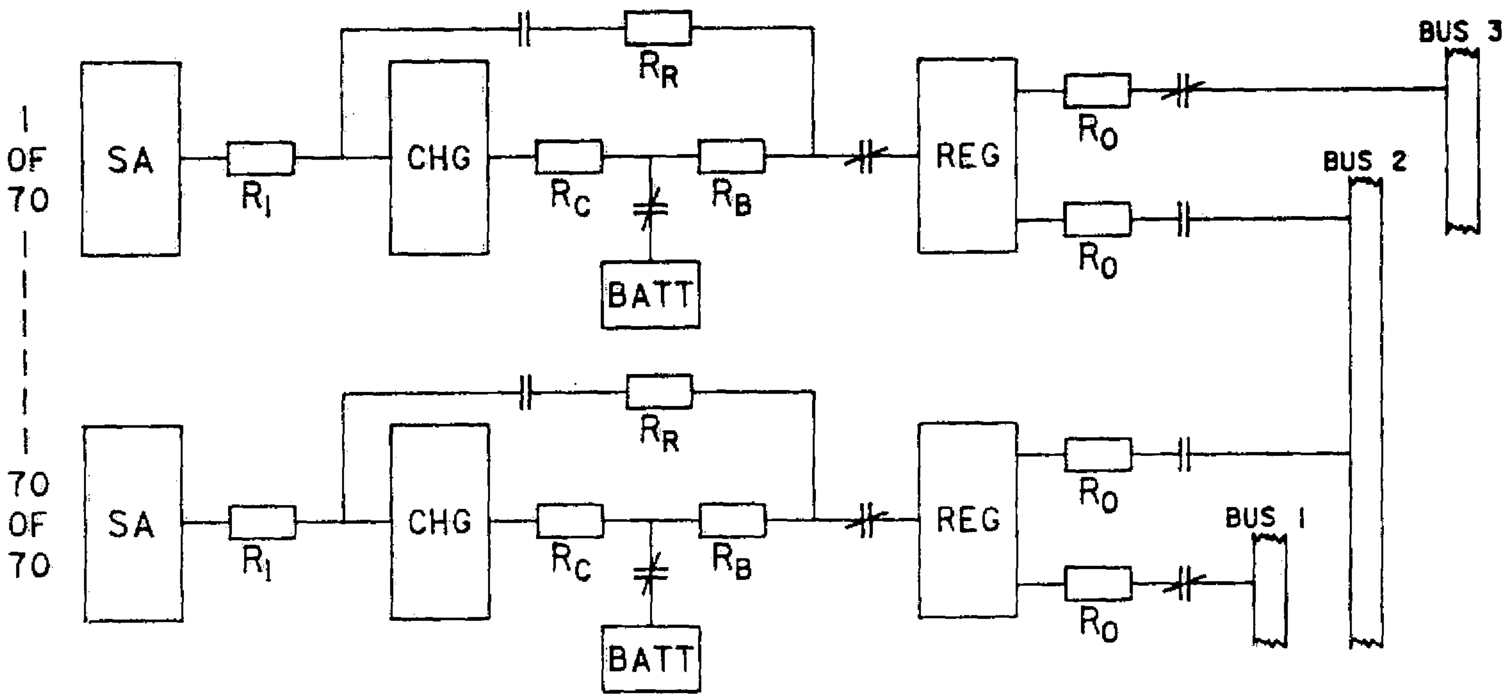
Figure 4. Power Module September 1977 baseline electrical power system.



LEGEND:

SA – SOLAR ARRAY
R_I-R_O – CABLE RESISTANCES
P_3 – PROGRAMMABLE POWER PROCESSOR
BATT – BATTERY
RPC – RECONDITIONING & PROTECTION CIRCUITS
REG – REGULATOR
RCCB – REMOTE CONTROLLED CIRCUIT BREAKERS

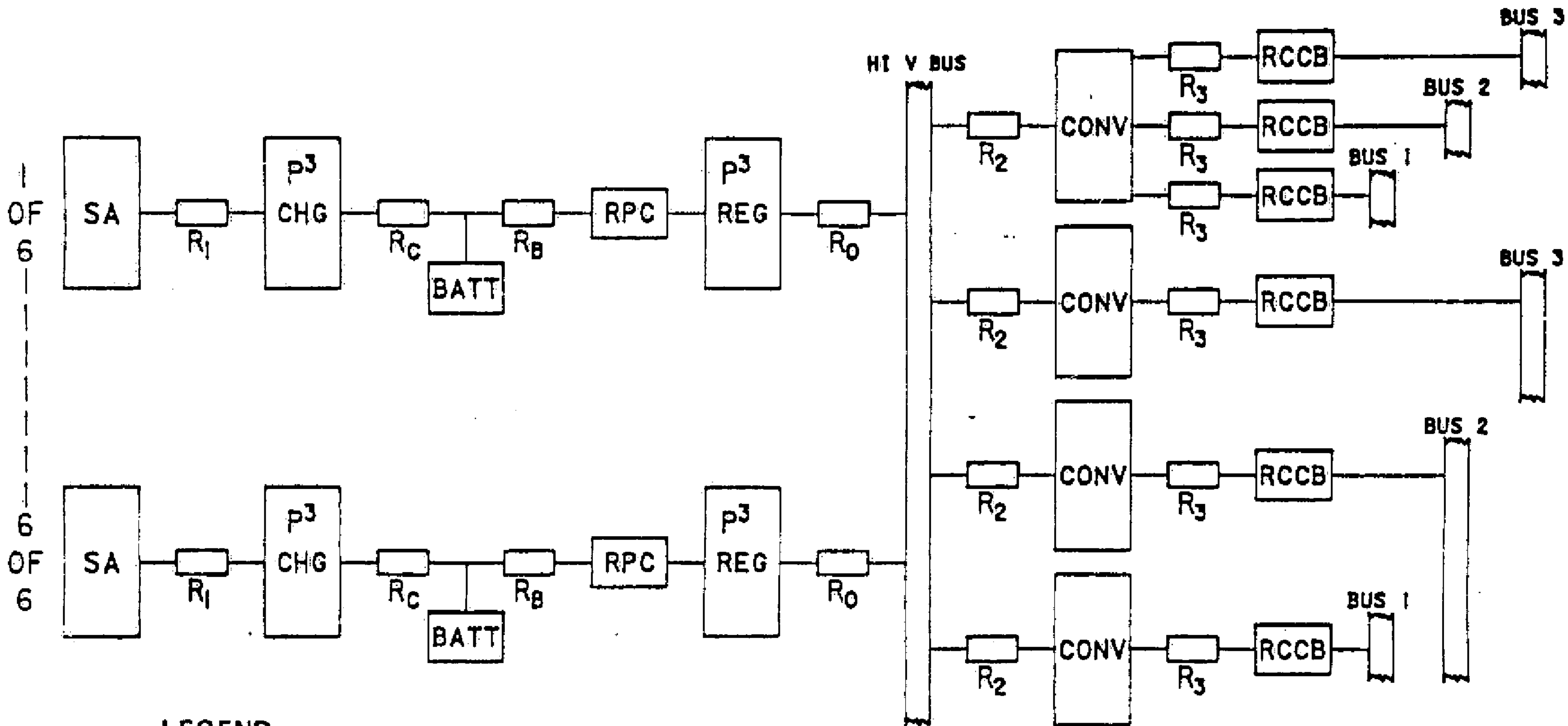
Figure 5. Power Module Alternate No. 1 — 12 PPG EPS (proposed December 1978 baseline).



LEGEND:

SA — SOLAR ARRAY
R_I-R_O — CABLE RESISTANCES
CHG — CHARGER
BATT — BATTERY
REG — REGULATOR

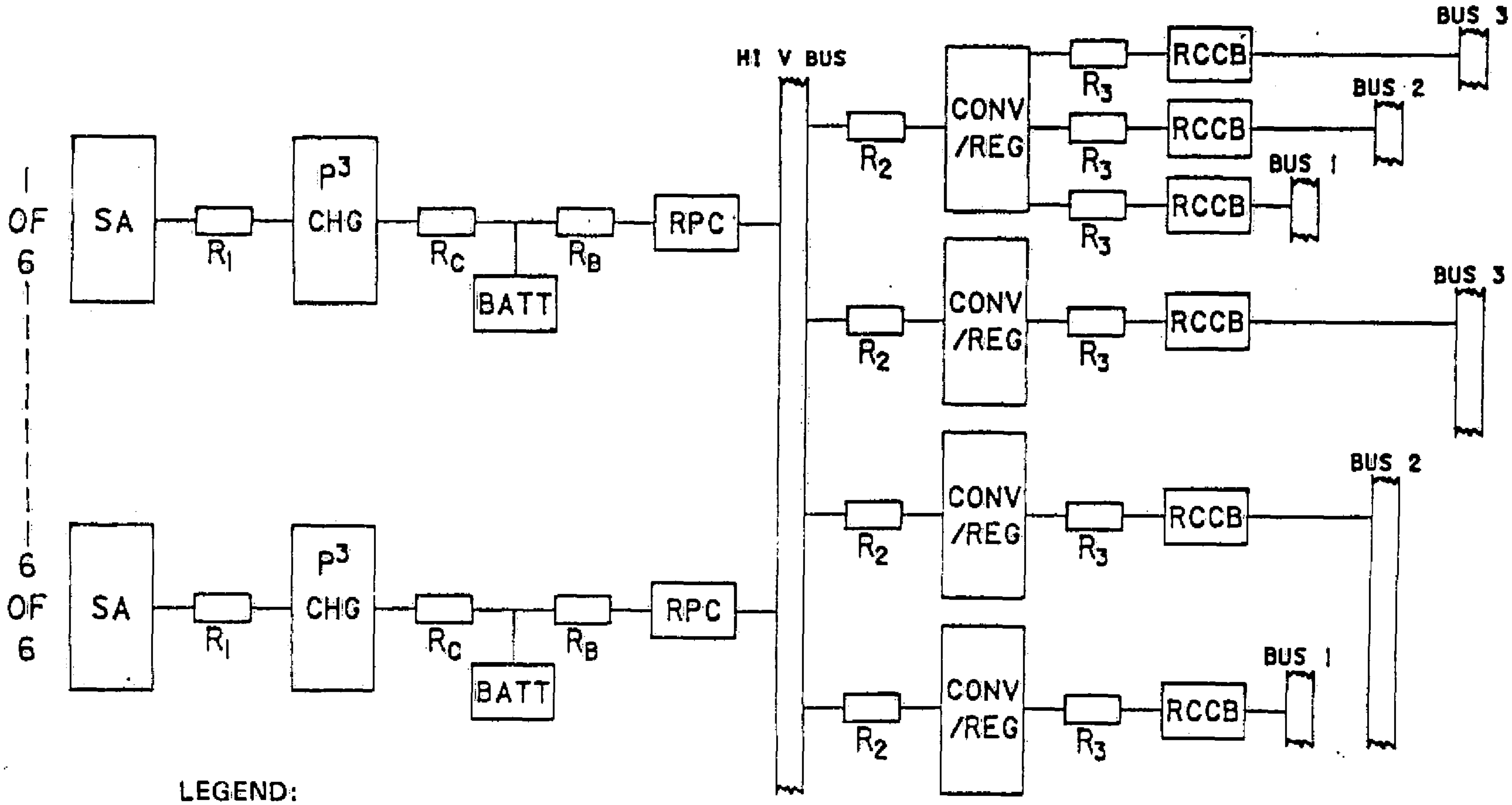
Figure 6. Power Module Alternate No. 2 — EPS with Skylab (airlock module) components.



LEGEND:

SA — SOLAR ARRAY
R_1 - R_3 — CABLE RESISTANCES
P_3 — PROGRAMMABLE POWER PROCESSOR
CHG — CHARGER
BATT — BATTERY
RPC — RECONDITIONING & PROTECTION CIRCUIT
REG — REGULATOR
HI V BUS — HIGH VOLTAGE
CONV — CONVERTER
RCCB — REMOTE CONTROLLED CIRCUIT BREAKERS

Figure 7. Power Module Alternate No. 3 — six PPG/four converter EPS.



LEGEND:

SA — SOLAR ARRAY
R_1-R_3 — CABLE RESISTANCES
P_3 — PROGRAMMABLE POWER POWER PROCESSOR
CHG — CHARGER
BATT — BATTERY

RPC — RECONDITIONING & PROTECTION CIRCUIT
HI V — HIGH VOLTAGE
CONV — CONVERTER
REG — REGULATOR
RCCB — REMOTE CONTROLLED CIRCUIT BREAKERS

Figure 8. Power Module Alternate No. 4 — six P^3 charger/four converter-regulator EPS.

processing and distribution subsystems have a growth capability, with no change in the hardware, to over 100 kW by reprogramming the power output to 120 V.

The following subsections of section 3.2 are allocated primarily to the major elements of the EPS. Subsection 3.2.2 is a brief, but relatively detailed, description of the solar array proposed for the new baseline EPS. Subsection 3.2.3 depicts both the utility and cost-saving versatility of the MSFC developed programmable power processor, wh'le subsection 3.2.4 discusses the batteries and the battery reconditioning and protection circuits envisioned for the new baseline. Subsection 3.2.5 details yet another MSFC developed EPS item, the standard load center converter, an item which has been accepted as a NASA standard. Subsections 3.2.6 through 3.2.9 discuss proposed switching, distribution, control indication, and on-orbit maintenance intentions.

3.2.2 Solar Array Electrical Design Description. The proposed 25 kW Power Module electrical design employs two array wings each composed of split array, flat-fold blanket strips (Fig. 9). Each of the four blanket strips provides 15 kW end-of-life (EOL) at the base of the array at a temperature of 70°C. The blanket strips are made up of panels hinged together. Each panel contains two electrical modules. Flat conductor cable power harnesses are attached at the two edges of each strip and are connected to the module power and return electrical pads. The modules utilize a lightweight, flexible printed circuit substrate of the SEPS solar array design. Wraparound contact silicon solar cells that are 2 by 4 cm are used. The cells are 8 mils thick and are covered with 6 mil fused silica. The module's 1530 solar cells are interconnected with 3 cells in parallel and 510 cells in series to provide 180 V at the base of the array, EOL 70°C. The covered cell assemblies are 12.4 percent efficient beginning-of-life (BOL). With a requirement for an operational lifetime in space of 2.5 years and for 12 power sources, each blanket strip will consist of 51 panels. Analyses on solar cell temperature versus time in orbit and power degradation due to trapped radiation were performed in support of the proposed array. Table 1 contains preliminary, summary level data on the electrical design of the array.

3.2.3 Programmable Power Processor. To take advantage of the rapidly developing technology area of microprocessors, MSFC EPS engineers began developing the programmable power processor, a multipurpose device [3] which, with different programming, can function as both a regulator and a charger. In fact, the processing power of the microprocessor also allows it, quite readily, to incorporate a peak power tracking characteristic. This

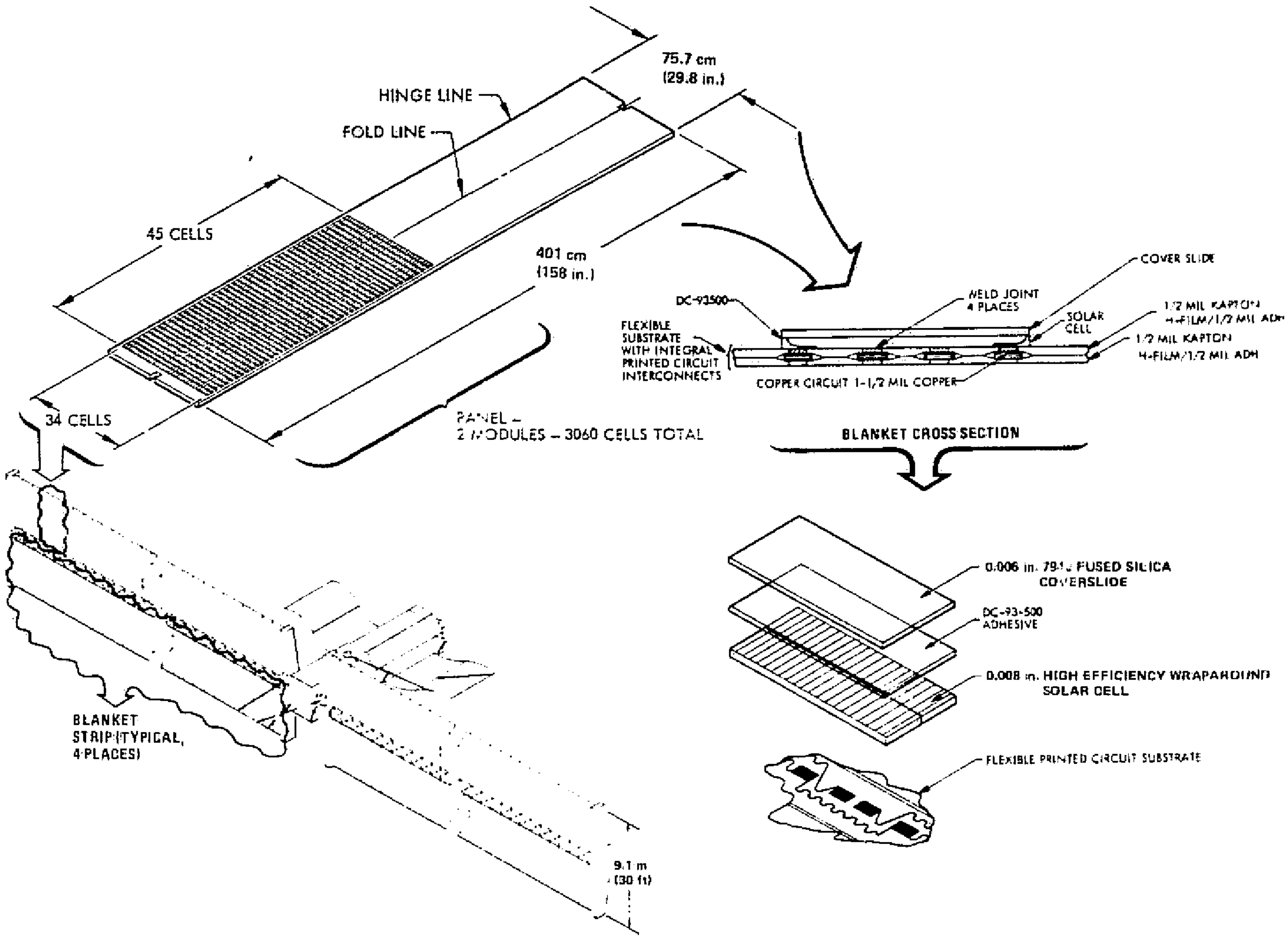


Figure 9. Standard solar array panel (with blanket and cell details).

TABLE 1. 25 kW POWER MODULE ELECTRICAL DESIGN (PRELIMINARY DATA) – DECEMBER 1978

<u>Solar Cell</u>

2 by 4 cm, wraparound contact, 8 mil silicon solar cell; base resistivity is 2 ohm-cm AMO, 28°C cell assembly efficiency = 12.4%

<u>Cell Cover</u>

6 mil fused silica, UV filter, 350 nm cut-on

<u>Cover Adhesive</u>

DC-93-500

<u>Substrate</u>

Laminated printed circuit, 1 oz copper rolled annealed interconnect. Insulation is two sheets of 0.5 mil kapton/0.5 mil high temperature polyester adhesive.

<u>Electrical Module</u>

510 series string cells
3 parallel string cells
1530 cells/module
2 modules/panel
3060 cells/panel

BOL module V_{mp}, base of array (B = 0°, temperature = 70.4°C) = 93.6 V
Two modules in series, V_{mp} = 187.2 V

EOL module V_{mp} base of array (B = 0°, temperature = 70.4°C) = 90.9V
Two modules in series, V_{mp} = 181.8 V

BOL V_{oc}, base of array (B = 0°, temperature = 71°C) = 189.3 V
Two modules in series, V_{oc} = 378.6 V

Voltage temperature coefficient = 0.45%/°C

TABLE 1. (Concluded)

<u>Radiation Losses, 2.5 years</u>

$P/P_o = 0.955$ (4.5% loss)

$V/V_o = 0.971$

<u>Sizing Assumptions — Power Losses</u>

Thermal cycling losses, 1.5% (I)
UV and thermal control losses, 1.2% (I)
Assembly loss, 1.5% (I) + 1.5% (V), 8.2%
Harness loss, 2% (V)
Diode loss, 0.5% (V)
Total power loss in 2.5 years — 8.2% + 4.5% = 12.7%

$$\text{BOL cell power} = \frac{60\ \text{W}}{0.873} = 68\ 729\ \text{W}$$

$$\frac{\text{No. of cells}}{\text{Array system}} = \frac{68\ 729\ \text{W}}{0.1353\ \text{W/cm}^2 \times 8.088\ \text{cm}^2 \times 0.124 \times 0.81\ \text{temperature factor}}$$

$$= 625\ 304$$

Four blankets, cells/blanket = 156 326

$$\frac{\text{No. of panels}}{\text{Blanket}} = \frac{156\ 326}{3060} = 51$$

multipurpose device has been designed, developed, and extensively tested to ensure that it performs as expected. Although it has not been flight packaged and undergone flight qualification testing, it has been developed to a "brass-board" stage -- a point that allows incorporation with assurance of this device into the new baseline EPS.

Technically, the programmable power processor is a high voltage, high power switching regulator whose output or input characteristics are rigidly controlled by a microprocessor. Control is implemented by the microprocessor selecting a particular feedback scheme within the switching regulator and then varying the reference voltage in the feedback loop. Determination of the reference voltage can be by external command or based on calculations performed by the microprocessor using data from analog or discrete inputs. For example, to function as a battery charger with peak power tracking, the microprocessor would select the mode that gave it control of the input current. By incrementing the input current, the microprocessor can sample the input power at different points and determine the optimum operating point. It simultaneously monitors the battery to determine when to reduce the input power so as to control the battery charging.

In reality, the possible applications for the P^3 are limited only by the hardware capabilities of the switching regulator. However, this is not a rigorous limitation since a buck regulator configuration is used that gives efficient and controllable operation over a wide range of input and output conditions. The regulator output is capable of regulating 100 A at any output voltage from 22 Vdc to 200 Vdc with input voltages as high as 375 Vdc. This versatility and programmability allows a wide range of applications not only for the Power Module but for other high power space vehicles as well.

As noted above, for the Power Module the same hardware may be used for both charger and regulator, thus eliminating the development cost of a major power processing element and resulting in a significant cost savings. In the charger mode, the P^3 will nominally operate with approximately 180 V input and 160 V, 15 A output at greater than 95 percent efficiency. In the regulator mode, it will nominally operate with 140 to 180 V input and 30 V, 75 A output at greater than 87 percent efficiency. This provides 27 kW to a 30 V bus from a 12-group system. As a note of interest, the same power processing hardware, with adequate source and energy storage, programmed for a 120 V bus could supply 108 kW.

3.2.3.1 Chargers. In this baseline, the chargers are P^3's programmed to track solar array peak power and charge the high voltage batteries during sunlit portions of the orbit. They are capable of performing this function while connected to one or two solar array sources. To keep up with the depth of discharge, the P^3 will maintain a running integration of battery percent state of charge and, in conjunction with appropriate battery voltage temperature curves, will be programmed to help assure optimum battery charging throughout the battery life. The programmable capability of the P^3 will allow the optimum battery charge characteristics to be selected and programmed at any point in the battery development program. The P^3 charger, with an output capability of 100 A, has the capacity to charge batteries with up to 100 A-h capacity.

3.2.3.2 Regulator. In this baseline, the regulators are P^3's programmed to provide a nominal 30 Vdc that is compatible with the Shuttle Orbiter electrical power system. Programmed slopes and selectable open circuit voltage settings will be provided to assure proper sharing between the 12 regulators of the new baseline EPS.

3.2.4 Batteries and Battery Reconditioning and Protection Circuits. For the new baseline EPS five 60 A-h multicell NiCd batteries have been selected for use because of the many favorable performance characteristics they exhibit. The five NiCd batteries are connected in series to provide a high voltage battery. Unfortunately, even these batteries in this configuration exhibit "memory" (also called "fading") and "cell reversal" characteristics — less favorable battery characteristics which can seriously weaken power systems. However, MSFC experiences with the ATM batteries, and subsequent analysis of Skylab data combined with laboratory investigations, have led to the development of ways to largely circumvent both problems and greatly increase the reliability and useful life of the batteries.

With nominal operation, the usable capacity (a certain amount of current for a specified length of time) of the NiCd battery will invariably decrease because of memory or fading. If the battery voltage is not maintained above a certain level, it becomes unusable as part of the entire system. As the system is operated and the battery capacity fades, the rate at which it approaches this level increases, thus the usable capacity decreases. Testing has proven that if each cell in the battery is discharged to approximately 0.5 V and held at this level for 2 to 3 days with no cell reverse voltage, then "reconditioning" is accomplished and capacity is restored [4]. A circuit, the BRPC, to do this has been breadboarded and tested, and has proven to be very effective in restoring the battery capacity.

Sometimes, even during normal operation, a cell may also degrade to the point that it collapses in voltage and the remainder of the cells in the battery will force a reverse potential on that cell. If the voltage on a cell is reversed, permanent damage can and does occur. In addition to performing a restoration function, the BRPC will also maintain that cell at approximately 0.5 V and prevent reversal from occurring. Each output of the circuit is diode isolated from the cell so that it will only supply power when the cell voltage approaches 0. V; otherwise, only low standby power is needed by the circuit.

In view of past experiences, a BRPC seems very much needed on future power systems that will employ NiCd batteries. When the Skylab space station was launched, severe loading was induced on the batteries on the ATM due to the failure of the airlock module (AM) power system. When the situation was corrected, a severe loss of usable capacity had resulted on the ATM batteries. If the BRPC circuit could have been employed on these batteries, the capacity would have been restored. Now, during the Skylab reactivation activities, one battery has become unusable due to damage resulting from reverse voltage on some cells. Again this circuit would have prevented this. Pragmatically, a circuit of this type was considered an integral, and mandatory, part of this baseline EPS. To illustrate why, a reliability analysis for a high voltage battery with and without a battery protection circuit indicated a battery reliability of 0.9995 and 0.2508, respectively, for 5 years of operation.

3.2.5 Standard Load Center Converter (SLCC). An SLCC developed by MSFC is now qualified and available as NASA standard hardware for the Power Module. The SLCC accepts inputs from an unregulated source and provides tightly regulated outputs at voltage levels required by a user. The modular design, consisting of a mainframe and plug-in output regulator modules, allows the user to select up to four discrete output voltage levels with power levels up to 100 W per plug-in module (400 W maximum per SLCC). One, two, and four module configurations are available. Output regulator modules are presently available with the more common outputs of ±5 Vdc, ±15 Vdc. Other voltages between 4 Vdc and 80 Vdc may be obtained by selecting the proper output transformer.

The SLCC specifications are as follows:

Input Voltage: 21 Vdc to 36 Vdc
Output Voltage: One to four discrete levels, 4 to 80 Vdc
Output Power: 100 W to 400 W

Voltage Regulation: ±0.25 percent (adjustable to +0.1 percent)
Output Ripple: 1.0 percent maximum p-p
EMI: Per SL-E-0002 and MIL-STD-461A
Isolation: 50 megohms at 50 Vdc at following points:

Input Common to Output Common
Input Common to Chassis
Output Common to Chassis

Operating Temperature: -20°C to +85°C
Weight: 400 W (four module unit) — 6.6 kg (14.6 lb)
200 W (two module unit) — 4.1 kg (9.0 lb)
100 W (one module unit) — 2.7 kg (6.0 lb)

Other features include remote startup/shutdown capability, overvoltage and short circuit protection, and remote sense capability.

On the 25 kW Power Module the hardware could provide (a) housekeeping voltage sources, (b) precision voltages, and (c) isolated power sources and auxiliary buses.

3.2.6 Switching. The necessary contactors, remote power controllers (RPC), and remote controlled circuit breakers (RCCB) will be provided as shown in Figures 5 and 10 to deadface, interrupt, and route power as required.

3.2.7 Distribution. The electrical power distribution system will be a three bus system, with the Orbiter and free-flyer interfaces as shown in Figure 10. The Power Module distribution will be a two bus system, with the capability to switch the third bus to either power module distribution bus.

The three main buses shall be packaged in individual distributors. The bus shall be circuit protected and have the capability to be isolated in the distributor.

The distributors shall be of the same design and only the positive wires will be routed through the distributor. All negative feeders will be bussed together in the vicinity of the three main bus distributors (Fig. 11). The negative bus will be isolated from the structure and contactors provided to connect the bus to structure. Minimum connectors will be utilized to route the main feeders. The distributors considered should have the capability to plug wires directly to the internal devices. Each distributor shall be designed for a continuous peak of 15 kW.

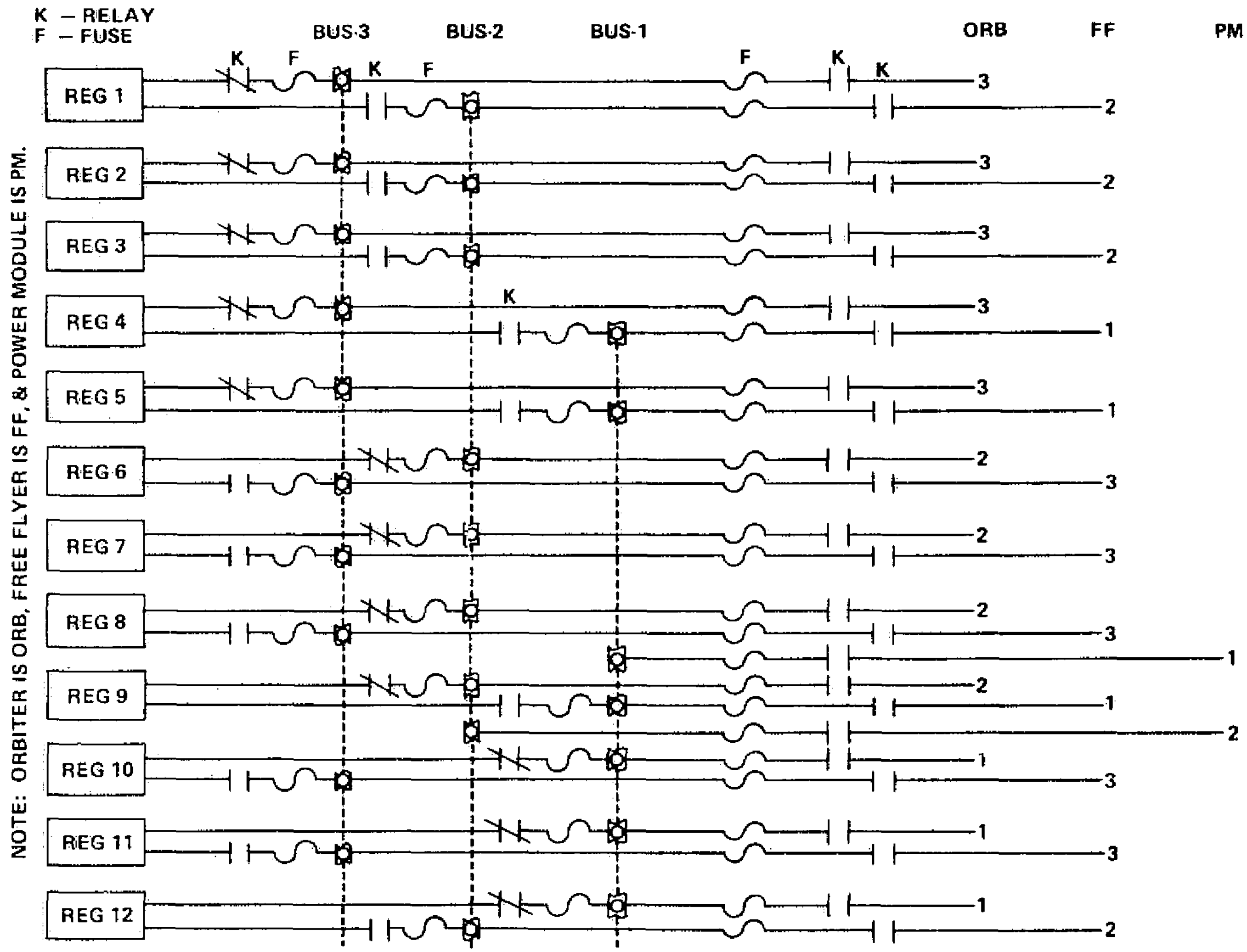


Figure 10. Power distribution system.

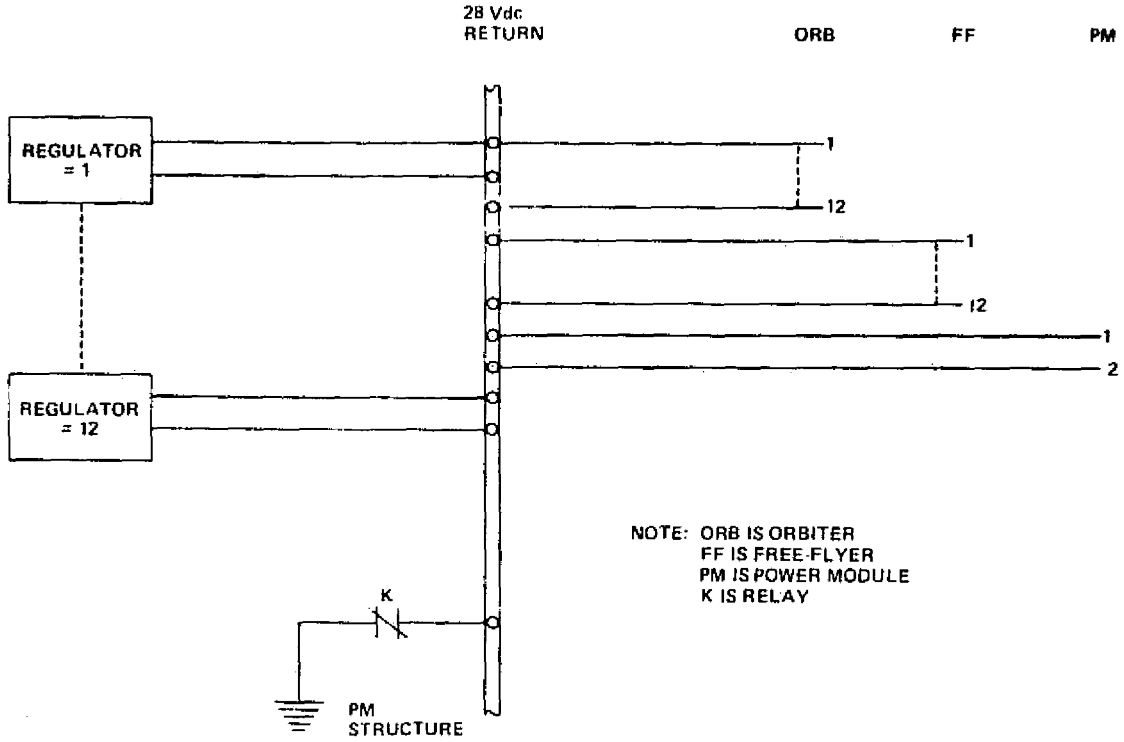


Figure 11. Power system grounding concept.

The PM internal distribution system shall utilize ATM distributors (main, auxiliary, control, and measuring distributors).

The distributors shall provide the capability of switching each P^3 to two of the three main buses. Each feeder in and out of the distributor shall be circuit protected with switching capability. Dead facing for free-flyer and Orbiter interfaces shall be provided.

The networks shall provide the capability for switching all loads and buses and providing wire protection to all components.

3.2.8 Control and Indication. Control and indications of all network distribution systems, power processors, and sources shall be provided to the Orbiter for power management by the crew and also provided to the ground for power management.

Caution and warning (C&W) will be provided for a fail-safe system.

3.2.9 On-Orbit Maintenance. It is anticipated that battery modules (22 cell packs) and P^3's (chargers or regulators) could be replaced in case of a failure. However, no planned maintenance would be required; all components in the system would be designed for the entire 5 year mission life including the batteries.

3.3 Attitude Control System

3.3.1 Introduction. The ACS proposed for the new baseline for the 25 kW Power Module will provide attitude control in all modes of operation. When operating in the sortie mode, the Orbiter VRCS will be used to execute most maneuvers and assist in desaturation of the CMG's when required. The proposed ACS functional block diagram is shown in Figure 12. The ACS will consist of three Skylab CMG's for actuation, and nine Skylab rate gyros and two Skylab acquisition Sun sensors for attitude/attitude rate sensing. The digital flight computer will contain the bending filters, control law, N-tuple steering law, momentum management scheme, and other maneuvering, stabilization, and control functions. Comparative trades on steering laws as well as trades on CMG reliability were conducted, and the results were used to develop the baseline ACS proposed. Other trades (such as on a configuration using additional CMG's or magnetic torquers as a backup ACS) were conducted, and while the preliminary results look promising, additional analysis is required.

3.3.2 Basic Assumptions. The basic assumptions and guidelines are the same as for the phase A study with the exception of a NASA Headquarters guideline to provide a basis for the development of larger, more capable orbiting systems. Those basic assumptions and guidelines are:

a) Use existing Skylab equipment to the fullest extent possible. Rationale — To minimize cost.

b) Maintain active control during the storage mode. Rationale — To ensure trickle charge to batteries.

c) The solar arrays will have one degree of freedom in rotation. Rationale — One degree allows a simple system, and, if limited to ±180 deg or less, there will be no need for slip rings.

ATTITUDE CONTROL SYSTEM BASELINE

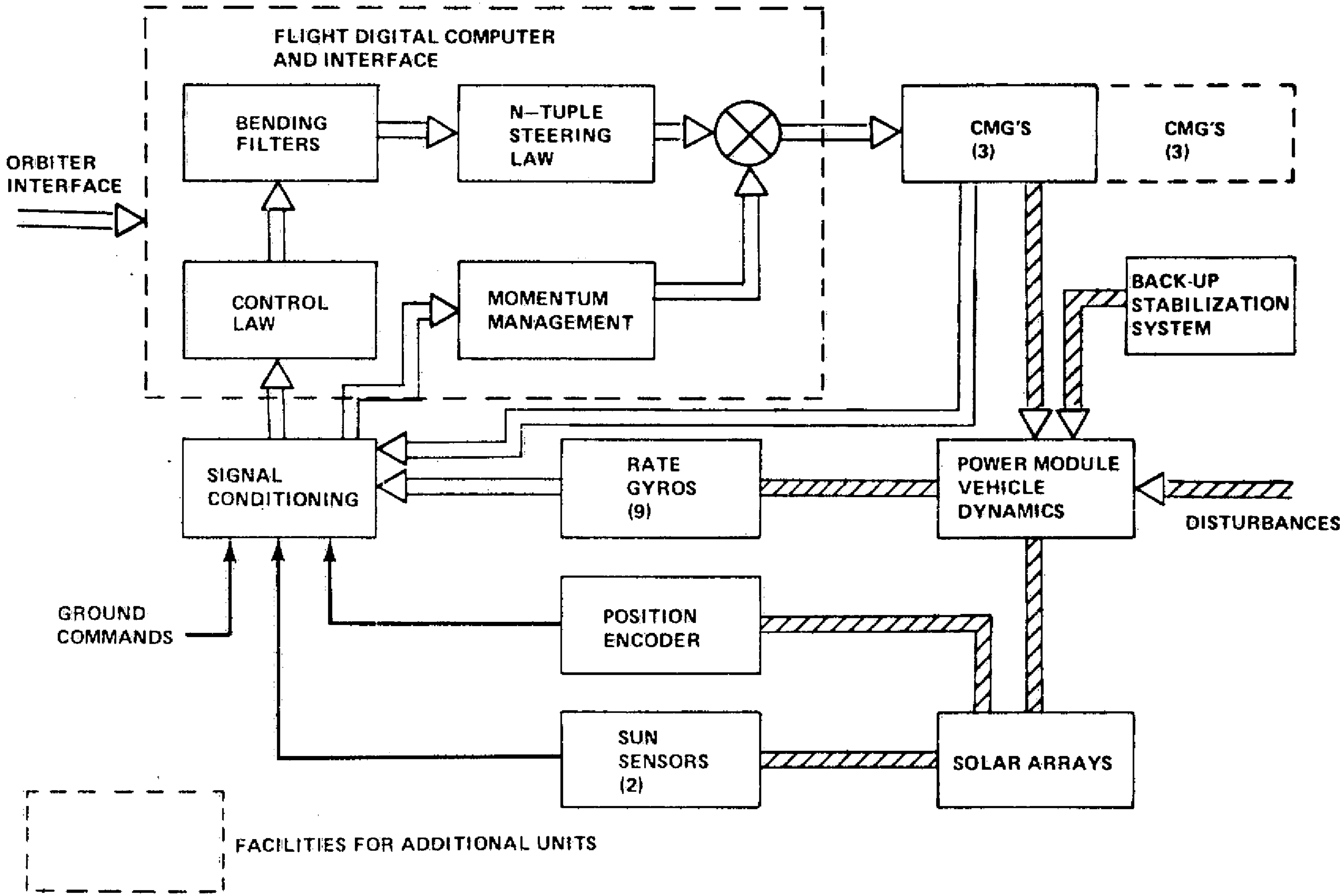


Figure 12. 25 kW Power Module attitude control system — September 1978.

d) Provide limited maneuvering capability in the sortie mode. Rationale — To provide savings of Orbiter VRCS consumables.

e) The PM will provide control for the Orbiter in the standard reference orientation relative to the orbital plane (X-POP, X-LV, etc.). Rationale — Reference systems requirements document.

3.3.3 Maximum Capability. System performance is essentially fixed by the ground rule to use three Skylab CMG's; therefore, the momentum buildup profiles are the same as the phase A study, which are shown in Figure 13 for the sortie mode and Figure 14 for the free-flying mode. However, trades are being conducted to study the effects of increasing the number of CMG's as well as looking into backup ACS's: passive stabilization, a reaction control system, electromagnetic torquing, and a six CMG configuration.

The nominal orientation for the sortie mode will be X-POP (X-axis perpendicular to the orbital plane). The nominal orientation in the free-flying mode will be dependent on the attached payload, but the free-flyer will be capable of assuming any of the standard orientations with respect to the orbital plane coordinates as shown in Figure 14.

Maneuvering of the PM as a free-flyer will be provided by the CMG's. In the sortie mode, only limited maneuvering will be possible with three CMG's. It is proposed, however, to execute most maneuvers in the sortie mode by commanding the CMG's and then letting the Orbiter VRCS be activated as the CMG's approach saturation. The VRCS will be programmed to be activated to desaturate the CMG's as a normal operation. Utilizing the CMG's in this manner for maneuvering will conserve VRCS propellant. Maneuvering rates for both the sortie and free-flyer modes are shown in Figure 15.

Pointing and stability accuracies in the sortie mode will be comparable with Skylab, i.e., ±6 arc min pointing accuracy and ±9 arc min for 15 min for stability. In the free-flyer mode, the stability accuracy will probably be maintained for longer periods of time because of the absence of Orbiter crew disturbances.

3.3.4 Attitude Control System Hardware. The hardware units for the attitude control system are shown in the function block diagram of Figure 12. The major components are the CMG's, rate gyro processors (RGP), Sun sensors, solar array articulation controller and position encoder, and the signal conditioning interface unit (SCIU). A backup stabilization system has not been

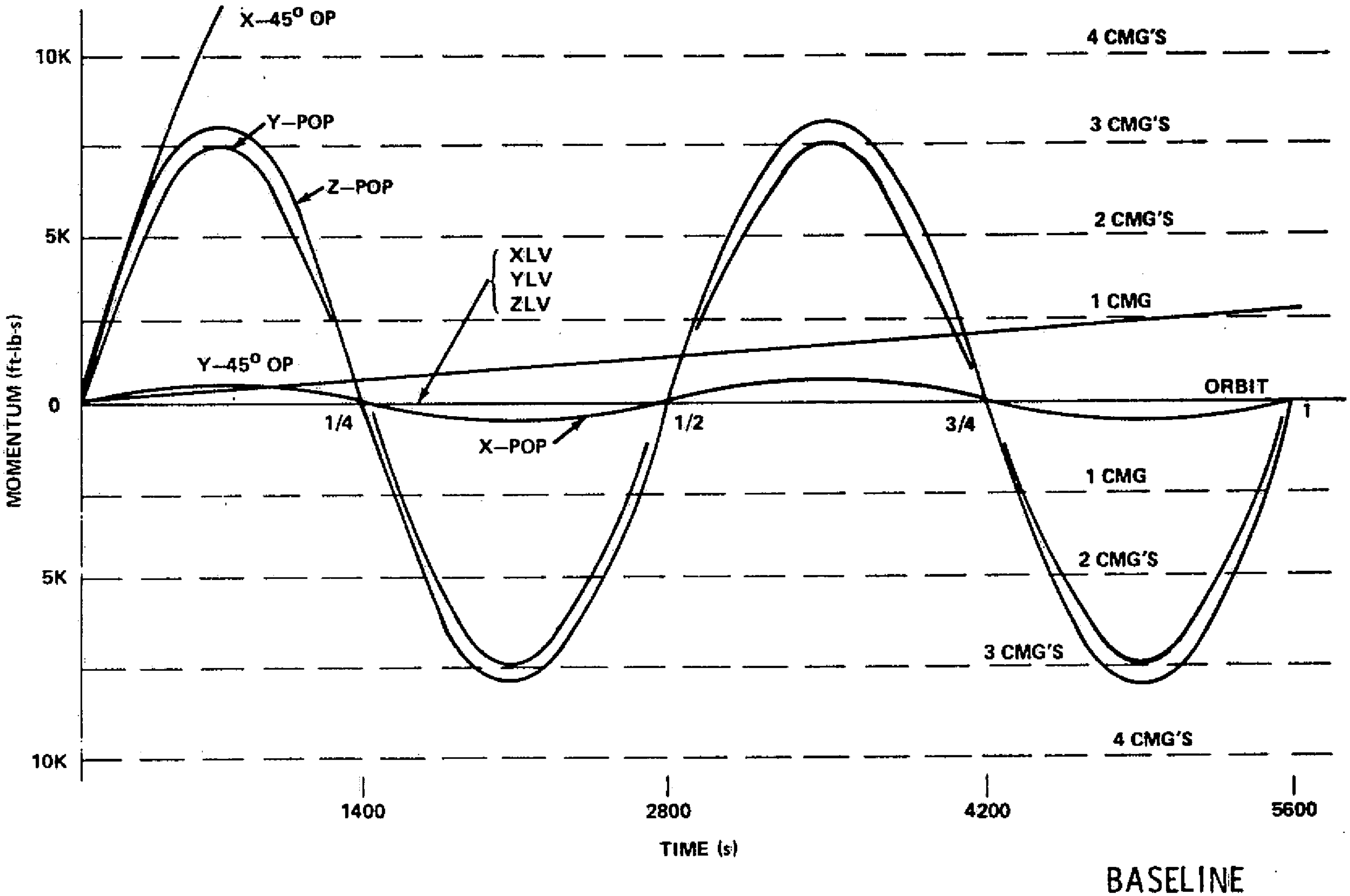


Figure 13. Power Module/Orbiter/payload momentum buildup profiles.

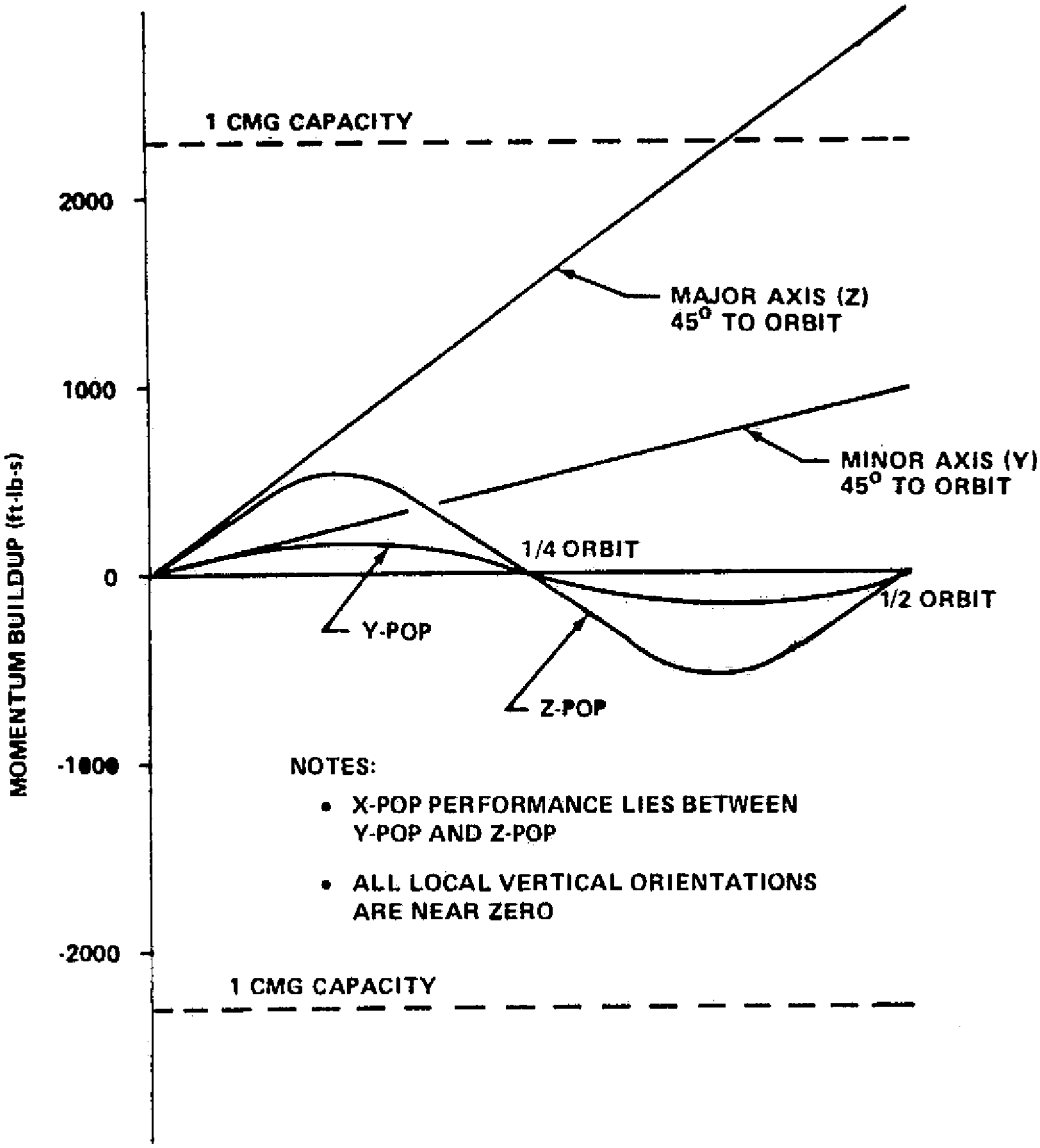


Figure 14. Typical momentum buildup on Power Module with free-flyer payload.

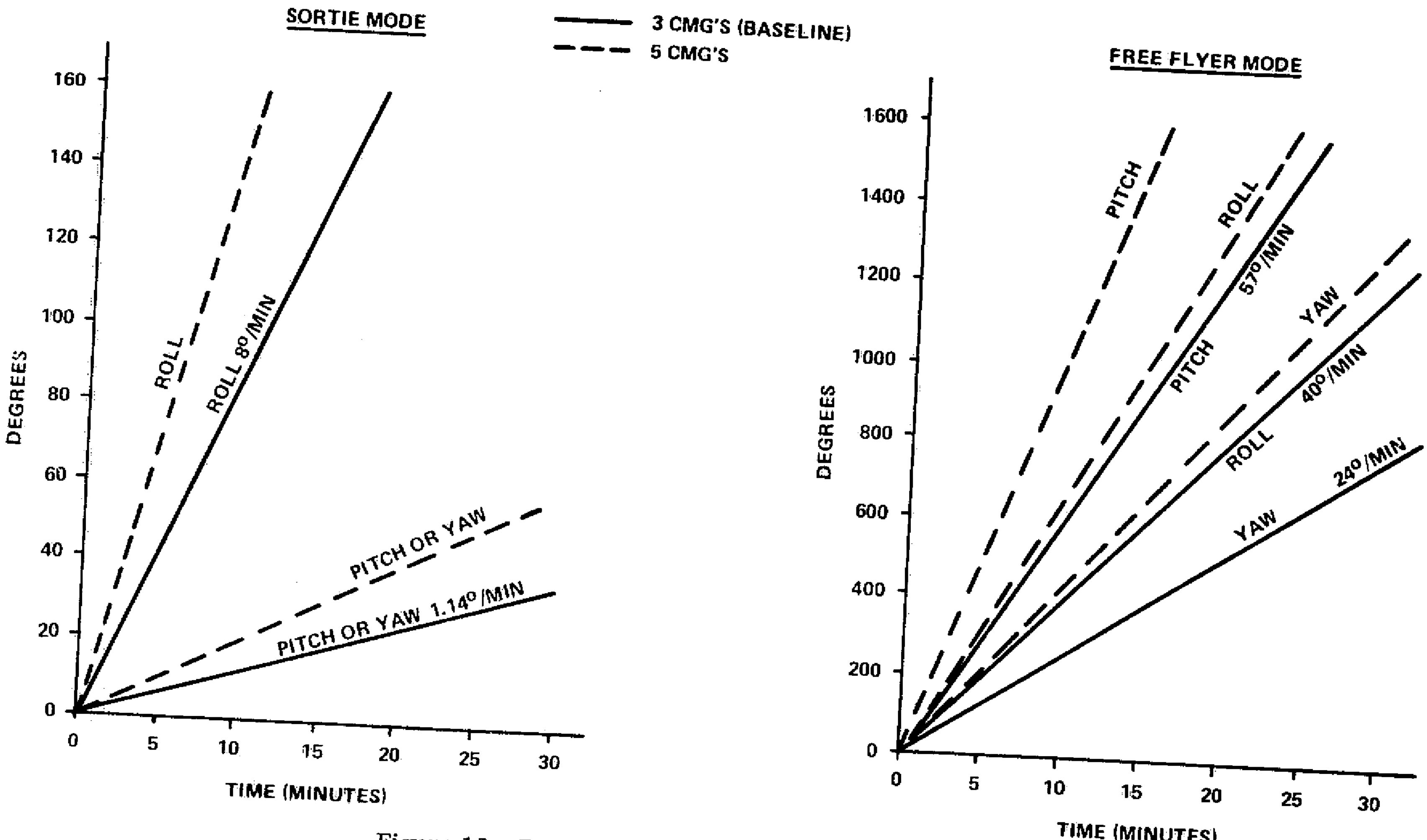


Figure 15. Power Module maximum maneuver capability.

defined or selected; therefore, there is no specific data on that equipment. A brief description of the foregoing hardware, its current status, recommended modifications, and planned testing is given in the following subsections.

3.3.4.1 Control Moment Gyros. The surplus ATM CMG's were specified to be used as the principal actuator for attitude control.

3.3.4.1.1 Hardware Description. The ATM CMG consists of a motor-driven rotor that is gimbaled to provide two degrees of freedom. The instrument, by utilizing its rotation momentum, is capable of producing torques on a vehicle proportional to the angular rate of the gimbals. By controlling the CMG gimbal rates, attitude control of a vehicle can be achieved.

3.3.4.1.1.1 Inner Gimbal (IG) and Rotor Assembly. The 21-in. diameter rotor rotates at a speed just under 9000 rpm and has an angular momentum of approximately 3000 Nms. The rotor is supported at each end by a single angular contact bearing and is constrained such that the correct bearing preload is maintained regardless of orientation during its operation within a wide environment range. Lubricant is supplied to each bearing by means of a lubrication nut that also locks the bearing inner race to the shaft. The IG and rotor assembly is instrumented so that bearing temperature, rotor speed, and cavity pressure can be monitored.

3.3.4.1.1.2 Motors. The rotor is rotated by two identical double squirrel cage induction motors. The motor rotors are an integral part of the rotor shaft, and the stators are mounted in the gimbal. The two motors provide redundancy as well as a symmetrical design for balanced heat dissipation. A single motor is capable of maintaining sufficient rpm if a failure occurs.

The motors are driven at 8900 rpm by 130 V, 455 Hz three-phase electrical power supplied from a CMG inverter assembly (CMGIA). Synchronous speed for the six-pole motor is 9100 rpm. The motors develop a relatively constant accelerating torque of 0.127 Nm up to the region near synchronous speed. Just below synchronous speed, the motors provide a steep torque versus speed characteristic that makes them behave as synchronous motors for good speed regulation with varying load. Input power is relatively constant during acceleration and drops sharply near synchronous speed, resulting in high operating efficiency.

The double cage induction motor uses two squirrel cage windings placed in slots one above the other. The outer cage winding has a high resistance and high leakage inductance, while the inner cage winding has low resistance and high leakage inductance. This combination tends to maintain a constant power transfer to the rotor and a relatively constant level of developed torque during acceleration. When the motor reaches the normal operating speed, the reactances of the cage windings are small compared to the resistances. The lower resistance inner cage then provides the major share of torque-producing capability to the motor. The characteristics of a low resistance rotor at low slip are high efficiency and good speed regulation through a range of varying load torques.

3.3.4.1.1.3 Motor Braking. The motor brake can be actuated by a crewman or by ground command to disconnect a CMG in case of a failure. Electrical means for providing braking toque to stop the CMG wheel consist of applying a dc voltage (from the CMGIA) between two phases of the wye-connected motor stator; the third phase is left open.

3.3.4.1.1.4 Gimbal Seals and Covers. The rotor and inner gimbal assembly are sealed so that during Earth testing, the gimbal cavity pressure can be maintained at a level of 100 μm of mercury or less. This is necessary to minimize windage friction losses on the high speed rotating wheel. During orbital operation, the evacuation valve will be opened and the gimbal cavity vacuum maintained by outer space.

External covers are provided and are attached to the upper and lower surfaces of the CMG frame. A thermal insulating blanket is attached to the CMG covers to assist in maintaining the proper thermal environment for the IG and rotor assembly.

3.3.4.1.1.5 Actuator Pivot Assembly. Both inner and outer gimbal rotation is accomplished by identical actuators. The actuator also serves as a gimbal pivot. Each actuator consists of a housing assembly, a dc torque motor, gear train, output shaft, and rate feedback dc tachometer.

3.3.4.1.1.6 Sensor Pivots. The sensor assemblies serve as gimbal pivots and provide electrical readouts that are used for gimbal position information. Each sensor consists of a housing, ball bearing mounted pivot shaft, resolver assembly, cam-operated microswitches, and a flex-lead assembly.

3.3.4.1.1.7 CMG Electronics Assembly (CMGEA). The CMGEA contains the rate servo electronics (amplifiers, modulator, detectors, etc.) to drive the CMG gimbals. The gimbal torquing rates are controlled by rate servoloops with a dc tachometer output for feedback. The rate servoamplifiers for either the inner or outer gimbal are functionally identical, differing only in servo characteristics.

3.3.4.1.1.8 CMG Inverter Assembly. Each CMG derives its gyro wheel power from an individual solid state inverter. The CMGIA also supplies 4.8 kHz to the CMG resolver.

3.3.4.1.2 Modifications to Existing Hardware. Experience with the CMG's during their extensive operation during the primary Skylab mission plus additional requirements imposed by the Power Module mission have resulted in a number of trade studies. These trade studies have resulted in the following:

a) Unlimited freedom of both the inner and outer gimbals was chosen, including bringing out the inner gimbal resolver cosine winding. This eliminates the gimbal stops and limit switches on both axes. (The N-tuple steering law requires unlimited freedom for the outer gimbal only.)

b) The added slip rings will be gold wire brushes, on gold over nickel over brass substrate rings. The base material will be sodium-aluminum-silicate filled epoxy. The housing will be 303 stainless steel. For reliability, the signal rings will have two wipers (where one is sufficient), and the power rings will have four wipers (where two are sufficient). Tests have shown that the originally selected Bray 815Z oil will not be suitable for lubricating the slip ring assembly. Present thinking is that a wet form of Vackote by Ball Brothers will be used. The slip rings will be of the cartridge form, having their own 440 steel sealed bearings with labyrinth housing seals. To reduce nonrecurring costs, the inner and outer axis slip ring assemblies will be identical, except that fewer rings will be machined into the inner axis assembly. The slip ring cartridges will be installed in the sensor pivots.

c) Only one of the three resolvers presently in each sensor pivot will be used, although the two remaining will probably be left in to provide mass balance.

d) Previous testing shows that no changes should be made in the torquer pivot design. The same motor, tachometer, gear train, bearings, seals, and lubricants will be used.

e) The rotor spin motors will remain unchanged. The spin bearings will incorporate an improved retainer design for better lubrication. The continuous-feed, spin-bearing lubrication system will remain essentially the same with the possible exception of an increased flow rate. All spin axes should be oriented normal to the flight direction at launch, and it may be necessary to have the rotors turning at a low rate (200 to 500 rpm).

f) A new rotor material has been tentatively chosen. Capenter Custom 455 stainless steel, precipitation hardened at 1000°F appears to be an excellent replacement for the highly stress corrosion susceptible 18 nickel 300 maraging steel formerly used. The major question remaining is whether or not the desired mechanical properties can be obtained in a forging this large, and whether or not a special forging die will be required.

g) The CMG vendor seems to have solved the reliability problem in the existing rotor speed pickoff. A screening process through thermal cycling is the apparent solution. An expanded speed range circuit will be added in the inverter assembly to measure small speed changes, thus permitting periodic monitoring of the spin bearing/lubrication system condition. Since speed information is also required in the steering law, it may be determined that redundant pickoffs should be employed. This would not be a significant mechanical change, since the original design accommodated dual-speed pickoffs.

h) The "as engineered" parts lists for the inverter assembly (IA) and the electronic assembly (EA) have been reviewed and given tentative approval for use even though "as built" lists are unobtainable on some units. The only electronic modification presently forseen for the IA is that of the expanded speed circuitry. The EA will be simplified somewhat due to removal of the gimbal stops and limit switch circuits.

3.3.4.1.3 On-Orbit Replacement Approach. Both the IA's and the CMG's with EA's will be replaced as units. No maintenance can be performed on these items while on-orbit. Preliminary concepts have been developed for orbital changeout. The IA will incorporate manually operated fasteners and electrical connectors and handles for manipulation by a crewman. The CMG with EA will require both the RMS and extra-vehicular activity (EVA) for changeout. For this purpose, a simple grapple fitting should be made a permanent part of each CMG structure. The changeout scenario could be as follows:

a) RMS attaches to CMG grapple fitting.

b) Fasteners and electrical connectors are released by EVA.

c) RMS moves CMG to tie-down point within the Shuttle cargo bay.

d) Fasteners are secured by EVA.

e) RMS attaches to replacement CMG in cargo bay.

f) Fasteners are released by EVA.

g) RMS moves CMG to its place on the Power Module.

h) Fasteners and electrical connectors are secured by EVA.

i) RMS is removed from CMG.

Because this scenario is complex and the CMG is a large mass, a neutral buoyancy simulation is planned to verify the orbital changeout design and procedure. Performance and design data are given in Table 2.

3.3.4.2 Rate Gyro Processor. System oriented trades have shown that the use of surplus ATM RGP's provides no clear cost advantage over using new RGP's. However, because of MSFC familiarity with them, surplus ATM RGP's are used as the attitude rate sensors for this Power Module baseline. The initial baseline called for the use of nine RGP's. As a result of the trade studies, it is now proposed to use either six in a dodecahedron configuration or four in a skewed configuration similar to that of the HEAO. Either of these configurations requires different mechanical/electrical layout, software, and alignment procedures.

3.3.4.2.1 Hardware Description.

3.3.4.2.1.1 General. The ATM RGP consists of a single Kearfott series 2519 rate integrating gyroscope and the electronics necessary to operate

TABLE 2. CMG PERFORMANCE AND DESIGN DATA

Angular Momentum Storage	2300 ft-lb
Rotor Operating Speed	8900 rpm
Number of Gimbals	Two
Gimbal Freedom	Both Unlimited
Control Torque Output	
Range	0 to 280 ft-lb
Threshold	0.25 ft-lb
Gimbal Rates	
Range	0 to 7.0 deg/s
Threshold	0.0057 deg/s
Precession Rate Bandwidth	4 to 10 Hz (Function of Gimbal Angles)
Electrical Power	28 Vdc
Rotor Spin Up	100 W for 14 hours
Rotor at Operating Speed	50 W
Bearing Heaters (16 to 27°C)	48 W
Size[a]	
CMG with Electronic Assembly	Within 42 in. Diameter Sphere
CMG Inverter Assembly	22 by 22.5 by 3.5 in.
Weight[a]	
CMG with Electronic Assembly	420 lb
Rotor Alone	150 lb
CMG Inverter Assembly	50 lb

a. Does not include modifications for orbital replacement.

it in the rate mode, and is contained in a rectangular enclosure 30.28 by 21.74 by 14.60 cm. The RGP weighs 5.58 kg. The RGP is attached to the spacecraft by four gusseted mounting feet. External connectors, a lapsed-time indicator, and the vent port are located on one end of the enclosure. The external finish is a flat black, high emissivity paint.

The RGP electrical components include: power supply, heater control, 4800 Hz generator, three-phase inverter, ac amplifier, demodulator, and torque driver.

3.3.4.2.1.2 Operation. A block diagram of the RGP is shown in Figure 16. The commands and input voltages of the RGP are shown in Table 3. The commands and output voltages are shown in Table 4.

The RGP has two modes of operation. The fine mode gives a scale factor of 450 V/deg/s for inputs up to ±0.1 deg/s, while the coarse mode gives a scale factor of 45 V/deg/s for inputs up to ±1.0 deg/s. The mode of operation is selected by gain control commands applied by the flight computer to the ac amplifier (Fig. 16). This command then determines the mode by selecting appropriate gains for the ac amplifier, demodulator, and torquer driver. Except for different gains, both modes operate in the same manner. All RGP's are commanded to change modes simultaneously.

The three-phase, 400 Hz excitation to the gyro spin motor is provided by the three-phase inverter assembly. The output of the inverter is phase-locked to the output of the 4800 Hz generator assembly and was provided with an inhibit capability. The outputs of the inverter provide phases A, B, and C in the proper amplitude and phase separation to drive the gyro spin motor.

The 4800 Hz generator assembly provides a regulated, stable, low-distortion excitation to the gyro pickoff primary. In addition, a reference output to the demodulator, phase shifted to compensate for the gyro shift, is provided. There is also a 4800 Hz square wave clock and a regulated +5 Vdc output which are used by the inverter assembly.

In addition, a signal conditioning circuit produces an output proportional to the temperature of the gyro. A temperature sensor in the gyro provides the input signal to the telemetry signal conditioner. The output ranges between 0 and 5 V and represents a temperature range from 62.2 to 73.3°C.

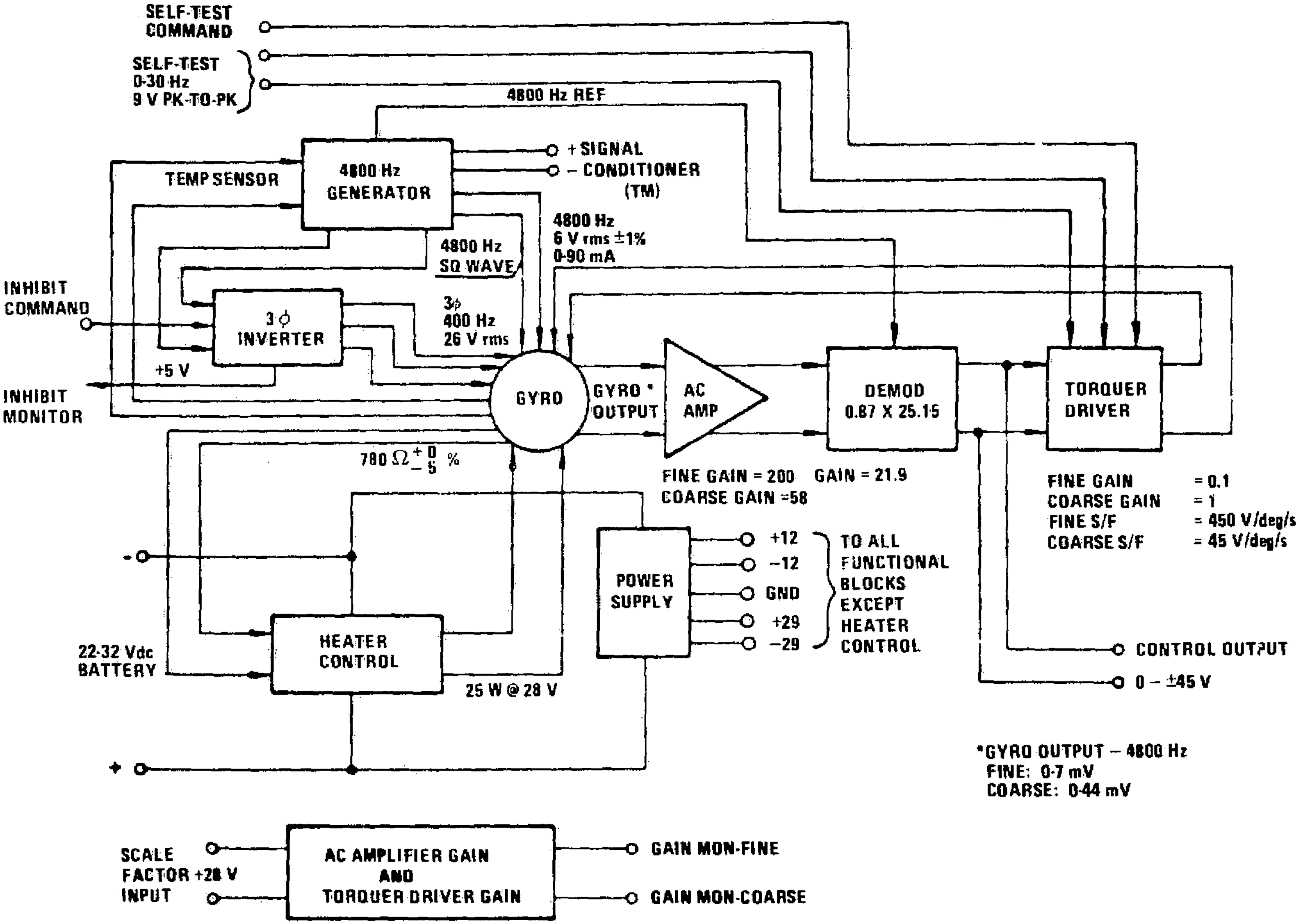


Figure 16. RGP functional block diagram.

TABLE 3. RATE GYRO INPUT

Inputs	Function
22 to 32 Vdc	Battery Voltage to Package
28 +4, -8 Vdc Gain Control No. 1 (Fine)	Selects Fine Mode of Operation
28 +4, -8 Vdc Gain Control No. 2 (Coarse)	Selects Coarse Mode of Operation
28 +4, -8 Vdc Self-Test Command	Energizes Relay that Applies Test Signal Source to Input of Torquer Driver
28 +4, -8 Vdc Wheel Inhibit Command	De-Energizes Wheel Excitation
Torque Test Input	Applies Test Voltage to Torquer Driver

TABLE 4. RATE GYRO OUTPUT

Outputs	Function
0 to ±45 Vdc Control Output	Output Proportional to Rate Input of Gyro and Used to Correct Attitude of ATM
Gain Monitor (Fine)	Discrete Signal Indicating RGP is in Fine Mode
Gain Monitor (Coarse)	Discrete Signal Indicating RGP is in Coarse Mode
Signal Conditioner	0 to 5 Vdc Indicating Temperature of Gyro
Wheel Inhibit Monitor	28 Vdc Indicating Wheel Inhibited

The secondary of the gyro pickoff produces an output that is proportional to the rate input to the gyro. This signal is applied to the ac amplifier where it is amplified by one of two precisely controlled gains. The gain used is selected by an external command signal and is a function of the operating mode selected.

The output of the ac amplifier is applied to the demodulator, which converts ac output to a proportional dc voltage. The polarity of the dc voltage indicates the phase of the input signal relative to the reference signal from the pickoff driver.

The output of the demodulator provides an error signal to the vehicle control system, the telemetry, and the torquer driver which provides the gyro torquer feedback current. The torquer driver, like the ac amplifier, has two gain levels so that operation in two different modes is possible. The gain is switched by an external command signal. Auxiliary inputs to the torquer driver permit preflight checkout as well as system bandpass measurements during manufacturing test.

A heater control circuit is provided to maintain the temperature of the gyro at $67.8 \pm 0.5°C$.

The power supply receives a bus voltage and produces +12, -12, and -29 Vdc for the RGP.

3.3.4.2.2 <u>Modifications to Existing Hardware</u>. Basically the RGP's can be used in their present configuration. However, if the engineering tests show the need for design changes, there are several modifications that will make the units easier to use. These are:

a) If the dual range outputs are not required, one of them can be made inoperable.

b) The output voltage is now ±45 V. If it is shown that new electronics are required, it is proposed that the output be modified to interface directly with the data handling system.

c) Mechanical and electrical modifications as defined by the dodecahedron or skewed mounting configuration.

d) Make provision for on-orbit changeout by a suited crewman on EVA.

3.3.4.2.3 <u>On-Orbit Replacement.</u> The RGP's will be replaced as units since no maintenance can be performed on these items while on-orbit. The RGP will incorporate manually operated fasteners and electrical connectors plus provisions for handling the units. Alignment of the units requires precise placement; therefore, the fasteners and mounting block will have provisions plus self-alignment.

3.3.4.2.4 <u>Test and Verification.</u>

a) A contract (NAS8-32837) was let to Kearfott for review and assessment of data on existing gyros and future data on existing RGP's.

b) A data system was procured to make a complete test station for the gyros and RGP's. The station has been completely checked and is operational.

c) Gyro motor data were taken at MSFC on the four gyro sensors in-house. The motor data were taken on gyro serial numbers (SN's) 006, 023, 104, and 121. The data were obtained by following a mutually agreeable test plan established by MSFC and Kearfott. It has been provided to Kearfott for their review and assessment. The data looked reasonably good with no obvious problems that would prevent use of the gyros on the 25 kW Power Module.

d) Four RGP's have been received from control storage (SN's 107, 109, 110, and 114). These RGP's will be run to determine the validity of performance in an engineering test. The remaining six RGP's will be investigated also.

e) It is recommended that one of the four sensors be disassembled at Kearfott for a thorough investigation. This is to include spin bearing lubricant, floatation fluid, O-ring seals, encapsulation materials, and all details that contribute to a reliable gyro assembly.

3.3.4.3 <u>Solar Array Articulation and Control System.</u> The solar array control system (Fig. 17) shall be capable of fully automatically positioning and tracking the Sun vector normal to the solar array panel upon initiation command. The system shall also have a higher rate acquisition mode. Each solar array wing shall have an independent actuation system. The control system shall have a continuous rate loop closed through a tachometer around the drive electronics and actuator. The output position loop, closed through the Sun sensor, shall be a tri-stable network configured such as not to require power within an acceptable

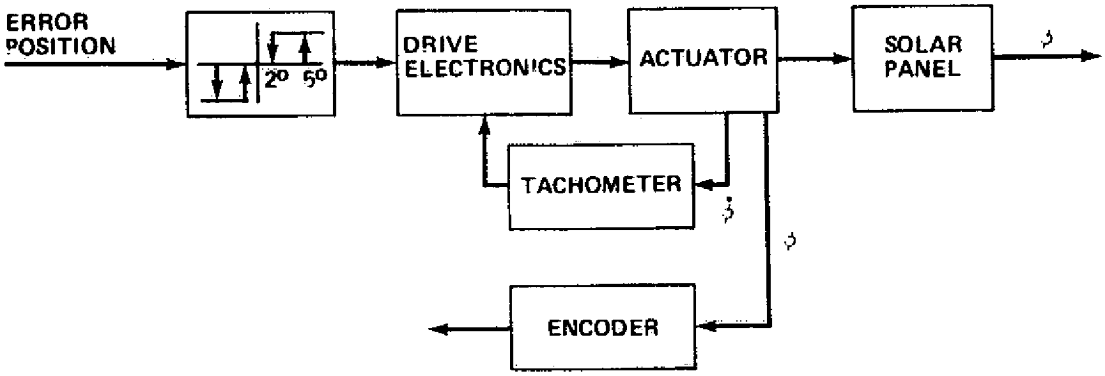


Figure 17. Solar array articulation control system.

error angle. The actuator shall consist of a redundant direct drive brush dc torque motor [2], a tachometer [2], and a double-enveloping worm gear set. The torque motor, tachometer, and drive redundancy electronics shall be active-standby. An encoder shall be integrated into the output shaft position to measure the relative position of the solar array with respect to the vehicle position. The gear train, helix angle, etc., shall be sized to prevent back-drive from normal external torques in the down mode; that is, when the control system has pointed the solar array within the acceptable error angle. The basic control requirements are:

ϕ — gimbal angle, ±180 deg

$\dot{\phi}_c$ — normal control rate, ±0.080 deg/s

$\dot{\Phi}_{max.}$ — maximum slew rate, ±0.30 deg/s

T_w — wire torques, 180 deg = ±6.0 ft-lb

T_A — actuator stall torque, ±15.0 ft-lb

$\xi\phi_{on}$ — error angle required to activate system, ±5.0 deg

$\xi\phi_{off}$ — error angle required to deactivate system, ±2.0 deg

W_a — actuator weight, ≤ 50 lb.

3.3.4.4 Acquisition Sun Sensor. The ATM acquisition was selected as the basic sensor for pointing the solar array. It is used in conjunction with the solar array articulation control and position encoder to control the direction of the solar with respect to the Sun. One is located on each wing.

3.3.4.4.1 Hardware Description. The acquisition Sun sensor consists of two major subassemblies: the optical assembly (OA) and the electronics assembly. The optical assembly uses a family of five photovoltaic cells arranged in a geometric configuration to generate voltage analogs of the Sun's angular position about two orthogonal axes, linear to ±5 deg, and a discrete signal any time the Sun is within the field-of-view of the error detectors (9 ± 1 deg circular). The EA uses conventional electronics to shape and scale the signals from the OA and then transmits the angular error signal and the Sun presence signal to the flight digital computers for subsequent processing and use in the attitude and pointing control. The OA optical axis will be aligned perpendicular to the Y-axis of the vehicle and will measure angular errors about the Y- and Z-axes.

The OA contains two pairs of fine pointing detectors (one pair per axis) and a target detector. Each pair of detectors is connected so that the output currents are differenced and functioned as an energy balance null sensor. When the sensor is pointed directly at the Sun, both photocells receive the same amount of energy and the electrical output is at a null. When the sensor is not pointed directly at the Sun, the optics (lenses and baffles) produce a difference in the solar energy reaching the two cells in the pair. The electrical output from the cells is proportional to the angle between the optical axis of the sensor and the Sun line. The target cell drone level detecting electronics that triggers a discrete when the energy enters a 9 ± 1 deg circular field-of-view is equivalent to one-half solar constant.

The subsystem is completely redundant. There are two independent systems operating at all times (Fig. 17).

3.3.4.4.2 On-Orbit Maintenance. Since no maintenance can be performed on the Sun sensor units while in orbit, they will be replaced as units only. The Sun sensor mountings will incorporate manually operated fasteners and electrical connectors plus provisions for handling the units in EVA modes. The sensor units require precise orientation with respect to the solar array; therefore, provisions will be made for self-alignment.

3.3.4.5 Signal Conditioning Interface Unit. Several of the instruments selected from ATM surplus for use in the attitude control system are not directly compatible with the Remote Interface Units (RIU) of the C&DH system. The CMG's, RGP's, and Sun sensors are typical of this incompatibility. Either the instruments have to be redesigned to provide the required output characteristics, or an interface unit will need to be provided to do this. In the ACS case, as well as for other systems, the SCIU is considered to incorporate the necessary characteristics to be compatible with the RIU's.

3.3.4.5.1 Hardware Description. The SCIU will condition command and data signals between the CMG's, RGP's, and Sun sensors and the flight data system as follows:

Data (Scale, Buffer, and/or Isolate)

- Analog (A) and discrete (D)
  - Sun sensors — 5 A, 2 D
  - Rate gyros — 18 A, 12 D
  - CMG's — 40 A, 36 D
- Analog-to-digital (A/D) conversion — 12-bit serial word (12 data plus sign)
  - Rate gyro — 3 [samples per second (SPS)]
  - CMG — 12 (1 SPS)

Commands

- Discrete — maintain or pulse stretch
  - Sun sensors — 4
  - Rate gyros — 27
  - CMG's — 36
- Serial digital
  - Digital-to-analog (D/A) conversion 12 bit + sign to 0 — ±5 Vdc
  - CMG torquers — 6

The SCIU will provide the capability for increasing the data rate of a selected measurement or a group or measurements for troubleshooting and redundancy management from ground control. A block diagram of the SCIU is shown in Figure 18.

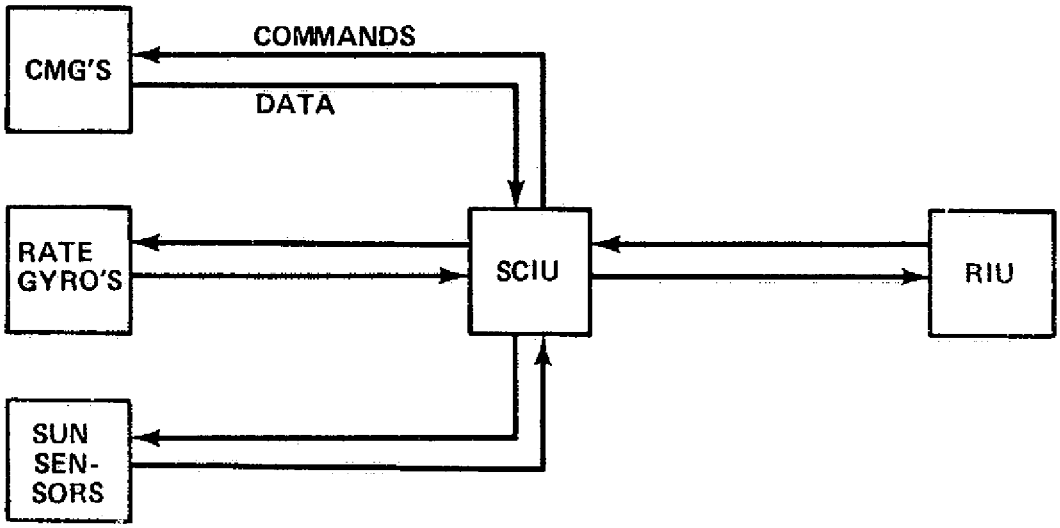


Figure 18. SCIU block diagram — CMG's command data.

3.3.4.5.2 Modifications to Existing Hardware. The SCIU is new equipment.

3.3.4.5.3 On-Orbit Replacement. The SCIU will be designed for replacement as a unit on-orbit by a suited crewman using only hand tools. The SCIU will incorporate manually operated fasteners and electrical connectors, and handles for manipulation by a crewman on EVA. No particular alignment requirement is involved; however, the thermal interface must be maintained.

3.3.5 On-Orbit Maintenance. On-orbit maintenance and replacement, to the depth considered for this baseline system, are covered in subsections 3.3.4.1.3, 3.3.4.2.3, 3.3.4.4.2, and 3.3.4.5.3.

3.4 Thermal Control System

3.4.1 Introduction. The thermal control system (TCS) proposed for the new baseline system is designed to maintain the 25 kW Power Module thermal requirements of approximately 9 kW and also to provide limited heat rejection capabilities for users of approximately 12 kW. Basic PM equipment heat rejection requirements and temperature limits are estimated in Table 5. As shown, the total PM power loss is 9.0 kW at maximum PM output and 1.7 kW at minimum output. The proposed PM TCS must be sized, therefore, to account for this parasitic heat load and to offer some additional heat rejection capability for PM users. The proposed new baseline TCS concept is shown in Figure 19. Note that this configuration is a departure from the phase A TCS concept given in Figure 20. The current baseline, changing to the dual fluid loops, was adopted because trade studies showed the phase A approach to be marginal in providing heat rejection for users. In addition, the current TCS concept offers system redundancy; a failure in one loop will not completely disable the vehicle, thereby possibly allowing mission continuation. Also, in the PM orbital storage mode or on low heat load missions, the TCS may require operation of only one loop, thereby extending TCS life. Adoption of the current baseline concept has also increased system reliability because the required pump operating speed has been reduced (2500 lb/hr as opposed to 3000 lb/hr).

3.4.2 Radiators/Interface Descriptions. The radiator configuration in the September 1977 baseline TCS is given in Figure 21. After studying several alternatives to the 1977 baseline radiator configuration, the four Orbiter double-surfaced radiators (Fig. 22) were selected for the new baseline. Adoption of the new radiator baseline was necessary not because of any thermal requirement, but to facilitate radiator packaging in the launch configuration. Among current TCS equipment shown in Figure 19, in addition to the Orbiter radiators, is a ground support heat exchanger incorporated to provide cooling during prelaunch. This is Orbiter part No. SV755511 and has a rated capacity of 0 to 111 640 Btu/hr with a temperature range of 32 to 120°F. The interface heat exchanger is the same as the interloop exchanger on Spacelab and has a capacity of 8 kW with a wide temperature and flow range. All PM components are mounted on Spacelab pallet type cold plates. The freon pumps are identical to Orbiter pumps, each having a flow capability of 2500 to 3000 lb_m/hr and a 20 000 hr lifetime. A detailed PM TCS equipment list is given in Table 6.

The September 1978 baseline TCS heat rejection capability varies from 14 to 21 kW depending on vehicle orientation. These values were calculated assuming a radiator freon inlet temperature of 98°F and an outlet temperature of 40°F as shown in Figure 19. These system capabilities are applicable when the PM is in a docked configuration with the Orbiter. Values for heat rejection increase from 10 to 12 percent in a free-flyer mode. Because the parasitic PM

TABLE 5. POWER MODULE THERMAL CONTROL SYSTEM REQUIREMENTS — DECEMBER 1978

	Docked		Free-Flyer	
	At Minimum PM Power	At Maximum PM Power	At Minimum PM Power	At Maximum PM Power
Heat Rejection				
Power Module Losses	0.75	7.1	0.75	7.1
Equipment	0.95	1.9	0.95	1.9
	1.70 kW_t	9.0 kW_t	1.70 kW_t	9.0 kW_t
Temperature Limits	°C		°F	
Communication	-10 to 50		14 to 122	
Data Handling	-18 to 60		-3 to 140	
Attitude Control	-10 to 55		14 to 131	
Electric Power	-18 to 55		-3 to 131	
Batteries	5 to 20		41 to 68	

Note: For higher reliability and long life, all avionic equipment should operate in the lower range level.

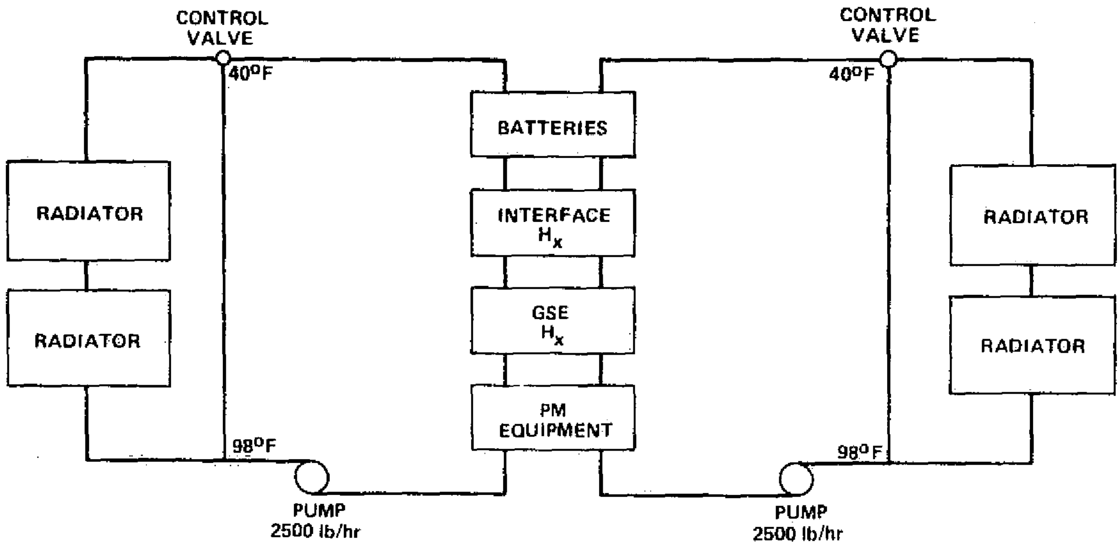


Figure 19. Power Module baseline thermal control system — December 1978.

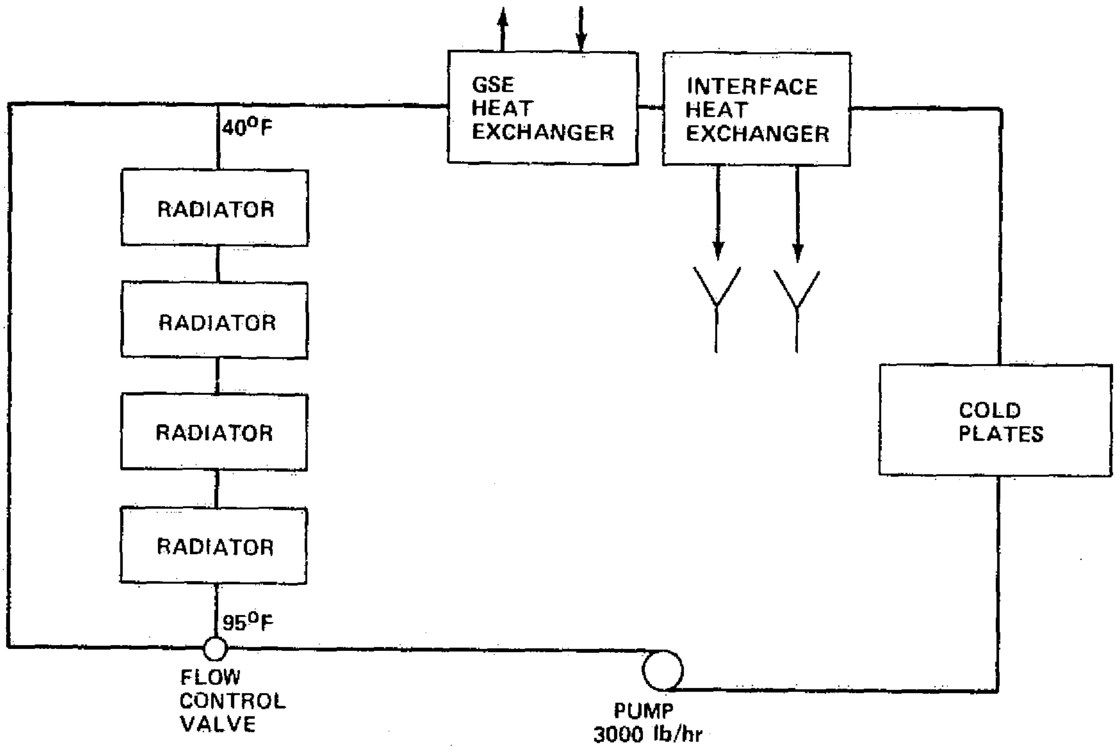


Figure 20. Phase A thermal control system configuration — September 1977.

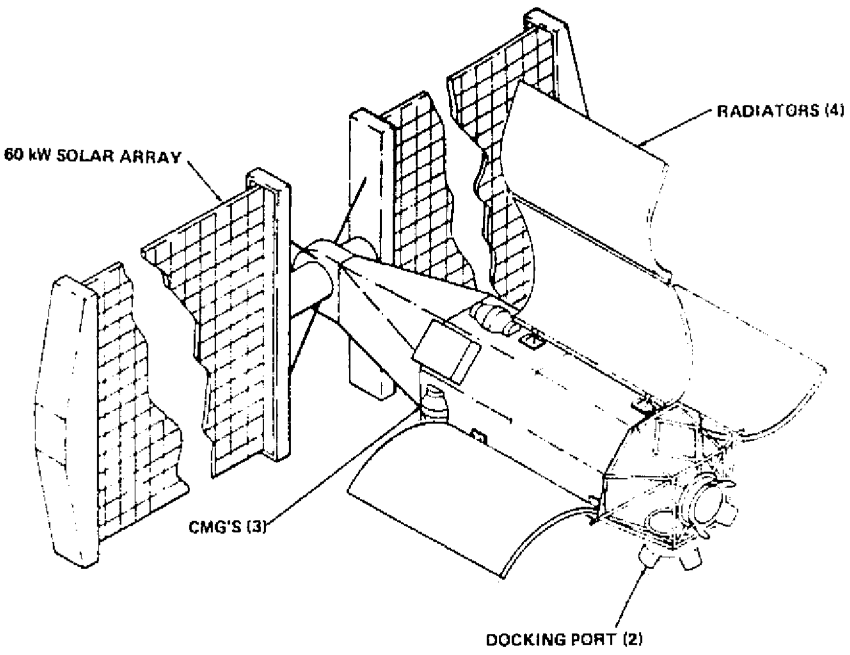


Figure 21. Phase A radiator configuration — September 1977.

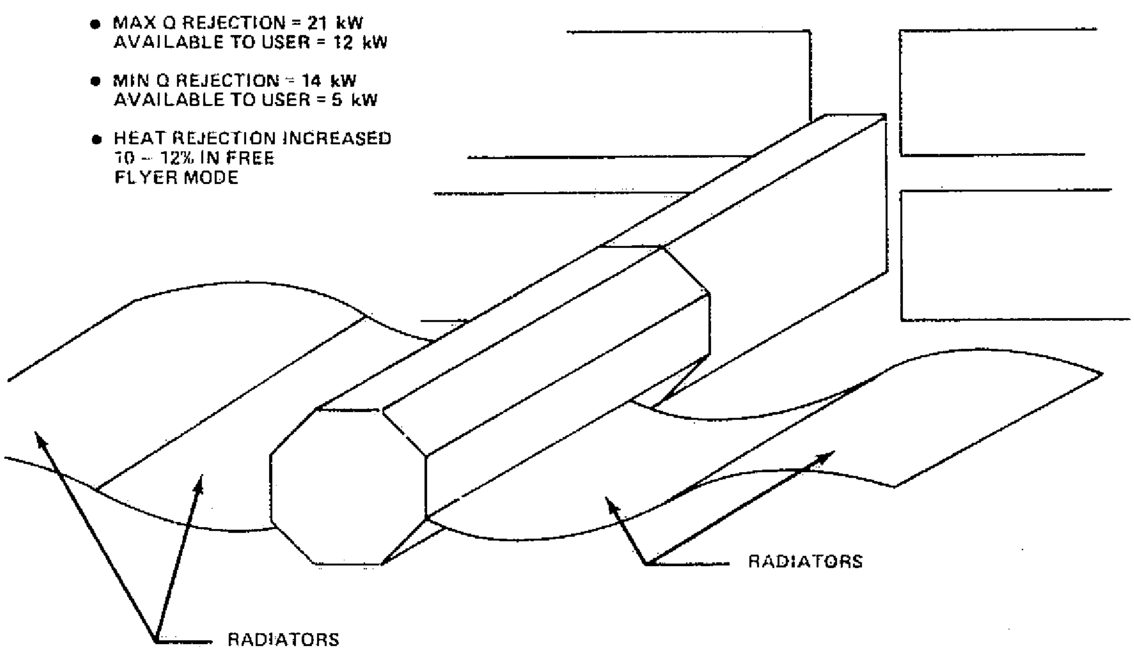


Figure 22. Power Module baseline radiator configuration (deployed) — December 1978.

TABLE 6. POWER MODULE THERMAL CONTROL SYSTEM EQUIPMENT LIST — DECEMBER 1978

No.	Hardware	Program	Baseline Weight (lb)
64	Cold Plates	Spacelab	520
4	Radiator Panels	Shuttle	747
2	Pump and Accumulator Assemblies	Shuttle	124
2	Pressure Regulators	Shuttle	20
2	Flow Control Assemblies	Shuttle	33
1	Interface Heat Exchanger	Shuttle	10
1	GSE Heat Exchanger	Shuttle	14
	Liner, Fasteners, etc.	Typical	43.5
	Insulation	Typical	100
TBD	Heaters	Typical	TBD
Total			1611.5

heat load varies from 1.7 to 9.0 kW and because user power and heat dissipation requirements will vary significantly, the heat rejection capability available for PM users will be highly mission dependent, but will never be less than 5 kW while docked with the Orbiter (worst case) nor greater than 22 kW while in a free-flying mode (best case).

3.4.3 PM User Fluid Interface. The PM user fluid interface has not been determined. Two apparently feasible concepts were identified, one using heat exchangers and the other using valves. Either option may be considered as the baseline. The two concepts are shown in Figure 23. Heat exchangers offer the advantage of having no direct coupling to the PM fluid loop. Valves might be a more efficient system, but their use would increase the possibility of leaks and

TWO OPTIONS AVAILABLE TO AUGMENT USER HEAT REJECTION

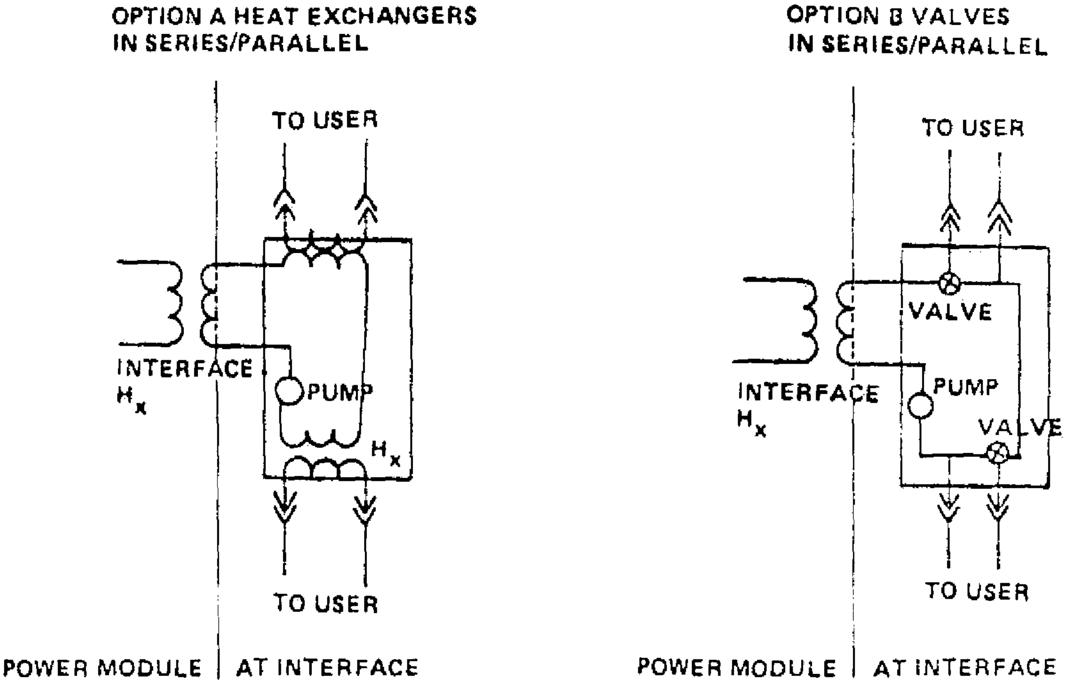


Figure 23. Power Module/user fluid interface options — December 1978.

would cause a very complicated fluid interface when more than one user is docked. Since this interface is crucial to TCS performance, phase B design efforts should result in an efficient method of integrating user and PM heat rejection requirements.

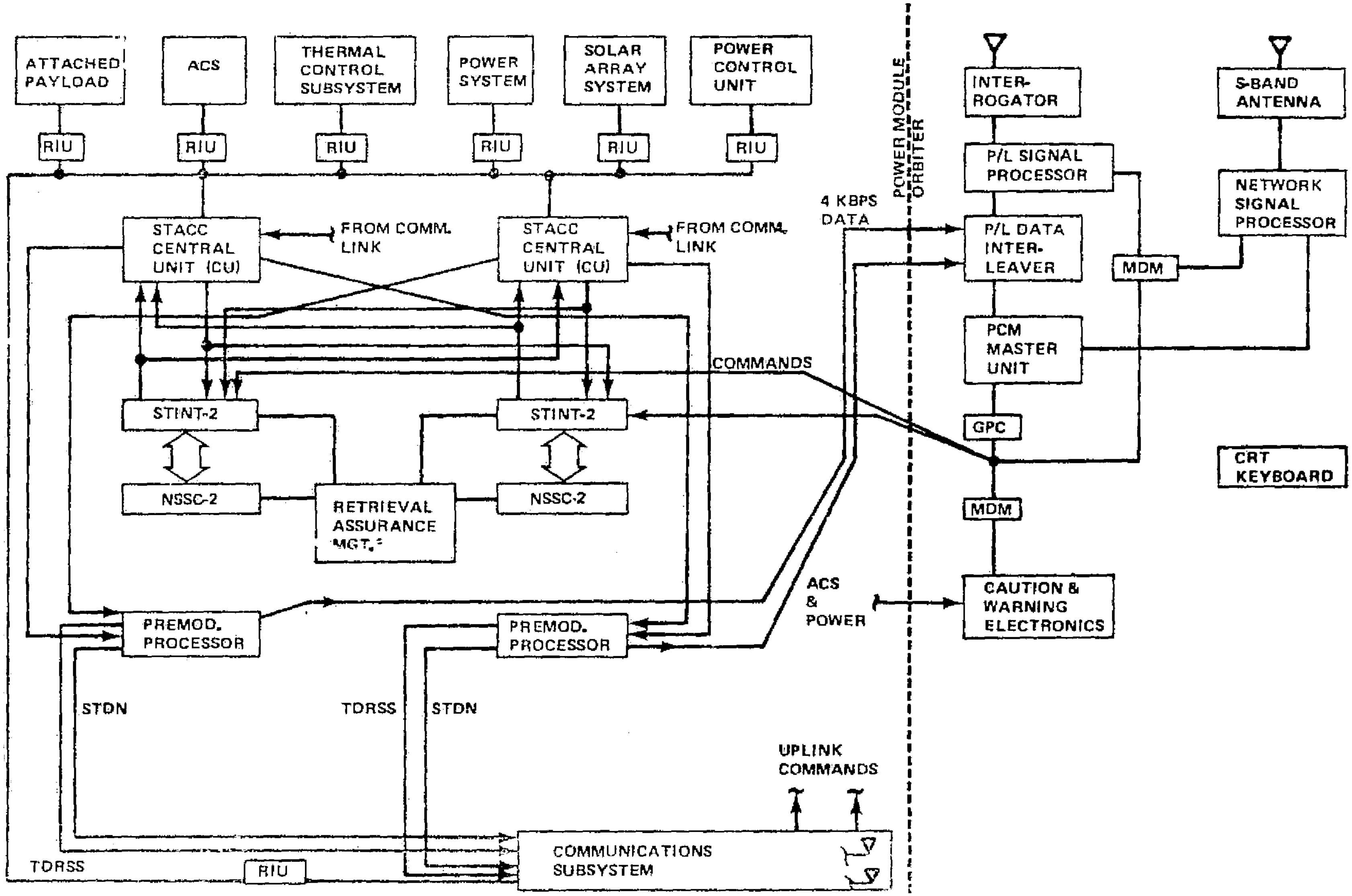
3.4.4 <u>On-Orbit Maintenance.</u> On-orbit replacement of TCS components is a highly desirable feature. Since most components will be modular Shuttle and Spacelab hardware, on-orbit replacement via EVA should be feasible. Replacement of radiator panels may be difficult but should also be possible. This capability should be incorporated into the design of critical fluid interfaces through the use of quick-disconnects such as those being developed by MSFC.

3.5 Communication and Data Handling System

3.5.1 Introduction. MSFC investigations, since the September 1977 baseline C&DH system was formulated, have revealed some problems with the on-board system depicted earlier. For example, use of the ATM digital computer and the ATM interface unit was determined to be an unsuitable choice primarily because of the difficulty of obtaining replacement electronics. Also, the September 1977 on-board C&DH system, which utilizes the NASA MMS C&DH equipment, will require modification to be compatible with the Orbiter; for example, the central unit accepts non-return to zero-level (NRZ-L) and non-return to zero-mark (NRZ-M), while the commands from the Orbiter are bi-phase-level (BI-ϕ-L).

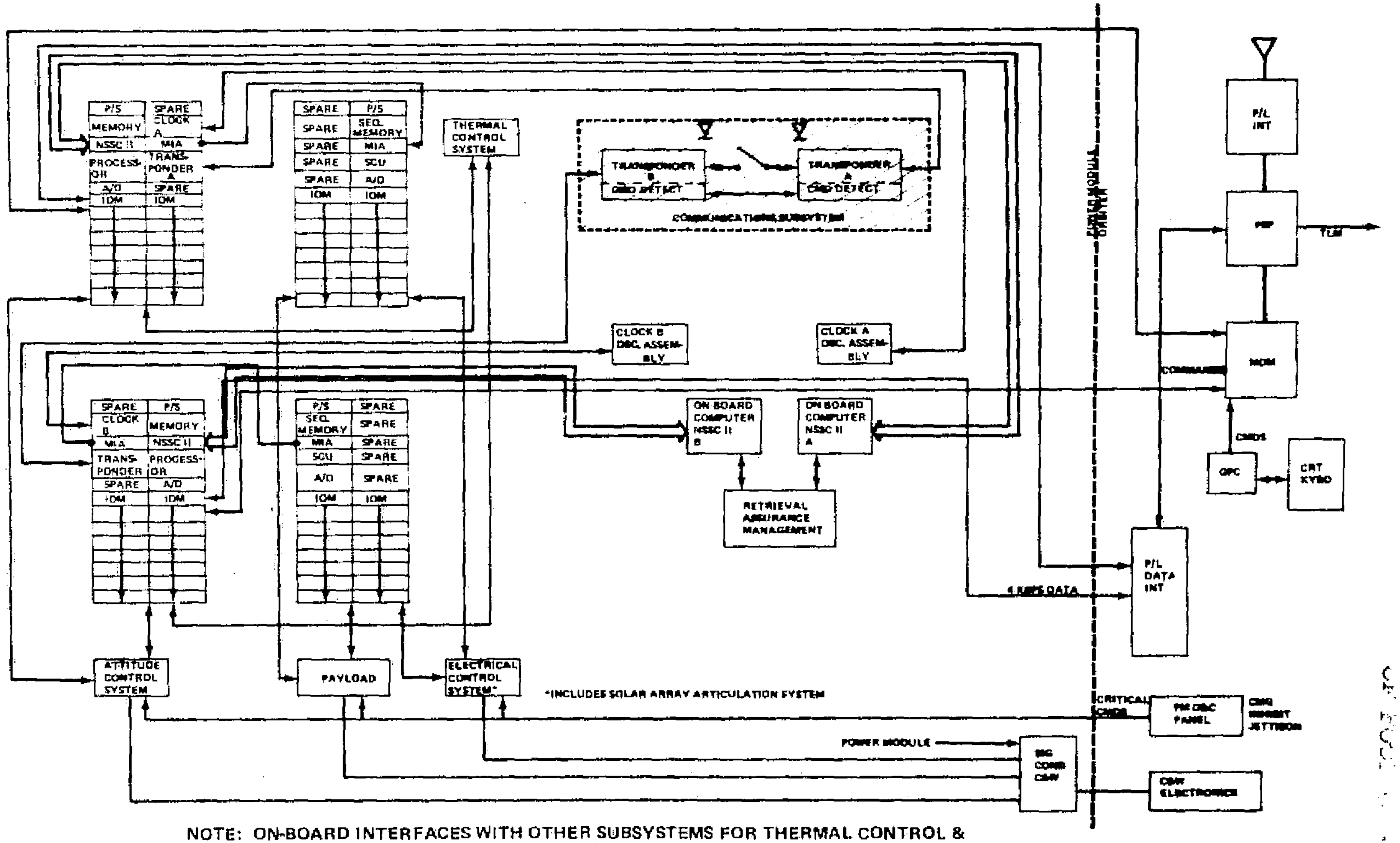
The data system activities have concentrated first on developing and refining the various flight and ground system data requirements, and then on defining a new baseline on-board system. The intent was to more fully understand the requirements and to develop two systems that seem to be reasonable (not necessarily optimum) to implement and operate. The reason that two C&DH systems were conceptually designed was to emphasize that no one system should be considered as superior in the early stages of this program development. In fact, the selection of these two is not meant to infer that other data systems could not work as well, or better. Factors considered in selecting these two systems include cost, probable availability in 2 years, box-level qualification status, development risk, growth potential, and reliability (although admittedly a rigorous quantification of all of these factors was not attempted in the recent activities). Two options that seem to be workable are depicted in Figure 24 (the modified MMS C&DH approach) and Figure 25 (an Orbiter compatible FMDM approach). Figure 26 is a functional block diagram of the communications, or RF portion, of the on-board data system and may be used with either option. Tables 7 and 8 contain "equipment list" type information on options 1 and 2, respectively.

In developing the two options for the new C&DH baseline system, the decision was made that the other flight systems, and not the C&DH system, are responsible for implementing the signal conditioning necessary to enable their respective system to interface with the standard RIU, or with a similar unit, such as the Spacelab remote acquisition unit. All signals, whether they are for telemetry, control, command, or on-board checking, etc., were assumed to flow through such a unit. Signals routed to or from systems other than over a data bus, and through an RIU, would be considered the exception rather than the



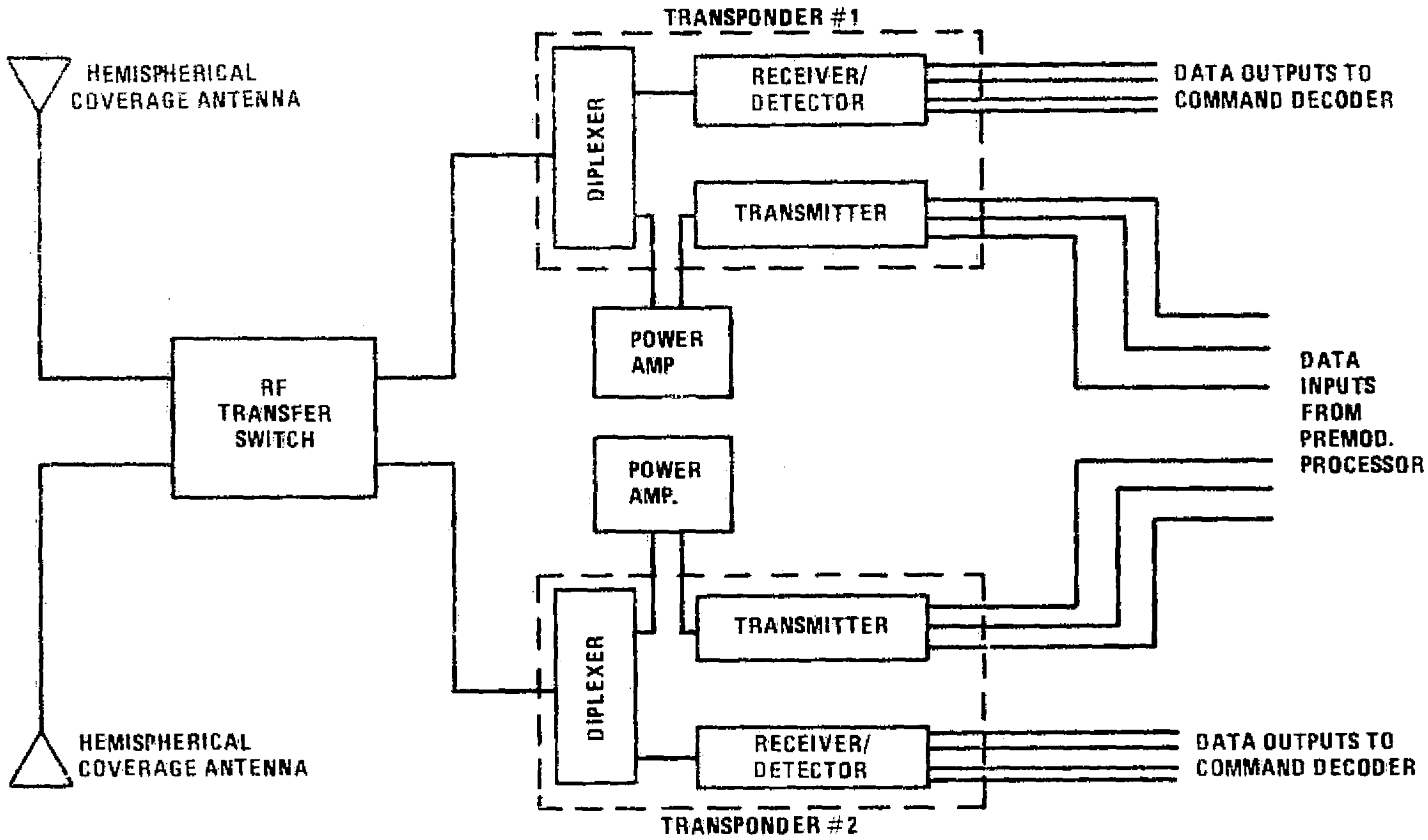
NOTE: ON-BOARD INTERFACES WITH OTHER SUBSYSTEMS FOR THERMAL CONTROL & ELECTRICAL CONTROL ARE NOT DEPICTED HERE. *INCLUDES AUX. MEMORY

Figure 24. On-board communications and data handling system for 25 kW Power Module — Option 1 (December 1978).



NOTE: ON-BOARD INTERFACES WITH OTHER SUBSYSTEMS FOR THERMAL CONTROL & ELECTRICAL (POWER & NETWORKING) CONTROL EXIST BUT ARE NOT DEPICTED HERE.

Figure 25. On-board communications and data handling system for 25 kW Power Module — Option 2 (December 1978).



2 EA.	NASA STANDARD TRANSPONDERS, DUAL MODE TDRSS/STDN (TRANSPONDER INCLUDES DIPLEXER AND COMMAND DETECTOR)
2 EA.	POWER AMPLIFIERS, 14 WATT OUTPUT
1 EA.	RF TRANSFER SWITCH
2 EA.	ANTENNAS, HEMISPHERICAL PATTERNS
1 SET	COAXIAL CABLES

Figure 26. Communications functional block diagram for 25 kW Power Module (December 1978).

TABLE 7. 25 kW POWER MODULE COMMUNICATIONS AND DATA HANDLING EQUIPMENT LIST — OPTION 1 (DECEMBER 1978)

Item No.	Item Name Civil Service Supplier and Responsible Engineer	One Manufacturer	No. Required	Unit Weight (lb) Size (in.) Volume (in.^{3})	Storage Range (°C) Min. Turn On (°C) Operating Range (°C)	Unit Power Standby (W) Unit Power Operation Unit Power Dissipation	Notes/Comments
1	Premodulation Proc. (PMP) EF Parker, G.	Fairchild	1	19 9 × 8 × 6.5 468	S-700-80 S-700-80 0 to 40	N/A 6 at 28 V 6	MMS design. Two functional units come in the one housing. S-700-80 is a GSFC document covering temperature ranges.
2	STACC Central Unit (CU) EF Parker, G.	SPACETAC	2	5.3 4.5 × 7.0 × 6.8 215	S-700-80 S-700-80 -10 to 50	0.9 10.9 10.9	MMS design. S-700-80 is a GSFC document. Slight unit modification may be necessary.
3	STINT-2 EF Parker, G.	TBD	2	2.4 1.6 × 9.1 × 7.1 102	S-700-80 S-700-80 -10 to 50	N/A 2.5 at 5 V 2.5	Similar to STINT-1 of MMS but is to interface with NSSC-2. STINT-2 will contain MIA and will be a new design.
4	NSSC-2 with 32K Memory EF Panciera, R.	IBM	2	26.6 9.9 × 11.8 × 5.5 640	-30 to 75 -30 -30 to 55	N/A 171 171	MMS design. Has fault tolerant memory (not yet designed and built).
5	Bus Coupler EF Parker, G.	SPACETAC	8	5.5 oz 2.5 × 2.3 × 0.6 4.0	S-700-80 S-700-80 -10 to 50	0.1 0.5 0.5	MMS design. S-700-80 is a GSFC document.
6	Remote Interface Unit (RIU) EF Parker, G.	SPACETAC	8	[illegible].7 2.6 × 8.1 × 7.1 598	S-700-80 S-700-80 -10 to 50	0.55 3.0 3.0	MMS design. S-700-80 is a GSFC document.
7	RIU EXPANDER UNIT EF Parker, G.	SPACETAC	6	2.5 2.3 × 8.1 × 7.1 151.1	S-700-80 S-700-80 -10 to 50	0.075 0.175 0.175	MMS design. Each housing contains two functional units.
8	Power Control Unit (PCU) EF Parker, G.	SPACETAC	1	8.7 8 × 8 × 7.1 454.4	S-700-80 S-700-80 -10 to 50	N/A 16.6 at 28 V 16.6	MMS design. Each housing contains two functional units. S-700-80 is a GSFC document.
9	Retrieval Assurance Mgt. Unit EF Panciera, R.	TBD	1	6.0 3.5 × 8.1 × 7.1 200	S-700-80 S-700-80 -10 to 50	N/A 15 15	New design. S-700-80 is a GSFC document.

Note: RF or communications equipment not included in this table.

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TABLE 8. 25 kW POWER MODULE COMMUNICATIONS AND DATA HANDLING EQUIPMENT LIST – OPTION 2 (DECEMBER 1978)

Item No.	Item Name	Civil Service Supplier and Responsible Engineer	One Manufacturer	No. Required	Unit Weight (lb) / Size (in.) / Volume (in.³)	Storage Range (°C) / Min. Turn On (°C) / Operating Range (°C)	Unit Power Standby (W) / Unit Power Operation / Unit Power Dissipation	Notes/Comments
1	MIU Baseline Unit	EF Parker, G.	Sperry Rand	2				FMDM being used for OFT pallet.
2	MCU Baseline Unit	EF Parker, G.	Sperry Rand	2				FMDM being used for OFT pallet.
3	IOM – Discrete In Card	EF Parker, G.	Sperry Rand	7				FMDM being used for OFT pallet.
4	IOM – Analog In Card	EF Parker, G.	Sperry Rand	15				FMDM being used for OFT pallet.
5	IOM – Serial I/O Card	EF Parker, G.	Sperry Rand	3				FMDM being used for OFT pallet.
6	IOM – Discrete Out Card	EF Parker, G.	Sperry Rand	7				FMDM being used for OFT pallet.
7	MCU PROM Memory Card	EF Parker, G.	Sperry Rand	2				New development required. 48K, 16-bit words.
8	Clock Interface Card	EF Parker, G.	Sperry Rand	2				New development required.
9	Transponder Interface Card	EF Parker, G.	Sperry Rand	2				New development required.

TABLE 8. (Concluded)

Item No.	Item Name: Civil Service Supplier and Responsible Engineer	Item Name: One Manufacturer	No. Required	Unit Weight (lb) / Size (in.) / Volume (in.^{3})	Storage Range (°C) / Min. Turn On (°C) / Operating Range (°C)	Unit Power Standby (W) / Unit Power Operation / Unit Power Dissipation	Notes/ Comments
10	Spare Card EF Parker, G.	Sperry Rand	12				FMDM being used for OFT pallet.
11	Clock Osc. Assembly Unit EF Parker, G.	TBD	2				New development required.
12	Retrieval Assurance Mgt. Unit EF Panciera, R.	TBD	1				
13	NSSC-2 Interface Card EF Parkor, G.	Sperry Rand					New development not required. Card is expected to be developed on another program.
14	NSSC-2 with 32K Memory EF Panciera, R.	IBM	2				MMS design. Has fault tolerant memory (not yet designed or built).

Notes: RF or communications equipment not included in this table.

There are four different boxes in this configuration. These are (1) NSSC-2, (2) Clock Oscillator Assembly, (3) Retrieval Assurance Management Unit, and (4) Flex-MDM (FMDM).

Item No. 11 (Clock Oscillator Assembly Unit) estimates are 5 × 3 × 2 in. with weight equal to 2 lb and dissipating 2 W.

Items 1 and 2 estimates are 5.5 × 7.25 × 13.5 in. with weight of 20 lb each (there are 4) and dissipating 22 W each (both operating and standby).

rule. The on-board C&DH system provides monitoring of points from all spacecraft systems, provides sample/process/response for the attitude control system and for the solar array articulation part of the electrical system, and provides the routing of commands to all flight systems except structures. Additionally, the on-board C&DH system provides similar services to itself; e.g., it must monitor automatic gain control levels to determine which antenna to accept inputs from, and then initiate commands to perform the appropriate switching functions.

3.5.2 Assumptions/Requirements. In the September 1977 preliminary definition document, some estimates were made on data system requirements. More recently, engineers had a greater opportunity to examine the various flight systems, and thus gain a much better insight into the data requirements. By talking with the other system engineers, C&DH system engineers were able to refine the requirements on the on-board C&DH system to a much higher degree than was available for the September 1977 baseline. However, it soon became apparent that some basic assumptions would have to be made to allow meaningful requirement estimates to be made. The assumptions made are as follows:

a) Only the TDRSS S-band single access (SSA) link will be available. Rationale — Some internal guideline was necessary at this stage of development to allow an initial system design, and the judgment was made that only a single link would be affordable by the project. This assumption should be seriously questioned in future project considerations.

b) The TDRSS SSA link availability will be limited to 15 continuous minutes out of 270. Rationale — A short, continuous period of a scheduled TDRSS SSA link was assumed for two reasons. First, some timeline scenario was necessary to help establish the data system configuration. Secondly, and perhaps more importantly, this assumption provided "constraints" against which all systems, such as the ACS, could rigorously determine if they could function.

c) The Orbiter will be the only communications link used when docked. Rationale — Implied from systems requirements document.

d) Spaceflight tracking and data network (STDN) will be used as a backup mode of communications. Rationale — This "safety" mode of operation, should the TDRSS have significant outages, is automatically achieved since the dual-mode transponder, which is necessary for communications with both the TDRSS and the Orbiter, is compatible with STDN stations because the Orbiter and the STDN formats are the same.

e) The PM communications service to any payload is only for limited telemetry measurements of housekeeping data and for limited commands. Rationale — Telemetry per systems requirements document. The limited command capability was extended to the payloads as a recently recognized requirement necessary for minor, occasional payload operations.

f) One data rate will suffice for both the "docked" and "free-flying" mode of the PM. Rationale — To minimize the need for multiple formats and, thus, to lower costs.

g) No more than three rate gyros will be active simultaneously and require monitoring and/or control. Rationale — Minimum number assumed necessary.

h) No more than three CMG's will be active simultaneously and require monitoring. Rationale — Minimum number assumed necessary.

i) Redundancy management will be performed (on-board, on the ground, or both) at the place or places that will incur the lowest total project cost. The assumption here is that only a minimum will be done on-board to ensure retrievability. Rationale — Systems requirements document.

j) There will be data rate upper limits of 4 kbps return link (PM to ground) and 1 kbps forward link (ground to PM). Rationale — The 4 kbps is from the systems requirements document. However, both the 4 kbps and the 1 kbps were assumed as self-imposed constraints, again to provide the systems engineers with rate figures to "work against."

k) No on-board tape recorder will be used. Rationale — A self-imposed constraint used for the dual purpose of holding down cost and indirectly providing another "communications timeline" type constraint against which the several systems engineers could work to test their concepts.

l) Attitude control and electrical power will condition their outputs to be compatible with standard data interfaces as would be afforded by an RIU. Rationale — This assumption was made to simplify the system and subsystem interfaces; all systems can now work to a reasonably well-defined data interface.

Several iterations of collecting data requirements from the other flight systems and determining the impact on a data bus implementation scheme allowed MSFC to develop a listing of system data requirements that was firmer than the phase A listing reported in the September 1977 preliminary definition document. These "refined" system requirements entail (for telemetry) over 400 measurements for the electrical system, over 60 for the ACS, over 110 for the C&DH system (some of these are requested by the ACS), over 150 allocated for payloads, and over 80 allocated as spares. Control requirements (necessitating data flow to and from the central computer, all of which would not necessarily be transmitted as telemetry) are confined to the ACS and the solar array articulation portion of the electrical system, while command requirements have been identified from all flight systems except structures.

In addition to the requirements discussed in the preceding paragraph, several derived requirements pertaining specifically to communications have been used. As may be noted in the project [1] and systems [2] requirements documents, the communications are required to support the free-flying mode of the PM operation (when docked to the Orbiter, all PM communications will be through the Orbiter RF system). In the free-flying/payload support mode, command, telemetry, and ranging links are required. Initial, cursory studies showed that one would approach the return bit rate limit of omni antennas through the TDRSS, using two 14 W power amplifiers at 4 kbps. The forward link bit rate limit, under the same conditions, is 1 kbps. To go to higher bit rates would necessitate changing to significantly higher gain (and higher cost) antenna configurations. Consequently, the 4 kbps/1 kbps rate limits for most of the study were self-imposed (to be used as real constraints unless system or other flight system engineers showed reasons that these were unacceptable) as "derived" requirements. Also a requirement not specifically stated in the referenced [1,2] requirements documents, but that must be met by all projects, is conformance to the international agreements regarding the level of radiated flux density at the surface of the Earth.

3.5.3 Communications Descriptions. Three alternatives for the communication configurations were considered. These were (a) the C&DH module as configured for the MMS (GSFC document S-700-15) with power amplifiers added, (b) same as (a) except for a modified switching configuration to avoid the antenna interference pattern, and (c) same as (b) but with individually mounted components on cold plates with elimination of the C&DH module honeycomb enclosure. As a result of studying various aspects of these three alternatives, configuration (c), as depicted functionally in Figure 26, is recommended. The elimination of the module enclosure will result in savings of both cost and weight over the first two options. In addition, the subsystem components will be more accessible for on-orbit replacement. The recommended alternative also uses the PM thermal control subsystem rather than relying on the thermal characteristics of the enclosure.

Link margin calculations for the free-flying PM utilizing the TDRSS show that this approach is reasonable if one limits the return bit rate to 4 kbps and the forward link to 1 kbps. The PM assumes a time-shared usage of an SSA channel on the TDRSS, and most calculations are based on this. However, to investigate the chances of broadening possible modes of operation with more nearly continuous RF coverage, a short study was conducted on the possible use of the TDRSS S-band multiple access (SMA) link.

The communications design selected for this baseline C&DH system is composed (Fig. 26) of two omni antennas each of which provides near-hemispherical coverage, a latching RF switch that is operated only in case of an antenna or transponder failure, two NASA standard transponders, and two 14 W power amplifiers to provide the required link safety margins. The transponder design allows direct communications with both the TDRSS and the Orbiter, and, as was noted earlier, is STDN compatible. To receive commands, both paths remain active, or on-line, completely through the communications portion of the on-board data system. The transceiver automatic gain control circuits are monitored continuously to select the strongest signal to decode.

During the course of the study activities, TDRSS interference problems were reported to MSFC. PM/Orbiter rates were not expected to be significantly affected. However, interference was expected to affect the direct PM to TDRSS S-band communications by causing an additional 2.0 dB loss and a coverage loss of approximately 10 percent. These losses were considered in the link calculations, and the communications design suggested provides a 3.0 dB safety margin for the return link and 3.9 dB for the forward link.

As noted previously, any new program such as the 25 kW Power Module will be required to meet an Earth surface flux density limit equal to or less than -141 dBW/m^2/4 kHz. Calculations show that the conceptual design recommended has a 3 dB margin of safety against a flux density violation.

3.5.4 Description of the Proposed Baseline On-Board Data System — Option 1. The on-board C&DH system as depicted functionally in Figure 24 is a variation of the MMS C&DH module, which is based on the concept of remote multiplexing of data and remote distribution of commands. All data (telemetry, control, or other) flow on a multiplex data bus,[1] a set of redundant "party lines" — outgoing on a supervisory line and return messages from the various

1. Technically a serial, duplex, digital data bus.

subsystems on a reply line — transformer-coupled at both ends.[2] All functions are performed under the control of the Central Unit (CU) which houses the command decoder, format generator, bus control and interface unit, and the spacecraft clock. While several data rates are available to select from, 4 kbps has been selected for use in this study. The actual command rate received from ground is expected to be far less than the 42 commands per second capability afforded by the bus, and, in this case, will be limited by the 1 kbps forward link rate. C&DH redundancy is provided primarily in the form of two independent, simplex chains from the various flight systems to the PM data output port (RF antenna or Orbiter data system). A second data bus with its own complete set of boxes (CU, RIU's, etc.) parallels the active first data bus. If the first path fails, the second will provide a redundant path for the proper transfer of data. In reality, the C&DH system depicted is much more powerful than one having only two totally independent chains, since several components may be cross-strapped, as indicated in the functional block diagram.

There are some notable differences between the data systems of Figure 24 and the standard MMS C&DH. These are:

a) NSSC-2 is used instead of NSSC-1.

b) STINT-2 is used instead of the STINT depicted in existing reference documents.

c) STACC CU modification is required.

d) A retrieval assurance management unit has been added.

Preliminary investigations were conducted in each of these four areas.

NSSC-2 (a NASA standard computer, as is NSSC-1) was selected for use primarily because of its growth potential in both speed and memory size. The NSSC-1 will not provide the 100 percent growth figures desired.

The STINT-2 will have to be a new design to enable an interface with the NSSC-2. The existing STINT is designed to interface with the 18-bit NSSC-1. However, should the NSSC-1 have been used, the STINT still would have required modification to enable it to interface with the Orbiter data system.

2. Bus coupler units are used with each connection to the data bus, but were left off the functional block diagram to prevent crowding of the drawing.

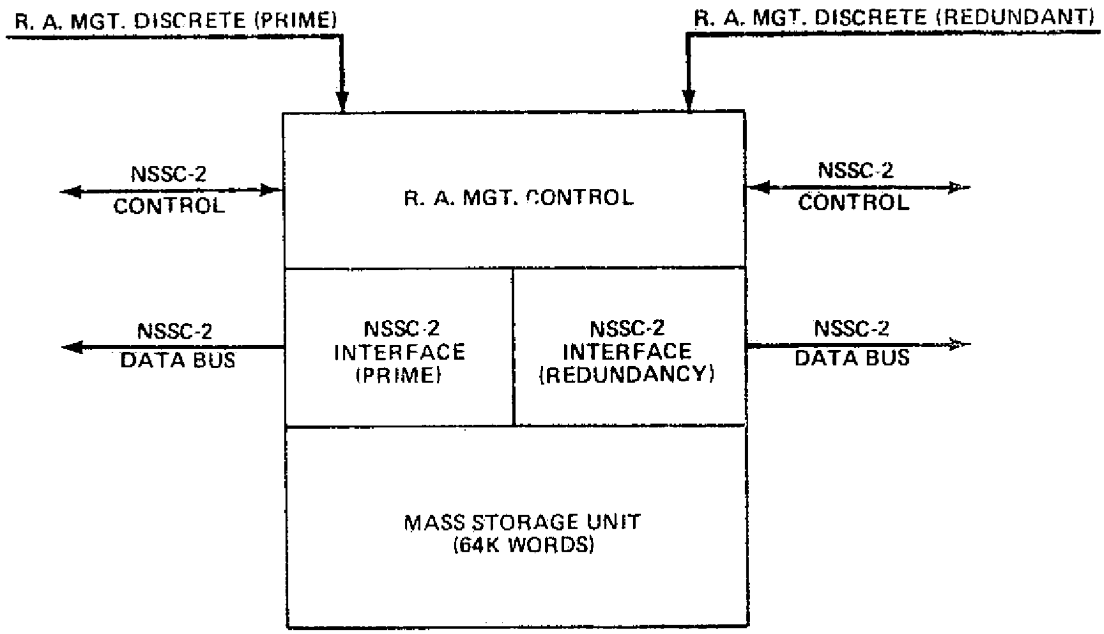
Since the MMS C&DH achieves redundant management essentially through continuous ground communications, a retrieval assurance management unit had to be added because the PM will experience ground contact only a small percentage of the time (assumed 15 minute block out of every 270). The retrieval assurance management unit was added to do what the name implies and is required primarily in case of an airborne computer failure. A functional depiction of the unit is presented in Figure 27. The mass storage unit is envisioned as a dynamic memory that is capable of loading the redundant computer with up-to-date data if the prime computer fails.

3.5.5 Description of the Proposed Baseline On-Board Data System — Option 2. Figure 25 is a functional block diagram of an on-board C&DH system that more closely parallels the Orbiter data system than the MMS C&DH system. Such a system has excellent growth potential and will interface well with the Orbiter. The Orbiter data system is a data bus approach centered around remote devices called MDM's (from multiplexer/demultiplexer). In an evolutionary process to extend the MDM capabilities to meet an even wider variety of needs, an FMDM has been developed for Orbiter flight test use. The PM data system option 2 uses the FMDM scheme. Two basic types of units are used; a module control unit (MCU) FMDM and a modular interface unit (MIU) FMDM. The MCU FMDM contains a microprocessor, program memory, and input/output (I/O) devices, while the MIU FMDM contains a bus adapter, sequence memory and control, and I/O devices. In essence, the MCU is in command or control of itself and one MIU unit. As in option 1, complete functional redundancy, except in the retrieval assurance management unit, is provided by a second MCU FMDM and MIU FMDM.

Some new cards would have to be developed to allow usage of the FMDM. Examples are a programmable read only memory (PROM) card, a transponder interface card (which is primary for command decoding), a NSSC-2 interface card, and a clock interface card. In addition, since no clock assembly is present, one would have to be acquired (and is envisioned as being a separate box).

3.5.6 Growth Potential. The current RF link is capable of the following maximum data rates:

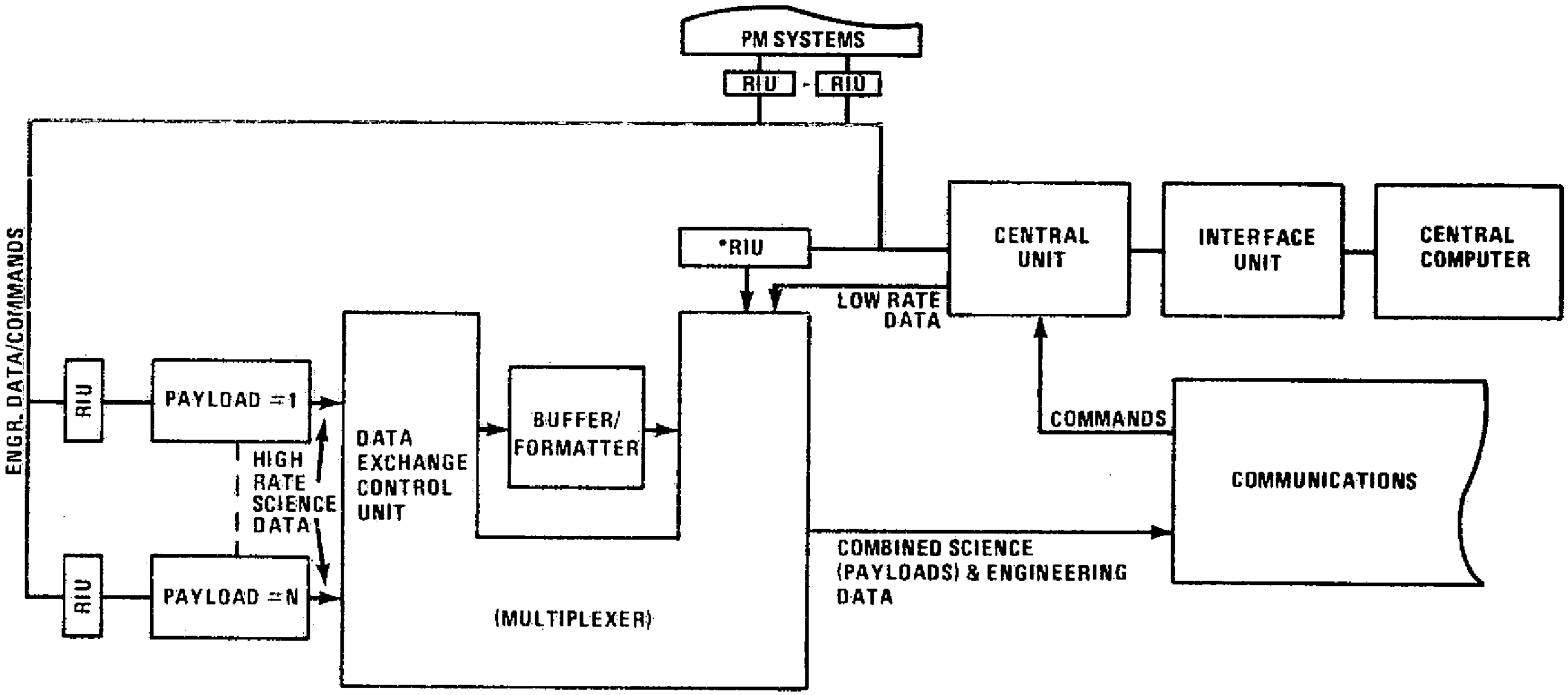
16 kbps to Orbiter from PM (only 4 kbps is planned in this baseline)
2 kbps from Orbiter to PM (only 1 kbps is planned in this baseline)
4 kbps from PM to ground via TDRSS only
1 kbps from ground to PM via TDRSS only



Note: The retrieval assurance management unit is proposed to be a fault tolerant unit monitoring the unit discretes to determine the condition of the active computer. If a computer failure occurs, the unit will command power to be removed from the active computer and applied to the standby computer. The unit shall then initiate a memory load to the newly active computer memory from the mass storage unit. Flight programs plus constantly updated critical flight parameters shall be stored in the mass storage unit.

Figure 27. Retrieval assurance management unit functional block diagram — December 1978.

The 16 kbps [3] and the 2 kbps are Orbiter-imposed constraints. The 4 kbps and the 1 kbps are seen as current maximum attainable rates with an RF system using the TDRSS and employing omni-directional antennas. Realistically, with omni antennas, there is no data rate growth potential. C&DH system studies have been conducted to determine the logical "growth" steps in both communications and data handling should higher transmission rates be necessary (Fig. 28).



*RIU'S ARE REMOTE INTERFACE UNITS

Figure 28. Functional block diagram of on-board communications and data handling system to handle high rate (1 Mbps and above) science data plus low (50 kbps or less) rate engineering data — December 1978.

3.5.7 Reliability. No attempt was made to calculate or design to a specific mean time between failure (MTBF) or reliability curve of the on-board communications and data handling system. However, most box-level reliability figures should be available from manufacturers for both data system options (Figs. 24 and 25). In a few cases in each system, no firm figures will be available since either modifications to existing equipment are required or new boxes must be developed. On the other hand, the modifications and new designs suggested are deemed to be straightforward and achievable, and, when completed, the units should be roughly comparable with other box-level reliability figures for the respective systems; no new techniques are necessary, thus development risks should be relatively low.

Reliability for the flight data system was considered from the viewpoint of providing some reliability through the addition of redundancy. With two exceptions, all functional components are redundant in both options. For example, the measurement requirements show a need for a certain number of RIU's (and extender units). Twice this number of RIU's (and extender units) are actually provided, yielding complete measurement redundancy. The two redundancy exceptions are in the communications area. The transponders are not functionally redundant; both must work to provide RF coverage in both hemispheres. Also, the RF switch linking the two omni antennas is not redundant. Should a transponder fail, the data could flow through the other, although the spacecraft attitude would dictate reception. Consequently, in this baseline no single point failure is expected to exist that would cause loss of the Power Module.

3.5.8 Redundancy Management. Within the guidelines of the systems requirements document [1], no rigid specific concept was devised for all on-board systems. The key was to develop redundancy management in such a way that the feature would not be too expensive, but would provide a high probability of retrieval of the PM should a major failure occur. The basic approach of the on-board communications and data handling system (in both options) was that two parallel, simplex chains would be used, one on-line and one redundant (and usually in standby). Only for the attitude control system does the data system (and computer) have the prime responsibility for detecting failures and taking subsequent corrective action. To detect and correct a computer failure with noncontinuous ground coverage, a retrieval assurance management box was added. It detects a failure and automatically switches to the redundant computer.

3.5.9 On-Orbit Maintenance. With the exception inferred in a brief discussion in the requirements concerning communications, no analysis into on-orbit maintenance or replacement within the on-board communications and data handling system has been conducted.

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APPROVAL

25 kW POWER MODULE UPDATED BASELINE SYSTEM

The information in this report has been reviewed for technical content. Review of any information concerning Department of Defense or nuclear energy activities or programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

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