Prepared for the
GEORGE C. MARSHALL
SPACE FLIGHT CENTER
Huntsville, Alabama

19 November, 1973

Contract No. NAS8-14000
MSFC-DRL-008A
Line Item No. 161S
IBM No. 73W-00315

Astrionics System Designers Handbook

Volume II

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IBM Federal Systems Division, Electronics Systems Center/Huntsville, Alabama

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SECTION 8

ELECTRICAL POWER COMPONENTS

'LECTRICAL POWER
MPONENTS

8.1.1 ATS-F SOLAR ARRAY

The electrical power subsystem derives power from two semicylindrical solar array panel assemblies that are located on the Y-axis of the spacecraft (Figure 8.1.1-1) and extend beyond the 30-foot reflector to avoid shadowing. The solar array output power is a nominal 600 watts initially, decreasing to a nominal 485 watts after 2 years. The solar arrays are deployed, after separation from the Titan III-C launch vehicle, by sequencer control pyrotechnics.

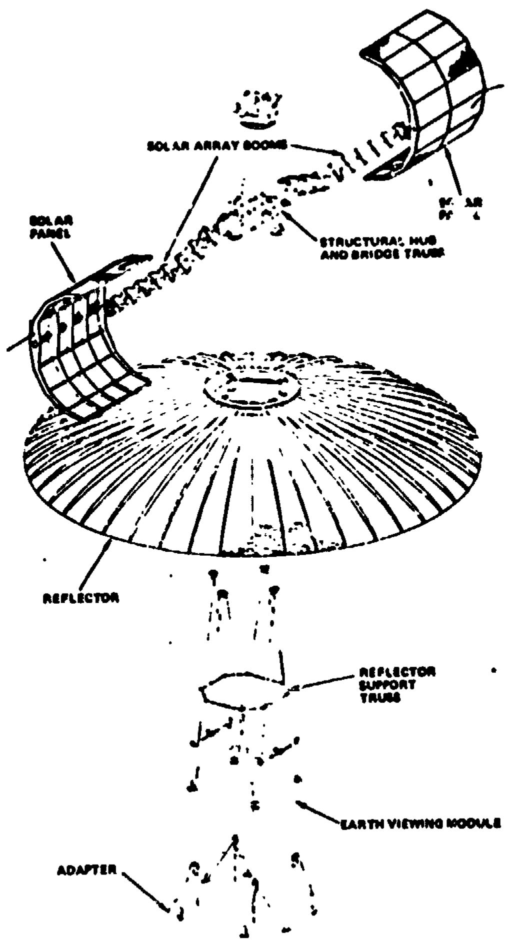


Figure 8.1.1-1. ATS-F Solar Arrays

8.1.1 ATS-F SOLAR ARRAY

General Description

Program: ATS-F
Vendor:
Part Number:

Performance Characteristics

Vehicle Position

Altitude: 19,323 NM
Inclination: 1.8°
Occultation Period: Synchronous Orbit
Attitude: Parallel to Local Vertical

Array

Type: Two semi-cylindrical solar array panel assemblies
Area, One Assembly: 137.2 cm (54 in) radius by 240.1 cm (94.6 in)
No. of Panels: 16 per assembly
No. of Independent Sources: 2 arrays

Panel

Area: $0.5M^2$ (5.8 sq. ft.) solar cell area (675 solar cells)
Watts: 15 to 19 watts (decreasing over 2 yr. life)
Configuration: 3 parallel cells per string, 75 series cells per string, 3 strings per panel

Cell

Type: N/P Silicon
Type A: 2 cm (0.8 in) by 4 cm (1.6 in) by .03 cm (.012 in)
Efficiency: 11%

Physical Characteristics

Size:
Weight:

References

GSFC ATS-F and -G Data Book, September 1972

Design Status

This component is scheduled to be flown as part of the planned mid-1974 ATS-F mission.

8.1.2 ERTS-B SOLAP ARRAY

The ERTS-B solar array (Figure 8.1.2-1) utilizes phosphorous-doped silicon N-on-P solar-cells to accomplish the energy conversion. With a solar-cell efficiency of 11.4 percent at air mass zero, a solar intensity of 140 milliwatts per square centimeter and considering the power-supply operating characteristics, the array will produce approximately 550 watts at a temperature of +35°C at the beginning of orbit life.

The solar array consists of two solar paddles. Each paddle consists of a solar cell mounting structure, solar cell array, a transition section, a latching assembly, a drive motor, with an associated gear-reduction unit, and a control shaft clamp. The attached figure shows the ERTS Observatory, including the solar array. Sun acquisition is simplified by requiring only one axis of rotation for the solar array. Each paddle is composed of nine solar cell circuits, two of which have voltage telemetry. There is also one thermistor for temperature telemetry on each paddle.

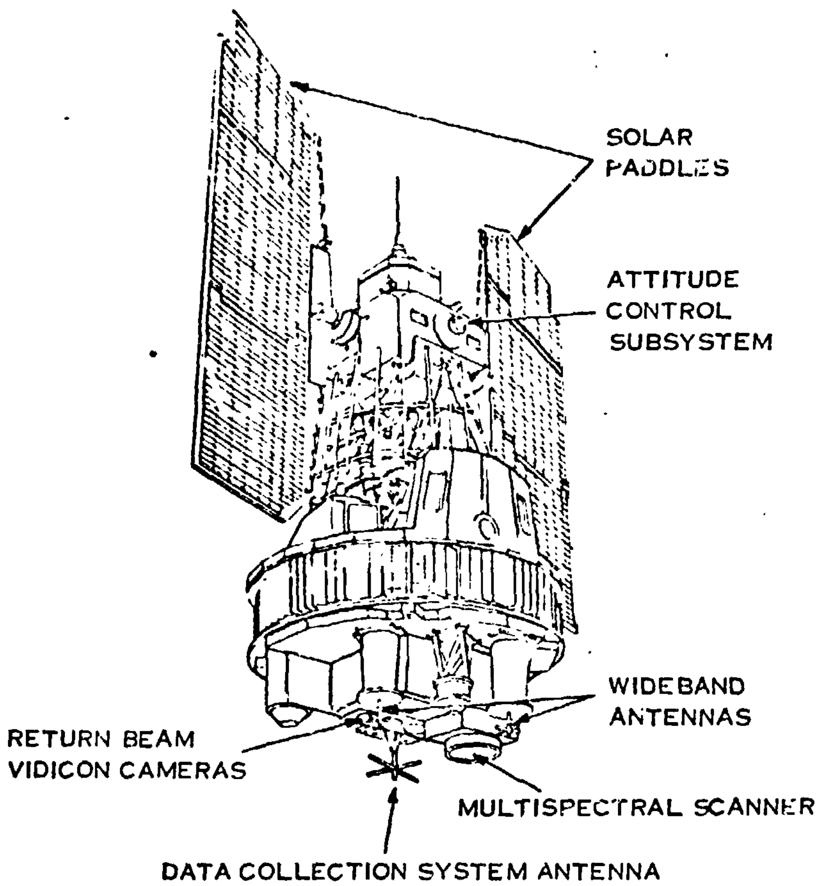


Figure 8.1.2-1. ERTS Observatory Configuration

8.1.2 ERTS-B SOLAR ARRAY

General Description

Program: ERTS-B
Vendor
Part Number:

Performance Characteristics

Vehicle Position
Altitude: 500 MI
Inclination: Polar orbit
Occultation Period: 30 minutes
Attitude: Parallel to local vertical, platforms canted 33 degrees with respect to the driveshaft axis.

Array
No. of Independent Sources: 2
Substrate Material: Aluminum Honeycomb

Platform (Paddle)
Area: $2.3M^2$ ($25.67\ ft^2$)
Voltage 38 volts max
Watts: 550 at $+35^\circ C$ at beginning of orbit life
Configuration: 9 circuits/ platform (paddle)

Cell
Type: N on P Silicon
Efficiency: 11.4%

Physical Characteristics

Size:
Weight:

References

ERTS Reference Manual, Space Division, General Electric Company Valley Forge Space Center, P. O. Box 8555, Philadelphia, Pa. 19101

Design Status

This component is scheduled to be flown as part of the planned late 1973 ERTS-B mission.

8.1.4 OSO-I SOLAR PANEL

The OSO-I solar panel (Figure 8.1.4-1) consists of a rectangular assembly of solar cells configured with a rectangular hole at bottom center to accommodate a pointed instrument assembly (PIA) Solar panel weight is less than 7.6 kilograms. The combined panel and PIA is called the "sail".

The solar panel provides power for non-eclipse operations. Minimum power supplied is 396 watts at 32 volts at the end of one year in orbit. The individual solar cells are N on P silicon, solder coated 2 cm x 2 cm, 0.03 cm thick, with 2 ohm-centimeter base resistivity cell strings. These cells are series-parallel connected and are diode OR'ed to the solar panel bus, using redundant wiring. The solar panel includes 4 temperature sensors.

Protection while in orbit includes a microsheet 0211 coverslide over each solar cell to provide protection from radiation damage and to reduce cell operating temperature. This cover slide has an ultraviolet reflecting filter on the undersurface and an antireflecting coating on the upper surface.

A	SOLAR PANEL
B	POINTED INSTRUMENTS
C	ELEVATION DRIVE ASSEMBLY
D	BALLAST ARMS
E	VHF WHIP ANTENNAS (8)
F	S-BAND ANTENNA
G	ADAPTOR

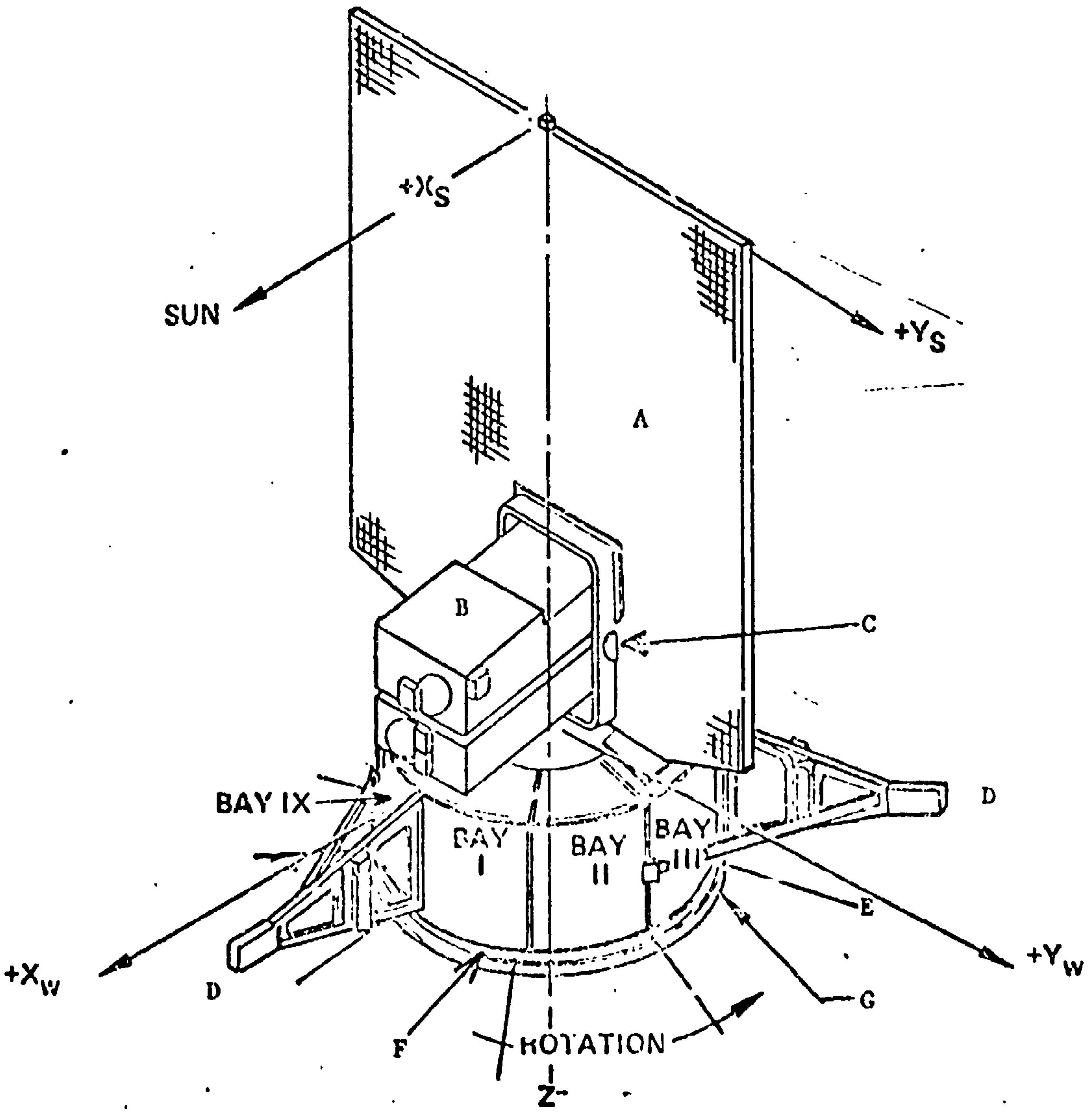


Figure 8.1.4-1. OSO-I Solar Panel Configuration

8.1.4 OSO-I SOLAR ARRAY

General Description

Program: OSO-I
Vendor:
Part Number:

Performance Characteristics

Vehicle Position

Altitude: 300 nautical miles
Inclination: 33 degrees
Occultation Period: 24 to 36 minutes each orbit
Attitude: See Figure 8.1.4-1

Array

No. of Independent Sources: One bus
Substrate Material: Aluminum honeycomb core covered with fiberglass facing sheets.
Area: 4.5 square meters (50 square feet)
Voltage: 32 volts
Current: 12 amps (approx)
Watts: 396 (minimum)
Configuration: Series - parallel groups with bus bar diode isolation

Cell

Type: Silicon N on P
Dimensions: 2 cm (0.8 in) by 2 cm (0.8 in) by .03 cm (.012 in)
Voltage: 0.475 + 0.002 volts
Milliwatts: 59.38 average, 54.64 min (Air Mass zero & solar radiation 135.3 mw/cm^2)
Mounting: RTV 566 (cell groups & strings soldered)

Cover Glass

Material: Optical grade Dow Corning 0211 microsheet
Thickness: .015 cm (.006 in)
Mounting: Dow 93-500
Coating: UV reflecting filter on undersurface and antireflecting coating on the upper surface.

Physical Characteristics

Size:
Weight: 7.1kg (16.7 lbs) (excluding substrate, design goal max)

8.1.4 OSO-I SOLAR ARRAY (continued)

References

1. Development Specification, Solar Panel, DS31331-121, Hughes Aircraft Company, September 29, 1972.

2. Procurement Specification, Solar Cell Cover, Bar Contact PS31331-131, Hughes Aircraft Company, September 29, 1972.

3. OSO-I Observatory Specification, SS31331-100, Hughes Aircraft Company, September 29, 1972.

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

8.1.6 OAO-C SOLAR ARRAY

The solar array converts solar energy into electrical energy to provide the spacecraft power load during the light portion of an orbit and to recharge spacecraft batteries.

The solar array consists of eight solar paddles, six main and two auxiliary. A total of 55,544 solar cells, one by two centimeters in size, are flat-mounted directly to the "A" side of the solar paddle surface, giving a total solar cell area on the array of 10.8 square meters. The main paddles comprise 8.9 square meters of this total area. Figure 8.1.6-1 shows the general arrangement of the solar paddles and the distribution of the cells on each.

Each solar cell assembly consists of a one by two centimeter, twelve mil thick, boron diffused, nominal 2 ohm-centimeter, silicon N/P cell; a blue filter and a six mil thick fused silica cover-glass. An anti-reflective coating (to minimize reflection losses) is deposited on the top (exposed) side of each cover-glass (Figure 8.1.6-2).

In addition to the solar cells used for spacecraft power, also included on the solar array are two radiation detector circuits used to determine solar array performance and to indicate experiment pointing proximity to the sun.

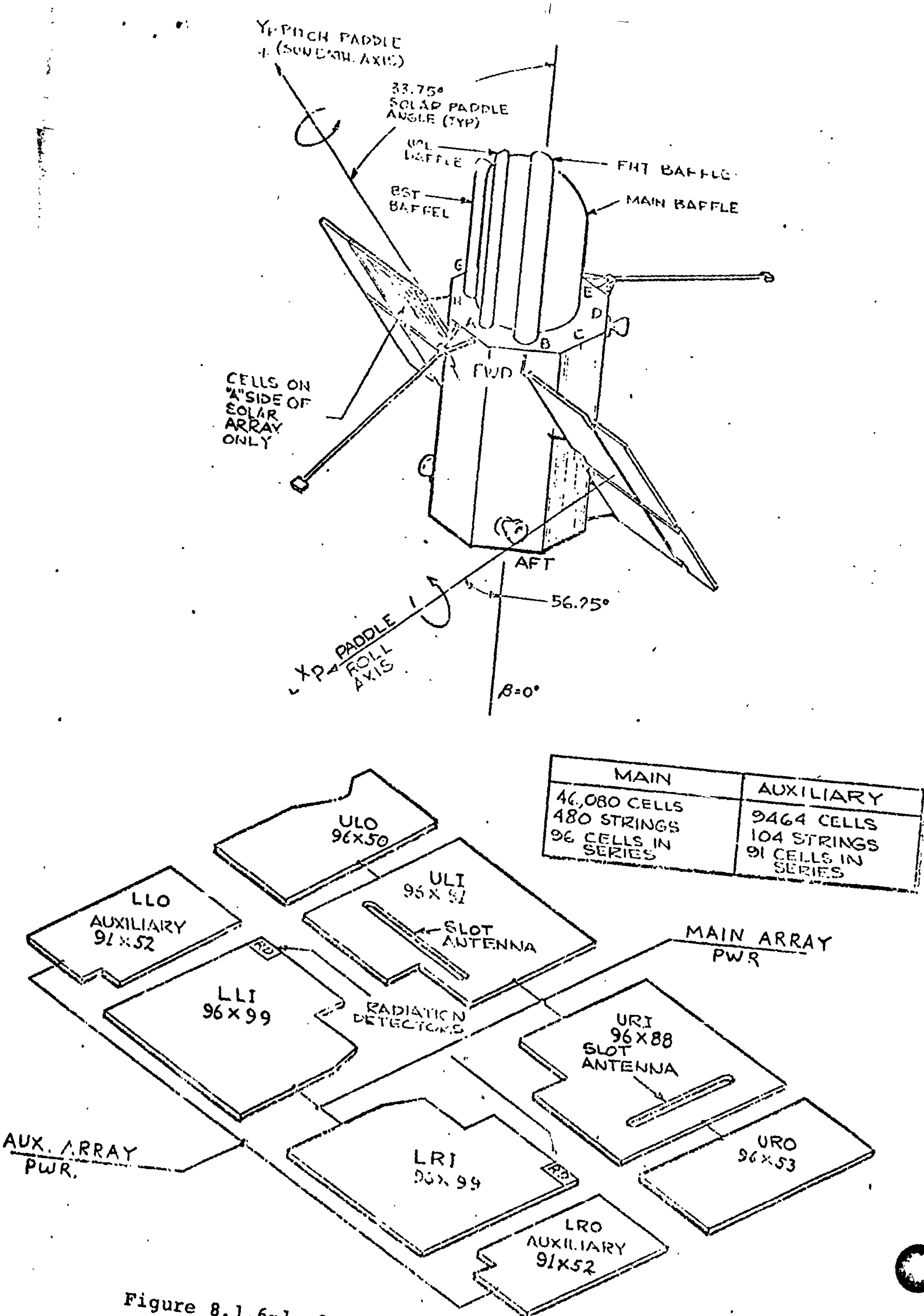
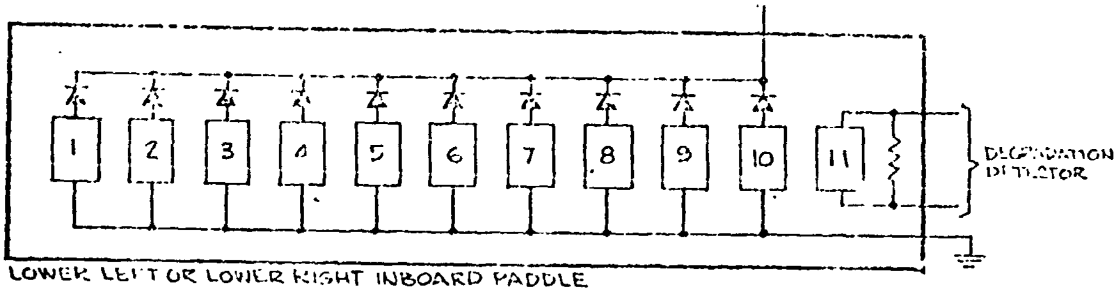
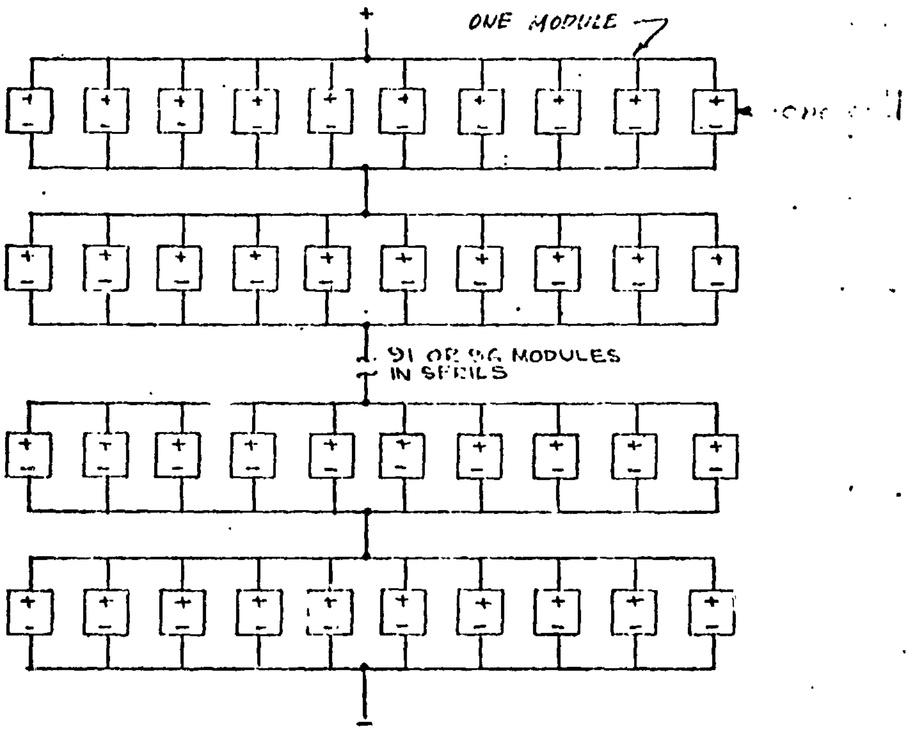


Figure 8.1.6-1 OAO SOLAR ARRAY



CIRCUIT	1	2	3	4	5	6	7	8	9	10	11	
NO. OF CELLS IN MODULE	10	10	10	10	10	10	10	10	10	9	1	
MODULES IN SERIES	96	96	96	96	96	96	96	96	96	96	20	TOT 9504
% OF TOTAL PER PADDLE	9.6	9.6	9.6	9.6	9.6	9.6	9.6	9.6	9.6	8.5		



SOLAR ARRAY CELL MODULE

The cells are made up into modules of 6, 8, 9 or 10 parallel cells.

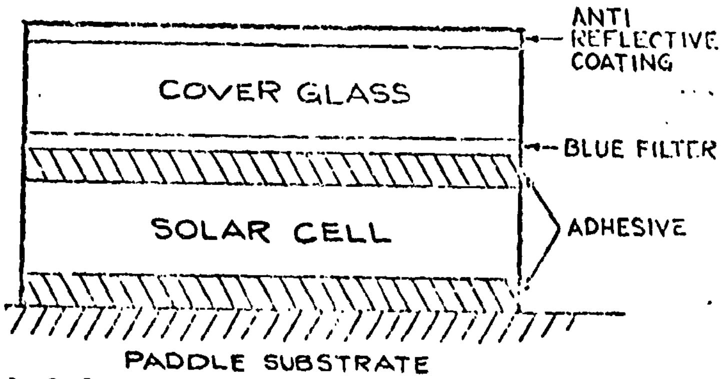


Figure 8.1.6-2 SOLAR CELL, MODULE AND PADDLE

8.1.6 OAO-C SOLAR ARRAY

General Description

Program: OAO-C
Vendor: Grumman
Part Number:

Performance Characteristics

Altitude: 479 mile circular orbit
Array
- Type: Paddle
- Area: 11.1 square meters (119.5 square feet)
- Number of paddles: 6 main and 2 auxiliary
- Number of independent sources: 2
- Substrate Material: Bonded aluminum honeycomb

Paddles
- Area, Main Total: 9.2 square meters (99.2 sq.ft.)
- Area, Auxiliary Total: 1.9 square meters (20.3 sq.ft.)
- Voltage: 23 to 34 volts when connected to load circuits.
- Watts: 610 watts max. (all paddles combined)
- Configuration: 6 main paddles connected in parallel and 2 auxiliary paddles connected in parallel

Modules
- Configuration A: 6 single cells in parallel
- Configuration B: 8 single cells in parallel
- Configuration C: 9 single cells in parallel
- Configuration D: 10 single cells in parallel

Cell
- Type: .03cm (12 mil) thick boron diffused silicon N/P
- Dimensions: 1 cm by 2 cm

Voltage: Voc = 1.3 Vmp
Current: Isc = 1.1 Amp
Efficiency: 13%
Cover Glass
- Material: Fused silica
- Thickness: 6 mil
- Coating: Anti-reflective
- Filter: Blue

Deployment Mechanism: Spring loaded deployment into locked fixed position

Physical Characteristics

Size:
Weight:
Cooling Method: Passive

Reference

Orbiting Astronomical Observatory Functional Operations Manual-Power Subsystem Document No. FO-G-0127-C dated Aug. 1972 Volume B, Goddard Space Flight Center

Design Status

This component was flown as part of the 1972 OAO-C mission.

8.1.7 SKYLAB AM/MDA SOLAR ARRAY

The primary power source for the Airlock Module (AM) electrical power system is the S-IVB solar cell array (Figure 8.1.7-1). A total of 360 solar modules are used to fabricate the array. The total array is divided into eight separate groups of 45 solar modules each, one for each of the eight Power Conditioning Groups.

Positioning of the solar array is accomplished by two DC drive motors for each solar array deployment boom (one primary and one secondary). Motor drive commands are generated by a logic unit, which responds to inputs from a sun sensor mounted on the array fairing, astronaut commands or ground commands. The sun sensor will maintain the solar array normal to the sun's ray within $\pm 10^{\circ}$. Gimbal range of the solar array fairing is from -60° to $+90^{\circ}$ wing position angle.

The degradation in the performance of each S-IVB array group is more than 6 percent per year due to exposure to solar ultraviolet light, proton and electron bombardment, micrometeorite impact, and all other sources of environmental degradation. This degradation is considered to be linear with time in orbit and to consist entirely of a decrease in short circuit current. Open circuit voltage degradation is negligible.

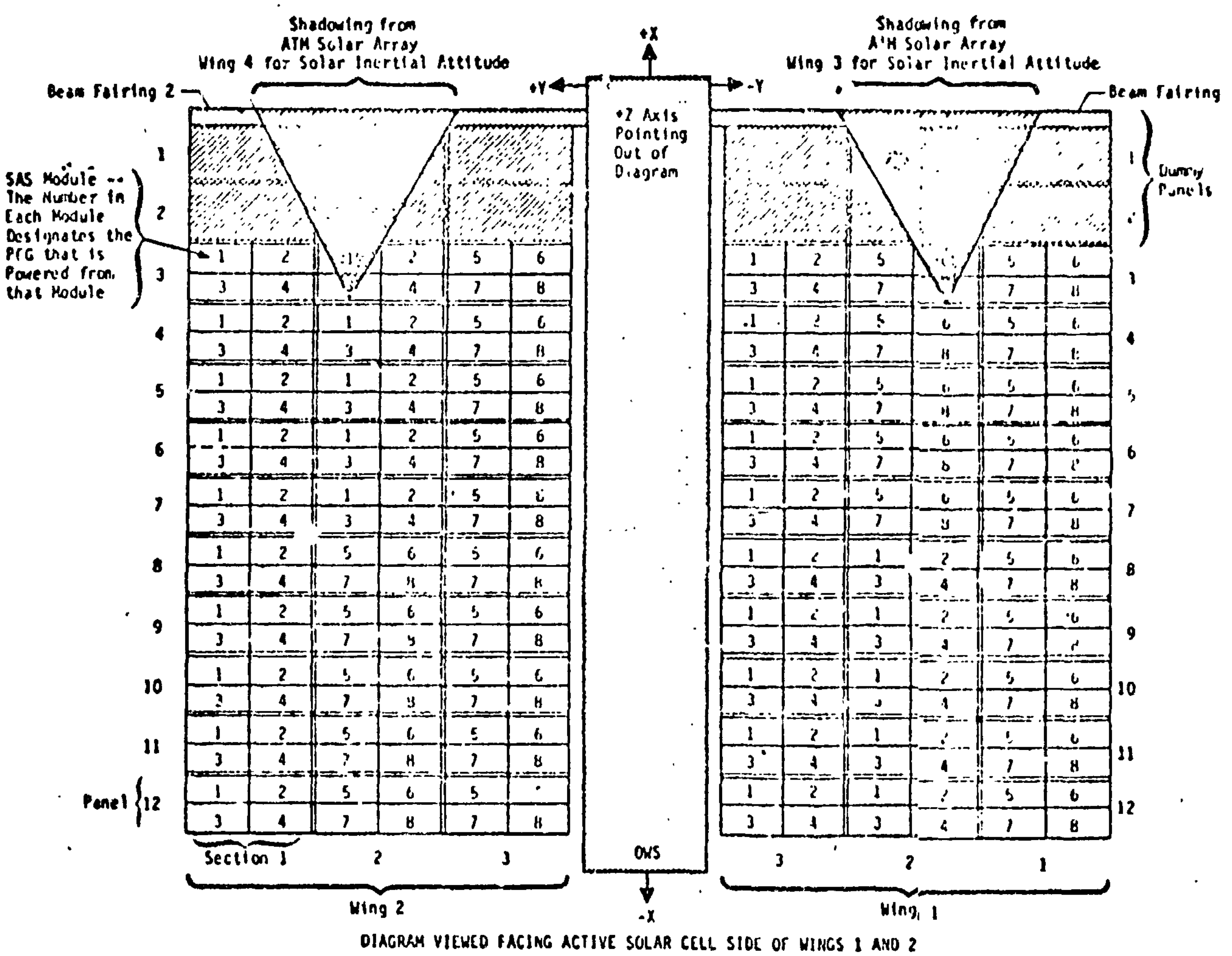


Figure 8.1.7-1. OWS/AM Solar Panels

8.1.7 SKYLAB AM/MDA SOLAR ARRAY

General Description

Program: Skylab AM/MDA
Vendor: McDonnell-Douglas East
Part Number:

Performance Characteristics

Vehicle Position
- Altitude: 235 nm circular
- Inclination: 50 degrees
- Attitude: Solar-inertial

Array
- Type: 2 wings
- Area, Side One: 60.3 square meters (670 sq. ft.)/wing (active area)
- Area Side Two: N/A
- No. of Panels: 30 panels per wing
- No. of Modules: 4 modules per panel
- Independent Sources: 8 total (30 modules per source)
- Power Output: 10,497 watts at 55°C at end of mission
- Substrate Material: Aluminum facesheet with .002 in. Kapton insulation
- Substrate Type: Honeycomb

Panels
- Area: 2.05 square meters (22.74 sq. ft.) per panel
- Watts: 67.0 watts at 29°C per module
- Configuration: Modules within a panel are not interconnected.

Module
- Configuration A: 4 parallel by 154 series cells
- Configuration B: N/A

Cell:
- Type: N/P Silicon
- Dimensions:
  - Type A: 2cm(0.8 in) by 2 cm (0.8 in) by .03 cm (.014 in)
  - Type B: N/A

Efficiency: 11.1%
Cover Glass:
- Material: Fused silica cover
- Thickness: .015 cm (0.006 in).
- Mounting: Fused

Restraint/Release Device: Tension strap/link assembly
Deployment Mechanism: Explosive expandable tube

Physical Characteristics

Size: 9.46 M (31 ft) by 8.33 M (27.3 ft) per wing
Weight: 1841.4 kg (4056 lbs)
Cooling Method: Passive

References

Skylab Data Handbook, Martin Marietta, ED-2002-1399, Dec. 1971

Design Status

This component was flown as part of the 1973 Skylab Mission.

8.1.8 SKYLAB ATM SOLAR ARRAY

The ATM solar array system (SAS) consists of 18 independent solar array power sources mounted on four solar wings. The wings are orient d at 45 degrees to the vehicle longitudinal axis (X-axis). Figure 8.1.8-1 shows the fully deployed array in the orbital configuration. Each solar array wing consists of four complete power sources containing 20 solar cell modules and one-half of a power source containing ten modules. The four one-half power sources are paired to form two power sources, making up a total of 18 independent sources in the complete array. Each solar panel is an independent power source and consists of 20 solar cell modules electrically connected in parallel. Each panel is capable of providing 650 watts at 49 volts and 30 degrees centigrade under Air Mass Zero (AMO) illumination.

The basic building block of the solar panel is the solar cell module. Two configurations of the module are used; one configuration consists of 684 2 cm by 2 cm solar cells with six cells in parallel and 114 cells in series; the other configuration consists of 228 2 cm by 6 cm solar cells with two cells connected in parallel and 114 cells connected in series. Figure 8.1.8-1 also shows the configuration for both types of solar cell modules. Half of the ATM solar array will consist of 2 cm by 2 cm cell modules and the other half 2 cm by 6 cm cell modules.

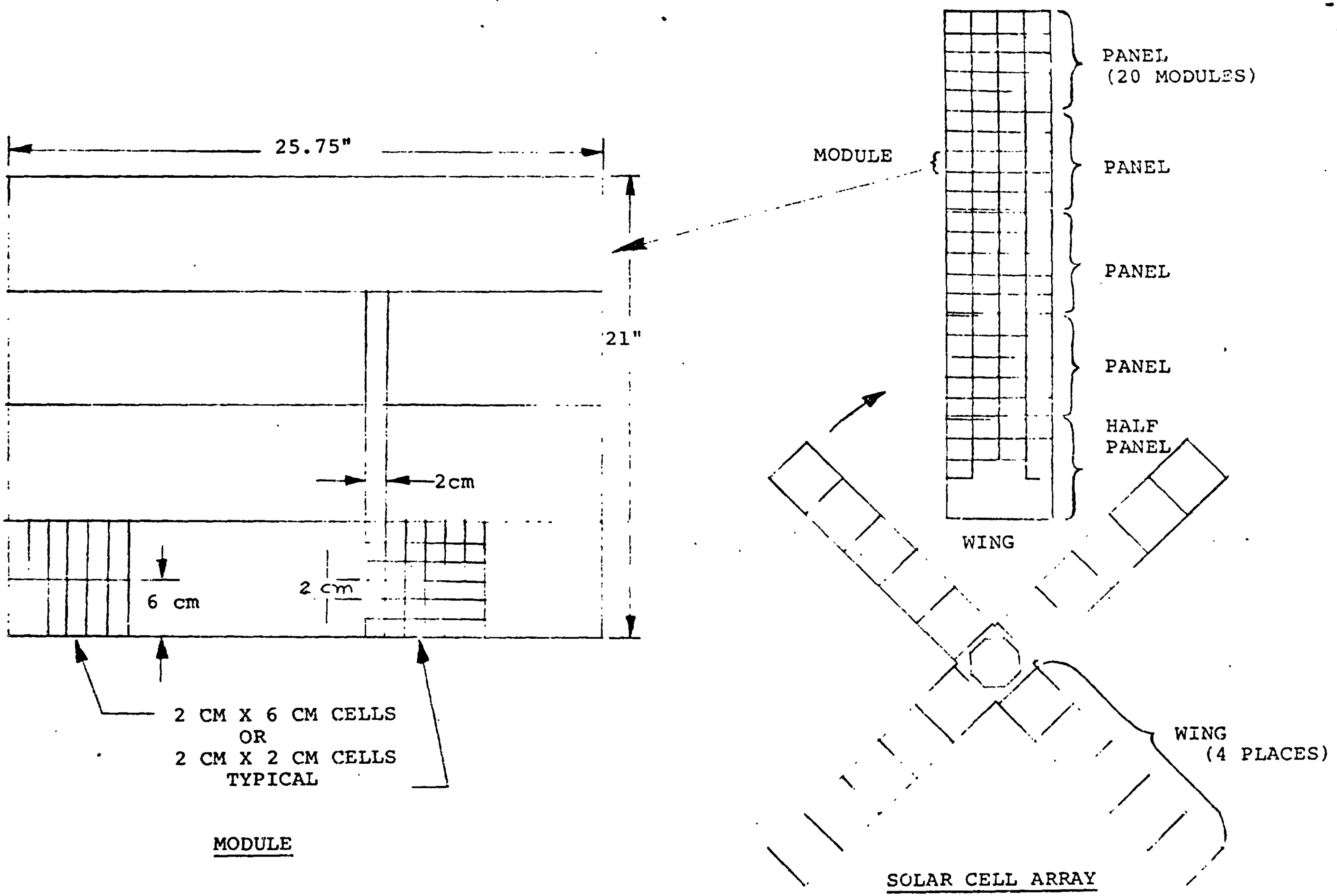


Figure 8.1.8-1. ATM Solar Array

8.1.8 SKYLAB ATM SOLAR ARRAY

General Description

Program: Skylab ATM
Vendor: MSFC
Part Number:

Performance Characteristics

Vehicle Position

Altitude: 235 nm circular
Inclination: 50 degrees
Occultation Period: 36 minutes
Attitude: Solar inertial

Array

Type: 4 wings
Area, Side One: 27 square meters (300 sq. ft.) per wing
Area, Side Two: N/A
No. of Panels: 4.5 per wing
No. of Independent Sources: 18

Panel

Area: 6.1 square meters (68 sq. ft.)
Voltage: 49 V at 30° C under AMO illumination
Current: 13.27 A at 49V, 30°C
Watts: 650 W at 49 V, 30°C
Configuration: 20 modules in parallel

Module

Dimensions: 62.5 cm (24.6 in.) by 50.8 (20 in.) by 1.3 cm (.5 in.)
Configuration A: 2 parallel cells by 114 series cells
Configuration B: 6 parallel cells by 114 series cells

Cell

Type: Shallow diffused N on P silicon
Dimensions:
Type A: 6 cm (2.4 in.) by 2 cm (.8 in.) by .0355 cm (.014 in.)
Type B: 2 cm (.8 in.) by 2 cm (.8 in.) by .0355 cm (.014 in.)
Voltage: 0.43 Vdc with 140 mw/cm^2 at 30°C
Current: 120 ma with 140 mw/cm^2 at 30°C
Efficiency: 10% at AMO

Cover Glass

Material: Dow-Corning 7940 fused silica
Thickness: .03 cm (0.012 in.)
Mounting Adhesive: Dow-Corning Sylgard 182
Coatings: Antireflective coating on top. Reflective UV coating on bottom.

8.1.8

Restraint/Release Device: Released by pyrotechnics
Deployment Mechanism: Spring loaded scissors and motor drive

Physical Characteristics

Size:
Weight: 1725 kg (3800 lbs.)
Cooling Method: Passive

References

40M26423 MSFC Development Report; ED-2002-1045-1 Martin Marietta Electrical Power Sys. Definition Document.

Design Status

This component was flown as part of the 1973 Skylab Mission.

8.1.11 MARINER MARS 71 SOLAR ARRAY

Each of four solar panels is divided into six electrically isolated sections (Figure 8.1.11-1). Each section consists of seventy-eight series by nine parallel cell combinations. Each is connected to a common bus using diode isolation to prevent reverse current flow between sections. The maximum array voltage is limited to 50 volts by a zener string connected across each panel section. The solar panels utilize 2 cm by 2 cm, n on p solar cells of nominal 2 ohm-cm resistivity and 18 mil thickness. The cells are covered with 20 mil thick fused silica cover glass with an anti-reflective optical coating, and an optical coating that limits the blue end of the solar spectrum to a 0.410 micron cut off.

The input solar radiation to the solar array varies during the mission. A maximum of 137 mw/cm^2 can be expected near Earth. A near Mars magnitude will be 67 mw/cm^2 dropping to 60 mw/cm^2 90 days after encounter.

The unregulated solar array power available is 475 watts near Mars, considering no solar flare degradation or reduction in power due to planet albedo. The output voltage of the solar array varies from 38 to 50 volts during the mission.

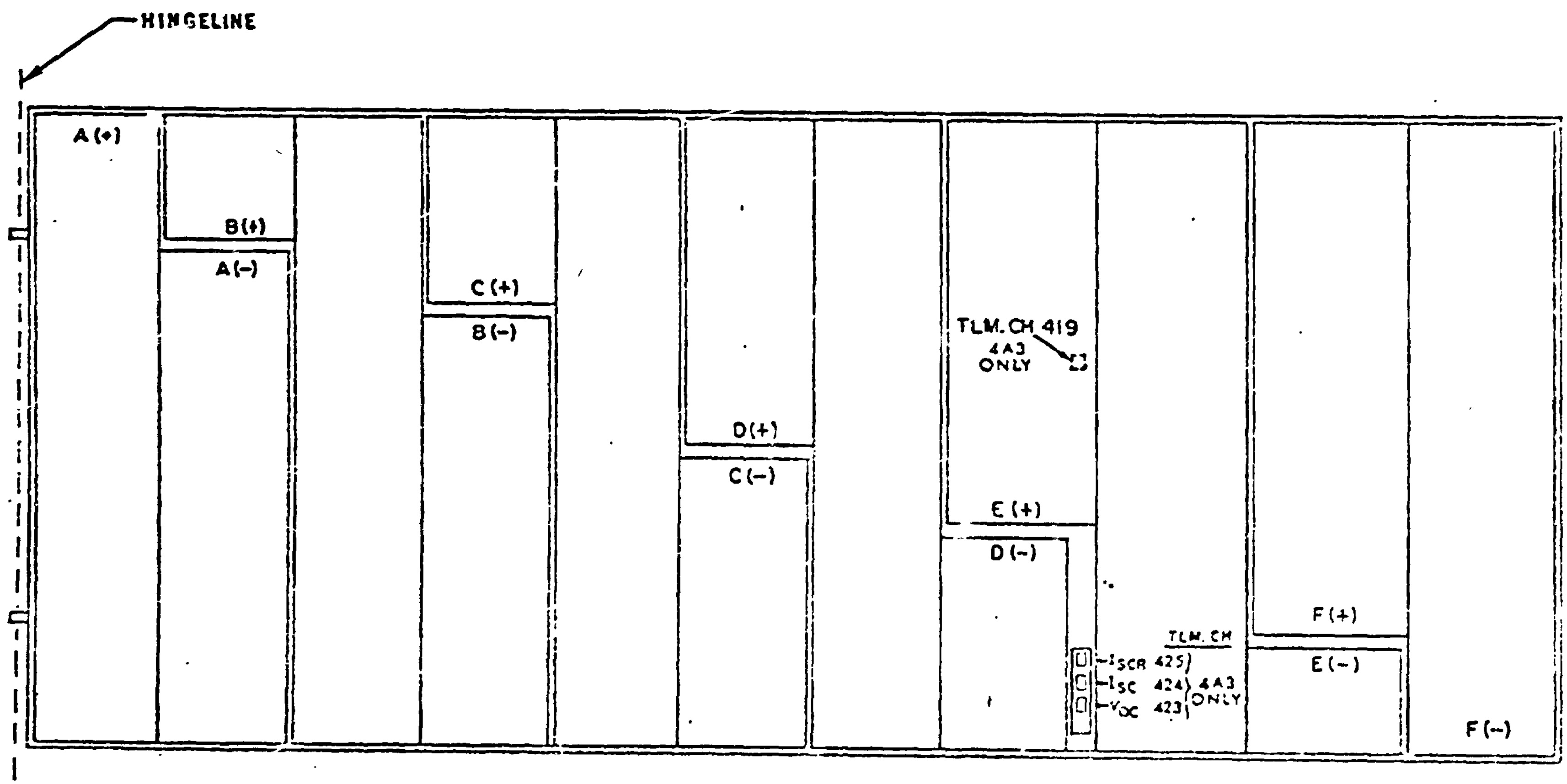


Figure 8.1.11-1. MM71 Solar Panel Design Showing the Six Submodules and Special Telemetry Solar Cell Locations

8.1.11 MARINER MARS 71 SOLAR ARRAY

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Array

Type: Four panels connected in parallel
Area, Side One: 7.5 square meters (83 square feet) maximum
Area Side Two: N/A
No. of Panels: 4
No. of Independent Sources: 1

Panel

Voltage: 38 to 50 volts during mission
Watts: 119 watts/panel near Mars
Configuration: 78 series by 56 parallel cells

Submodules: 6 per panel

Configuration A: 78 series by 9 parallel cells/submodule

Cell:

Type: N on P
Dimensions:
Type A: 2 cm (.8 in) by 2 cm (.8 in) by .046 cm (.018 in)
Type B: N/A

Cover Glass

Material: Fused silica cover glass
Thickness: .051 cm (.02 in)
Coating: Anti-reflective and limits blue end of the solar spectrum to a 0.410 micron cutoff

Physical Characteristics

Size:
Weight: Solar array assembly excluding substrate structure 22.7 kg (50 lbs.)
Cooling Method: Passive

8.1.11

References

Mariner Mars 71 Flight Equipment Power Subsystem JPL M71-2004-1 dated 6 November 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.2.1 ATS-F BATTERY

Energy storage is provided by two 19-cell, 15 A-H, nickel-cadmium batteries which are discharged in parallel. The batteries provide electrical power during occult portions of the ascent phase, eclipse periods and peak loading which exceeds the solar array capability. When excessive power is available from the solar array, the battery chargers recharge the batteries. Each battery has separate charge control circuits and is normally charged at a current limited (1.5 Amps) rate with temperature compensated voltage control. The battery is removed from charge when the battery temperature reaches $35^{\circ}C \pm 2^{\circ}C$.

The batteries are sized to provide sufficient power for the occult and load sharing modes without exceeding a 50% depth of discharge.

8.2.1 ATS-F BATTERY

General Description

Program: ATS-F Battery
Vendor:
Part Number:

Performance Characteristics

Number of Cells: 19
Voltage Per Cell: 1.2 (Min), 1.5 (Max)
Battery Output Voltage: 22.8 (Min), 28.5 (Max)
Amp/Hrs.: 15

Physical Characteristics

Size:
Weight:

References

GSFC ATS-F and -G Data, Book, September 1972.

Design Status

This component is scheduled to be flown as part of the planned mid-1974 ATS-F mission.

8.2.2 ERTS-B STORAGE MODULE

Each of the eight storage modules contains a series string of 23 nickel-cadmium storage cells each rated at 4.5 ampere-hour capacity. The charge current available during satellite day is determined by the solar-array output and the spacecraft subsystem load power requirements. It is regulated as a function of battery voltage and temperature with a closed-loop charge controller. The maximum charge current is also limited to a preset value. Battery voltages, current and temperature are telemetered.

8.2.2 ERTS-B STORAGE MODULE

General Description

Program: ERTS-B
Vendor:
Part Number:

Performance Characteristics

Number of Cells: 23
Voltage Per Cell: 1.22 volts/cell
Battery Output Voltage: 28.0 Vdc
Amp/Hrs.: 4.5

Physical Characteristics

Size:
Weight:

References

Earth Resources Technology Satellite Reference Manual, General Electric Co., Valley Forge Space Center, P.O. Box 8555, Philadelphia, Pennsylvania 19101.

Design Status

This component is scheduled to be flown as part of the planned late 1973 ERTS-B mission.

8.2.3 HEAO-A BATTERY

The HEAO-A batteries shall normally store electrical energy during the sunlit portion of the orbit and discharge during the eclipsed portion of the orbit. However, capability shall be provided for other modes of operation such as launch and system startup, orbit transfer, and off-solar vector pointing. There shall be a minimum of four (4) batteries. Sufficient power and capacity shall be available so that the two year mission can be completed with the loss of one battery. The batteries shall be nickel cadmium rechargeable batteries capable of satisfying the present estimated load of 640 watts (including 20% growth factor) for a minimum mission life of two years and/or a minimum of 12,000 cycles. The maximum normal cyclic depth of discharge is 15%.

8.2.3 HEAO-A BATTERY

General Description

Program: HEAO-A
Vendor:
Part Number:

Performance Characteristics

Number of Cells: 22
Voltage Per Cell. 1.1 V
Connection Arrangement: Series
Battery Output Voltage: 24.2 Vdc Nominal
Amp/Hrs.: 30

Physical Characteristics

Size: 38.1 cm (15 in) by 50.8 cm (20 in) by 17.8 cm (7 in)
Weight: 30.9 kg (68 lbs)

References

High Energy Astronomical Observatory Preliminary Reqts. Review Vol. III-D Subsystems Reqtes and Data HEAO Electrical Subsystem Doc 17622-301-001-001, TRW Systems Group (DR No. CM-05), 24 July 1972.

Notes

Thermal capacity: 2.3 by 10^{-4} cal/gm °C
Thermal Resistance: 0.13°C/W (max)
T (Hottest cell to baseplate): 3°C for nominal subsystem operation

Design Status

The HEAO-A program is currently in a state of redefinition. This component is a viable candidate for application in the 1977 HEAO-A mission.

8.2.4 OSO-I BATTERY

The battery provides power for the spacecraft subsystems and experiment instruments during eclipse and other periods when power required exceeds solar array output. To meet these demands, two twenty-one cell, twelve ampere-hour nickel-cadmium batteries are provided. The normal average depth of discharge is 15%, at average charge and discharge rates of 3.2 amperes. Nominal battery voltage is 28 Vdc. Battery weight is less than 33.0 lbs. Battery life is in excess of one year. Each battery consists of two cell packs, one ten-cell pack and one eleven-cell pack. Each battery contains:

a) One charge control sensor per cell pack

b) One telemetry sensor per cell pack

c) One over temperature switch per cell pack

8.2.4 OSO-I BATTERY

General Description

Program: OSO-I
Vendor:
Part Number: 3280221 & 3280222

Performance Characteristics

Number of Cells: 21
Voltage Per Cell: 1.20 V
Connection Arrangement: One 10 cell and one 11 cell pack
Battery Output Voltage: 25.2 V
Amp/Hrs.: 12
Additional Electrodes: One signal-electrode cell

Physical Characteristics

Size:
Weight: 15 kg (33 lbs) max.
Cooling Method: Ambient and conduction via mount surface

References

Hughes Aircraft, Development Specification for Battery DS 31331-123, Revision A , January 15, 1973

Hughes Aircraft, Procurement Specification for Battery Cell, PS 31331-132, Revision A , November 11, 1971

Hughes Aircraft, Environmental Test Requirements, SS 31331-003, Revision B , January 8, 1973

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

8.2.6 OAO-C BATTERY

The rechargeable nickel-cadmium storage batteries (Figure 8.2.6-1) are used to furnish power for launch and initial stabilization, for all loads during the dark period and for peak power requirements during light periods. The OAO battery configuration consists of three (3) twenty-two cell batteries. Each twenty-two (22) cell battery consists of twenty-one (21) nickel-cadmium storage cells and one (1) nickel-cadmium auxiliary electrode cell connected in series. The nickel-cadmium auxiliary electrode cell is the most negative cell of the twenty-two (22) cell battery. The total of sixty-six (66) cells is divided into two (2) battery assemblies of thirty-three (33) cells each. One assembly contains thirty-three (33) nickel-cadmium storage cells and the other assembly contains thirty (30) nickel-cadmium storage cells and three (3) nickel-cadmium auxiliary electrode cells.

The battery assemblies are designed and constructed so that a 99.8 percent of probability of successful operation, within the performance requirements specified herein, is achieved during the launch and one year orbital lifetime. All components and component parts have an operating life of 17,500 hours as well as a charge-discharge capability of 10,000 cycles during this period.

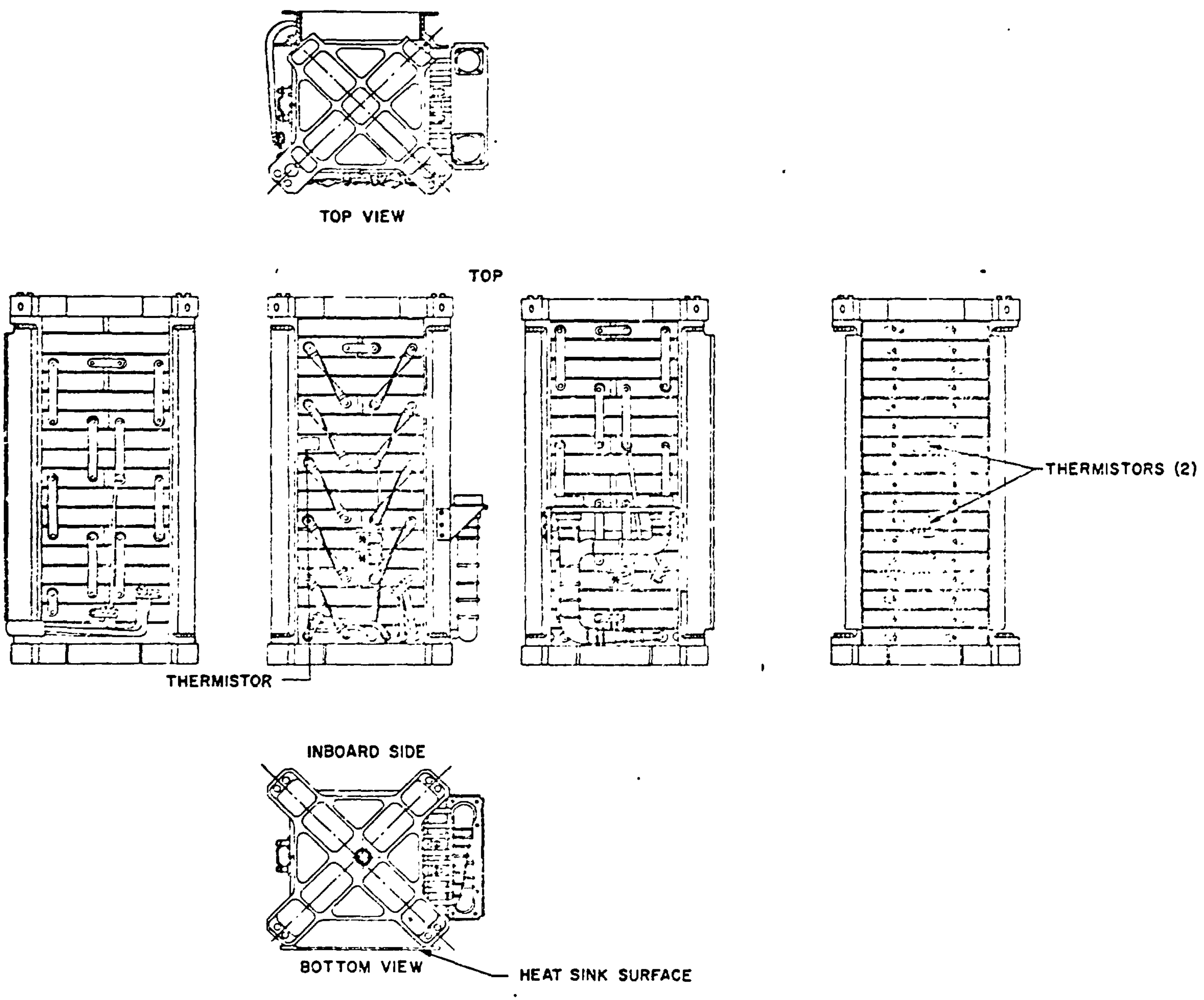


Figure 8.2.6-1. OAO-C Nickel Cadmium Storage Battery Assembly

8.2.6 OAO-C BATTERY

General Description

Program: OAO-C
Vendor:
Part Number:

Performance Characteristics

Cells
Number: 22
Voltage Per Cell: 1.5 V (20 Amp/Hr.)
Material: Nickel-Cadmium

Output
Voltage: 26-32V
Amp-Hrs: 20

Physical Characteristics

Size:
Weight: 37.2 kg (82 lbs) max

References

Nickel Cadmium Storage Battery Assembly, Power Supply Subsystem Specification AV-252CS-26D, 4 September 1969, Grumman Aircraft Engineering Corp., Bethpage L.I., N.Y. 11714.

Design Status

This component was flown as part of the 1972 OAO-C mission.

8.2.7 SKYLAB AM/MDA BATTERY

The Skylab AM/MDA battery consists of 30 series-connected Ni-Cd cells. Rated capacity is 33 amp-hr. Discharge voltage ranges from 40 V at zero current and full charge to 37 V at 23 amp at a 50% depth of discharge and 20°F battery temperature. Thermistors for amp-hr compensation and charge rate are connected to the charger control logic. An over temperature switch and a pressure relief valve are provided. The battery retains 65% of rated capacity after 4000 charge/discharge cycles at a 30% depth of discharge. Eight such batteries are required.

8.2.7 SKYLAB AM/MDA BATTERY

General Description

Program: Skylab AM
Vendor: Eagle-Picher Company
Part Number: SAR-8055-17

Performance Characteristics

Number of Cells: 30
Voltage Per Cell: 1.4 open-circuit voltage
Cell Dimensions: 4.6 cm (1.8 in) by 10.2 cm (4 in) by 190 cm (7.5 in)
Connection Arrangement: Series
Battery Output Voltage: 30 to 42 V
Amp/Hrs: 33 at 18 Amp discharge rate (120°F)
Additional Electrodes: None

Physical Characteristics

Size: 68.6 cm (27.25 in) by 21.6 cm (8.5 in) by 19.6 cm (7.75 in)
Weight: 54.5 kg (120 lbs)
Cooling Method: Mount on active coolant system coldplate

References

McDonnell Astronautics Co., Procurement Specification for Eagle-Picher, 61B769004 Nickel-Cadium Storage Battery, September 21, 1971.

Notes

Cell relief valves opens at 200 ± 25 psi and reseats at 100 psiq minimum. The battery case relief opens at 35 ± 5 psig and reseats at 20 psig, minimum.

Thermal switch opens at 125°F to provide signal to stop battery charging, switch recloses when temperature drops to 117°F.

Thermistors for charge rate control.

Thermistors for A-hr. meter compensation.

Design Status

This component was flown as part of the 1973 Skylab mission.

8.2.8 SKYLAB ATM BATTERY

The Skylab ATM battery contains 24 series-connected hermetically-sealed Ni-Cd cells of the four electrode type. Each battery unit weighs approximately 48 pounds. One of the design criteria is that per orbit depth of discharge of the battery be limited to an average of 30 percent of rated capacity during the solar inertial mode of operation.

Charging of the batteries is controlled by use of a charge control system which monitors the third electrode signal voltage of three cells in each battery and terminates the charge when a 200 mV level is reached. Oxygen gas pressure created during charging causes current to flow in the circuit formed by the external connection of the third electrode and the cell negative electrode through a resistive load. Over a limited range there is a linear relationship between oxygen pressure and the third electrode voltage across this load. The fourth electrode is provided to recombine the excess oxygen or some hydrogen that may be developed during the charging cycle. This reduces excessive pressure build-up within the cells.

8.2.8 SKYLAB ATM BATTERY

General Description

Program: Skylab ATM
Vendor: General Electric
Part Number: AB12G

Performance Characteristics

Number of Cells: 24
Voltage Per Cell: 1.25 to 1.45 V
Cell Dimensions: 18.12 cm (7.1 in) by 7.62 cm (3 in) by 2.29 cm (.9 in)
Connection Arrangement: Series
Battery Output Voltage: 30 to 35 volts
Amp/Hrs: 20 A-Hrs
Additional Electrodes: 3rd for charge sensing; 4th for oxygen recombination

Physical Characteristics

Size:
Weight: 22.7 kg (50 lbs)
Cooling Method: Passive

References

MSFC Development Report 40M26995; Martin-Marietta Electrical Power System Definition Document ED-2202-1045-1, July 21, 1972.

Notes

This unit is an integral part of the total CBRM (charger battery regulator-module).

This battery contains an electric heater to maintain temperature above 0°C.

Design Status

This component was flown as part of the 1973 Skylab mission.

8.2.9 APOLLO-17 CMS BATTERY

A secondary source of energy storage is provided by five silver oxide-zinc batteries located in the CSM. Three rechargeable entry and postlanding batteries supply sequencer logic power at all times, supplemental dc power for peak loads, all operating power required for entry and postlanding, and can be connected to power either or both pyro circuits. Two pyro batteries provide energy for activation of pyro devices throughout all phases of a mission.

Each entry and postlanding battery is mounted in a vented plastic case and consists of 20 silver oxide-zinc cells connected in series. The cells are individually encased in plastic containers which contain relief valves that open at 35 ± 5 psig, venting during an overpressure into the battery case. The three cases can be vented overboard through a common manifold, the battery vent valve, and the ECS waste water dump line.

Each battery is rated at 40-ampere hours (AH) minimum and will deliver this at a current output of 35 amps for 30 minutes and a subsequent output of 2 amps for the remainder of the rating.

At Apollo mission loads each battery is capable of providing 45 AH and will provide this amount after each complete recharge cycle. However, 40 AH is used in mission planning for inflight capability, and 45 AH for postlanding capability of a fully charged battery.

Open circuit voltage is 37.2 volts. Sustained battery loads are extremely light (2 to 3 watts); therefore a battery bus voltage of approximately 34 V c will be indicated.

The two pyrotechnic batteries supply power to initiate ordnance devices in the SC. The pyrotechnic batteries are isolated from the rest of the EPS to prevent the high-power surges in the pyrotechnic system from affecting the EPS, and to ensure source power when required. These batteries are not to be recharged in flight. Entry and postlanding battery A, B, or C can be used as a redundant source of power for initiating pyro circuits in the respective A or B pyro system, if either pyro battery fails.

8.2.9 APOLLO-17 CSM BATTERY

General Description

Program: Apollo 17
Vendor:
Part Number:

Performance Characteristics

Number of Cells: 20
Voltage Per Cell: 1.9V
Battery Output Voltage: 37.8V (Open Circuit)
Amp/Hrs.: 40 Amp/Hrs.

Physical Characteristics

Size:
Weight:

References

Apollo Operations Handbook SM2A-03-Block II (1) Vol. 1 Spacecraft Description. Contract NAS9-150 Exhibit I, Para. 10.3 Published under authority of NASA Spacecraft Sys. Operations Branch, Flight Crew Support Division.

Design Status

This component was flown as part of the 1972 Apollo mission.

8.2.10-1 APOLLO-17 LUNAR MODULE ASCENT BATTERY

The two ascent stage batteries (Figure 8.2.10-1) are identical. Each battery is composed of silver-zinc plates, with a potassium hydroxide electrolyte. Each battery weights 125 pounds, and has a 296 ampere-hour capacity (50 amperes at 28 volts dc for 5.92 hours, at +80°F) when discharged in accordance with the nominal power profile of the mission. The batteries can operate in a vacuum while cooled by ECS cold rails to which the battery heat sink surface is mounted. The nominal operating temperature of the ascent stage batteries is approximately +80°F. Two ascent stage batteries ordinarily supply the dc power requirement from normal staging to final docking of the ascent stage with the orbiting CSM or during any malfunction that requires separation of the ascent and descent stages. However, if one ascent stage battery fails, the remaining battery provides sufficient power to accomplish safe rendezvous and docking with the CSM during any part of the mission.

The ascent batteries do not have low-voltage taps; inherent high-voltage characteristics are eliminated because the batteries operate with sufficient loading to bring the batteries to the proper partial discharge voltage. Normally, the ascent stage batteries are paralleled to ensure that both batteries discharge evenly. The terminal voltages of an ascent stage battery are as follows:

Nominal voltage: 30.0 volts dc

Minimum voltage (with two ascent batteries): 28.0 volts dc

Minimum voltage (with one ascent battery, abort): 27.5 volts dc

Maximum voltage (under load): 32.5 volts dc

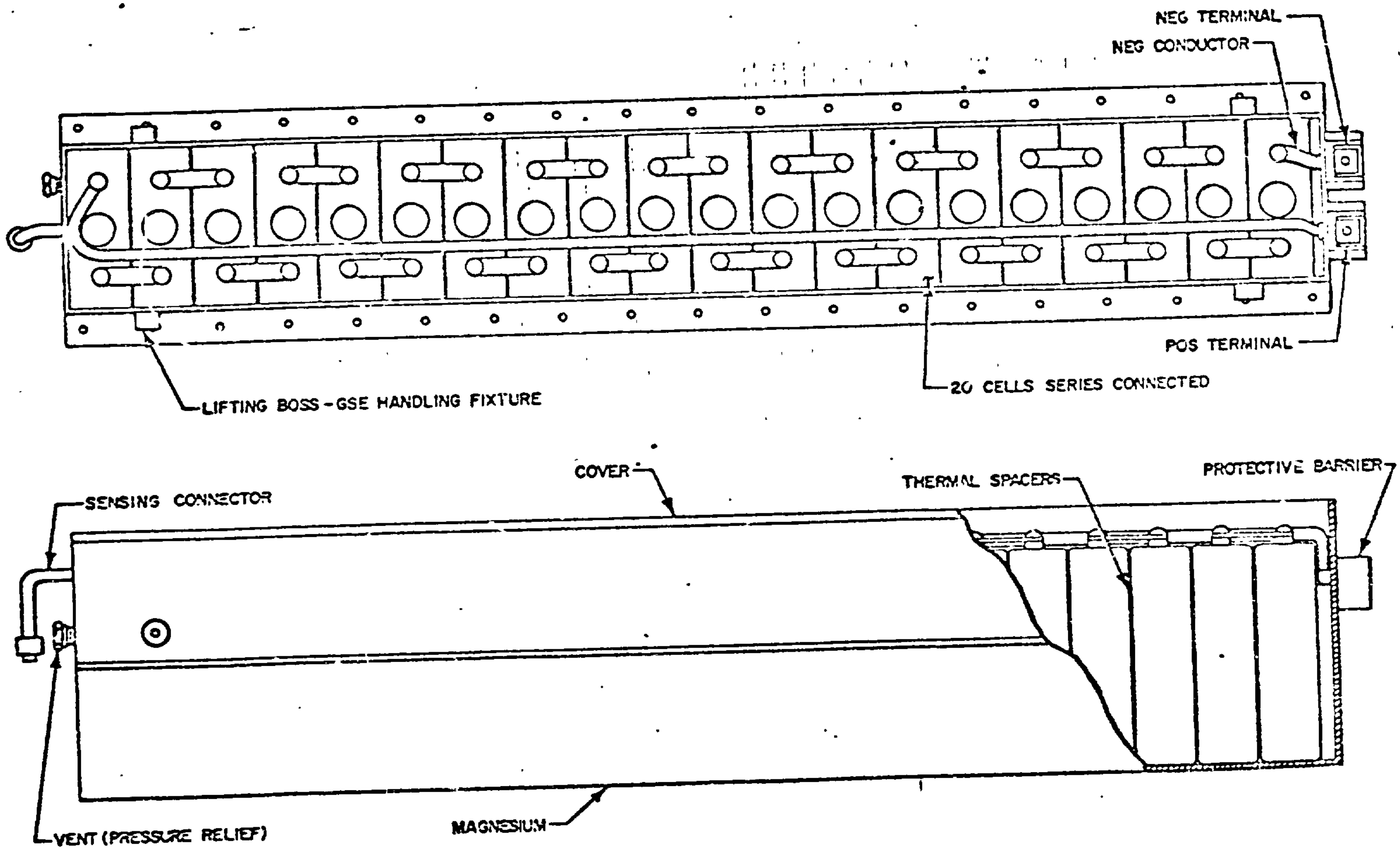


Figure 8.2.10.1. Lunar Module Ascent Battery

8.2.10-1 APOLLO-17 LUNAR MODULE ASCENT BATTERY

General Description

Program: Apollo 17 Lunar Module
Vendor: Eagle Pitcher
Part Number: LSC 390-21000 (control dwg)

Performance Characteristics

Number of Cells: 20
Voltage Per Cell: 1.5V
Connection Arrangement: Series
Battery Output Voltage: 30.0 Vdc nominal 27.5 Vdc min
32.5 Vdc max
Amp/Hrs.: 296 min
Electrolyte: Potassium Hydroxide

Physical Characteristics

Size:

Weight: 59.5 kg (131 lbs) maximum
Cooling Method: Battery heat sink cooled by ECS

References

Grumman Aircraft Specification LSP-390-21C dated May 12, 1967; Electrical Power System Study Guide, Lunar Module LM-5 & Subs, LSG 770-154-4.

Design Status

This component was flown as part of the 1972 Apollo 17 mission.

8.2.10-2 APOLLO-17 LUNAR MODULE DESCENT BATTERIES

The descent batteries include the lunar battery, which is identical with descent batteries No. 1 through 4. Each battery is composed of silver-zinc plates, with a potassium hydroxide electrolyte. Each battery has 20 cells, weights 135 pounds, and has a 400-ampere-hour capacity (25 amperes at 28 volts dc for 16 hours, at +80°F) when discharged in accordance with the nominal power profile of the mission. The batteries can operate in a vacuum while cooled by an ECS cold-rail assembly to which the battery heat sink surface is mounted. If one descent stage battery fails, the remaining descent stage batteries can provide sufficient power for a curtailed mission. Five thermal sensors monitor cell temperature limits (+145 $\pm$ 5°F) within each battery. The batteries initially have high-voltage characteristics; a low-voltage tap is provided (at the 17th cell) for use from T-30 minutes through transposition and docking. The high-voltage tap is used for all other normal LM operations. (Manual switchover from low to high voltage usually occurs when the battery has discharged to approximately 90% of capacity, less than 27 volts dc bus voltage.) Normally, the descent stage batteries are paralleled so that all batteries discharge evenly. The terminal voltages of a descent battery are as follows:

Nominal voltage: 30.0 volts dc

Minimum voltage during 40-ampere discharge: 28.8 volts dc

Maximum voltage (under load): 32.5 volts dc

Figure 8.2.10-2 shows the descent battery.

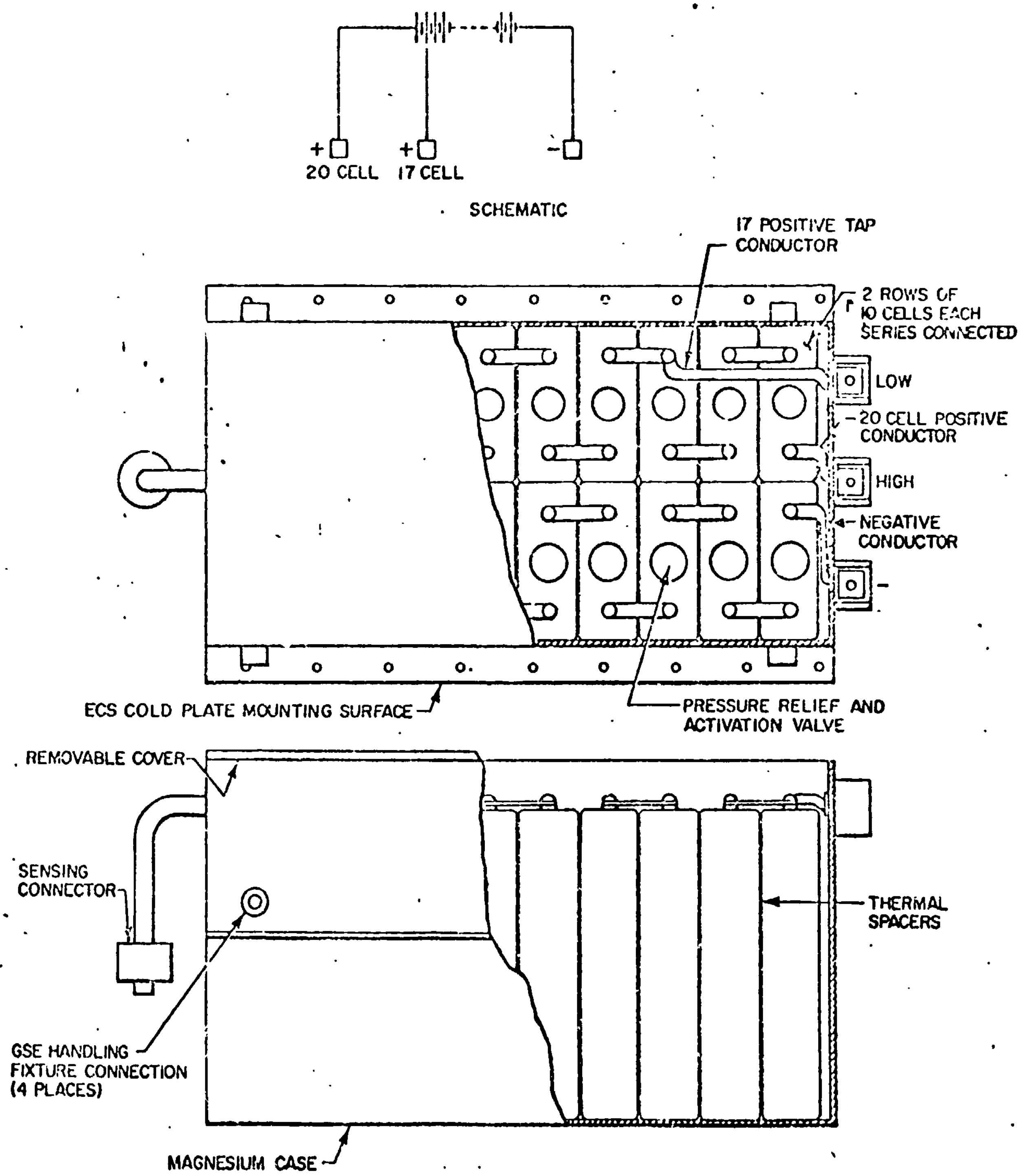


Figure 8.2.10- 2 Lunar Module Descent Battery

8.2.10-2 APOLLO-17 LUNAR MODULE DESCENT BATTERY

General Description

Program: Apollo 17 Lunar Module
Vendor: Eagle Pitcher
Part Number: LSC-390-22000

Performance Characteristics

Number of Cells: 20
Voltage Per Cell: 1.5V nominal
Connection Arrangement: Series
Battery Output Voltage: 30.0 + 2.5
- 2.0
Amp/Hrs.: 400
Additional Electrodes: None
Electrolyte: Potassium Hydroxide

Physical Characteristics

Size:
Weight: 63.6 kg (140 lbs) maximum
Cooling Method: Battery heat sink-mounted on a coolant cooled plate.

References

Lunar Module Cockpit Simulation Training Course (LM-10-EPS-2).
Specification, Battery, Primary Power, Descent Stage, No. LSP-390-22C Grumman Aircraft Engineering Corp. dated May 26, 1967.

Design Status

This component was flown as part of the 1972 Apollo 17 mission.

8.2.11 MARINER MARS 71 BATTERY

The battery provides spacecraft power for the launch to the sun acquisition phase, maneuvers, orbit insertion and orbit trims, and the boost out of a share mode. The battery is kept fully charged when not in use but kept on the line throughout the mission to provide a backup for the solar panels. The battery is a nickel-cadmium battery with a nominal capacity of 20 amp-hrs and an operating temperature range of 30° to 90°F. The battery consists of 26 series connected rechargeable cells. The discharge voltage ranges from 36 to 27 volts at the battery terminals. The charger rate is 2 amps for the high rate and 0.65 amps for the low rate. The battery contains two normally open temperature switches connected in parallel to provide a closure at 100°F for the purpose of switching the battery charger from the high to low rate charge. Switching to the low rate charge is also brought about when the battery voltage reaches 37.5 volts during high rate charge.

8.2.11 MARINER MARS 71 BATTERY

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Number of Cells: 26
Voltage Per Cell: 1.0 to 1.4V
Connection Arrangement: Series
Battery Output Voltage: 27 to 36 volts
Amp/Hrs.: 20

Physical Characteristics

Size:
Weight: 28.1 kg (62 lbs)

References

Mariner Mars 71 Flight Equipment Power Subsystem, JPL M71-2004-1, dated 6 November 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.3.2 ERTS-B PULSE WIDTH MODULATED REGULATOR

Spacecraft loads are supplied by redundant pulse width modulated (PWM) regulators in the power control module. During satellite night the PWM regulators input is derived from the discharge of nickel -cadmium batteries. When the spacecraft enters sunlight solar array power is supplied to the regulators.

The PWM regulators, while more efficient than series regulators, have a variable conversion efficiency, with higher efficiency when operating from low voltages such as batteries than from high voltages such as solar arrays.

As each battery charge current is cut back to a minimum, the array-bus voltage increases. The voltage operating point on the array I-V curve increases as the battery charge current and/or spacecraft load decreases. When the array voltage reaches the threshold voltage of a shunt dissipator the array power is dissipated in this shunt load thereby limiting the array voltage to approximately 38 volts.

When a change of more than ± 2 volts from the PWM regulator output of -24.5 volts is sensed by a comparator, the standby PWM regulator is switched on and the operating regulator is turned off.

The PWM regulator itself is current limited to approximately 22 amperes. With further load increases, the regulated bus voltage decreases to the battery tap voltage, approximately -16 to -18 volts. The regulators are not switched while under overload.

Redundant PWM's are also located in the Payload Regulator Module for additional spacecraft loads. These regulators, however, are current limited to 26 amperes.

8.3.2 ERTS-B PULSE WIDTH MODULATED REGULATORS

General Description

Program: ERTS-B
Vendor:
Part Number:

Performance Characteristics

Input Voltage:	38 volts max.
Output Voltage:	-24.5 Vdc $\pm$0.5 Vdc
Output Current:	22 amps (Max) for power control module regulators 26 amps (Max) for payload regulator module regulators
Watts (Max):	550 watts for power control regulators 650 watts for payload regulators

Physical Characteristics

Size:
Weight:

References

ERTS Reference Manual, GE Space Division, Valley Forge Space Center, P. O. Box 8555, Philadelphia, Penn.

Design Status

This component is scheduled to be flown as part of the planned late 1973 ERTS-B mission.

8.3.4-1 OSO-I VOLTAGE REGULATOR

The voltage regulator provides a regulated 28.0 Vdc $\pm 2\%$ and an ON/OFF control for its associated experiment. It also provides experiment overload control with a selected output current limit level. The voltage regulator includes one command selectable output and one fixed output, with two regulator circuits (A and B) each selectable by ground command. Regulator telemetry outputs include regulated bus voltage monitoring as well as internal regulator circuit (A or B) select status.

8.3.4-1 OSO-I VOLTAGE REGULATOR

General Description

Program: OSO-I
Vendor:
Part Number: 3280235-100

Performance Characteristics

Input Voltage: 31.0 to 35.0 Vdc
Output Voltage: 28 Vdc $\pm 2\%$
Output Current: See attached Figure 8.3.4-1
Output Ripple: Included in 28 Vdc $^{+2\%}_{-0\%}$

Physical Characteristics

Size:
Weight: 0.6 kg (1.3 lbs)
Cooling Method: Radiation to surroundings and conduction to mounting surface

References

Hughes Aircraft Company Development Spec., Experiment Regulator DS-31331-129, November 16, 1972

Hughes Aircraft Company Subsystem Spec., Electrical Power Subsystem SS 31331-120 Rev A., November 16, L972

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

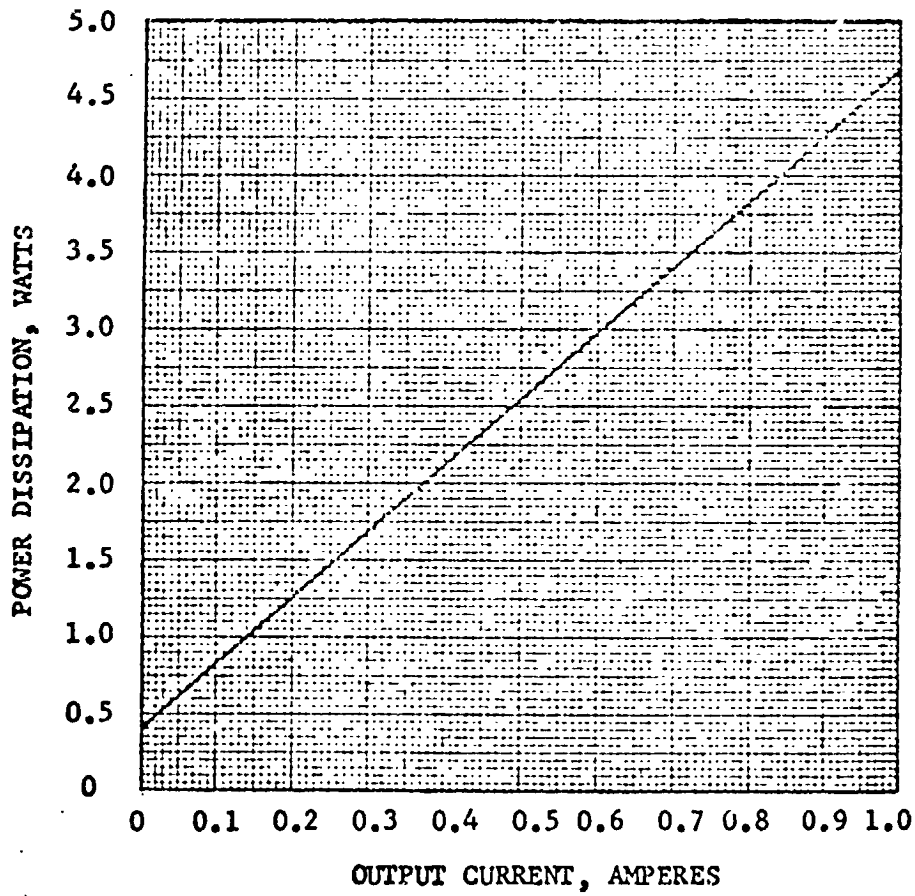


Figure 8.3.4-1 EXPERIMENT REGULATOR MAXIMUM POWER DISSIPATION

8.3.4-2 OSO-I VOLTAGE LIMITER

At the completion of battery charging, excess solar panel power is dissipated in eight bus voltage limiters. To minimize the effect on thermal control, these are adjusted in groups of 4 to turn on at two different bus voltage values. Each limiter assembly is capable of dissipating a minimum of 66 watts. The turn-on or set-point value range is 32.6 volts (full off) to 33.0 volts (full on). The voltage limiter provides temperature sensing and other data to telemetry. In addition it can be controlled through ground commands.

8.3.4-2 OSO-I VOLTAGE LIMITER

General Description

Program: OSO-I
Vendor:
Part Number: 3280238-100

Performance Characteristics

Input Voltage: 23.0 to 33.0 Vdc
Output Voltage: 32.6 to 33.0 Vdc clamp level
Output Current: 2.0 amps nominal
Watts (Average Load): 66.0 w @ 33 Vdc

Physical Characteristics

Size:
Weight: 0.41 kg (0.916 lbs) maximum
Cooling Method: Radiation & conduction to mounting surface

References

Hughes Aircraft Company Development Spec. Voltage Limiter, DS-31331-127, 16 Dec 1972

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

8.3.6 OAO-C REGULATOR CONVERTER

Figure 8.3.6-1 is a block diagram of the OAO regulator-converter. The power input to the regulator-converter is derived directly from the unregulated bus, the voltage of which may vary from 25 to 32 volts during normal operation. Two multivibrator type power oscillators (master and slave) convert the unregulated DC input to square wave pulses at 2KC. The combined outputs of the master and slave oscillators are added in an auto-transformer connection in the slave oscillator and a secondary connection in the master oscillator. The resulting output is full wave rectified and filtered. Separate outputs are provided at +28, -28, +10, -10 and +18 volts dc.

Voltage regulation is accomplished by varying the phase angle between the master and slave oscillators. With high output loads and low input voltage, the oscillators are in phase (minimum phase shift). With the input-output conditions reversed, the phase shift approaches a maximum (180° out of phase). The phase angle between the two oscillators is controlled by a magnetic amplifier. The output voltage of the regulator-converter is sensed through a precision resistor divider network, compared to a zener network reference voltage and applied to the control winding of the magnetic amplifier. The drive power for the magnetic amplifier is supplied by the master oscillator. The control current determines the reset time for the magnetic amplifier thereby controlling the firing angle between the master oscillator and magnetic amplifier. The magnetic amplifier output is used as the base drive for the power transistors in the slave oscillator. Oscillator frequency control is accomplished through the use of a saturable reactor in the feedback circuit of the master oscillator.

The voltage level for each of the regulator-converter output lines is adjusted, within preset voltage tolerance limits, to allow for average voltage drops in the OAO distribution network. Component redundancy is provided in each of the critical circuits. Each power oscillator contains two redundant transistors for each oscillator side. As with redundant diodes, each power transistor is fused so that if a transistor fails, the parallel transistor circuit will assume the entire switching function without performance degradation.

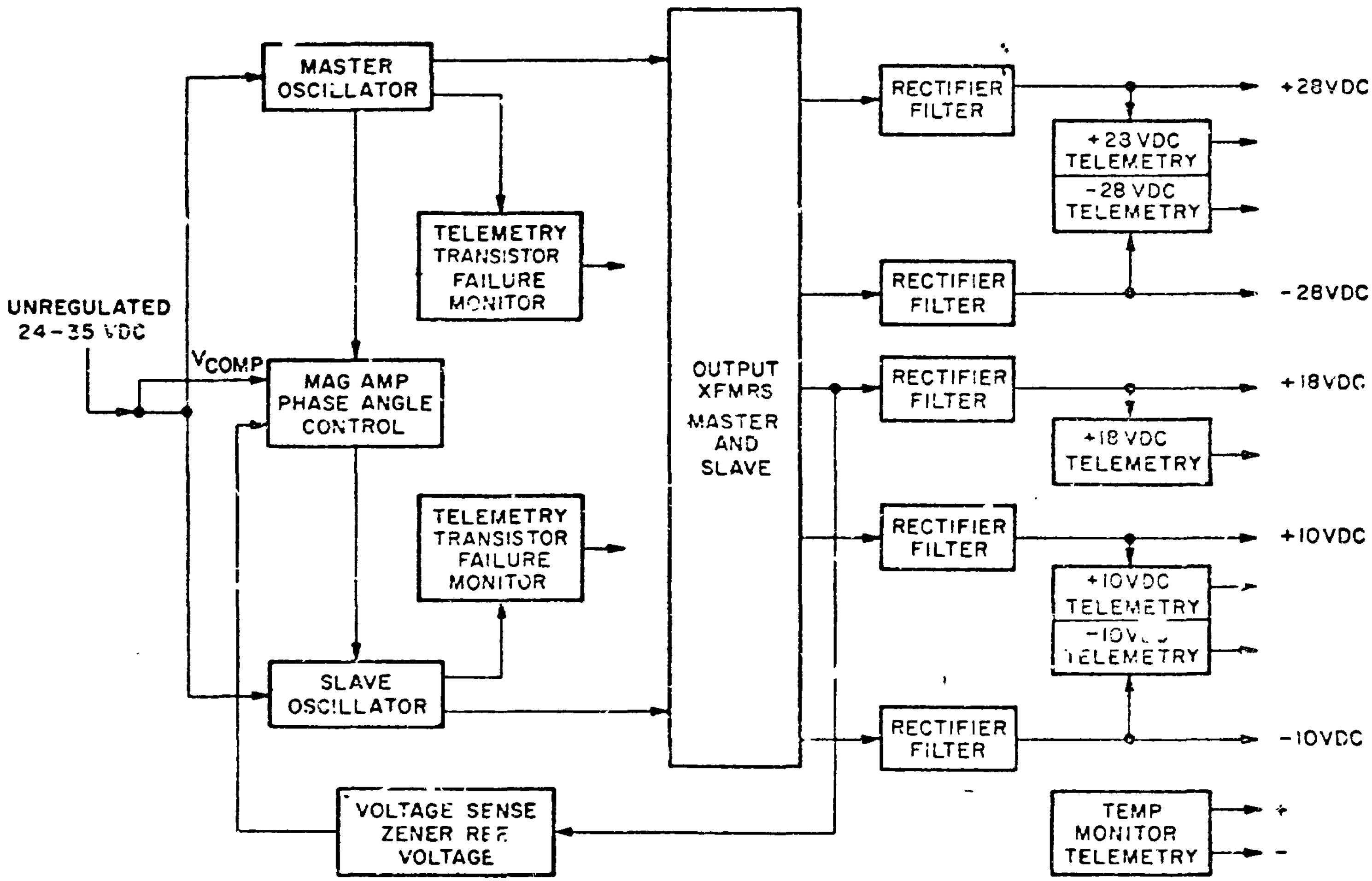


Figure 8.3.6-1 OAO-ᴗ REGULATOR-CONVERTER

8.3.6 OAO-C REGULATOR-CONVERTER

General Description

Program. OAO-C
Vendor:
Part Number:

Performance Characteristics

Input Characteristics

Voltage: 25-32 volts
Current: 10 to 15 Amps with nominal loads
Power: 360 watts nominal

Output Characteristics

Voltages:	+28 ± 2%,	-28 ± 2%,	+10 ± 1%,	-10 ± 1%,	+18 ± 1%
Power (Ave):	135.7w,	11.5w,	48.7w,	7.6w,	81w

Physical Characteristics

Size:
Weight:

References

OAO Functional Operations Manual, Power Subsystems, Goddard Space Flight Center, Greenbelt, MD. Document No. FO-G-0127-C, dated August 197[illegible].

8.3.7 SKYLAB AM/MDA VOLTAGE REGULATOR

The voltage regulator is a buck type with a manually adjustable open circuit voltage range of 26.0 to 30.0 V. Individual Power Conditioning Group (PCG) trimming adjustments are provided to equalize load sharing among the eight PCGs. A master (common) adjustment allows load sharing between the AM and ATM power systems in the paralleled mode. The regulator voltage drop is 40 mv/amp. The maximum output is 50 amperes.

8.3.7 SKYLAB AM/MDA VOLTAGE REGULATOR

General Description

Program: Skylab AM/MDA
Vendor: Engineering Magnetics, Inc.
Part Number: 514800-13

Performance Characteristics

Input Voltage: 33 to 125 Vdc
Output Voltage: Adjustable from 24 to 30 Vdc ±0.1 Vdc
Output Current: 0 to 50 Amps
Output Ripple: 0.3 Volts (p-p) at 35 to 125 Vdc Input
Efficiency: 92%
Watts (average load): 1600 Watts
Voltage Drop: 40 MV/amp

Physical Characteristics

Size: 27.7 cm (10.85 in) by 25.4 cm (10 in) by 11 cm (4.3 in)
Weight: 6.4 kg (14 lbs)
Cooling Method: Mount on Active Coolant System Coldplate

References

McDonnell Astronautics Co., Procurement Specification for Engineering Magnetics, D.C. Voltage Regulator, 61B769005, September 2, 1971.

Design Status

This component was flown as part of the 1973 Skylab mission.

8.3.8 SKYLAB ATM VOLTAGE REGULATOR

The voltage regulator employs a dump-store method for regulation (Figure 8.3.8-1). Energy is stored in inductor L_1 when Q_1 is switched on. This connects the battery or the solar panel output to the inductor. When Q_1 is shut off, L_1 dumps its stored energy into the load. This circuit allows the power source to be isolated from the buses in such a way that a single-point failure in the regulator cannot result in a high voltage on the buses. If a failure occurs so that Q_1 is either continually on or continually off the regulator output voltage will drop to zero.

The load regulator is designed to regulate power from either the battery or the solar panel. When the solar panel voltage is greater than the battery voltage, all of the input power to the load regulator is received from the solar panel. The input power to the regulator is supplied by the battery during the eclipse portion of the orbit and by the solar panel during the sunlight portion of the orbit. Hence, the input voltage varies from 80 to 25.5 volts. The load regulator maintains the output voltage at 30.0 plus or minus 0.9 volts direct current (Vdc).

The load regulator efficiency is 89 percent. The efficiency of the regulator is maintained relatively constant over its load range. The output current of the load regulator under peak loads is 15.5 amperes. The output voltage is 29.0 Vdc minimum under this condition and 31.0 Vdc maximum under no load conditions. The output voltage is linear between 0 and 15.5 ampere loads. Short circuit protection is provided by limiting regulator output current to 20.0 amperes maximum. The output voltage will start to drop off above 15.5 amperes and decay to zero at 20 amperes.

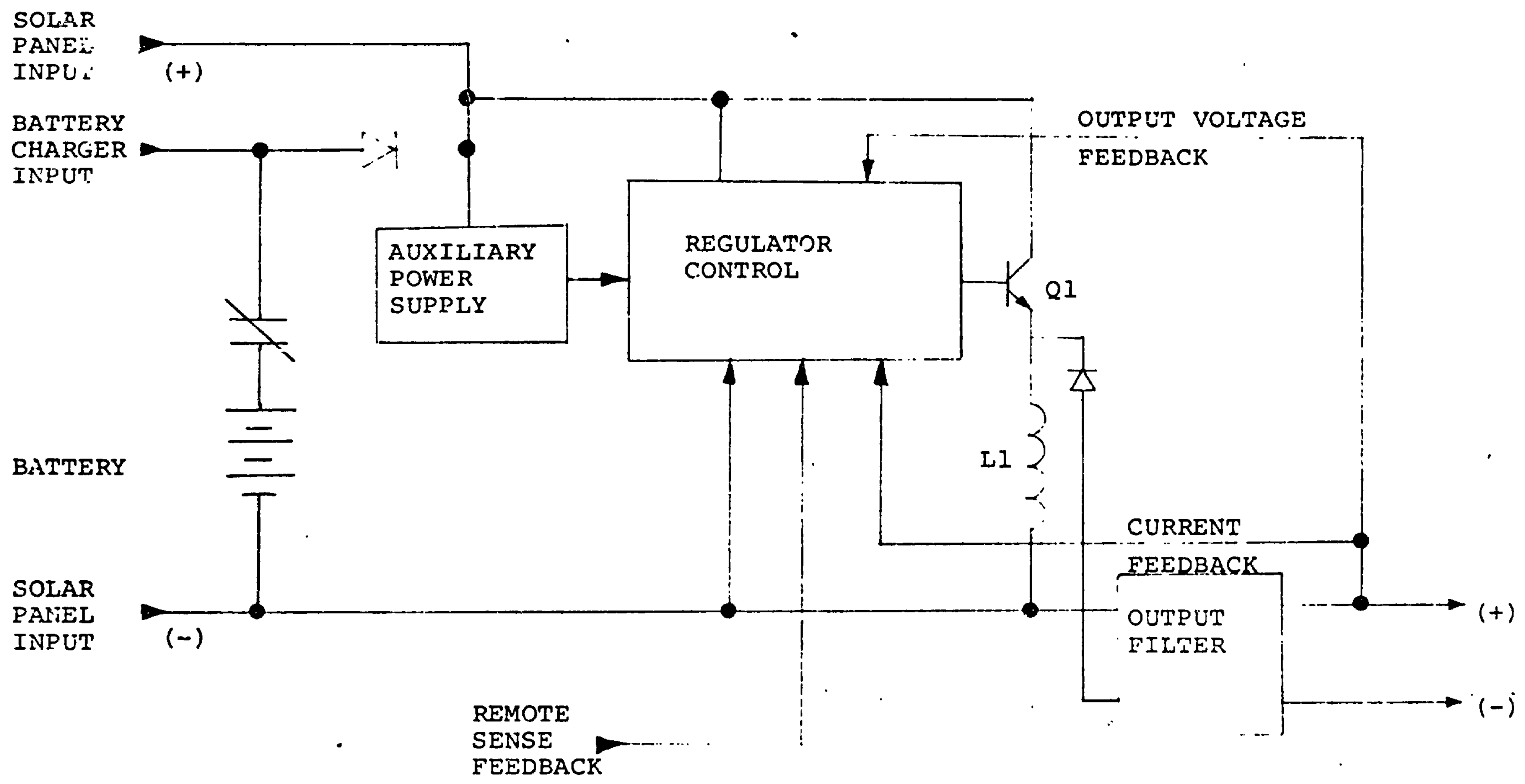


Figure 8.3.8-1. Skylab ATM Charger Regulator Battery Module (CBRM) Block Diagram

8.3.8 SKYLAB ATM VOLTAGE REGULATOR

General Description

Program: Skylab ATM
Vendor: MSFC
Part Number: 40M26203

Performance Characteristics

Input Voltage: 25.5 Vdc to 80 Vdc
Output Voltage: 29.1 Vdc to 30.9 Vdc
Output Current Limit: 20 Amps. Max
Output Ripple: 0.10 Volts peak to peak
Efficiency: 89% at peak load
Watts (average load): 235 watts
Watts (peak transient load): 415 watts

Physical Characteristics

Size:
Weight:
Cooling Method: Passive

References

MSFC Development Report 40M26995, July 1972, Martin-Marietta Electrical Power System Definition Document, ED-2002-1045-1, November 1970.

Notes

This unit is an integral part of the total CBRM (Charger Battery-Regulator-Module)

Design Status

This component was flown as part of the 1973 Skylab mission.

8.3.11-1 MARINER MARS 71 BOOSTER REGULATOR, MAIN AND STANDBY

The booster regulator (BR) modules transform the unregulated power of the dc power bus to regulated power at 56 Vdc. The standby unit is electrically identical to the main unit, and is switched on-line automatically in the event of a main power chain failure. The BR block diagram is shown on Figure 8.3.11-1.

The input to the BR assemblies is 25 to 50 Vdc. The BR output has the following characteristics.

a. Output voltage: Regulated 56 Vdc, ±1 percent

b. Maximum power: 295 watts

c. Minimum power: 50 watts

d. Conversion efficiency: The minimum BR conversion efficiency at an output of 295 watts is 90 percent with a 50 Vdc input, and 85 percent with a 23.5 Vdc input.

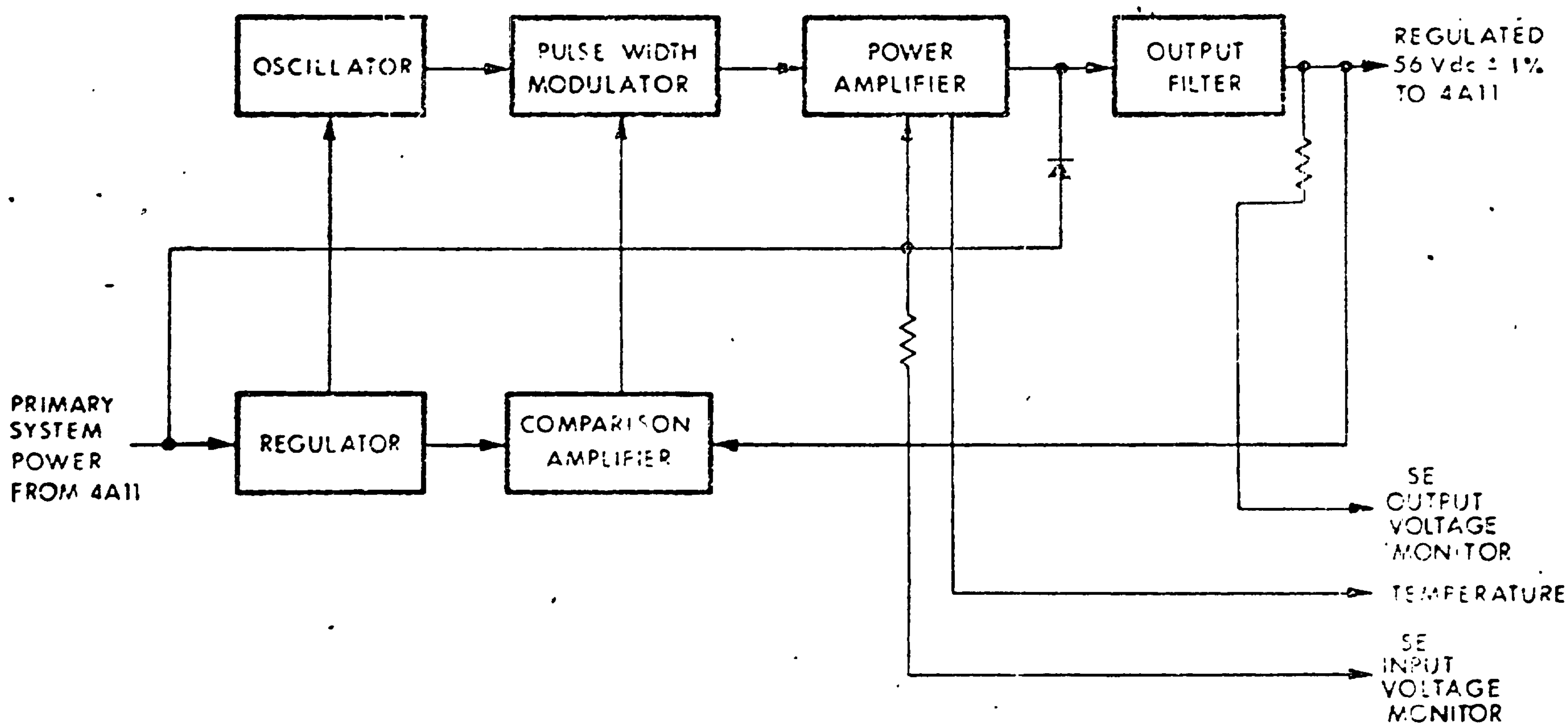


Figure 8.3.11-1 MAIN/STANDBY BOOSTER REGULATOR BLOCK DIAGRAM

8.3.11-1 MARINER MARS 71 VOLTAGE BOOSTER REGULATOR, MAIN AND STANDBY

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 25 to 50 Vdc
Output Voltage: 56 Vdc $\pm 1\%$
Output Ripple: 400 Millivolts peak-to-peak maximum
Efficiency: 85 to 90% minimum
Maximum Power: 295 watts
Minimum Power: 50 watts

Physical Characteristics

Size:
Weight: 2.8 kg (6.2 lbs)

References

Mariner Mars 71 Flight Equipment Power Subsystem, JPL M71-2004-1, dated 6 Nov 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.3.11-2 MARINER MARS 71 30 VDC REGULATOR

The dc regulator (DCR) module regulates the primary raw power voltage at 30 Vdc. This regulator is available to power the gimbal actuators and propulsion valve upon command. A block diagram is shown on Figure 8.3.11-2.

The input to the DCR is dc power at 25 to 50 Vdc. The DCR is capable of operating with a minimum input voltage of 23.5 Vdc. The input current ripple will not exceed 400 ma maximum nor the in-rush current exceed 6 amperes maximum. Approximately 3 watts of 2.4 kHz power is required.

The DCR is activated upon command from the Attitude Control System (ACS). The ACS command energizes a relay that applies input power to the DCR and connects the output to the gimbal actuators and valves.

DCR outputs have the following characteristics:

a. Nominal output voltage: Regulated 30 Vdc, ± 5 percent.

b. Nominal loads: 12 watts minimum, 90 watts maximum.

c. Absolute load limits: 6 watts minimum, 150 watts maximum. Maximum load duration not to exceed one minute.

d. Transient characteristics: $30 {}^{+12}_{-15}$ Vdc with load transients between 6 and 150 watts. Recovery to 30 Vdc ± 5 percent within 150 millisecond after termination of the load transient.

e. Maximum ripple: 600 mv p-p at nominal max load.

f. Maximum spikes: 3 volts peak at nominal max load.

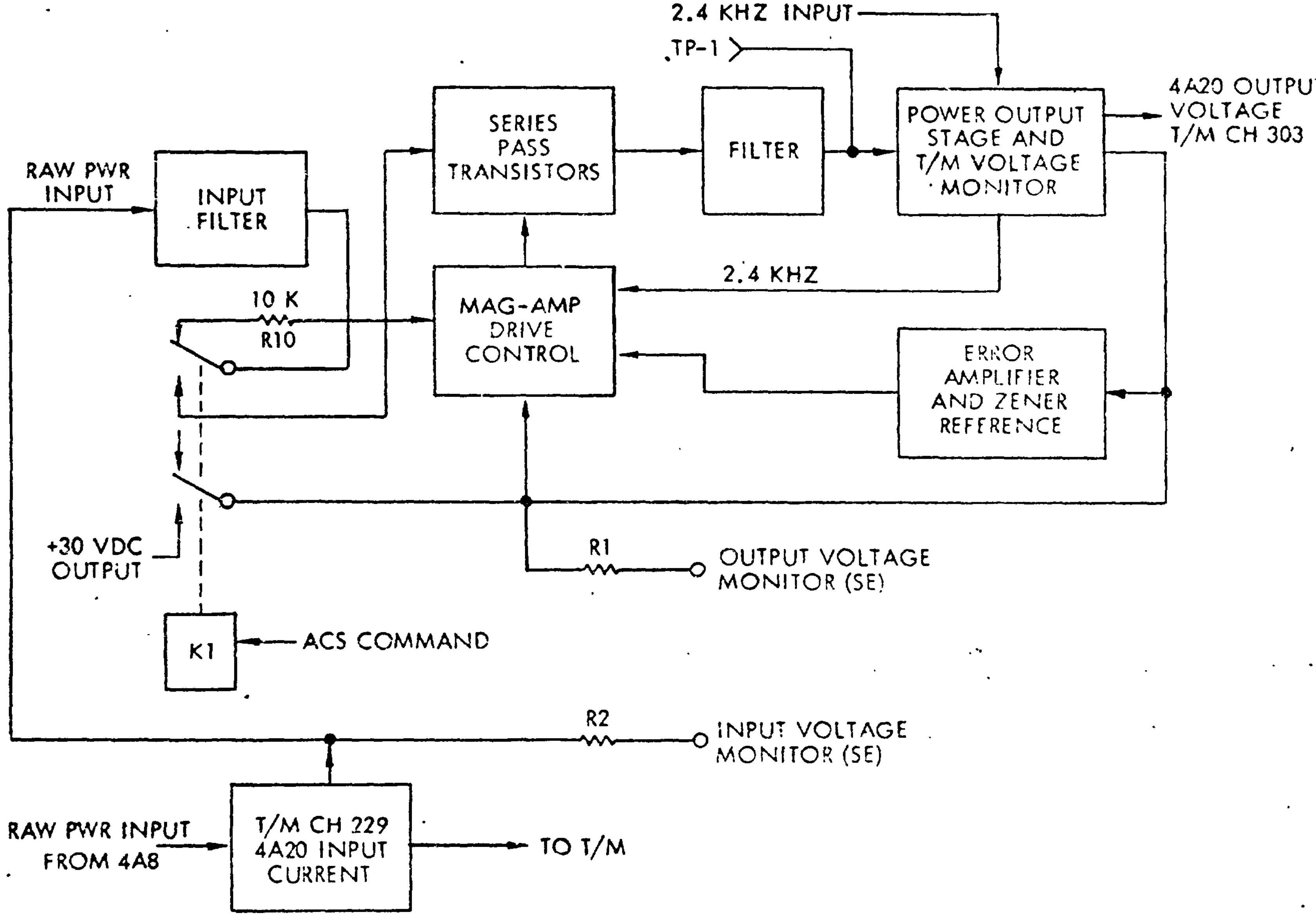


Figure 8.3.11-2 +30 Vdc REGULATOR BLOCK DIAGRAM

8.3.11-2 MARINER MARS 71 30 VDC VOLTAGE REGULATOR

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 23.5 to 50 Vdc and 50 Vrms at 2.4 kHz
Output Voltage: 30 Vdc ±5%
Spikes: 3 volts peak at nominal maximum load
Output Ripple: 600 mv p-p at nominal maximum load
Load Limits: 6 watts minimum, 150 watts maximum. Maximum load not to exceed one minute.

Physical Characteristics

Size:
Weight: 2.7 kg (6 lbs)

References

Mariner Mars 71 Flight Equipment Power Subsystem, JPL, M71-2004-1, dated 6 Nov., 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.4.6 OAO-C VOLTAGE INVERTER

The voltage inverter (Figure 8.4.6-1) accepts the unregulated dc voltage generated by the solar cell array-battery supply and provides regulated 400 Hz ($\pm10\%$) ac voltage to the utilization equipment. Output voltages are:

Nominal (Volts)	Voltage Regulation (%)
26 (1Ø)	±1
26 (2Ø)	±5
26 (3Ø)	±5

The voltage inverter operates at efficiencies of 62% minimum for the LOW load range, 70% minimum for the AVERAGE load range and 68% minimum for the MAXIMUM load range.

A completely redundant (standby) inverter is provided in the same package with the main unit. Both units are identical in every aspect. Transfer from the operating (main) inverter to the standby unit is accomplished by a motor driven switch.

The voltage inverter is designed for 98 percent probability of operation, within the specified performance requirements, from fabrication to launch and during the launch and one year orbital lifetime. All components and component parts are selected for an operating life of 12,000 hours, which shall be representative of the time accumulated during operation in orbit for one year.

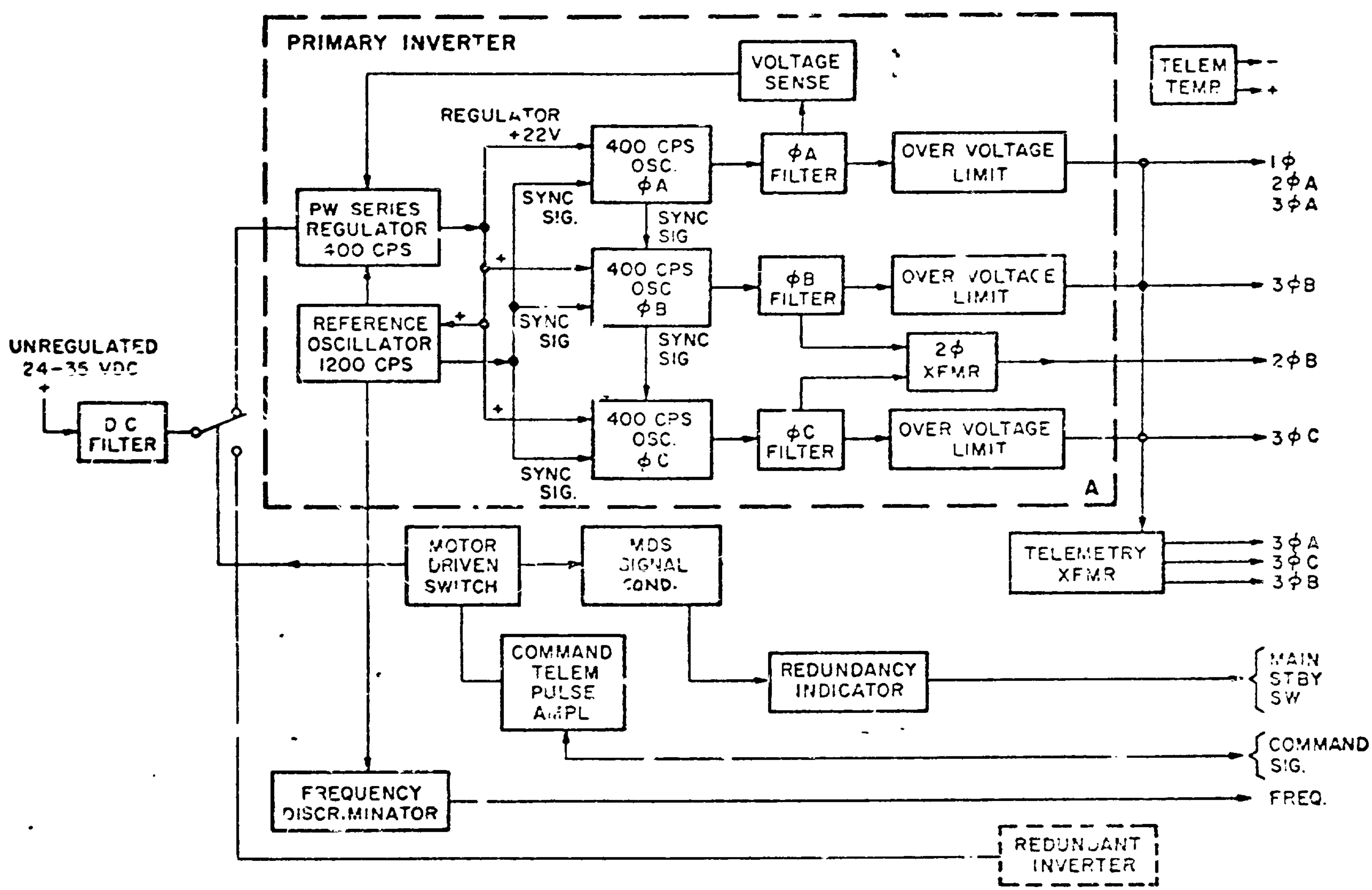


Figure 8.4.6-1 INVERTER BLOCK DIAGRAM

8.4.6 OAO-C VOLTAGE INVERTER

General Description

Program: OAO-C
Vendor:
Part Number:

Performance Characteristics

Voltage Input: 23 to 35 Vdc
Voltage Output: Phase 1 26 Vac ±1% regulation
Phase 2 26 Vac ±5% regulation
Phase 3 26 Vac ±5% regulation
Maximum Load Current: 6.75 amps
Output Frequency: 400 Hz ±1 % (3 phase delta)
Power Factor: 90 lagging
Voltage Transient: Recovery time to ±5% of regulation band less than 300 milliseconds
Frequency Transient: Recovery time to ±5% of regulation band less than 300 milliseconds
Voltage Transient Excursion: No greater than 55 volts peak high. Not less than 30 volts peak low
Operating Life: 12,000 hours

Physical Characteristics

Size:
Weight: 16.7 kg (37 lbs)
Cooling Method: Passive system - heat sink within the inverter & radiation mode to external environment. Maximum internal temperature variations not greater than 20°F.

References

OAO Voltage Inverter Power Supply Subsystem Specification No. AV-252CS-21D, Grumman Aircraft Corporation, Bethpage, L.I. 1 Dec. 1969

Design Status

This component was flown as part of the 1972 OAO-C mission.

8.4.9 APOLLO-17 CSM INVERTER

The inverter is composed of the major elements shown in Figure 8.4.9-1. The inverter normally receives a 6.4 kHz square wave synchronizing signal from the central timing equipment (CTE) to maintain the inverter output at 400 Hz. If this external signal is completely lost, the free running oscillator within the inverter will provide pulses that will maintain inverter output within ±7 Hz. The internal oscillator is normally synchronized by the external pulse.

The eight-state digital countdown section is synchronized by the 6.4 kHz signal and produces eight 400 Hz square waves, each mutually displaced one pulse-time from the preceding and the following wave. One pulse-time is 156 microseconds and represents 22.5 electrical degrees. The eight square waves are applied to the eight-stage power inversion section for amplification. The amplified square waves, still mutually displaced 22.5 electrical degrees, are next applied to the harmonic neutralization transformer where they are transformed into a 3 phase, 400 Hz 115 volt signal.

The harmonic neutralization section consists of 31 transformer windings on one core. The manner in which these transformers are wound on a single core produces flux cancellation which eliminates all harmonics up to and including the fifteenth of the fundamental frequency. The 22.5 degree displacement of the square waves provides a means of electrically rotating the square wave excited primary windings around the 3 phase, wye connected secondary windings, thus producing the 3 phase 400 Hz sine wave output. This 115 volt signal is then applied to the ac output filter where the remaining higher harmonics are eliminated. The ac filter also produces a rectified signal whose amplitude is a function of the amplitude of ac output voltage. This signal is applied to the Zener diode reference bridge for voltage regulation. After filtering, the 3 phase 115 volt ac 400 Hz sine wave is applied to the ac buses through individual phase current-sensing transformers.

The current-sensing transformers produce a rectified signal whose amplitude is a direct function of inverter output current magnitude. This dc signal is applied to the Zener diode reference bridge to regulate inverter current output; it is also sent to an overcurrent sensing circuit.

Inputs to the Zener diode reference bridge vary as functions of the inverter voltage and current outputs. A variance in voltage output unbalances the bridge, providing an error signal to the buck-boost amplifier via the magnetic amplifier. The buck-boost amplifier voltage output controls voltage variations through its bias to the power inversion section. When inverter current output reaches 200 to 250 percent of rated current, the bridge unbalances and provides an error signal causing the buck-boost amplifier to operate in the same manner as during an overvoltage

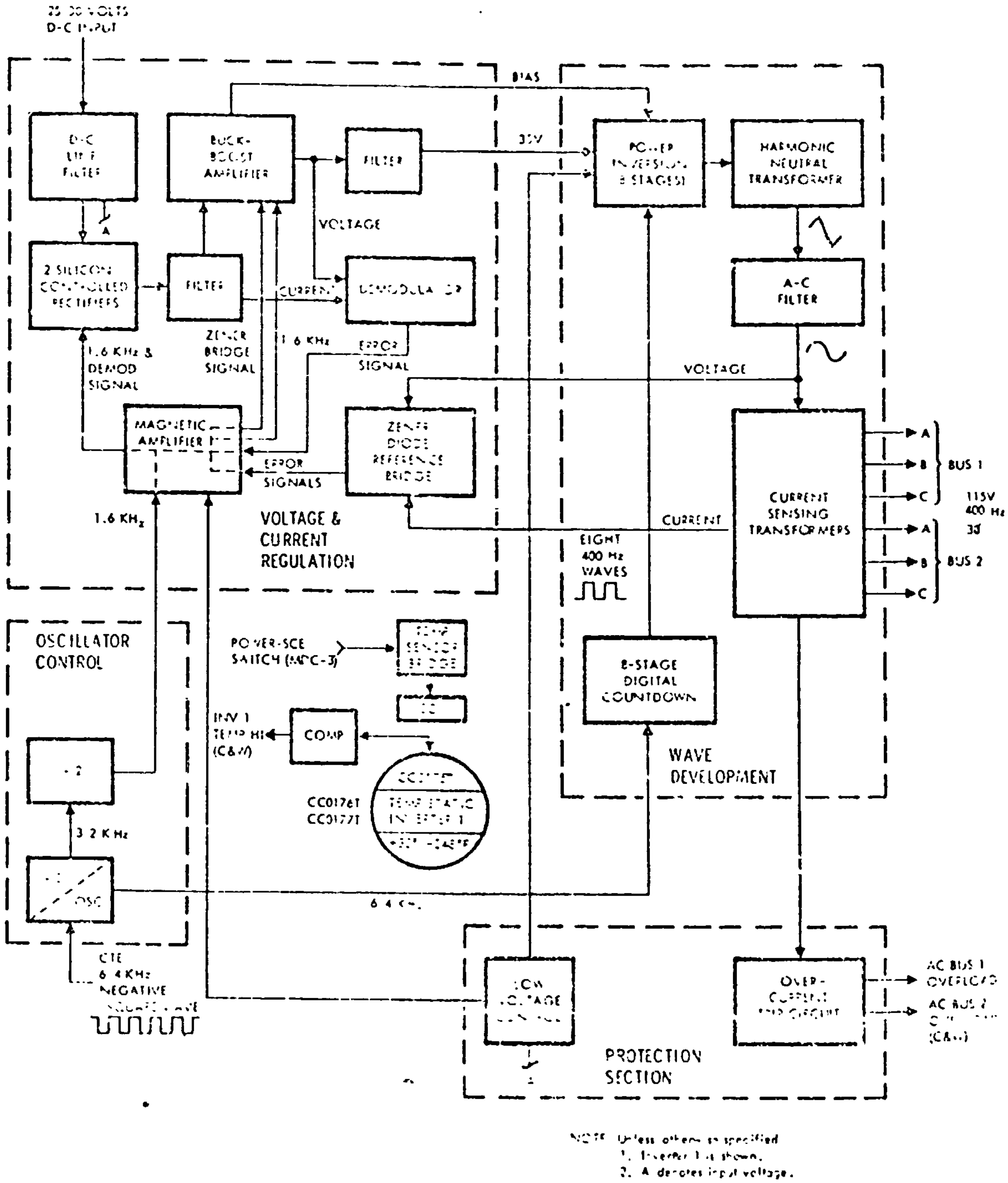


Figure 8.4.9-1 INVERTER BLOCK DIAGRAM

condition. When inverter current output exceeds 250 percent of rated current, the overcurrent sensing circuit is also activated.

The overcurrent sensing circuit activates warning lights when the output current exceeds limits. When total inverter current output exceeds 250 percent of rated current, this circuit will illuminate an overload lamp in 15 $\pm$ 5 seconds. If current output of any single phase exceeds 300 percent of rated current, this circuit will illuminate the overload lamp in 5 $\pm$ 1 seconds.

The 6.4 kHz square wave from the CTE is also applied to the internal oscillator. The oscillator has two divider circuits which provide a 1600 Hz signal to the magnetic amplifier.

The silicon-controlled rectifiers are alternately set by the 1600 Hz signal from the magnetic amplifier to produce a dc square wave with an on-time of greater than 90 degrees from each rectifier. This is filtered and supplied to the buck-boost amplifier where it is transformer-coupled with the amplified 1600 Hz output of the magnetic amplifier to develop a filtered 35 volts dc which is used for amplification in the power inversion stages. The buck-boost amplifier also provides a variable bias voltage to the eight-stage power inversion section to regulate inverter voltage and maintain current output within tolerance.

The demodulator circuit compensates for low-frequency ripple (10 to 1000 Hz) in the dc input to the inverter. The high-frequency ripple is attenuated by the input filters. The demodulator senses the 35 volt dc output of the buck-boost amplifier and the current input to the buck-boost amplifier. A drop or increase in the voltage input to the inverter will be reflected in a drop or increase in the 35 volt dc output of the buck-boost amplifier, as well as a drop or increase in current input to the buck-boost amplifier. A sensed decrease in the buck-boost amplifier voltage output is compensated for by a demodulator output, coupled through the magnetic amplifier to the silicon-controlled rectifiers. The demodulator output causes the SCRs to conduct for a longer time, thus increasing their filtered dc output. A sensed increase in buck-boost amplifier voltage output, caused by an increase in dc input to the inverter, is compensated for by a demodulator output coupled through the magnetic amplifier to the silicon-controlled rectifiers causing them to conduct for shorter periods, thus producing a lower filtered dc output to the buck-boost amplifier. In this manner, the 35 volt dc input to the power inversion section is maintained at a relatively constant level irrespective of the fluctuations in dc input voltage to the inverter.

The low-voltage control circuit samples the input voltage to the inverter and can terminate inverter operation. Since the buck-boost amplifier provides a boost action during a decrease in input voltage to the inverter, in an attempt to maintain a constant 35 volts dc to the power inversion section and a regulated 115 volt inverter output, the high boost required during a low voltage input would tend to overheat the solid state buck-boost amplifier. As a precautionary measure, the low-volta e control will terminate inverter operation by disconnecti..g operating voltage to the magnetic amplifier and the first power inversion stage when input voltage decreases to between 16 and 19 volts dc.

8.4.9 APOLLO-17 CSM INVERTER

General Description

Program: Apollo-17 CSM
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 25-30 volts dc
Output Voltage: 115.5 (+1, -1.5) VAC 3 phase
Outpu Frequency: 400 ± 3 Hz
Rating: 1250 VA

Physical Characteristics

Size:
Weight:

References

Apollo Operations Handbook SM2A-03-Block II- 1 dated 16 July 1969.

Design Status

This component was flown as part of the 1972 Apollo-17 mission.

8.4.10 APOLLO-17 LUNAR MODULE VOLTAGE INVERTER

Two inverters are used in the electrical power subsystem. These inverters (Figure 8.4.10-1) receive their power from the dc busses and transform the dc power into ac power to supply two ac busses. The inverters output is 115 volts ac, 400 Hz. Each inverter is capable of delivering 350 volt-amperes steady state. Both inverters can be powered and running at the same time, but ac power can be used from only one inverter at a time.

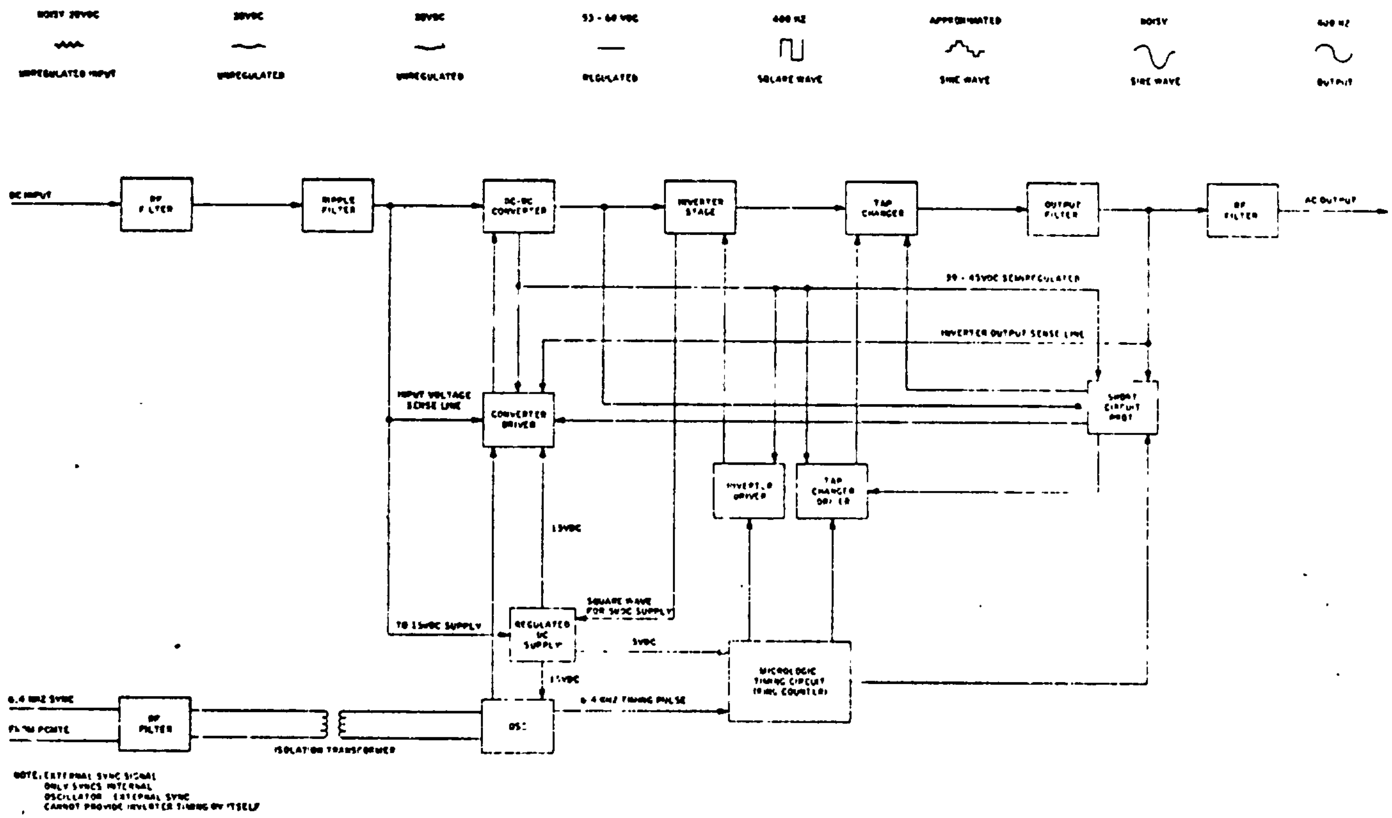


Figure 8.4.10-1 LM INVERTER

8.4.10 APOLLO-17 LUNAR MODULE VOLTAGE INVERTER

General Description

Program: Apollo-17 Lunar Module
Vendor:
Part Number:

Performance Characteristics

Output Voltage: 115 Vac ±2 Vac
Steady State Output: 350 VA
Peak Output (peak transient load): 525 VA (10 sec)

Physical Characteristics

Size:
Weight:

References

Electrical Power System, Lunar Module Study Guide LM-5 & Subs. Contract NAS9-1100 Exhibit E; Paragraph 3.7.4 Document No. LSG 770-154-4-LM-5 and Subs., dated May 1969.

Design Status

This component was flown as part of the 1972 Apollo-17 mission.

8.4.11-1 MARINER MARS 71 VOLTAGE INVERTER, 400 HERTZ

The voltage inverter (Figure 8.4.11-1) converts 2400 Hz square wave power to single phase 400 Hz power and to three phase 400 Hz power, utilizing redundant inputs of 55 Vdc. The single phase output is a 28 volt (rms) square wave. The three phase output is a 27 volt (rms) line to line, quasi-square wave.

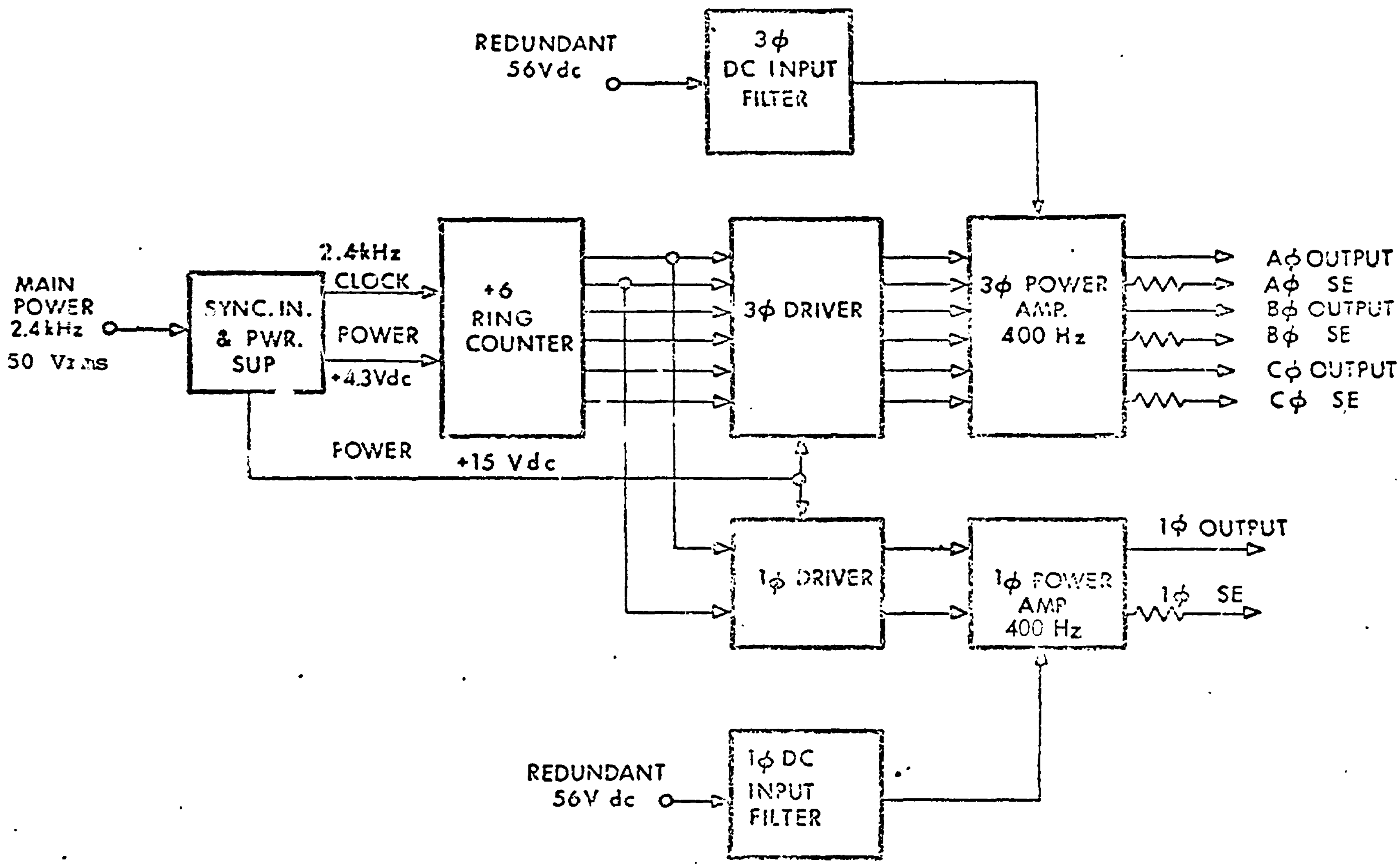


Figure 8.4.11-1 MARINER MARS 71 SINGLE PHASE AND THREE PHASE INVERTERS

8.4.11-1 MARINER MARS 71 VOLTAGE INVERTER, 400 Hz

General Description

Program: Mariner Mar 71
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 55.2 Vdc and 50 VRMS, 2.4 KHz
Output Voltage: 28 VRMS $\pm$6%, 400 Hz $\pm$0.01% single phase
27.2 VRMS $\pm$5%, 400 Hz $\pm$0.01% three phase
Spikes: 1.0 volts maximum, leading edge
Efficiency: 83% with 12 watts out (single phase)
85% with 12 watts out (three phase)
Watts (Average Load): 12 watts (single phase)
12 watts (three phase)
Watts (Peak Load): 15 watts for 60 seconds (single phase)
15 watts for 60 seconds (three phase)
Power Factor: 0.8 lagging or greater (single phase)
0.5 lagging or greater (three phase)

Physical Characteristics

Size:
Weight: 2.3 kg (5 lbs)

References

Mariner Mars 71 Flight Equipment Power Subsystem JPL M71-2004-1, dated 6 Nov 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.4.11-2 MARINER MARS 71 VOLTAGE INVERTER, 2400 HERTZ

The voltage inverter (Figure 8.4.11-2) converts regulated 56 Vdc to 50 volts (rms) 2400 Hz square wave power for distribution as prime power in the Mariner Mars 71 spacecraft. Frequency is maintained within ± 0.01 percent by an internal crystal oscillator so as to provide accurate timing signals to the Central Computer and Sequencer subsystem. The square wave output and its return are isolated from the chassis ground with 47.5 k ohm resistors.

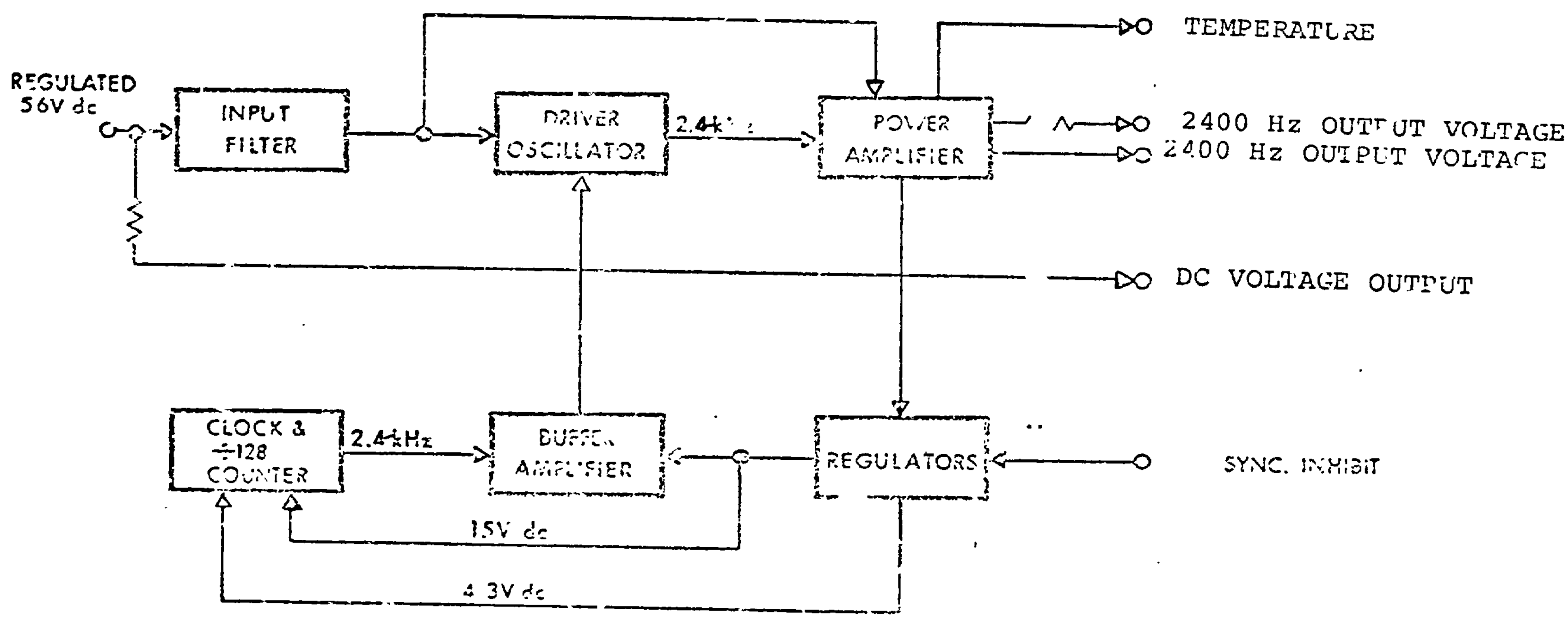


Figure 8.4.11-2 VOLTAGE INVERTER, 2400 HERTZ

8.4.11-2 MARINER MARS 71 VOLTAGE INVERTER, 2400 HERTZ

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 56 Vdc ±1%
Output Voltage: 50 Vrms +3%, -4% 2.4 kHz square wave
Spikes: 5 volts maximum with 5 μs maximum duration
Efficiency: 89% minimum at 150 watts output
Watts: 95 minimum, 250 maximum
Power Factor: 0.95 lagging or greater

Physical Characteristics

Size:
Weight: 1.8 kg (4 lbs)

References

Mariner Mars 71 Flight Equipment Power Subsystem, JPL M71-2004-1, dated 6 Nov 1970

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

8.5.2 ERTS B BATTERY CHARGE CONTROLLER

The battery charge controller (Figure 8.5.2-1) protects the battery from over-voltage and over-temperature conditions while it is being charged by the solar array. The control circuitry compares battery current, temperature and voltage with built-in references. Error signals are amplified and used to adjust the battery charging current, decreasing the error. The charge controller operates in three modes; constant voltage charge, current limited charge, and trickle charge.

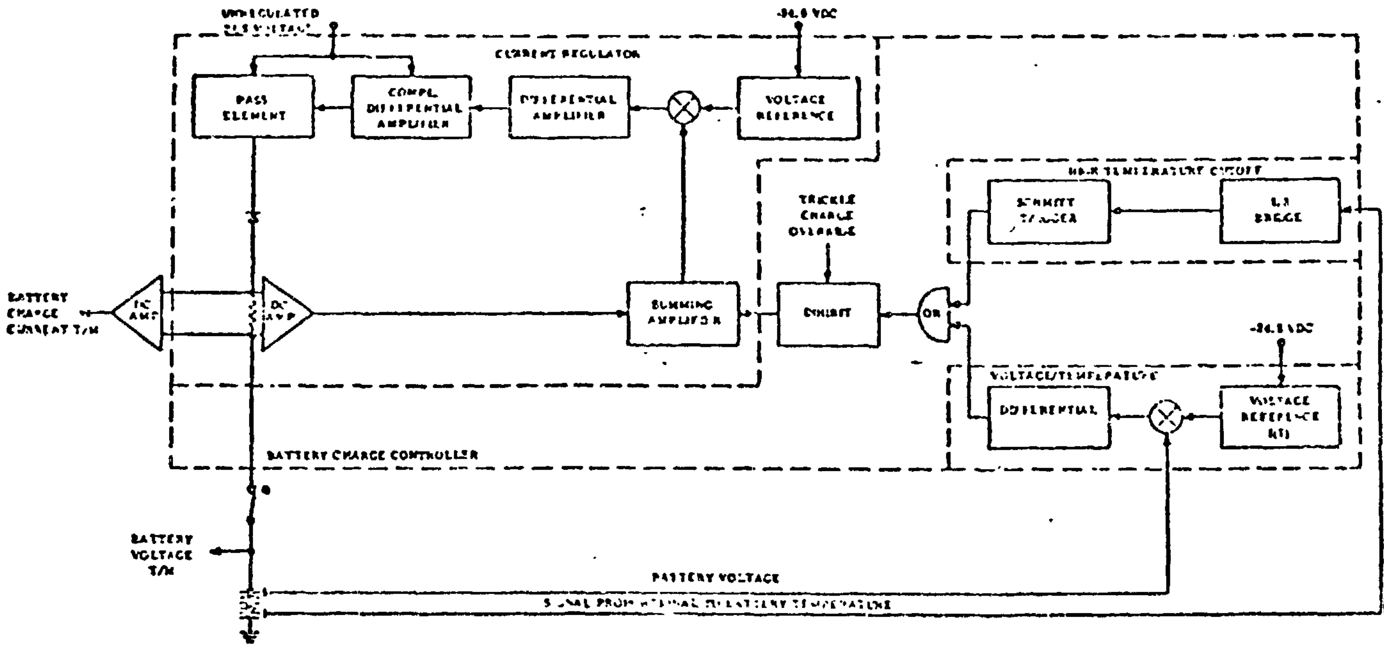


Figure 8.5.2-1 BATTERY CHARGE CONTROLLER
BLOCK DIAGRAM

8.5.4 OSO-I BATTERY CHARGE CONTROL

The battery charge control initiates charge in one of two charge modes to its associated battery following earth eclipse. When the battery temperature is below a selected control curve (temperature vs voltage curve), all the solar panel power available (in excess of the observatory loads) is utilized to charge the batteries at a maximum rate. When the battery voltage-temperature characteristic reaches the control curve, the battery terminal voltage is limited to a fixed level of 23.0 to 32.6 volts (depending on battery temperature). Battery current is permitted to decrease as the battery charge characteristic is satisfied.

Battery charge control design includes ground command capability and telemetry monitoring of critical operating parameters. A battery over-temperature switch provides complete charge cut-off. Battery charge is automatically re-enabled when the over-temperature switch opens.

8.5.4 OSO-I BATTERY CHARGE CONTROL

General Description

Program: OSO-I
Vendor:
Part Number: 3280231-100

Performance Characteristics

Input Voltage: 23.0 to 33.0 Vdc
Output Voltage: 23.0 to 32.6 Vdc (in the voltage limited mode)

Physical Characteristics

Size:
Weight: 1.4 kg (3.0 lbs) design goal maximum
Cooling Method: Radiation & conduction to mounting surface

References

Hughes Development Specification, Battery Charge Control DS-31331-125, 28 November 1972.

Hughes OSO-I Observatory Specification Development Specification, Battery Charge Control , SS31331-100, Rev. C, 27 September 1972.

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

8.5.6-1 OAO-C STATE OF CHARGE UNIT

The State of Charge Unit (SOCU) is designed for use as an auxiliary controller of the battery charge control subsystem. It is also designed for passive operation as a battery state-of-charge indicator. State of charge is obtained in ampere-hours by integrating the charge and discharge currents flowing through each of the three batteries. The current is monitored by measuring the voltage drops across shunts in the negative lead of each battery. State of charge can also be obtained by measurement of the potential of the adhydrode (third electrode) cell in each of the three batteries.

The SOCU contains three identical ampere-hour circuits and three circuits for measurement of adhydrode voltage plus associated command, telemetry, Power Control Unit (PCU) interface, and power conversion and regulation circuitry. A block diagram of the unit is shown in Figure 8.5.6-1. The three ampere-hour and the three adhydrode circuits are monitored continuously by telemetry and either group of circuits may be used for auxiliary control of the PCU.

When operated in one of the SOCU control modes, the first indication of full charge on a battery enables the PCU Battery Voltage Limit Switch (BVLS) relay coils, causing the PCU to transfer to the lower BVLS level. The PCU remains at that level until the end of the sunlight period.

The NITE signal resets the PCU relays to the highest BVLS level. If the NITE signal appears during a slew of the spacecraft when one or more batteries are at full charge and the highest level is enabled, the level is reduced as soon as the DAY signal reappears if operating in the adhydrode mode of control. When operating in the ampere-hour mode, one of the batteries must be fully recharged as indicated by all ONE's in an eight bit output register before the PCU is returned to the lower level.

Command circuitry controlled by one of the experiment command lines, fifteen bit time lines, and the command enable line are used for control of all internal SOCU functions.

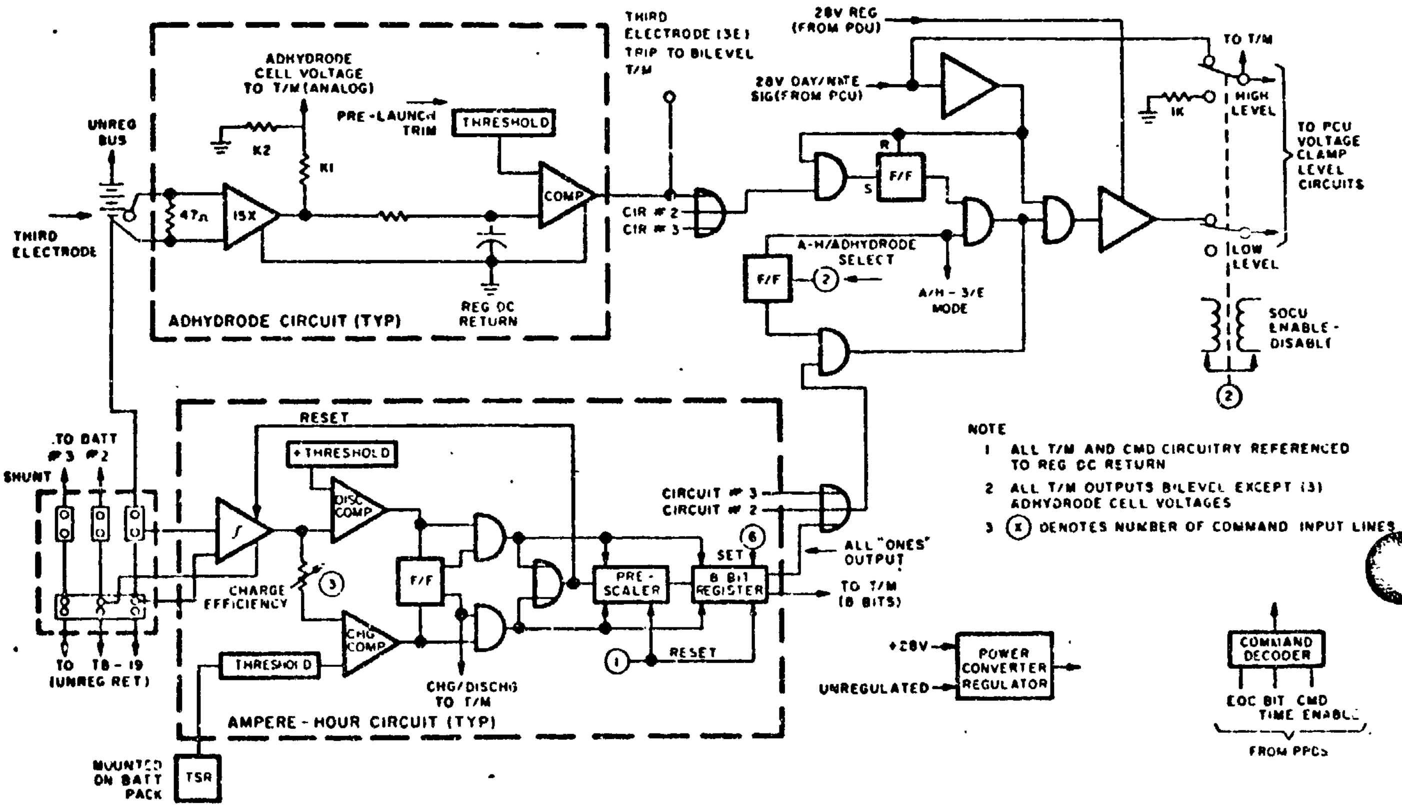


Figure 8.5.6-1 STATE OF CHARGE UNIT BLOCK DIAGRAM

8.5.6-2 OAO-C POWER CONTROL UNIT (PCU) AND POWER REGULATOR UNIT (PRU)

The operation of the PCU and PRU (Figure 8.5.6-2) can be more readily explained by considering them jointly, since their functions are interrelated. The State of Charge Unit (SOCU) is also used in conjunction with these two units to complete the battery charging function. This operation will be grouped into major functions as follows:

1. Solar array clamping circuit
2. Battery Thermostat
3. Single Battery Voltage - Temperature Detector
4. Regulator Control
5. Battery Control
6. Battery Voltage Limit Switch (BVLS) Logic
7. OBP Power Boost

Items two thru seven above are used for battery charge control.

1. Auxiliary Array Clamping Circuit

   The portion of the solar array referred to as the auxiliary array is connected directly to the unregulated bus to avoid the power losses in the PRU and diode box. Upon coming out of an eclipse (dark), the solar arrays are cold which results in high array voltage and power capability. In the event the spacecraft unregulated bus load demand is approximately equal to or less than the auxiliary array maximum power capability (P_{MP}), an excessive voltage condition (> 35 volts) could occur on the unregulated bus. To protect against this condition, a clamping circuit is incorporated.

2. Battery Thermostat Control Circuit

   The primary functions of the thermostat control circuit are the following:

   - Provide an output signal when a battery assembly reaches the limiting temperature of $90^{\circ} \pm 1^{\circ}F$. This signal will be used to switch the battery voltage limit switch (BVLS) to BVLS 2.

   - Provide an output signal when the battery assembly temperature drops below $85^{\circ}F \pm 1^{\circ}F$. This signal is used to switch to the nominal BVLS 4.

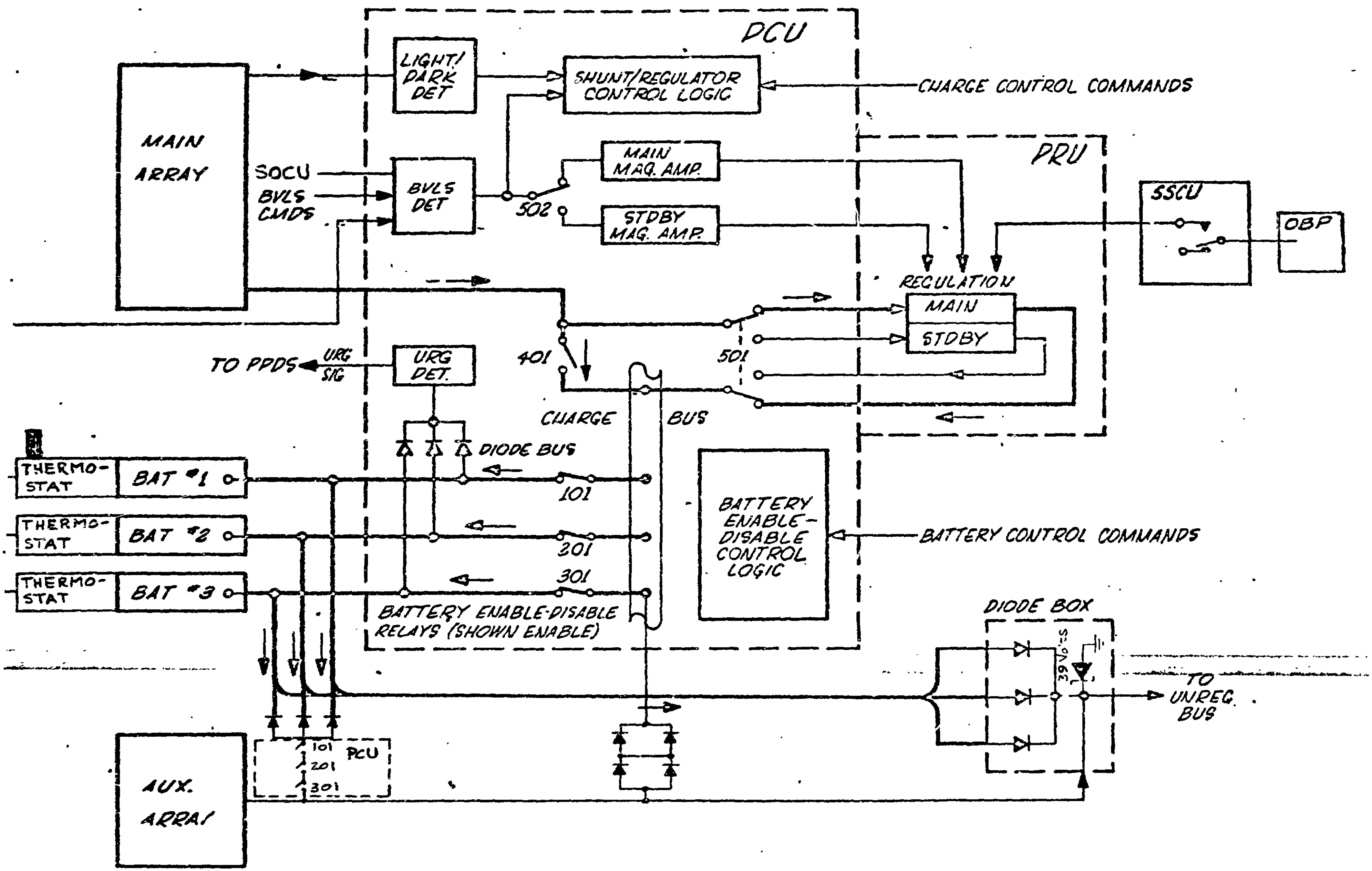


Figure 8.5.6-2 POWER CONTROL UNIT AND POWER REGULATOR UNIT

3. Battery Voltage - Temperature Detector

There are three such circuits one for each battery. The primary purpose of this circuitry is to provide a temperature compensated voltage limit control for battery charging.

4. Regulator Control

The primary functions of the regulator control are as follows:

- Cause battery charge voltage regulator turn-on when either of the following conditions exist:
  - Spacecraft command for either main or standby regulator ON
  - Any battery attains the pre-selected voltage-temperature limit (BVLS level)
- Cause regulator turn-off when either of the following exists:
  - Spacecraft shunt charge command
  - Entry into dark period
- Cause regulator transfer from main to standby:
  - In event of a main regulator failure
  - Spacecraft command for standby regulator ON
- Inhibit regulator ON, for duration of one light period by deliberately executing shunt charge command twice within 500 milliseconds.

5. Battery Control

There is no automatic control circuitry to place a battery ON or OFF the charge bus. There are six ground commands (enable and disable for each of three batteries) to perform these functions. The commands also enable/disable the associated battery detector circuitry.

6. Battery Voltage Limit Switch (BVLS)

The eight BVLS levels are utilized to limit and regulate the battery charging to a selected voltage temperature curve. The level selection is attained by switching resistors in the battery detector circuitry.

7. On Board Processor (OBP) Power Boost

The OAO is equipped with an additional charge regulation mode termed OBP Power Boost. In this mode, an output from the OBP is connected by an enabling switch to the auxiliary control windings in the magnetic amplifier of the power control unit. The OBP samples the PRU/PCU output current and varies the voltage on the auxiliary control windings so as to achieve maximum output current. However, during this mode the output to the batteries is limited to a maximum of 1000 watts. The OBP power boost mode is automatically terminated when the battery over-temperature, BVLS or the SOCU reduces the rate of charge.

8.5.7 SKYLAB AM/MDA BATTERY CHARGER

The battery charger consists of a peak power tracker, an amp-hr meter charge control, and a buck regulator, which track peak power voltage of the solar array group to provide maximum power transfer from array to battery. The control circuit protects the sealed nickel cadmium battery against excessive overcharge or discharge. The bypass circuit allows use of array power directly to AM/MDA loads via the load voltage regulator in case battery or battery charger fails.

Each of the eight AM/MDA batteries has its own battery charger. A battery charger recharges a battery by returning to the battery the same amount of amp-hrs removed from the battery, as determined by its amp-hr meter charge control circuit, plus an additional amount equivalent to battery internal losses.

In addition the battery charger also controls the transfer of power from the solar array via a voltage regulator to AM/MDA loads. If the solar array power is not sufficient to recharge the batteries and supply power to the AM/MDA loads the load requirements have priority.

8.5.7 SKYLAB AM/MDA BATTERY CHARGER

General Description

Program: Skylab AM
Vendor: Engineering Magnetics, Inc.
Part Number: 514823-15

Performance Characteristics

Input Voltage: 0 to 125 Vdc (max)
Input Current: 56.4 amps max
Output Current: 52.5 amps max
Output Voltage: 29 to 48 volts
Efficiency: 93% max

Physical Characteristics

Size: 28.9 cm (11.35 in) by 25.4 cm (10 in) by 18.6 cm (7.25 in)
Weight: 12.3 kg (27 lbs)
Cooling Method: Active coldplate mounting

References

McDonnell Astronautics Co., Procurement Specification for Engineering Magnetics, 61B769006 Battery Charger, September 1971.

Design Status

This component was flown as part of the 1973 Skylab mission.

8.5.8 SKYLAB ATM BATTERY CHARGER

The battery charger (Figure 8.5.8-1) is a non isolated step-down switching regulator type. The initial charging is performed at a constant current. When the battery terminal voltage reaches a predetermined value,which is a function of battery temperature, and is sensed by either of two redundant thermistors, the charge voltage drops by approximately 0.85 volt. Constant voltage charging is then initiated at the new voltage. As a result, the charging current drops sharply and then tapers off. The amount of battery overcharge (and hence, the battery thermal dissipation) is controlled by sensing the third electrode signals from three of the 24 cells in the battery package. Whenever the threshold level of 200 milli-volts (across a 200-ohm resistor) is reached by any one of the third electrodes in the three cells, charging is terminated. The battery voltage then drops to its open-circuit value for the remainder of the sunlight portion of the orbit.

The voltage trip point at which the charger changes to a constant voltage source is controlled as a function of temperature. As the battery temperature rises, the voltage trip point is lowered.

The current limit used for the constant current charging of the battery is 15 amperes. The amount of power the charger can deliver is actually dependent upon the solar array capability and the load on the ATM power buses.

The method of optimizing solar array operation is to control the charger so as to operate the array at its maximum power current.

The output of the charger is protected from short circuit by current limiting. The current limit is 15 amperes. The efficiency of the battery charger is 92 percent over a load variation of 50 to 500 watts.

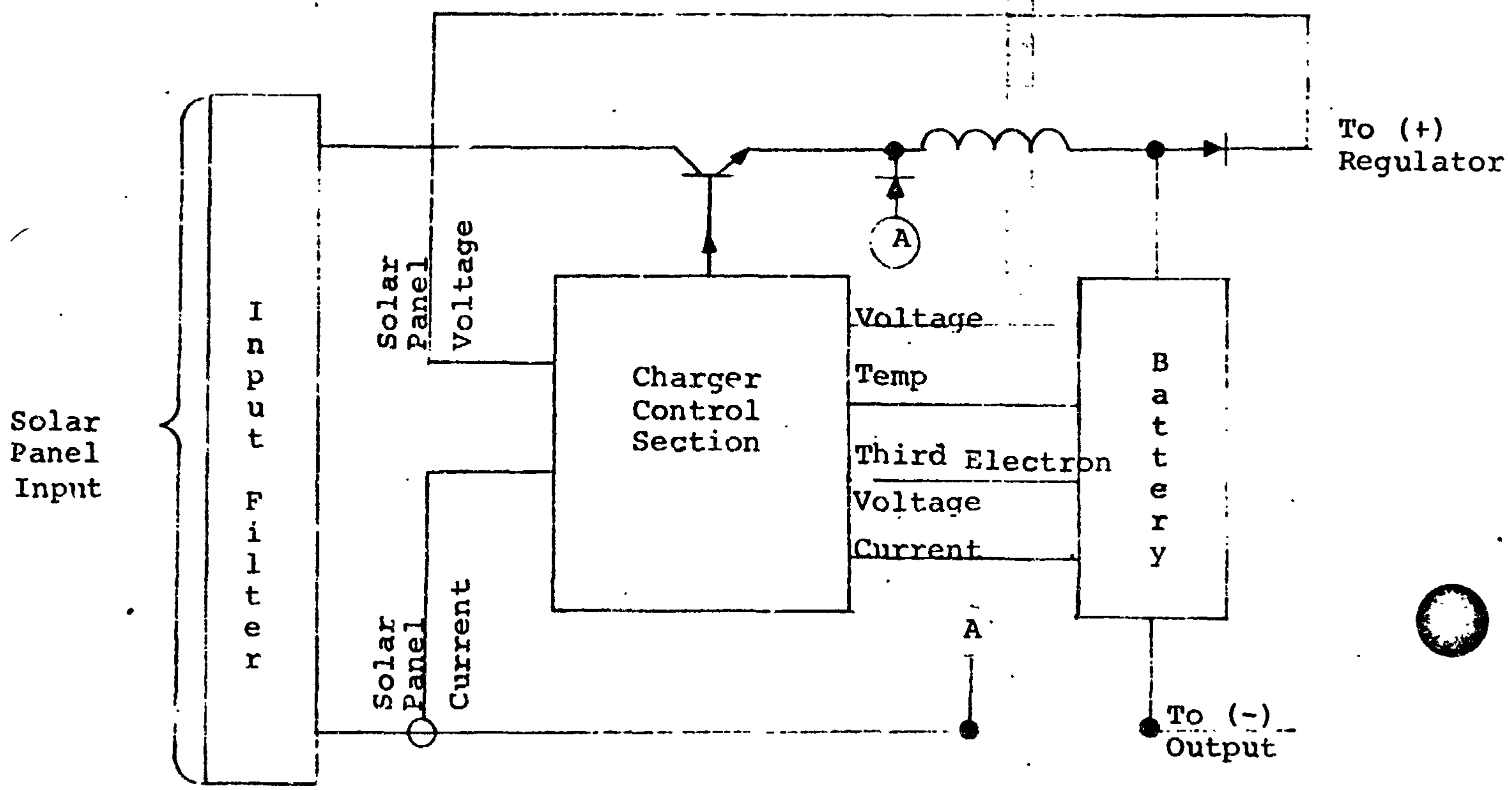


Figure 8.5.8-1 CBRM Battery Charger Block Diagram

8.5.8 SKYLAB ATM BATTERY CHARGER

General Description

Program: Skylab ATM
Vendor: MSFC
Part Number: 40M26203

Performance Characteristics

Input Voltage: 85 Vdc max.
Input Current: 15.2 Amps max.
Output Current: 14 Amps max.
Output Voltage: 33.85 to 35.65 during constant voltage charge;34.70 to 36.50 during constant current charge
Efficiency: 92%

Physical Characteristics

Size:
Weight:
Cooling Method: Passive

References

MSFC Development Report 40M26995; Martin-Marietta Electrical Power System Definition Document, ED-2002-1045-1, July 21, 1972.

Notes

This unit is an integral part of the total CBRM (charger battery-regulator-module)

Design Status

This component was flown as part of the 1973 Skylab mission.

8.5.9 APOLLO-17 CSM BATTERY CHARGER

Input power to the battery charger (Figure 8.5.9-1) is 25 to 30 volts dc and 115 volts, 400 Hz 3-phase ac. All three phases of ac are used to boost the 25 to 30 volt dc input and produce 40 volts dc for charging. In addition, phase A of the ac supplies power for the charger circuitry. The logic network in the charger, which consists of a two stage differential amplifier (comparator), Schmitt trigger, current sensing resistor, and a voltage amplifier, sets up the initial conditions for operation. The first stage of the comparator is ON, the second stage OFF, thus setting the Schmitt trigger first stage ON, and the second stage OFF. Maximum base drive is provided to the current amplifier which turns the switching transistor ON. With the switching transistor ON, current flows from the transformer rectifier through the switching transistor, current sensing resistor, and switch choke to the battery being charged. Current lags voltage due to switching choke action. As current increases, the voltage drop across the sensing resistor increases until, at a predetermined value, the voltage level causes the comparator to change states. This reversal sets the voltage amplifier OFF, and reverses the Schmitt trigger. These, in turn, set both the current amplifier and the switching transistor OFF. The switching transistor, when OFF, terminates power from the source, causing the field in the choke to collapse and discharge into the battery, then through the switching diode and the current sensing resistor to the opposite side of the choke. As the EMF in the choke decreases, current through the sensing resistor decreases, reducing the voltage drop across the resistor. At some point, the decrease in voltage drop across the sensing resistor reverses the comparator circuit, setting up the initial condition and completing one cycle of operation. The output load current, due to the choke action, remains relatively constant except for the small variation through the sensing resistor. This variation is required to set and reset the switching transistor and Schmitt trigger through the action of the comparator.

Battery charger output is regulated by the sensing resistor until battery voltage reaches approximately 37 volts. At this point, the biased voltage sensor circuit is unbiased, and in conjunction with the sensing resistor provides a signal for cycling the battery charger. As battery voltage increases, the internal impedance of the battery increases, decreasing current flow from the charger. At 39.8 volts, the battery is fully charged and current flow becomes negligible. Recharging the batteries until battery amp hour input equals amp hours previously discharged from the battery assures sufficient battery capacity for mission completion.

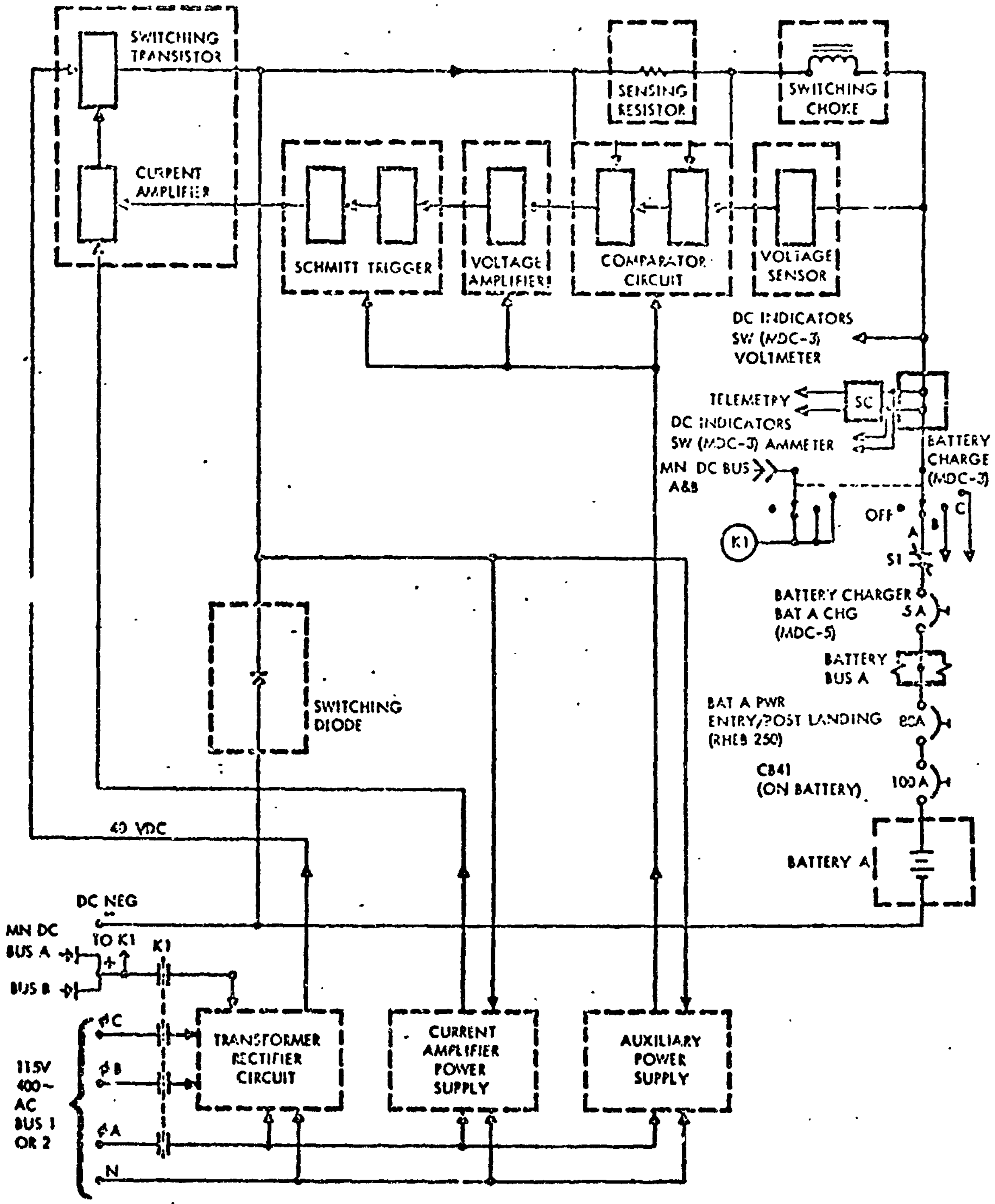


Figure 8.5.9-1. Battery Charger Block Diagram

8.5.9 APOLLO-17 CSM BATTERY CHARGER

General Description

Program: Apollo-17 CSM
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 25 to 30 Vdc and 115 volts, 400 Hz ac
Output Current: 2.7 Amps Max.
Output Voltage: 39 Volts Max.

Physical Characteristics

Size:
Weight:

References

Apollo Operations Handbook SM2A-03-Block II (1), Vol. 1, Spacecraft Description. Contract NAS9-150, Exhibit I, Paragraph 10.3, Published under authority of NASA Spacecraft System Operations Branch, Flight Crew Support Division.

Design Status

This component was flown as part of the 1972 Apollo-17 mission.

8.5.11 MARINER MARS 71 BATTERY CHARGER

A high and low rate (trickle) charger (Figure 8.5.11-1) is used to supply charging current to the battery when the panels are sun oriented. The high rate charger is comprised of a pulsewidth modulated series switching regulator utilizing a mag amp sensing and current element. The maximum output current is 2.0 ± 0.1 amperes. The low rate charger is designed to deliver a maximum current of 0.65 ± 0.15 ampere.

The battery charger incorporates a function which automatically switches from the high rate charge to the low rate charge under the following conditions:

a. Battery voltage at the output of the high rate charger exceeds 37.5 + 0.3 volts.

b. Battery temperature exceeds 100 +5°F as indicated by the closure of a temperature sensitive switch located on the battery.

The capability also exists to disable both functions of the automatic switchover circuit by ground command.

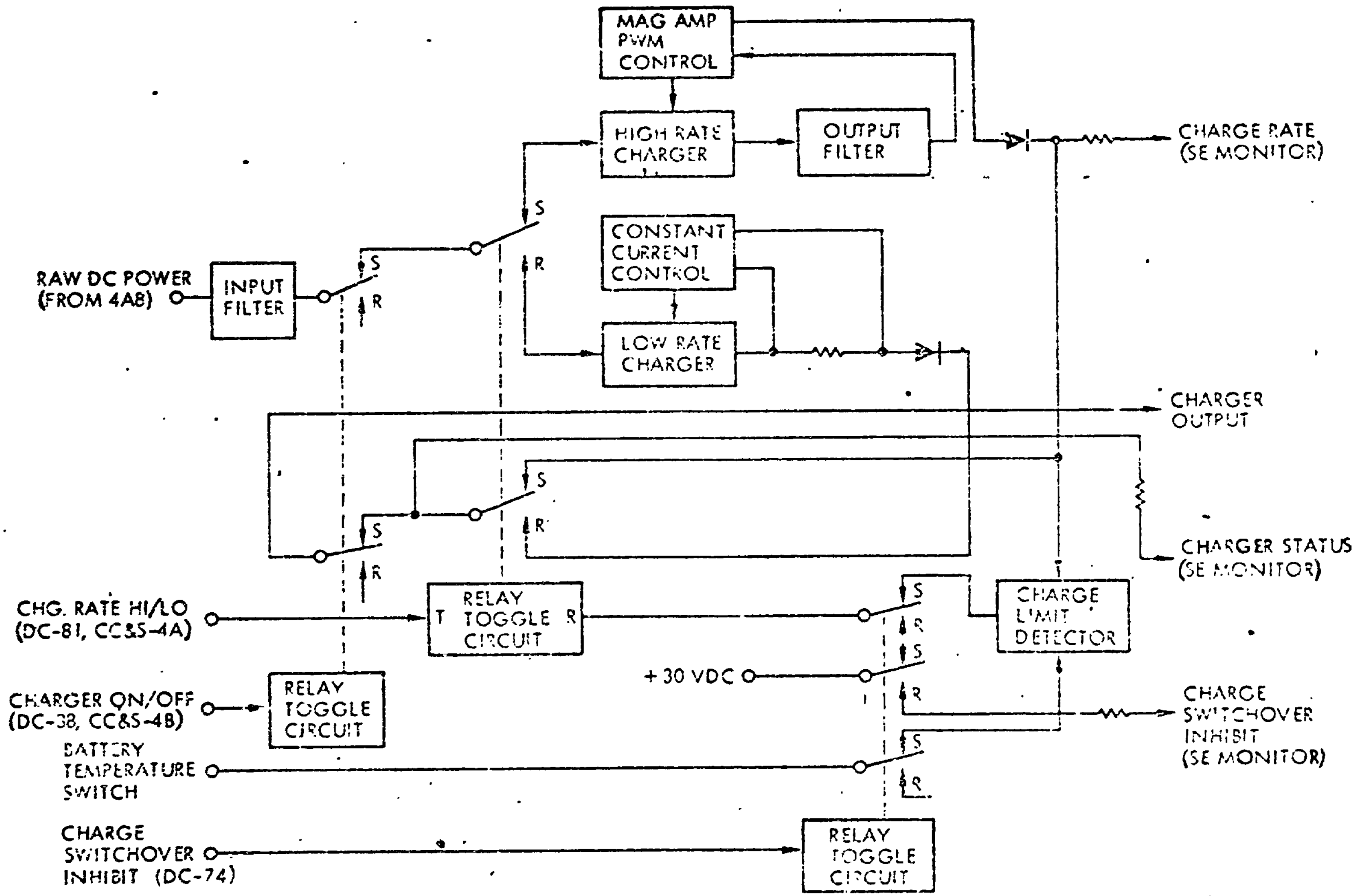


Figure 8.5.11-1 Battery Charger Subassembly Block Diagram

8.5.11 MARINER MARS 71 BATTERY CHARGER

General Description

Program: Mariner Mars 71
Vendor:
Part Number:

Performance Characteristics

Input Voltage: 25 to 50 Vdc
Output Current: High Rate 2.0 $\pm$0.1 Amps. Low rate 0.65 $\pm$0.15 Amps
Output Voltage: 26 to 39 Vdc

Physical Characteristics

Size:
Weight: 1 kg (2.3 lbs.)

References

Mariner Mars Flight Equipment Power Subsystem JPL M71-2004-1 dated 6 November 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

SECTION 9

ATTITUDE AND CONTROL COMPONENTS

ATTITUDE CONTROL
SUBSYSTEM COMPONENTS

9.1.5 VIKING LANDER INERTIAL REFERENCE UNIT (IRU)

The Viking Inertial Reference Unit (IRU) design (Figure 9.1.5-1) reflects the stringent weight and power limitations associated with the Viking Mars lander mission, as well as the specific environmental requirements imposed. The Viking IRU includes a redundant gyro and accelerometer to provide added reliability during the year-long mission. The redundant gyro is oriented such that its input axis forms an equal angle (54 deg), with each of the input axes of the other three system gyros, which are arranged in a conventional orthogonal triad. This orientation provides equal sensitivity to rotations about the vehicle principal axes. The redundant accelerometer is oriented parallel to the vehicle thrust axis accelerometer which is configured as part of an orthogonal triad.

Radiation is utilized as the primary mode of IRU waste heat dissipation, and is achieved by means of a light-weight finned cover design. The IRU is designed to operate continuously in the space environment in 0°F to 80°F ambient conditions, and to sustain high and low temperature transients.

An important feature of the Viking design is the requirement that all systems be sterilized to avoid biological contamination of the Martian surface and the introduction of spurious results in the scientific experiments which are to be performed. The ramifications of the cyclic exposure of the IRU to 280°F dry heat for a total of 380 hours were evaluated. The only inertial instrument change required was the substitution of a sterilizable damping fluid (polychlorotrifluorethylene, PCTF) in the HSSC RI-1139S gyro and an associated bellows redesign to accommodate the increased thermal expression of the new fluid.

The Viking IRU incorporates a self-test torquing capability which permits individual gyro and accelerometer loops to be periodically checked during the mission.

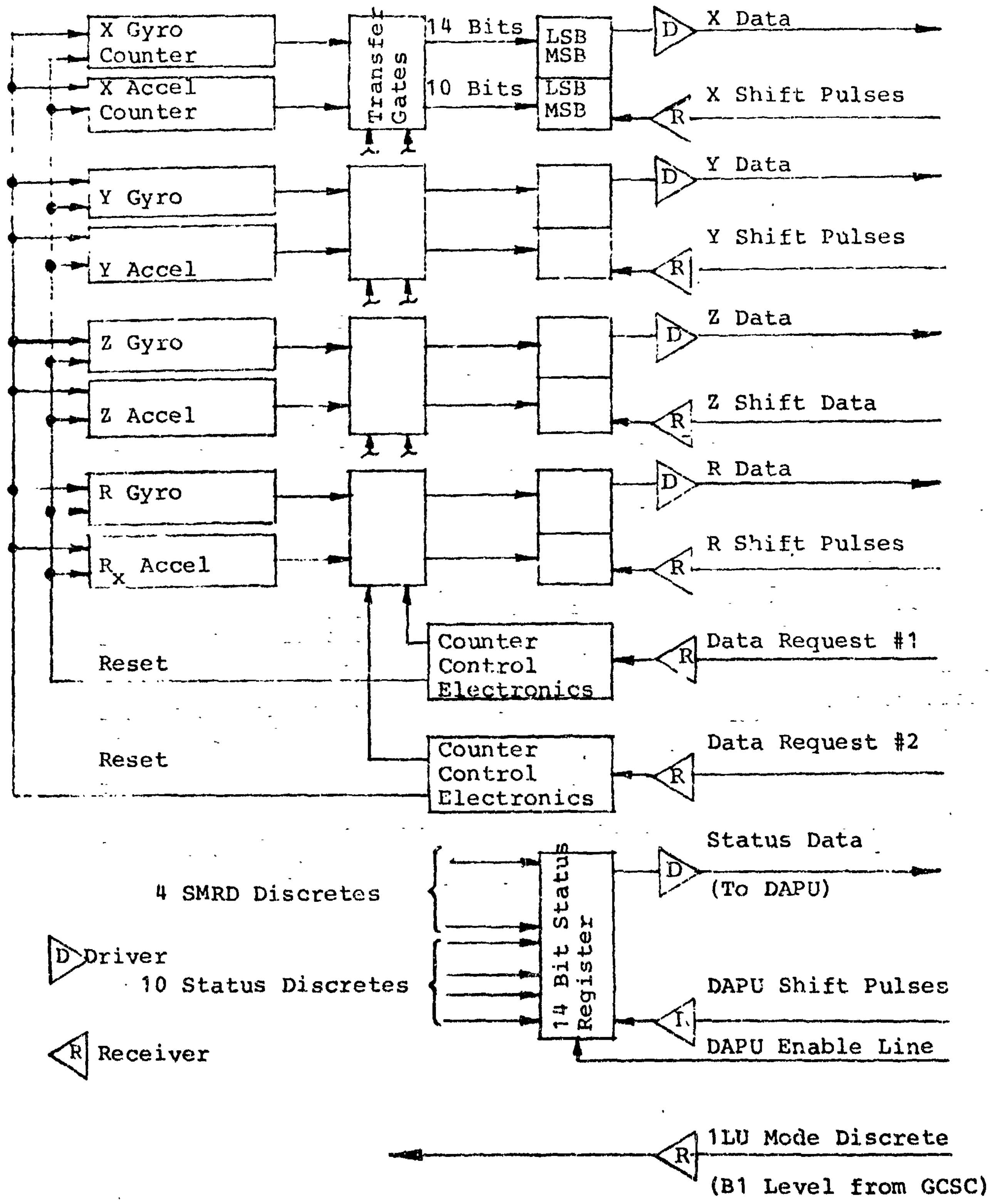


Figure 9.1.5-1 IRU Signal Data Transfer Mechanization

9.1.5 VIKING LANDER INERTIAL REFERENCE UNIT

General Description

Program: Viking MARS Lander (75)
Vendor: United Aircraft Corporate Systems Center
Part Number:

Performance Characteristics

Input Rate: 100 deg/sec (High Mode)
12.5 deg/sec (Low Mode)
Self Test Torque Rate: ±4.7 deg/sec
Input Acceleration: 20 g's Thrust Axis
5 g's Lateral Axis
Self Test: 20 g's Accel. ±7.5 g
5 g's Accel. ±1.9 g
Short Term (45 min) repeatability (run to run)
Mass Unbalance: ±.02 deg/hr/g (Requirement ±.05)
Bias: ±.02 deg/hr. (Requirement ±.05)
Scale Factor: ±0.5 PPM

Total System Probability of Success: .9991

Physical Characteristics

Size: 22.9 cm (9 in) by 27.9 cm (11 in) by 15.2 cm (6 in)
Weight: 10.7 kg (23.5 lbs)
Input Power: 28 volts, 85 watts
Cooling Method: Radiation/Convection

References

Hamilton Standard Strapdown Inertial
System Brochure, dated October 2, 1973

Comments

Design proposal information only. Thermal and vibration qualification was performed. No production data is available.

Design Status

This component is scheduled to be flown as part of the planned 1975 Viking mission.

9.1.6 OAO-C INERTIAL REFERENCE UNIT

The IRU is a three-axis, strapdown gyro system whose prime function is to provide short-term (i.e., hours as opposed to days) inertial angle reference with a 1-sigma error of less than 20 arc sec per 100 minute orbit. The references is maintained throughout sequential single-axis slews totaling less than 90 degrees. This eliminates the need for continuous monitoring by star trackers and thereby eliminates much of the ground control operation required to shift from one star tracker to another as the spacecraft is slewed or as it moves through its orbit. In addition to the primary function described above, two other functions are served; namely, providing rate information for establishing initial reference after separation and providing rate information and an inertial reference for orienting the solar paddles for maximum power.

The IRU consists of two packages an Inertial Package (IP) and an Electronics Package (EP). They are mounted in adjacent bays in the OAO spacecraft and interconnected by cables (Figure 9.1.6-1). The IP is temperature controlled and contains three precision gyros (type 2FBG-6F), their control and readout electronics, a frequency source assembly which provides the precision frequency for all the electronics in the IRU and temperature control electronics. Attached to the IP is a radiator plate which conducts heat away from the IP and radiates it to the OAO's skin.

The EP contains analog, digital, and hybrid electronics. There are 17 major electronic blocks including power supplies, control units, error conditioning units, and input/output units. Torque data from the gyros are processed as rates and rate integrals (i.e., angles) in the EP and converted to analog signals for use by the OAO's Stabilization and Control System. The IRU's reference position about each axis can either be reset or changed incrementally. The resets can be commanded to respond to individual axis reset commands or to individual axis optical sensors. Incremental changes in position are limited to one axis at a time and used when changes to the spacecraft attitude are commanded.

The Inertial Package, which contains the gyros and the IRU frequency source, must be rigidly temperature controlled to obtain the desired system accuracy. The gyros must be maintained at their design temperature setting of 135°F $\pm$1°F, and remain stable within 0.1°F of the initial set point. Gyro support electronics require similar temperature stability, and the frequency source (XTAL) must be held within 1°F. Temperature control is provided by operational and auxiliary heaters.

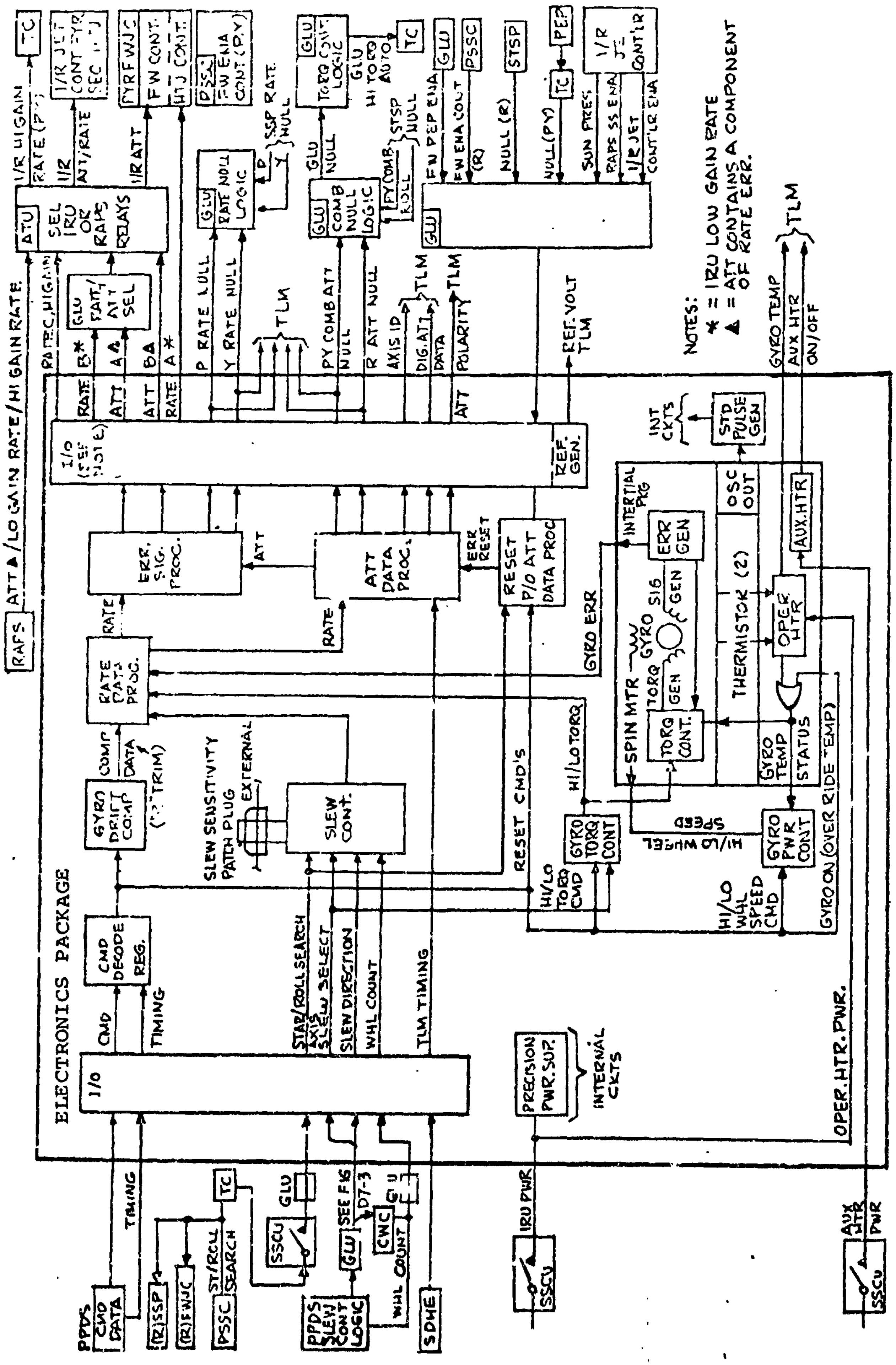


Figure 9.1.6-1 OAO-C INERTIAL REFERENCE UNIT

9.1.6 OAO-C INERTIAL REFERENCE UNIT

General Description

Program: OAO-C
Vendor: MIT
Part Number:

Performance Characteristics

Wheel Speed:	750 rpm (low)	12000 rpm (high)
Drift Rate, Uncomp.:	5.0°/hr	0.03°/hr
Drift Rate, Comp.:	0.5°/hr	0.01°/hr
Drift Stability:	0.01°/hr/day	2x10-4°/hr/day
Max Input (Correct Error):	2°/sec	480°/hr
Max. Input (Without System Degradation or Gyro Damage):	100°/sec	5°/sec
Stabilization Time:	30 min.	6 hrs.

Physical Characteristics

Size: 22.9 cm (9 in) cube IRU
22.9cm (9 in) by 38.1 cm (15in) by 15.2cm (6 in) electronics
Weight: 37 kg (81.5 lbs) IRU + heat sink + electronics
Input Power: 29W IRU, 30w heaters, 59w electronics
Cooling Method: Heat sink

References

Functional Operations Manual, Stabilization and Control Subsystem, GSFC, Doc. No. FO-G-0127-C, dated August 1972.

Stabilization and Control Equipment, OAO Specification GAC-AV-252CS-17E, dated June 15, 1968 amended December 17, 1970.

Inertial Reference Unit, Performance and Interface, Specification GAC-SYS-252PI-5, dated March 31, 1971.

Comments

Uses MIT Gyro Type 2FBG
Operating Life 12,000 hrs.

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.1.10-1 LUNAR MODULE INERTIAL MEASUREMENT UNIT (IMU)

The IMU is a gimbaled, three-degree-of-freedom gyroscopically stabilized device whose primary functions are:

a. To maintain an orthogonal, inertially referenced coordinate system for spacecraft attitude control and measurement.

b. To maintain three accelerometers in this coordinate system for accurate measurement of spacecraft velocity changes.

The orientation of the stable member, with respect to the spacecraft, is measured by resolvers on the gimbal shafts. The resolver gimbal angle output signals are supplied to the Coupling Data Unit (CDU). The resolver signals are also used as a total attitude input to the Flight Director Attitude Indicator (FDAI) for display to the astronauts.

The IMU (Figure 9.1.10-1) uses three "25 IRIG's" (inertial reference integrating gyros) to sense changes in the orientation of the stable member, and three 16 pulse integrating pendulums (16 PIP's) to sense changes in velocity. The 25 IRIG is a fluid and magnetically suspended, single-degree-of-freedom gyro with a 2.5 inch diameter case. The 16 PIP is a fluid and magnetically suspended pendulum-type device, which, when fixed in its associated electronics loop, becomes an integrating accelerometer.

The IMU gimbals consist of an outer gimbal mounted to the case, a middle gimbal mounted to the outer gimbal, and an inner gimbal or stable member mounted to the middle gimbal. All three gimbals have 360 degrees of freedom, and are positioned by torque motors.

The inner gimbal (stable member) is machined from a solid block of cold-pressed and sintered beryllium with holes bored in the block for mounting the IRIG's, PIP's, and small electronic assemblies.

The spherical middle and outer gimbals are assembled from two hydroformed, hemispherically shaped, aluminum alloy sections.

The support gimbal or case is spherical and also fabricated from an aluminum alloy. It provides a hermetically sealed environment containing air at atmospheric pressure. The case also contains a coolant passage formed by roll bonding two sheets of aluminum together except where the passage is located. After the case is formed the unbonded portion is inflated. This method forms a leak-free coolant passage.

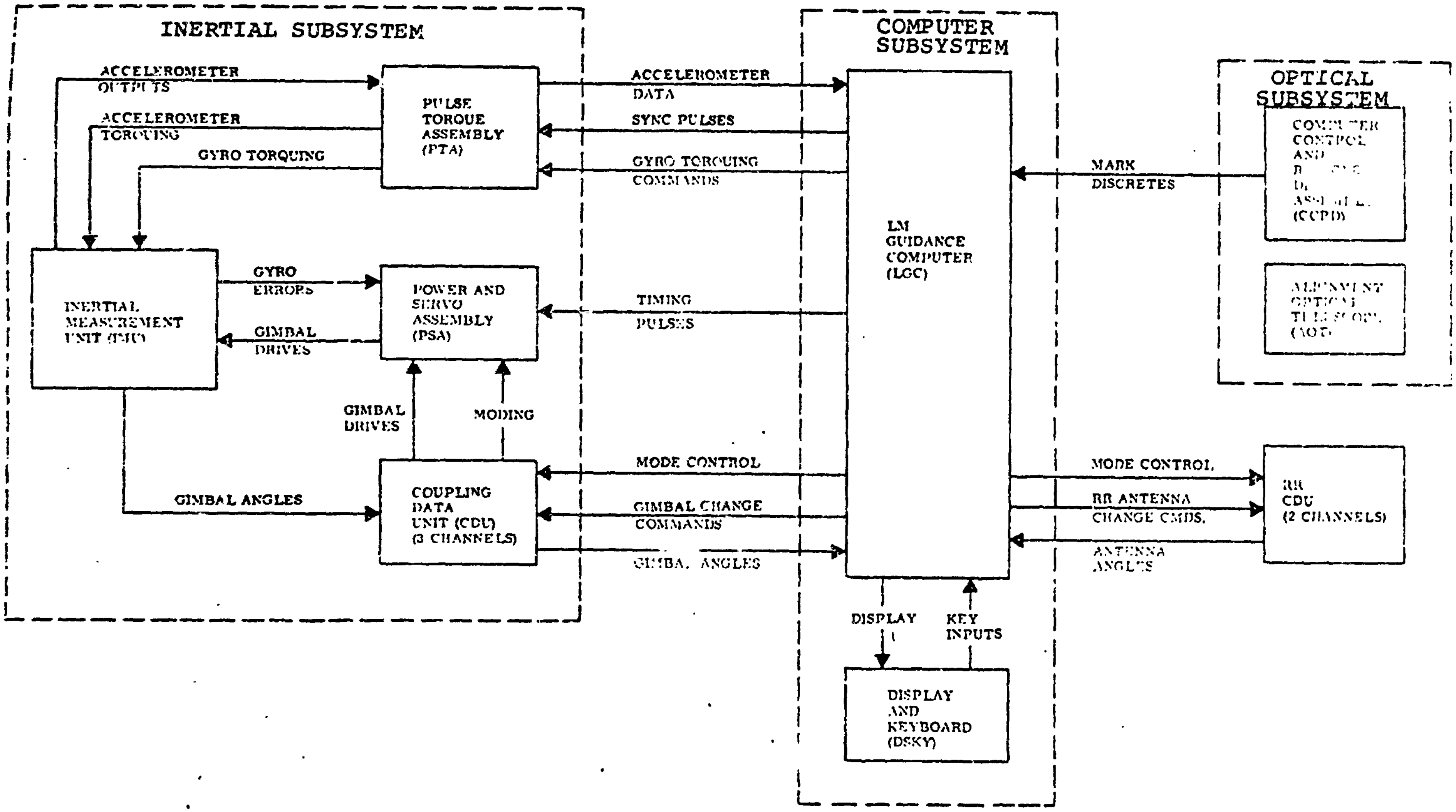


Figure 9.1.10-1 Lunar Module Primary Guidance System

9.1.10-2 APOLLO-17 LUNAR MODULE ABORT SENSOR ASSEMBLY

The lunar module abort sensor assembly (LM/ASA) is a self-contained assembly consisting of three gyros, three accelerometers, six pulse-torquing servo amplifiers (PTSA), a frequency countdown unit, warmup and fine-temperature controllers, interface electronics, a power supply and a housing subassembly including a non-hermetically sealed cover. The inertial instruments are mounted in an orthogonal triad in the beryllium housing which provides a rigid, stable mounting fixture as well as a mounting base for the unit and thermal path for the internally mounted electronics. An insulated cover eliminates spurious thermal and electrical effects.

The LM/ASA electrical interface consists of 28 volt dc power, 128 kHz clock input signals and six 64 kHz digital inertial sensor output pulse trains. A highly efficient power supply converts the dc input to various ac and precision dc voltages. The 128 kHz clock signal is processed by the frequency countdown to provide several digital timing signals for internal and external data signal conditioning. An interface electronics unit processes both input dc and clock voltages, thereby providing EMI and fault isolation conditioning. In addition, the six digital output data trains are processed to provide proper signal wave shape, amplitude, and timing.

The inertial instruments used in the LM/ASA unit consist of an orthogonal triad of RI 1139B gyros manufactured at Hamilton Standard System Center and three Kearfott 2401 accelerometers. The United Aircraft RI 1139 gyro is a 0.5 lb, floated, single-degree-of-freedom unit with an R-4 spin-axis ball bearing pair. Its angular momentum is 250,000 gm-cm^2/sec at 12,000 rpm, and its unique platinum-cobalt permanent magnet torquer is capable of generating 100,000 dyne-cm of torque at 1.3 watts of torquer power.

The "B" version of this gyro was specifically designed for the LM requirements. It is normally pulse-torque rebalanced for angular input rates of up to 28 deg/sec requiring 1.3 watts of torquer power.

The strapdown IMU has been qualified with the Kearfott C702401024-1 accelerometer. The Kearfott accelerometer is a single-axis, pendulous, permanent magnet, force rebalanced instrument of high precision and sensitivity. The instrument weighs approximately 0.3 pound.

The LM/ASA is qualified for the man-rated Apollo program and has a reliability design goal of 10,000 hours MTBF. The demonstrated bearing life of the RI 1139B gyro is more than 18,000 hours.

9.1.10-2 APOLLO-17 LUNAR MODULE ABORT SENSOR ASSEMBLY

General Description

Program: Apollo-17 Lunar Module
Vendor: United Aircraft Corporate Systems Center
Part Number:

Performance Characteristics

Mission Reliability: 0.991
Max. Angular Rate: 28 deg/sec
Gyro Drift Rate: 0.8 deg/hr max.
Max. Acceleration Range: 100 ft/sec^2
Accelerometer Bias: 70 g's (max)

Physical Characteristics

Size: 22.9 cm (9 in) by 29.2 cm (11 5 in) by 12.7 cm (5 in.)
Weight: 9.4 kg (20.7 lbs.)
Input Power: 28 Volts, 74 Watts
Cooling Method: Coldplate

References

1. Hamilton Standard Strapdown Inertial Systems, October 2, 1972

2. Abort Guidance Section Study Guide, Lunar Module LM-2, Contract NAS8-1100 Exhibit E; Paragraph 3.7.4, February 1968.

Design Status

This component was flown as part of the 1972 Apollo-17 mission.

9.2.3 HEAO-A STAR TRACKER ASSEMBLY

The star tracker is a strapped-down (non-gimballed) device comprising a sun shade, sun shutter, telescope optics, and image dissector tube and support electronics. It acquires and tracks fourth magnitude or better stars in its limited field-of-view and provides accurate indication of their coordinates about two axes in instrument coordinates to the transfer assembly.

9.2.3 HEAO-A STAR TRACKER ASSEMBLY

General Description

Program: HEAO-A
Vendor:
Part Number:

Performance Characteristics

FOV: $\pm$32 Degrees (2 axes)
Tracks to within 30° of sun
Acquires brightest star in FOV

Physical Characteristics

Size: 15.2 cm (6 in) by 15.2 cm (6 in) by 30.5 cm (12 in) Tracker
30.5 cm (12 in) diameter by 63.5 cm (25 in) Shade
Weight: 6.6 kg (14.5 lbs)
Input Power: 5 watts

References

"Preliminary Requirements Review", Volume III-C, Attitude Control and Determination Subsystem, TRW #17622-301-001-001.

Design Status

The HEAO-A program is currently in a state of redefinition. This component is a viable candidate for application in the 1977 HEAO-A mission.

9.2.4 OSO-I STAR SENSOR

The star sensor consists of four major elements:

(1) Electro-optical assembly
(2) Signal processor unit
(3) Power supply
(4) Shade

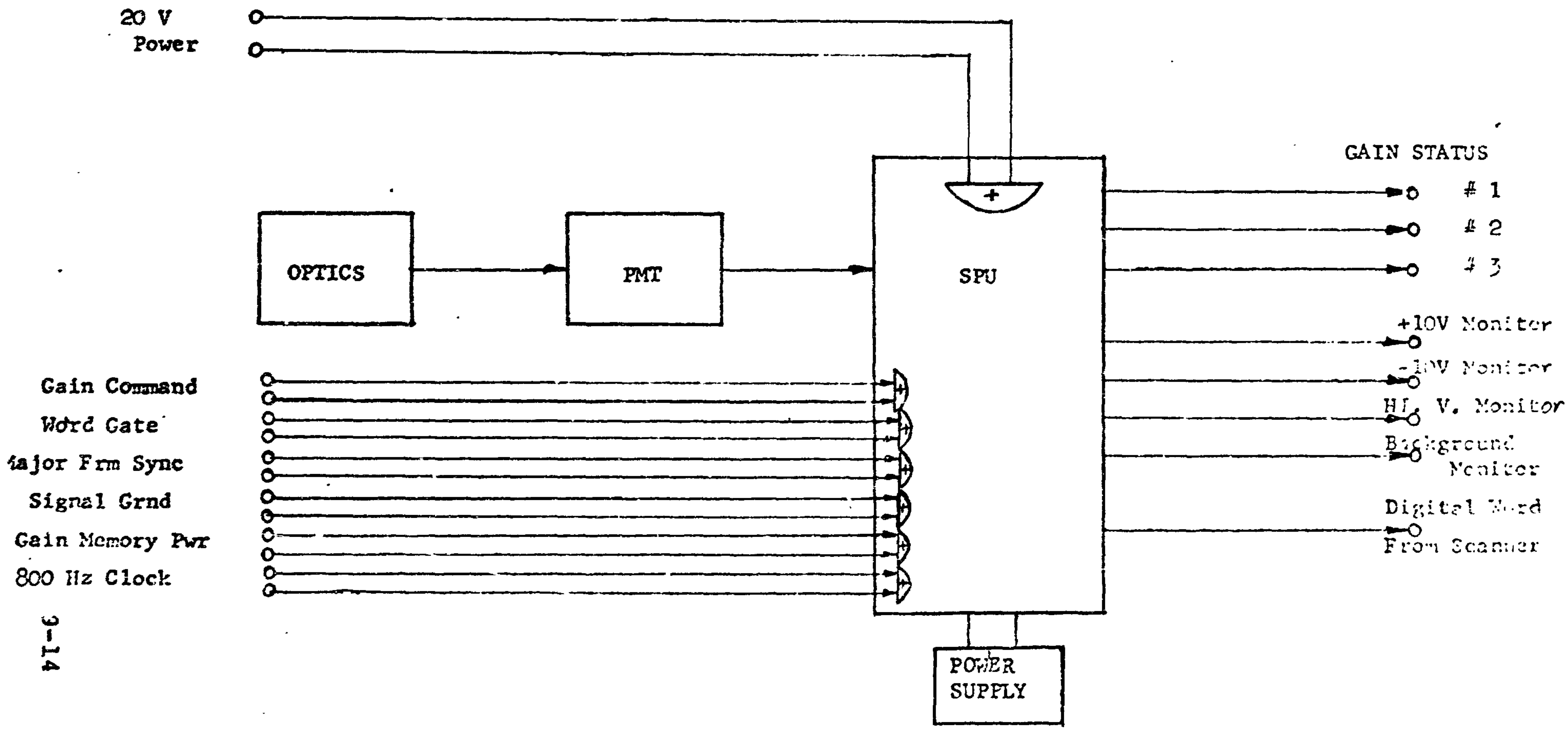
The electro-optical assembly consists of a sun shield, lens assembly, reticle, and a photomultiplier. The electronic processor consists of a preamplifier, video amplifier, level detector, logic, encoder, and telemetry interface. The power converter consists of a regulator, a low voltage converter, and a high voltage converter. Figure 9.2.4-1 depicts the signal interface.

The star scanner aspect system is used to determine the spacecraft attitude. The aspect system consists of two main components: (1) an on-board star scanner (hardware) and (2) a ground-based computer program (software) used to reduce the star scanner data.

The star scanner generates two pulses each time its field-of-view sweeps past a star brighter than a preselected level. Thus, during each wheel revolution, the scanner generates a series of pulse-pairs corresponding to the bright stars that pass through its field-of-view. The scanner electronics encode these data and multiplexes the information into the telemetry main frame for transmission to ground receivers.

The ground-based computer software is used to decode the telemetry data to identify the observed group of stars by comparing the telemetered position data with known star position data. The computer derives the orientation of the spacecraft spin axis in celestial coordinates and the instantaneous azimuth of any reference point on the spinning wheel in reference to the master telemetry clock.

Thus, the star scanner provides accurate information about star crossings in spacecraft coordinates, and the ground-based computer transforms this information into celestial coordinates to determine the instantaneous aspect of the spacecraft and wheel experiments.



OSO-I STAR SENSOR INPUT/OUTPUT DIAGRAM

Figure 9.2.4-1

9.2.4 OSO-I STAR SENSOR

General Description

Program: OSO-I
Vendor: Ball Brothers Research Corporation
Part Number: 3280020-100

Performance Characteristics

FOV: 4 degrees in azimuth
Accuracy: 3 arcmin
Stability: ± 30 arcsec

Physical Characteristics

Size:
Weight: 6.6 kg (14.4 lbs)
Cooling Method: Radiation

References

"Star Sensor/OSO-I Interface", Spec IS 31331-187, Hughes Aircraft Company.

Comments

Compatible with observatory spin rate of 6 ± 1 rpm.

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

9.2.6-1 OAO-C GIMBALLED STAR TRACKER

The star tracker equipment consists of a star tracker optical-mechanical assembly with a digital photoencoder, and a star tracker electronics controller assembly. The optical-mechanical assembly includes a sun shade, gimbaled telescope, shaft encoders, resolver, servo motors and related hardware. The gimbaled telescope is capable of $\pm 55°$ minimum freedom in each of two axes. The electronic controller assembly contains the circuitry necessary for the operation of the optical-mechanical assembly. The shaft encoders convert mechanical motion into digital gray code with a minimum resolution of 20 seconds of arc. Star radiation is modulated and applied to a servo to cause the telescope to track the star automatically, within 10% of the smallest increment of the digital photo-encoder. In addition, the gimbaled star tracker equipment has the ability to position itself in response to gimbal analog error signals. Each star tracker equipment is mechanically and electrically interchangeable. Selectable phasing is required external to the gimbaled star tracker equipment to permit its use in four (4) different mounting positions in the OAO. A resolver on the outer gimbal accepts the inner gimbal analog error signal and produces sine and cosine components of the inner gimbal error with respect to the outer gimbal axis. The zenith position of the telescope is defined as that position where the telescope is perpendicular to the plane of the optical tool mounting pads on the star tracker optical-mechanical assembly.

The star tracker encoder power supply consists of one assembly containing six separate power supplies which in turn supply power to the twelve shaft encoders.

Figure 9.2.6-1 is a pictoral drawing of the gimbaled star tracker control loop.

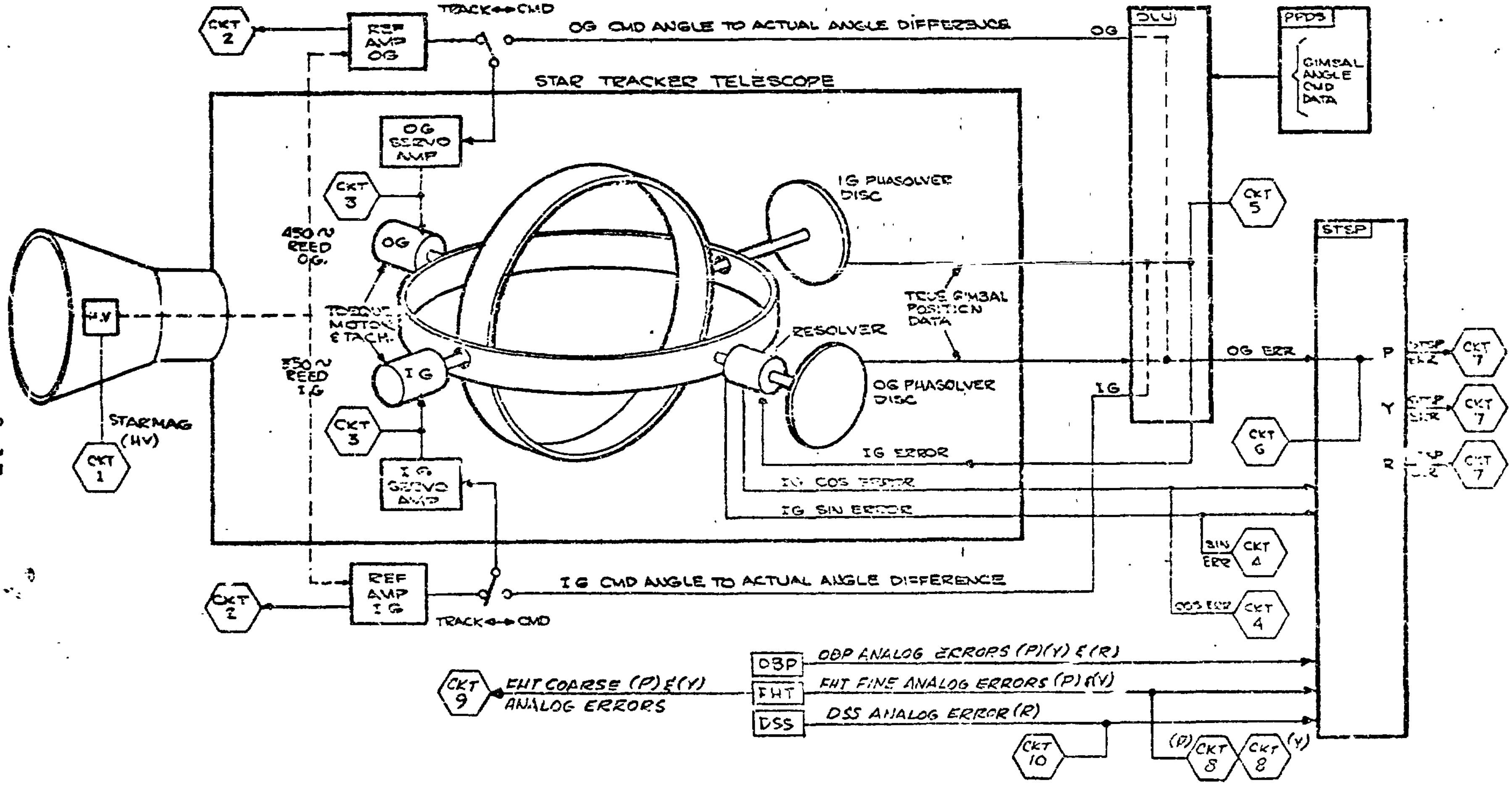


Figure 9.2.6-1 GIMBALLED STAR TRACKER PICTORAL CONTROL LOOP

9.2.6-1 OAO-C GIMBALLED STAR TRACKER

General Description

Program: OAO-C
Vendor:
Part Number: 252SCAV192 (ST) and 252SCAV191 (STE)

Performance Characteristics

Field of View (FOV): 1° x 1° square
Star Recognition: Type AO +2 magnitude
Tracking Null: 2 arc min
Track to Within 30 deg of sun
Gimbal Stops: ±55°

Physical Characteristics

Size:
Weight: 11.3 kg (25 lbs)
Input Power: 14.5 W peak, 9.8 W Av
Cooling Method: Radiation

References

OAO-C Functional Operations Manual - Volume D Stabilization and Control Subsystem FO-G-0127-C, dated August 1972

Gimbaled Star Tracker Equipment, GAC Spec AV-252CS-72A, dated July 2, 1964

Comments

Operating life > 12000 hrs.

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.2.6-2 OAO-C BORESIGHTED STAR TRACKER EQUIPMENT

The boresighted star tracker (BST) is a non-gimballed tracker with the capability of tracking sixth magnitude stars (class BO) or brighter. The BST provides accurate spacecraft attitude control for the pitch and yaw axes while the roll axis may be changed to present a better solar paddle orientation to the sun.

The BST consists basically of the following equipments:

- basic tracker and optics
- command offset logic
- BST switching logic unit (BSLU)
- earth sensor

The basic tracker generates pitch and yaw analog error signals for controlling observatory pitch and yaw attitude. Star images are focused on the photocathode of an image dissector phototube by an objective lens. The photocathode converts the optical image to an electron image which is electromagnetically scanned resulting in a series of pulses at the anode. When the star image lies on the optical axis the output pulses are of equal width. Displacement from the axis results in pulses of unequal width. The error information for each channel is extracted by a synchronous electronic gating circuit and filtered to provide d-c error signals to control the attitude of the observatory.

The BST and its associated circuitry provide the capability to electronically offset (readjust) the BST pointing axis alignment via commands. The range of accurate pitch and yaw offset is ±90 arc minutes about the original null (zenith) position.

Like the gimballed star tracker (GST) the BST optics section of the photo multiplier tube (PMT) is susceptible to damage by high levels of incident light. Unlike the GST the BST can suffer permanent damage from direct earthshine or moonshine. For this reason, the BST employs an earth presence sensor. The sensor has a field of view of ±3°, that energizes a non-latching relay (K40) external to the BST optical system when earth or moon albedo enters its view.

Figure 9.2.6-2 is a block diagram of the BST.

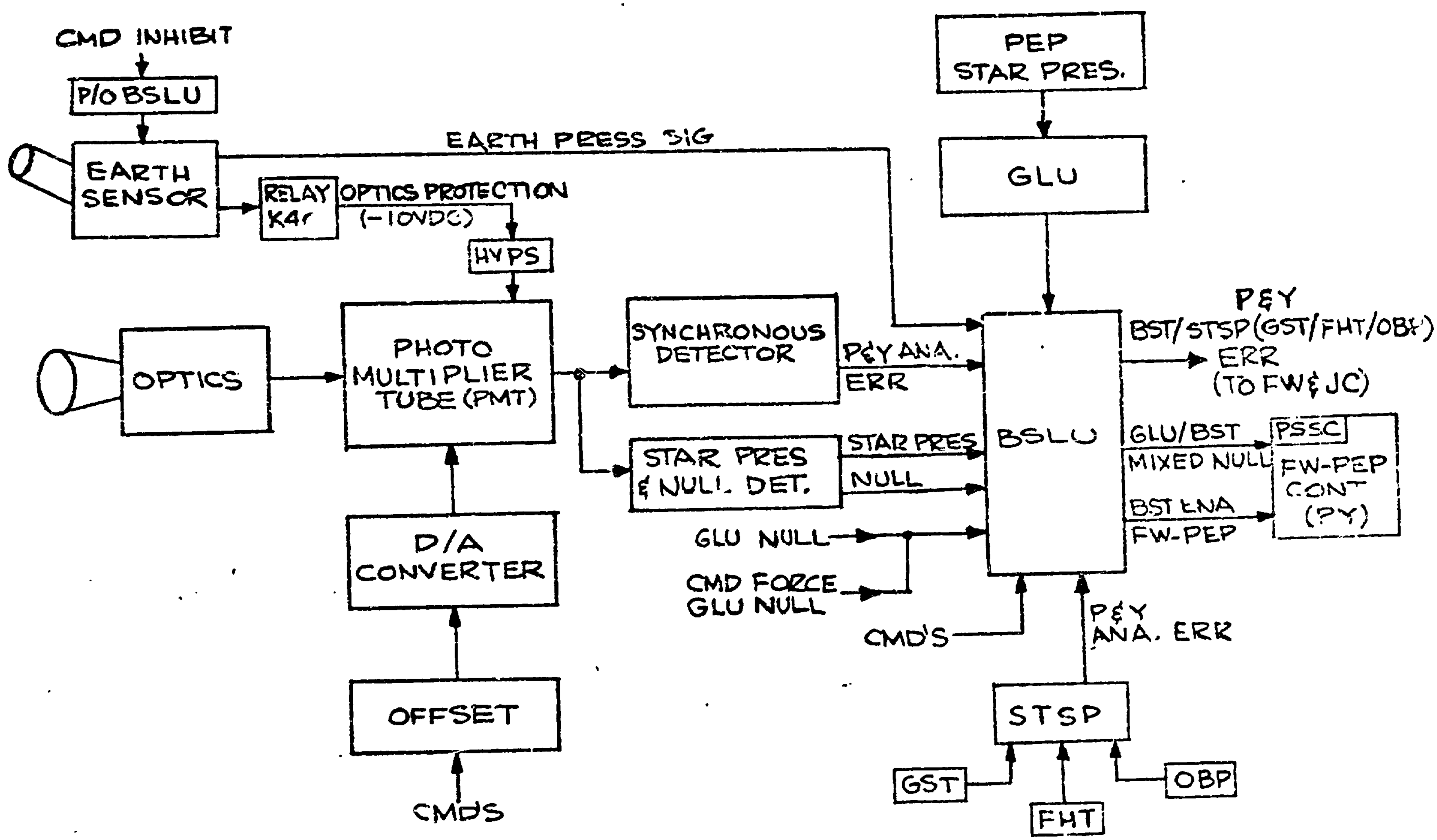


Figure 9.2.6-2 BORESIGHTED STAR TRACKER BLOCK DIAGRAM

9.2.6-2 OAO-C BORESIGHTED STAR TRACKER EQUIPMENT

General Description

Program: OAO-C
Vendor:
Part Number: 252SCAV117 (BST), 252SCAV181 (BSTE)

Performance Characteristics

Field of View (FOV): 10 arc min
Offset Range: ± 90 arc min (15 arc sec increments to 15 arc min, then 1 arc min)
Sensitivity: +6 magnitude, class BO star
Accuracy: $\pm$ 2 arc sec

Physical Characteristics

Size:
Weight:
Input Power: 9.2W max
Cooling Method: Radiation to skin

References

Boresighted Star Tracker Equipment, GAC Spec AV-252CS-51, dated June 28, 1965, amended September 20, 1969

OAO-C Functional Operations Manual - Volume D, Stabilization and Control Subsystem, FO-G-0127-C, dated Aug '72

Comments

Operational life >12000 Hrs.

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.2.6-3 OAO-C FIXED STAR TRACKER

Operating in earth orbit at altitudes greater than 100 miles, the tracker produces suitable error signals for stars to +3 magnitude. The tracker acquires the brightest target within its 8 degrees by 8 degrees square field of view. All other targets dimmer by at least one stellar magnitude and displaced at least 40 minutes of arc from the brightest star produce negligible effects.

The tracker provides high accuracy pitch and yaw error signals proportional to the angular errors in orthogonal pitch and yaw planes. The pitch or yaw angular error is defined as the component in the pitch or yaw plane of the angle between the null line of sight (line of sight with both axes at electrical zero) and the line of sight to the target.

9.2.6.3 OAO-C FIXED STAR TRACKER

General Description

Program: OAO-C
Vendor:
Part Number:

Performance Characteristics

Detection Capability: $\geq$ +3 magnitude star
Field of View: 8 deg by 8 deg square
Tracking Accuracy: $\pm$10 arcsec
Response Time (Acquisition to Null): $\leq$ 1 second (when relative motion $\leq$ 0.5 deg/sec)
Linear Region (each channel): $\geq \pm$16 arcmin
$\leq \pm$32 arcmin
Channel cross coupling: $\leq \pm$4 arcmin (when other channel at 4 deg)

Physical Characteristics

Size: 15.2 cm (6 in) by 20.3 cm (8 in) by 35.6 cm (14 in)
Weight: 9.1 kg (20 lbs.)
Input Power: 14 watts, 28 volts DC
Cooling Method: Radiation

References

OAO Fixed Star Tracker Specification, GSFC, SC-A-0034-C Rev. D
June 18, 1970

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.2.8 SKYLAB ATM STAR TRACKER

The ATM star tracker provides celestial position inputs (with respect to the sun, earth, and spacecraft) for calculating the vehicle roll reference, and to initially provide an aid for manual attitude alignment about the vehicle Z-axis.

The ATM star tracker design provides for manual and automatic search and track modes for locating and tracking suitable stars. Target stars are Canopus (primarily), Achernar, and Alpha Crux. Star tracker subassemblies are the optical-mechanical assembly (OMA) and the electronics assembly (EA).

The OMA (Figure 9.2.8-1) consists of a refractive telescope, photomultiplier tube detector, and telescope electronics with a high voltage power supply mounted on a double gimbal suspension. The EA (Figure 9.2.8-2) contains servo control circuits, power supplies, ac to dc converter, digital logic unit, mode selecting circuits, shutter drive amplifiers, and auxiliary electronics. All star tracker input-output functions to other systems are interfaced in the EA.

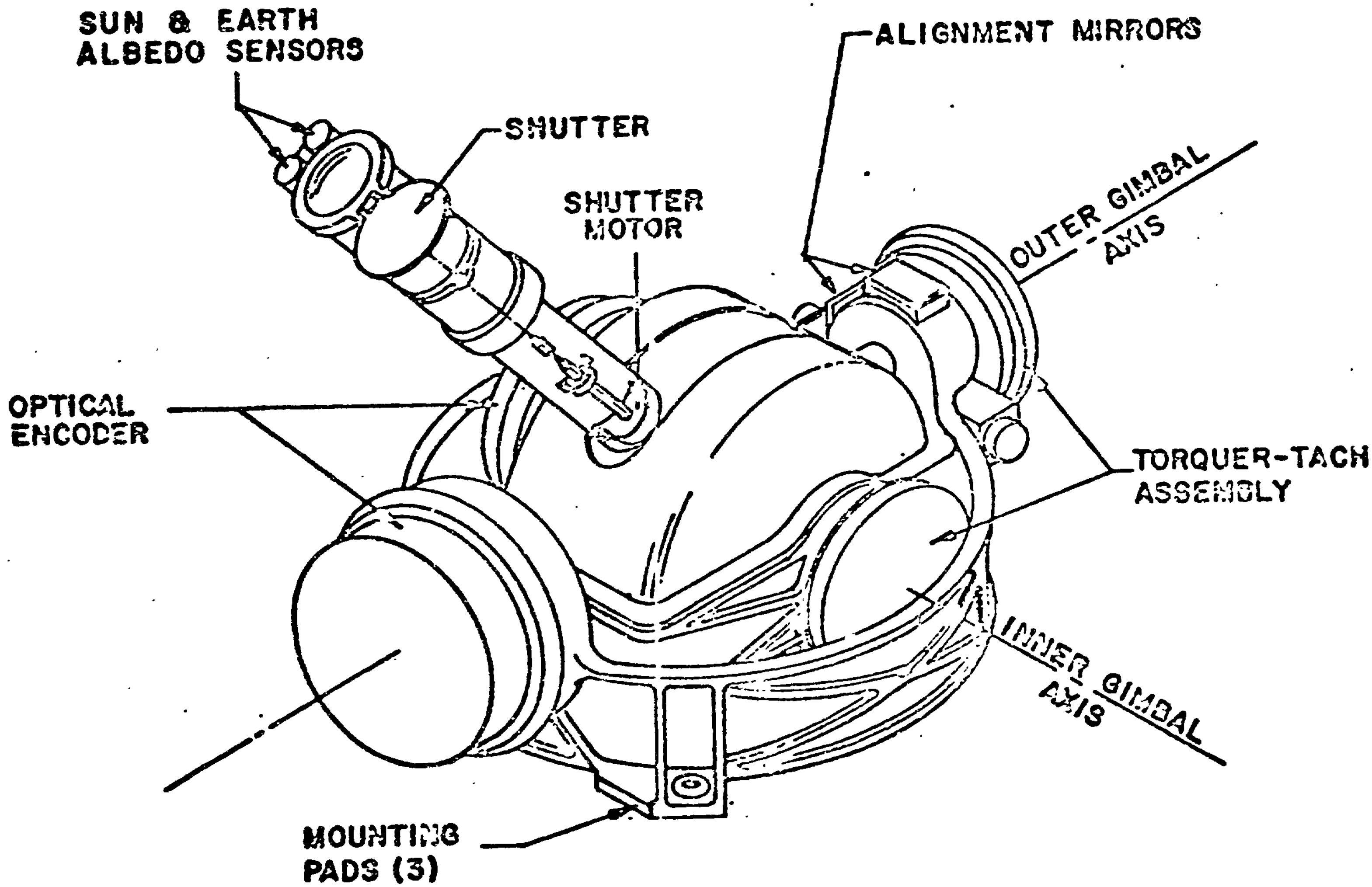


Figure 9.2.8-1 ATM STAR TRACKER

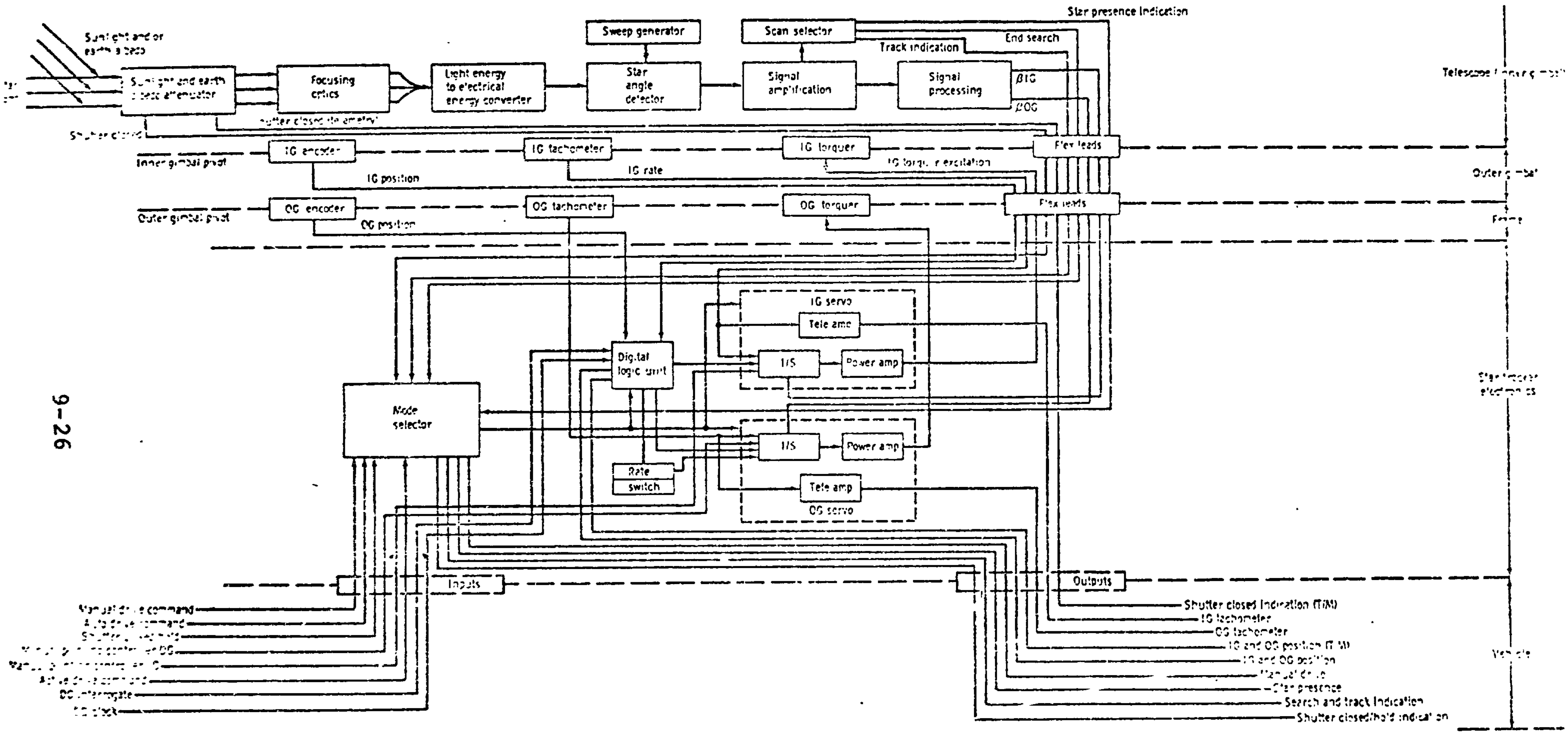


Figure 9.2.8-2 STAR TRACKER FUNCTIONAL DIAGRAM

9.2.8 SKYLAB ATM STAR TRACKER

General Description

Program: Skylab ATM
Vendor: Bendix Corp.
Part Number:

Performance Characteristics

Field-of-View
Acquisition: 1 arc degree (approx)
Track: 10 arc minutes (approx)
Tracking Accuracy: ±10 arc seconds
Capable of tracking to within 5 degrees of earth (0.35 albedo), 45 degrees of sun, 2 degrees of structure (black), and 5 degrees of OWS solar array.

Provides safety shutter for pointing toward the earth or sun.

Search Angles
Coarse: ±15 deg (outer); +5.25, -4.5 deg (inner)
Fine: ±2 deg (outer); ±1.5 deg (inner)
Gimbal Performance
Freedom
Outer Gimbal: ±87 degrees
Inner Gimbal: ±40 degrees
Readout: Serial binary to ATMDC, parallel binary to telemetry
Position Resolution: 30 arc sec (serial binary and parallel binary)
Position Accuracy: ±1 Bit

Capable of operating in automatic search/track, manual and shutter closed/hold modes.

Physical Characteristics

Size:
Weight: 32.7 kg (72 lbs)
Input Power: 69w (peak), 24w (average)

References

Environmental Design and Qualification Test Criteria for Apollo Telescope Mount Components -50M02408

Electromagnetic Compatibility Control Plan -50M12725

Apollo Telescope Mount Star Tracker Operational Requirements and Descriptions.

Design Status

This component was flown as part of the 1973 Skylab mission.

9.2.11 MARINER MARS STAR SENSOR (CANOPUS TRACKER)

The Canopus Tracker is an electro-optical device which provides a single axis error signal proportional to the angular deviation of the line of sight to the star from the null plane about the spacecraft roll axis. This signal is used to control the attitude of the spacecraft about the roll axis. Figure 9.2.11-1 depicts the tracker's field of view geometry. The Canopus Tracker design approach is that of electronic gimbaling and scanning of the electon image of a star. It provides a large field of view for acquisition while providing a small field of view and high signal-to-noise ratio for attitude hold. Stars other than Canopus are discriminated against on the basis of intensity and cone angle setting in the Canopus Tracker acquisition logic. Cone angle discrimination is obtained by limiting the tracker cone angle field of view to ± 5.9 degrees from the nominal value. The nominal setting can be incremented by external commands to accommodate seasonal variations of the Canopus cone angle. When a star satisfying the intensity gate logic is detected, the attitude control subsystem terminates the roll search.

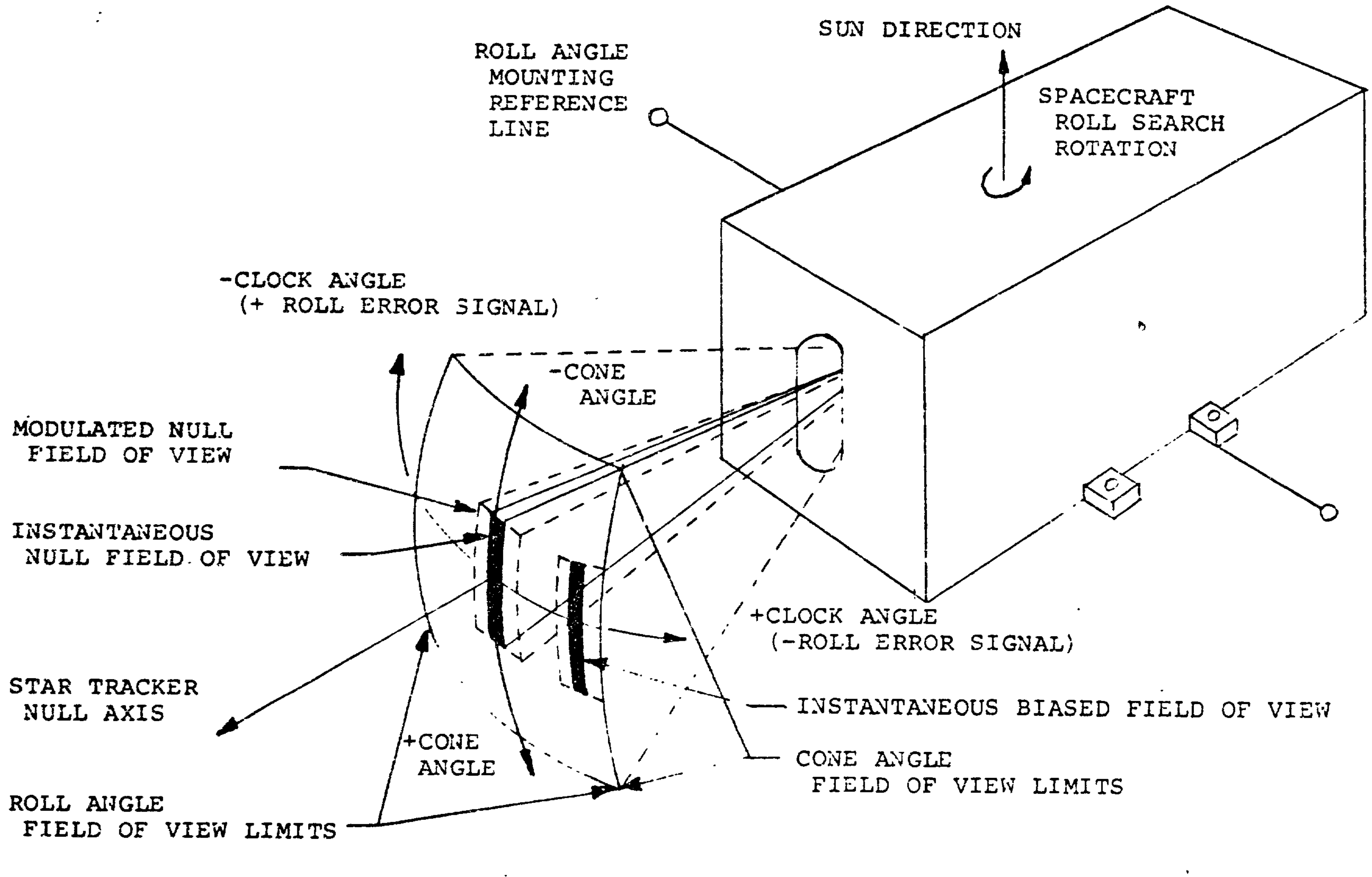


Figure 9.2.11-1 CANOPUS TRACKER FIELD OF VIEW GEOMETRY

9.2.11 MARINER MARS STAR SENSOR (CANOPUS TRACKER)

General Description

Program: Mariner Mars 71
Vendor: Barnes Engineering
Part Number:

Performance Characteristics

- An unobstructed field-of-view from 73 to 106 degrees in cone and -5 to +5 degrees clock is required.
- The instantaneous field-of-view is 1.0 $\pm$.075 degrees by 11.8 $\pm$0.4 degrees in clock roll and cone directions, respectively.
- The acquisition (scanned) field-of-view is 3.0 $\pm$0.325 degrees by 11.8 $\pm$0.4 degrees in clock and cone.
- Total field-of-view is $11.0^{+2.32}_{-1.82}$ by $36.36^{+1.0}_{-0.6}$ degrees in clock and cone directions respectively.

Physical Characteristics

Size:
Weight: 2.3 kg (5 lbs)
Input Power: 1.5 W

References

Mariner Mars 1971 Flight Equipment Attitude Control Subsystem, JPL-M71-2007-1, Rev. A, dated 19 May 1970

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

9.3.3 HEAO-A COARSE SUN SENSOR

The Bendix Wide Angle Sun Sensor-type 1771858 or 1818787-provides two direct current output signals proportional to the azimuth and the zenith angles of the sun's position relative to a predetermined reference on a spacecraft. The unit is used for spacecraft attitude control and orientation and telemetering vehicle attitude to ground stations. A single unit views 180 degrees in both azimuth and zenith, so that two units completely cover a sphere of 360 degrees. The two type numbers cover identical sensors, which differ only in the interconnection of outputs.

Each sensor has a dual channel, silicon-cell-detection structure that provides a five milliamp maximum signal through a 100 ohm load resistance for each channel in direct sunlight. The simultaneous output signals of the two channels reference the attitude of the spacecraft relative to the sun's rays. The magnitude of the angular deviation of the spacecraft from the sun in azimuth and zenith is defined by the magnitude of the signal from each channel. This definition is linear within the 20 degree cone of the boresight. For angles greater than +10 degrees, the output increases as an approximate cosine function up to +90 degrees.

When the sun sensor is applied to stabilization control and attitude orientation, the output of both channels is zero or null for an on target condition. Off target conditions produce error signals for control, correction, or re-servoing to null.

When the sun sensor is mounted co-axially with the roll axis of a spacecraft, a fractional deviation from zero in zenith produces sine waves at both outputs when the spacecraft rolls. The frequency of the sine wave signals corresponds to the rate of rotation about the roll axis. Phase relationship and signal amplitudes indicate the spacecraft's attitude.

No auxiliary power is required for operation, and sensors are normally supplied without an amplifier.

Four photovoltaic cells mounted on an aluminum casting behind a square-aperture plate sense the sun through a central, twenty-degree cone-shaped view. Each pair of diametrically opposed cells is interconnected, one pair for each channel. Eight photovoltaic cells mounted on the periphery of the aluminum casting, at right angles to the quad cell structure, sense the complete hemisphere beyond the central, twenty-degree cone. Shadowing bosses shield these cells during the first twenty degrees of rotation. The cell structures are hermetically sealed by a glass dome.

Diametrically opposed cells are interconnected, four to each channel, in parallel with two diagonal cells of the quad structure. In the type 1771858 sensor, the parallel outputs of the cell banks are internally connected in series to a four-pin output connector. In the 1818787 sensor, the parallel outputs are fed to an eight-pin connector for external interconnection.

Light entering the square aperture parallel to the sensor's zenith and azimuth centerline illuminates the quad calls equally, but does not impinge on the peripheral cells. This is the null and no current passes. When light enters at angles other than null, a proportional current is generated by the cells. The proportionality between an angle of deviation and the corresponding electrical output is linear through the central cone of twenty-degrees by the action of the aperture plate. As the sensor rotates beyond the twenty-degree cone, the response of the quad cells decreases and the peripheral cells increases, providing a smooth transition in output signals. Figure 9.3.3-1 is a block diagram of the sun sensor.

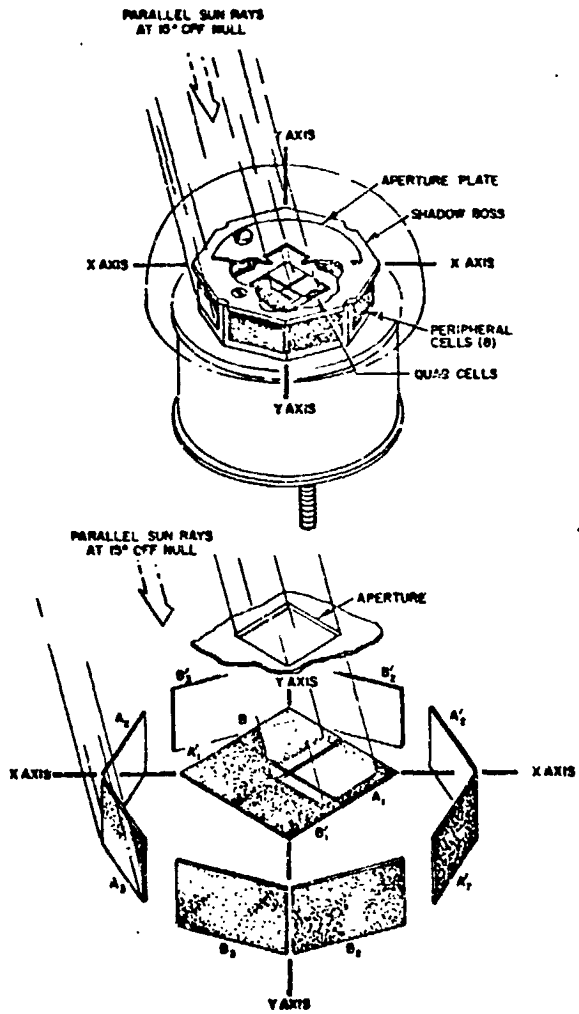


Figure 9.3.3-1   HEAO COARSE SUN SENSOR

9.3.3 HEAO-A COARSE SUN SENSOR

General Description

Program: HEAO-A
Vendor: Bendix Corporation
Part Number:

Performance Characteristics

Output: 0-5 ma
Sensitivity: 0.2 ma/deg
Load Impedance: 100 ohms
Field-of-View: 180°
Temperature Range: -70°C to +50°C

Physical Characteristics

Size: 4.8 cm (1.9 in) diameter by 5.1 cm (~2 in)
Weight: 57 gms (2.5 oz)

References

Preliminary Requirements Review Volume III-C, Attitude Control and Determination Subsystem, Doc. No. TRW 17622-301-001-001, 24 July 1972.

Design Status

The HEAO-A program is currently in a state of redefinition. This component is a viable candidate for application in the 1977 HEAO-A mission.

9.3.4-1 OSO-I WHEEL SUN SENSOR

The wheel sun sensor (WSS) provides the capability for onboard measurement of pitch angle and provides a spin synchronous reference for precession jet control and experiment utilization. The sensor assembly consists of three dual-fanbeam sensing sections. The center section is used for normal on-orbit sensing and control functions, while the upper and lower sections are normally used only during initial acquisition. The upper and lower sections have pitch FOV's which overlap the center section FOV to insure proper jet control reference transition during initial acquisition maneuvers. The center section provides redundant dual-fanbeam sensors, with each sensor dedicated to one of the Wheel Control Electronics (WCE) units, while the upper and lower sensors are non-redundant with signal fanout to both WCE units.

Each section consists of 4 sensor units mounted on a precision bracket. A silicon photovoltaic cell in each of the sensor units produces an output signal when the sunline is in (or very nearly in) its respective field of view. The field of view of each sensor unit is a fan-shaped beam roughly 1-3/4° by 90° (Figure 9.3.4-1).

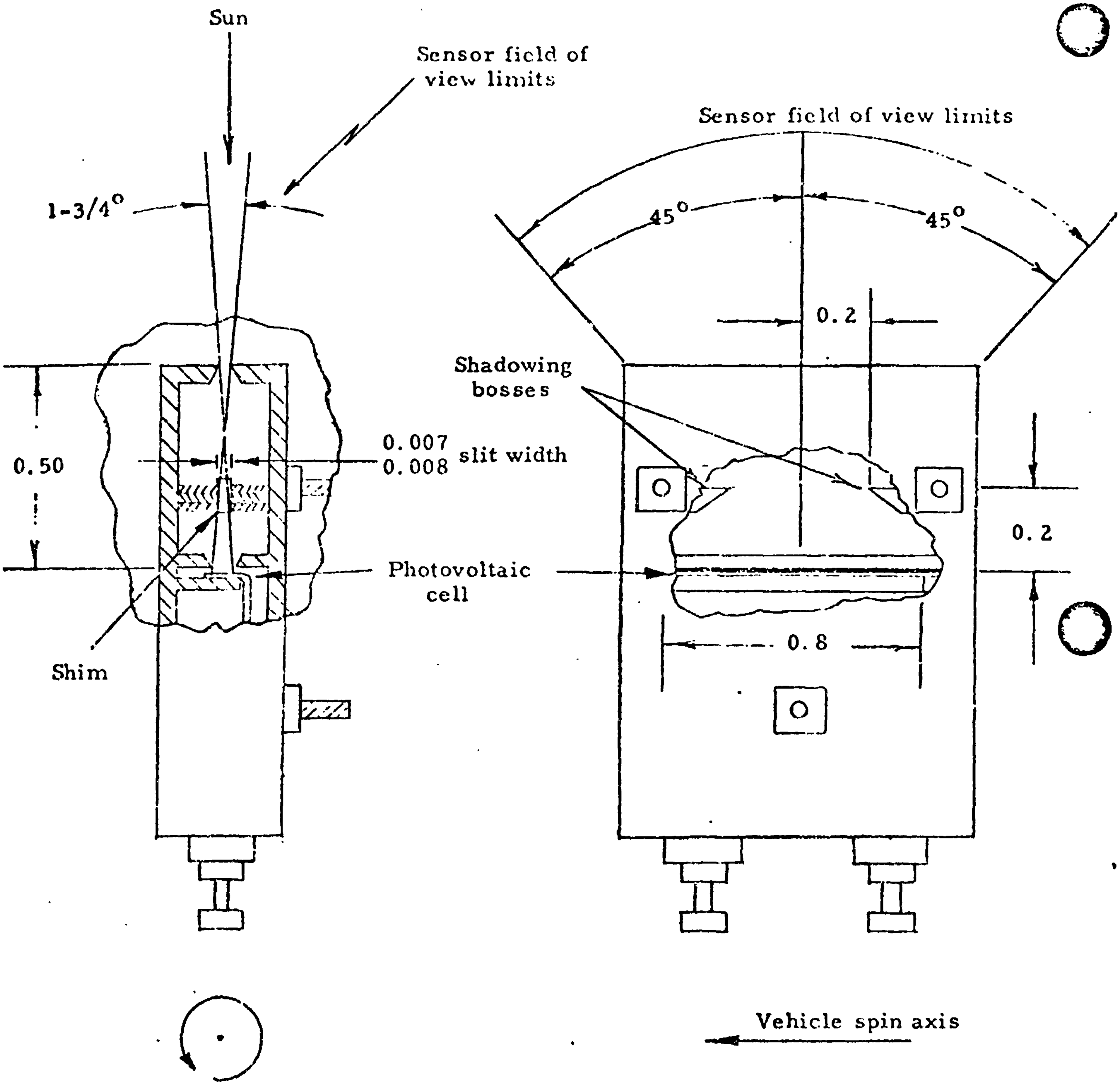


Figure 9.3.4-1 Individual Sun Sensor Schematic

9.3.4-1 OSO-I WHEEL SUN SENSOR

General Description

Program: OSO-I
Vendor:
Part Number:

Performance Characteristics

Sensor: Silicon photovoltaic cell
Sensor Assy: Dual-fanbeam sensor sections
Field of View: Fan shaped 1.75° by 90° approx., each sensor
Output Voltage: Sunlight, 0.250 volts to 0.500 volts
Pulse to Pulse Repeatability: ±0.05%

Physical Characteristics

Size:
Weight: 0.273 kg (0.6 lbs.) total assembly

References

Hughes Spec. DS 31331-142, Wheel Sun Sensor Assembly, November 20, 1972

Hughes Spec. SS 31331-140, Control Subsystem Specification, August 30,1972.

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

9.3.4-2 OSO-I POINTED INSTRUMENT ASSEMBLY (PIA) SUN SENSOR

The sun sensor generates azimuth and elevation information relative to the sun position and the instrument telescope position. The sun direction outputs may be used directly or as internally corrected for change in telescope alignment with the sun sensor. Corrections are made through an optical link between the sensor block and light sources in the telescope focal plane. The telescope focal plane source is illuminated by solar energy piped from a collection point to the focal plane apertures.

The sun sensor consists of a sensor block, a sun collector, a focal plane assembly and an electronics unit (Figure 9.3.4-2).

The sensor block is a single quartz block which contains coaligned electro-optical sensors. Three of the sensors have fields of view which are parallel to the telescope and in the same direction, while the other three have views into the telescope. The three outward viewing sensors are the elevation and azimuth control sensors and the control AGC sensor. The three inward looking sensors are the elevation and azimuth collimator sensors and the collimator AGC sensor, which view a pair of focal plane sources for detecting motions of the telescope relative to the sun sensor boresight axis.

The sun collector is a separate block containing a sun viewing collector lens, a light pipe connection, and an electronic light modulator, mounted to the front of the pointed instrument.

The focal plane assembly is an aperture plate fastened to a stainless steel tube containing a fiber optics bundle. The fiber optics connects the sun collector output energy to the two circular apertures on the aperture plate in the focal plane. These apertures are located equidistant from the experiment telescope focal plane slit. The energy at these apertures are the targets for the collimator eyes and the means of measuring the telescope axis position in both elevation and azimuth.

The solar sensor electronics unit contains the power conversion, power regulation, signal amplifiers and output buffers required for the sensors.

Sun sensor coordinate definitions and the control sensor signal range are shown in Figures 9.3.4-3 and 9.3.4-4 respectively. Sun sensor electronics are shown in Figure 9.3.4-5.

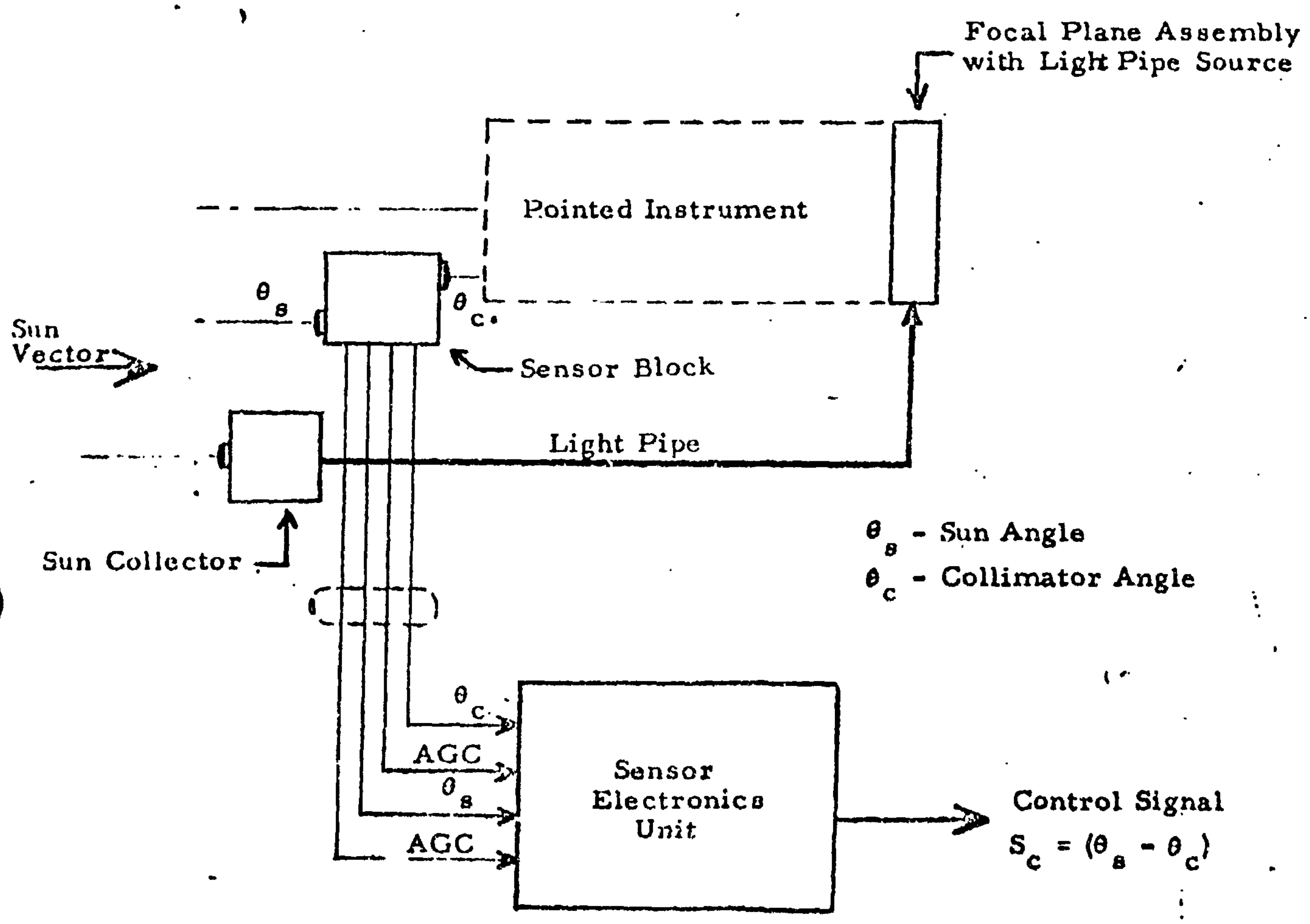
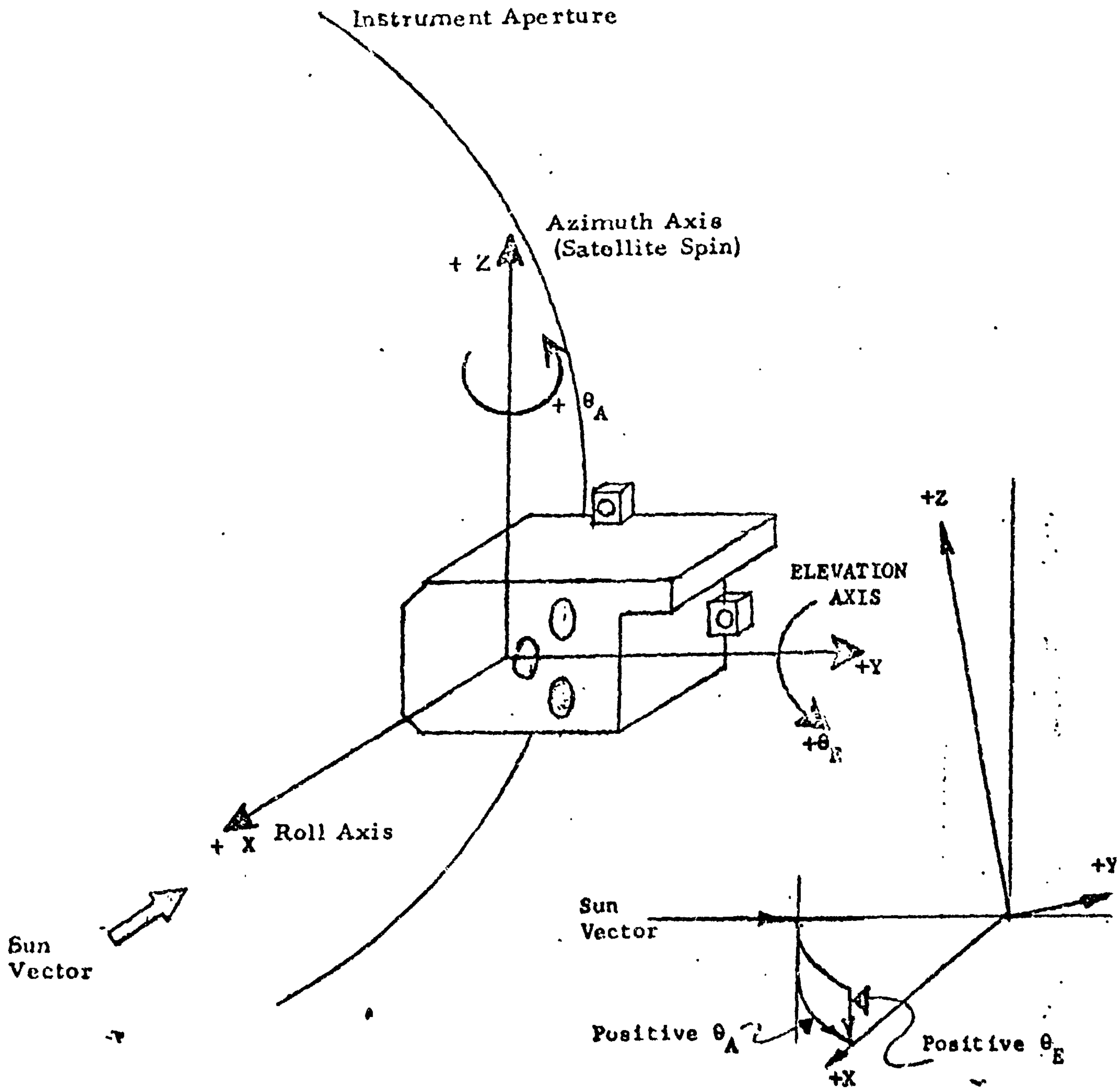


Figure 9.3.4-2 Illustration of OSO-I Sun Sensor



NOTE: Positive angular change in the sensor shall be a positive voltage.

Figure 9.3.4-3 OSO-I Coordinate Definitions

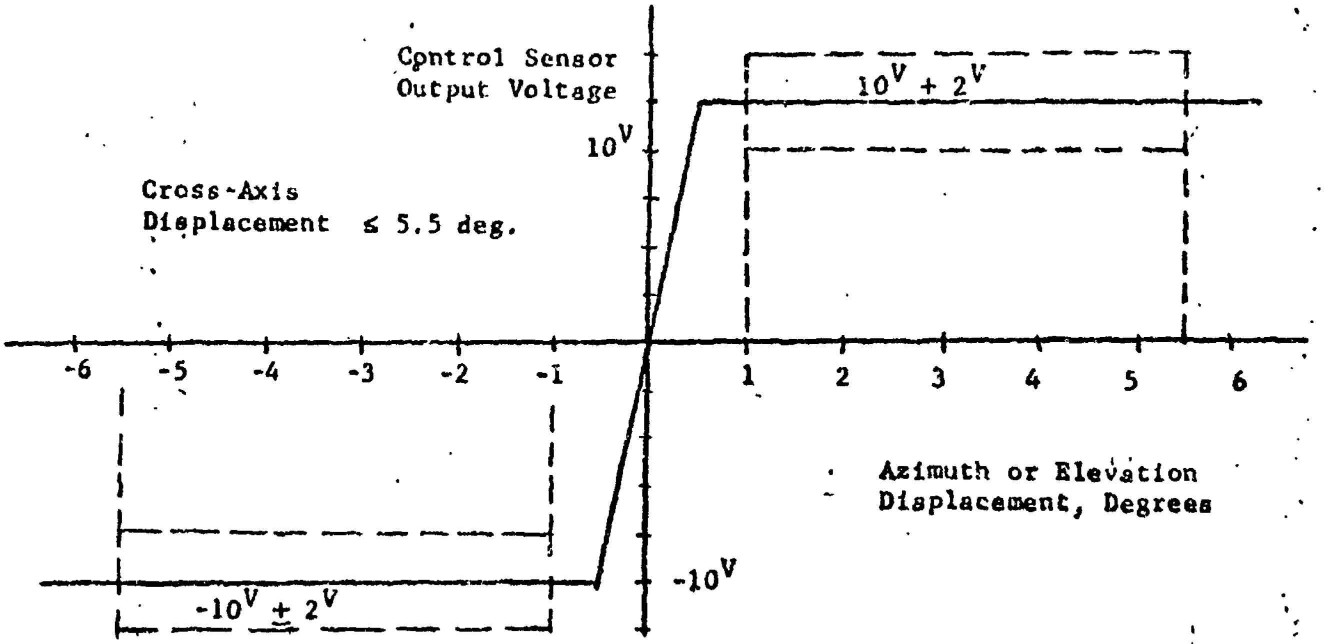


Figure 9.3.4-4 OSO-I Control Sensor Signal Range

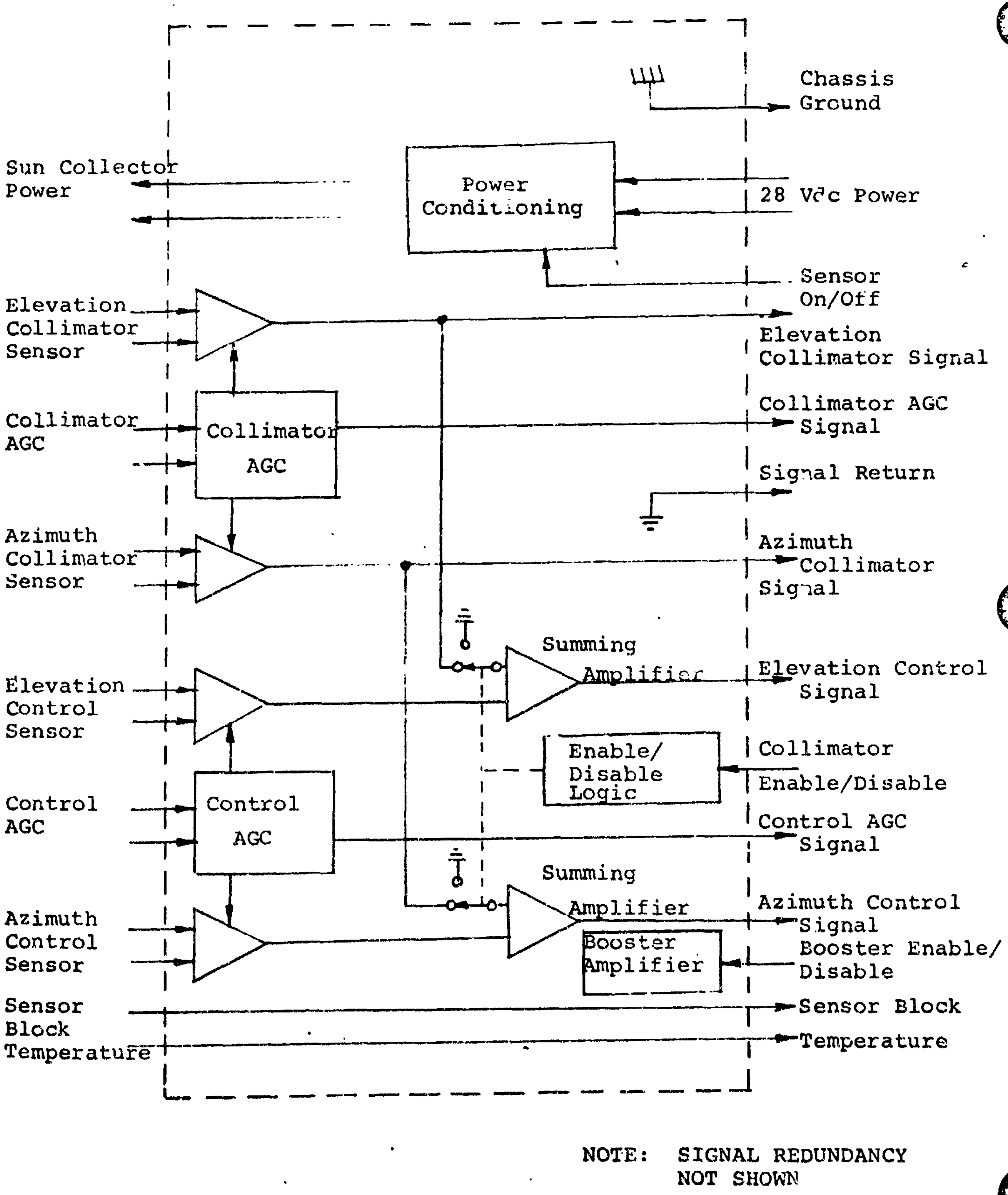


Figure 9.3.4-5 Sun Sensor Electronics

9.3.4-2 OSO-I POINTED INSTRUMENT ASSEMBLY (PIA) SUN SENSOR

General Description

Program: OSO-I
Vendor:
Part Number:

Part Number:	Sensor Block	255440
	Sun Collector	255437
	Focal Plane Assy	255439
	Electronics Unit	255438

Performance Characteristics

Sensor linear range: ± 30 arcmin
Accuracy: ± 4.5 arcsec (over ± 30 arcmin range)
Stability: ± 0.5 arcsec (5 min time period)
± 2.5 arcsec (70 min time period)
Repeatability: ± 2.5 arcsec

Physical Characteristics

Size: 147.6 cm^3 (9 in^3) Sensor block
95 cm^3 (5.8 in^3) Sun Collector
820 cm^3 (50 in^3) Electronics Unit
Weight: 1.8 kg (4 lbs) (total assembly)
Input power: 2.0 watts (total)

References

OSO PIA Sun Sensor Procurement Specification, PS 31331-141, Hughes Aircraft Company, dated July 1,1971

Design Status

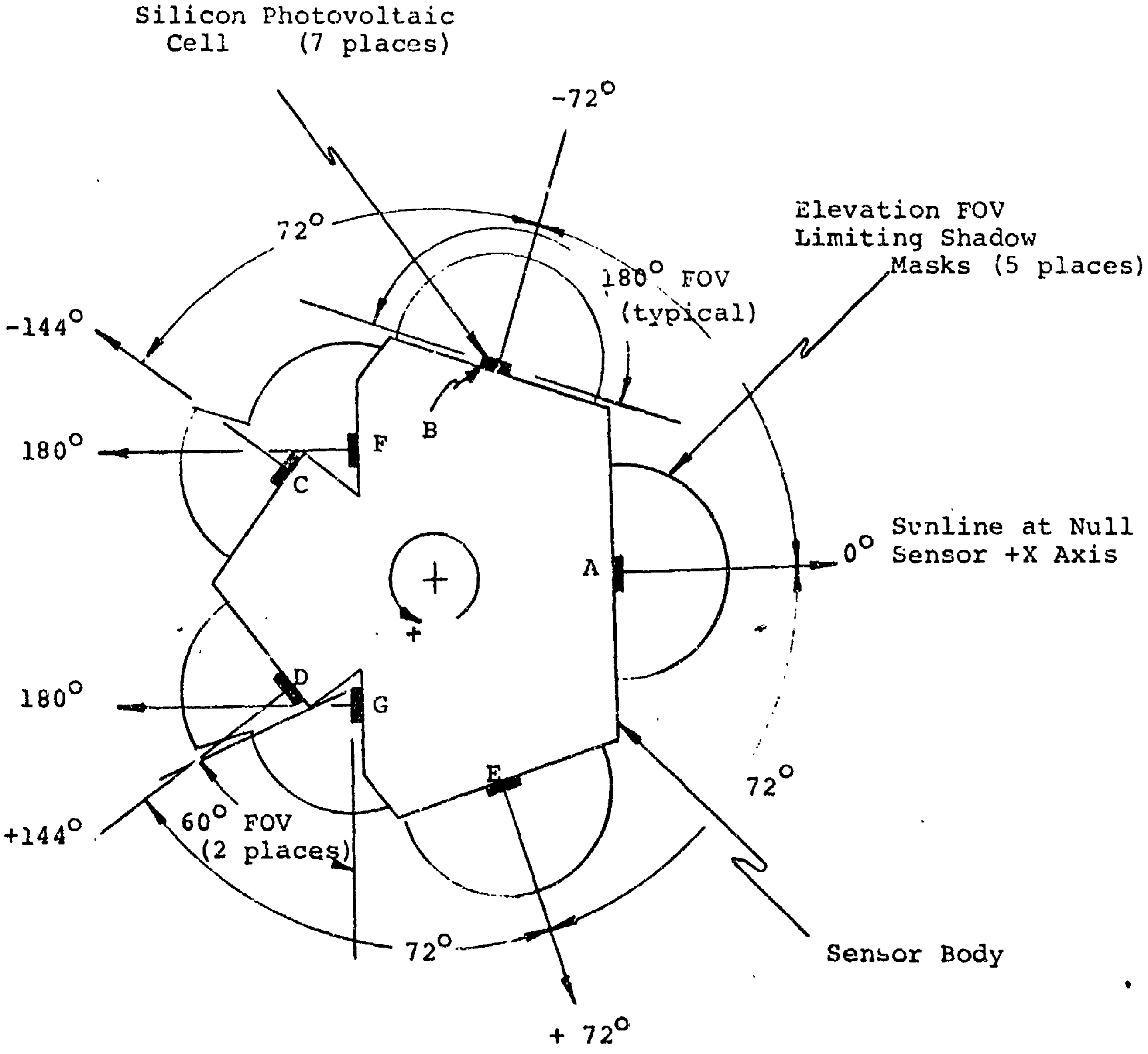
This component is scheduled to be flown as part of the planned 1974 OSO-I mission.

9.3.4-3 OSO-I SAIL SUN SENSOR

The Sail Sun Sensor Assembly for OSO is a fully redundant assembly which provides two sets of four output signals. One of the sets of four output signals is processed by one of the sail control electronic (SCE) units, while the other set is processed by the other SCE unit. The processed signals are used to provide (a) a day/night signal and (b) a coarse azimuth acquisition signal. The sail sun sensor assembly is mounted on top of the sail and has a clear field of view of 360 degrees in azimuth and ± 5 degrees in elevation.

The sail sun sensor assembly is made up of two sensor units mounted on a precision bracket. Each of the sensor units consists of a metal body upon which is mounted seven silicon photovoltaic cells and five sets of shadow masks which limit the field of view of the cells in elevation to reduce the effects of earth shine or other stray light inputs. The arrangement of cells is shown in Figure 9.3.4-6. Cells A, B, C, D, and E each have a full 180-degree field of view in azimuth and their output is nominally proportional to the cosine of the azimuth angle between the sunline and the normal to their respective sensitive surface. They are arranged in a pentagonal arrangement with 72 degrees between cell normals. The azimuth fields of view of cells F and G are restricted to obtain the desired output versus azimuth angle characteristics at large angles from null. Over their nominal 60-degree unrestricted fields of view output is nominally proportional to the cosine of the azimuth angle between the sunline and the normal to their respective sensitive surface. Both cells are oriented with their normals 180 degrees from the sensor null axis.

The output signal from each cell is developed across a nominal 97 ohm load resistor. To reduce the number of wires required between the sail sun sensor assembly and the sail control electronics, the two sets of seven cell output signals are combined into two sets of four sensor output signals. The electrical schematic of the assembly is shown in Figure 9.3.4-7. These outputs are used in combination to provide the day/night and azimuth acquisition signals. The sun intensity (day/night) signal is formed by combining the outputs of cells A, B, C, D, and E. Figure 9.3.4-8 shows the sun intensity output signal at various stages. The azimuth error (azimuth acquisition) signal is formed by combining the outputs of cells B, C, D, E, F, and G. Figure 9.3.4-9 shows the azimuth error output at various stages.



NOTE: View is with the spin (azimuth) axis perpendicular to the plane of the paper.

Figure 9.3.4-6 Sail Sun Sensor Unit Cell Arrangement

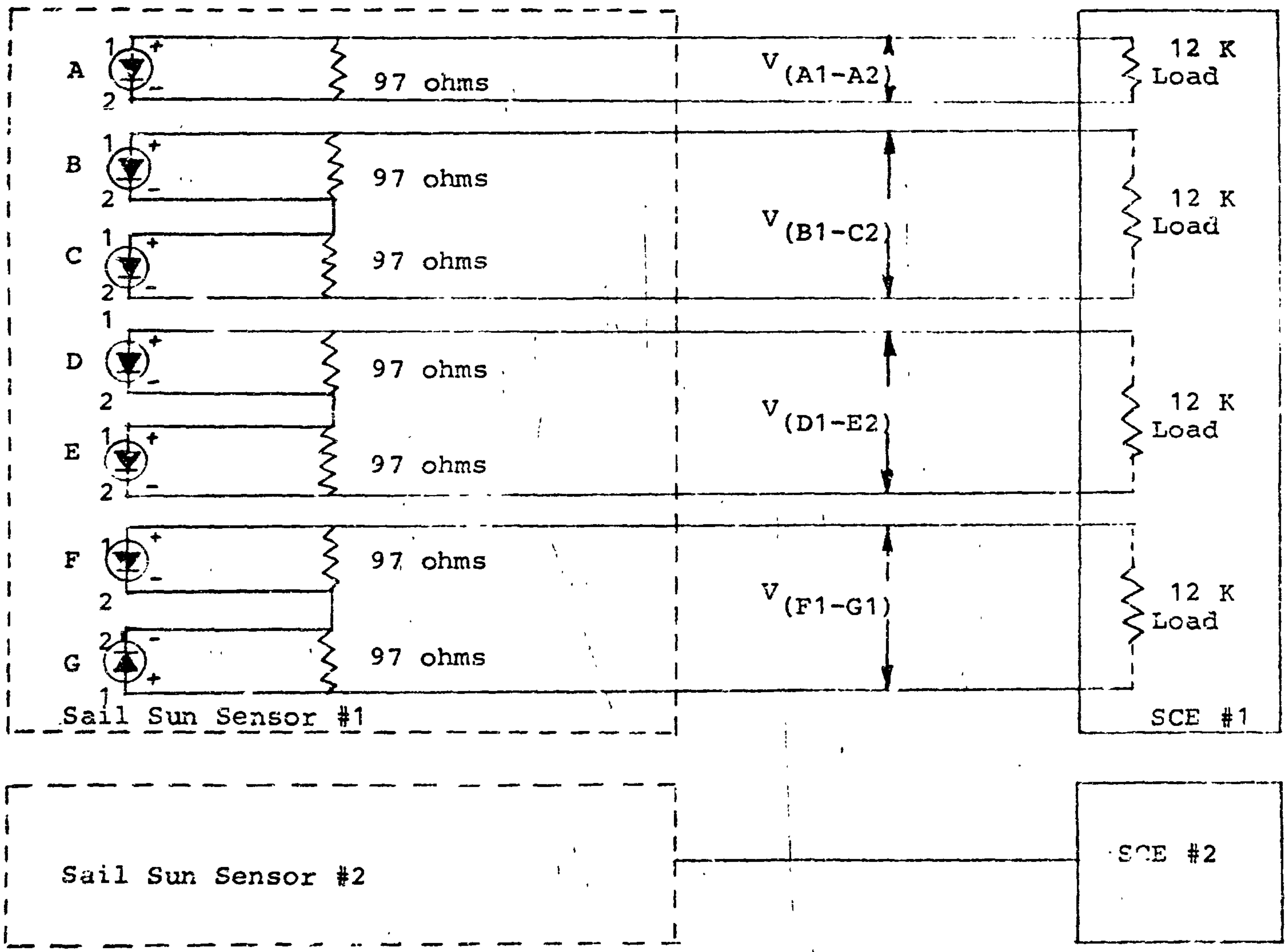
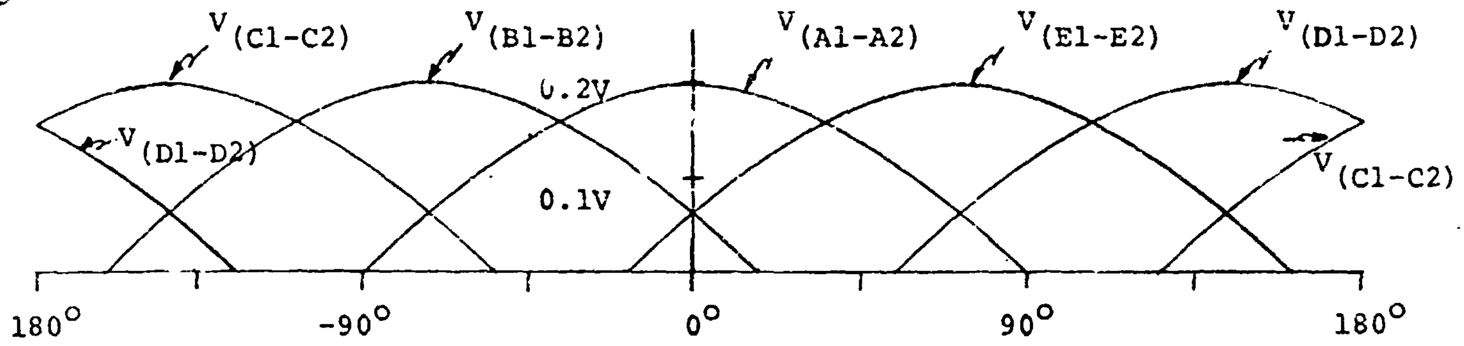
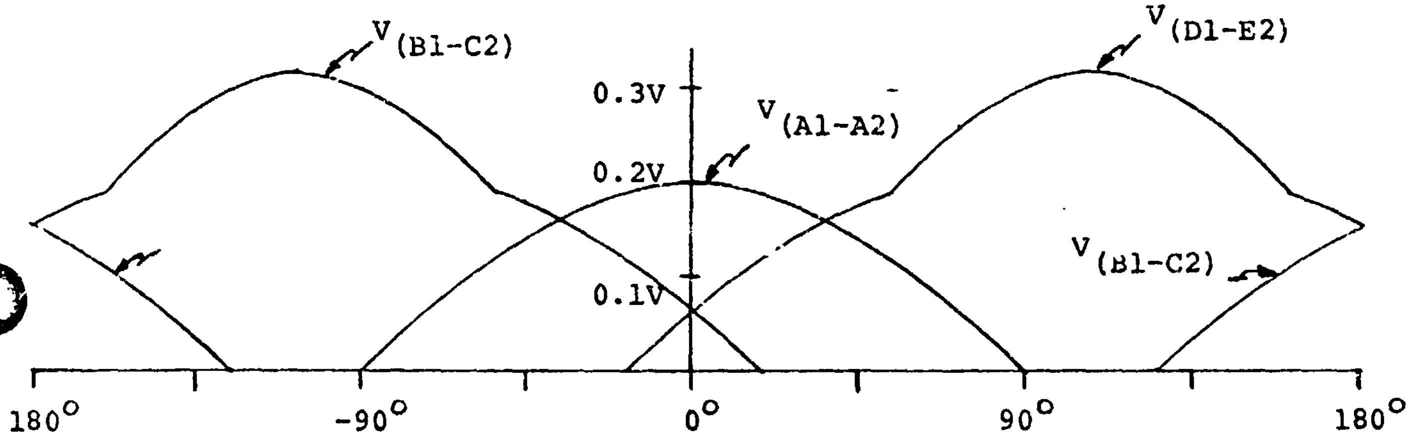


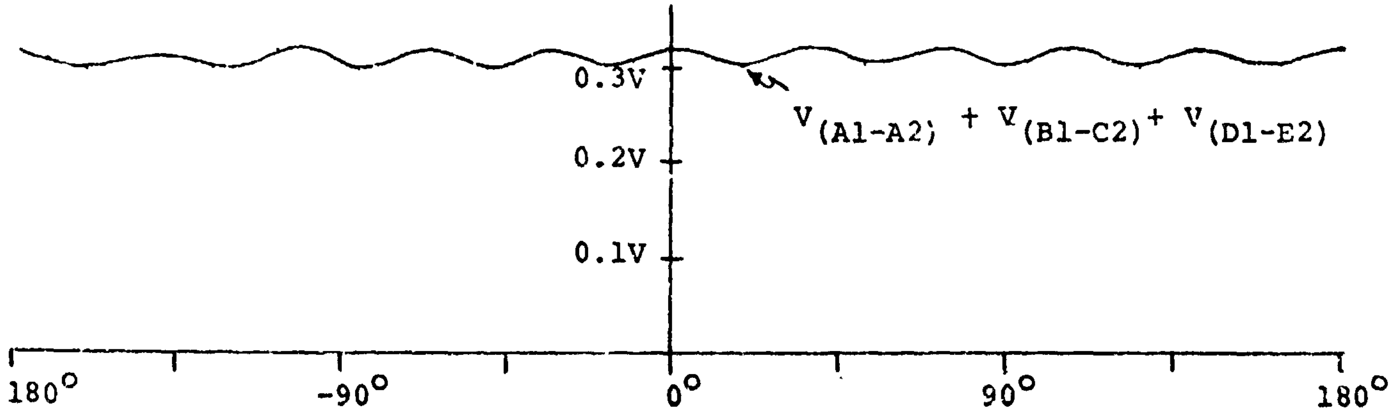
Figure 9.3.4-7 Sail Sun Sensor Electrical Schematic



a) Outputs of Individual Sensor Cells



b) Outputs of Sensor Assembly



c) Output of SCE

Figure 9.3.4-8 Sun Intensity Signal Vs. Azimuth Angle

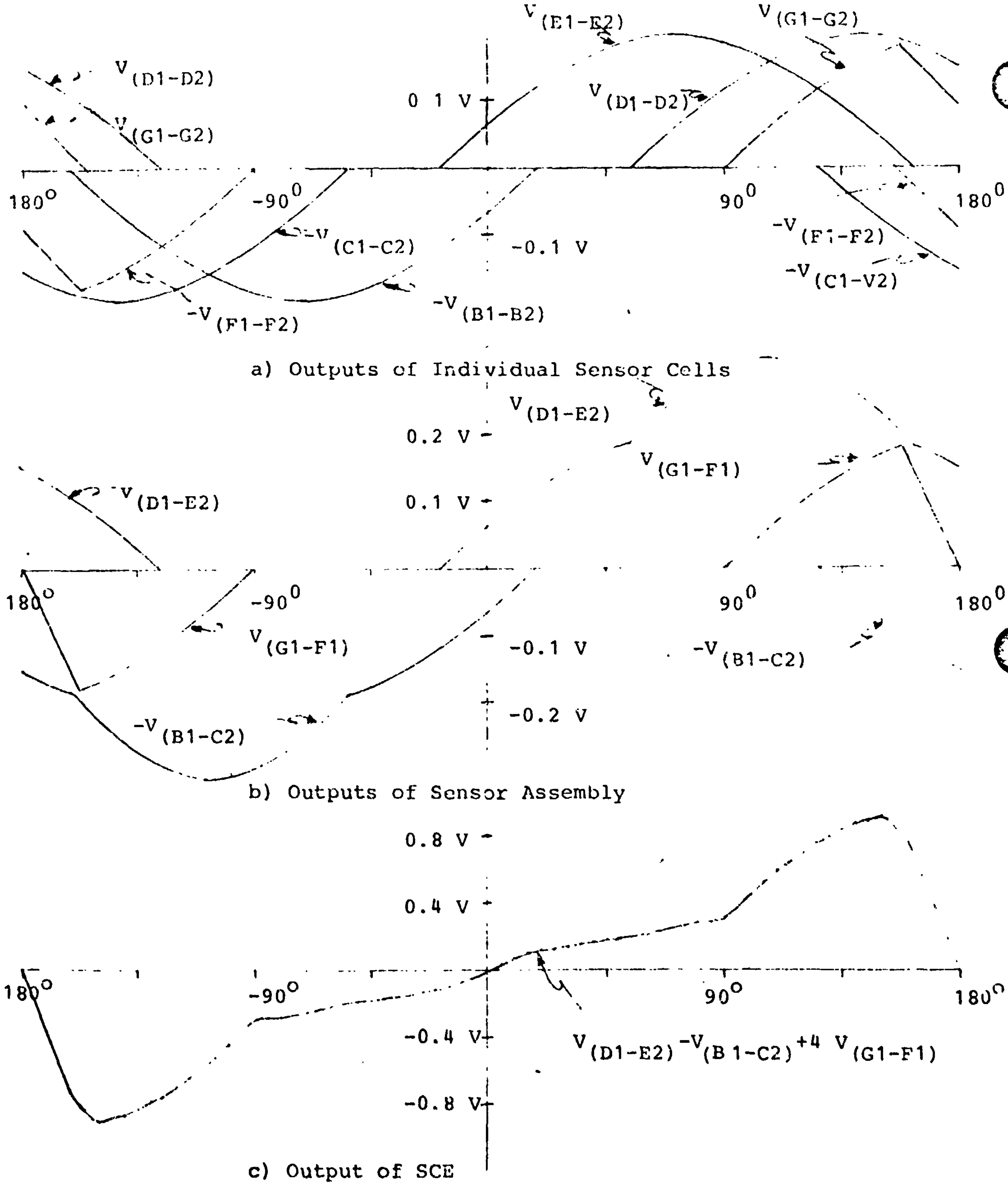


Figure 9.3.4-9 Azimuth Error Signal Versus Azimuth Angle

9.3.6 OAO-C SOLAR ASPECT SENSOR (SAS)

The solar aspect sensor equipment is used to determine two angles which describe the position of the sun vector with respect to the observatory. Information uniquely describing the sun vector in half degree increments (about sensor zenith) is presented by the electronics unit in digital form. In addition, an analog voltage roughly proportional to the cosine of the sun angle is used in establishing the threshold level of the sensor.

A sensor assembly consists of two reticle units mounted together as shown in Figure 9.3.6-1. Each reticle is a quartz block containing a slit and an encoder-type surface pattern (Figure 9.3.6-2). As sunlight impinges on the reticle, the slit forms a "plane of light" which is encoded as it passes through the surface pattern. The encoded light is detected by photo cells mounted below the reticle resulting in an eight bit data signal and a one bit automatic threshold adjust (ATA). The output of the ATA photo cell is used to adjust the threshold in the discrimination of the bit cell outputs and is an analog indication (labelled VATA) of the inclination of the sensor normal to the sun. With the two reticles at 90° to each other, two-axis sun direction is obtained.

The photo cell outputs are amplified and signal conditioned in the electronics unit for transmission to the ground.

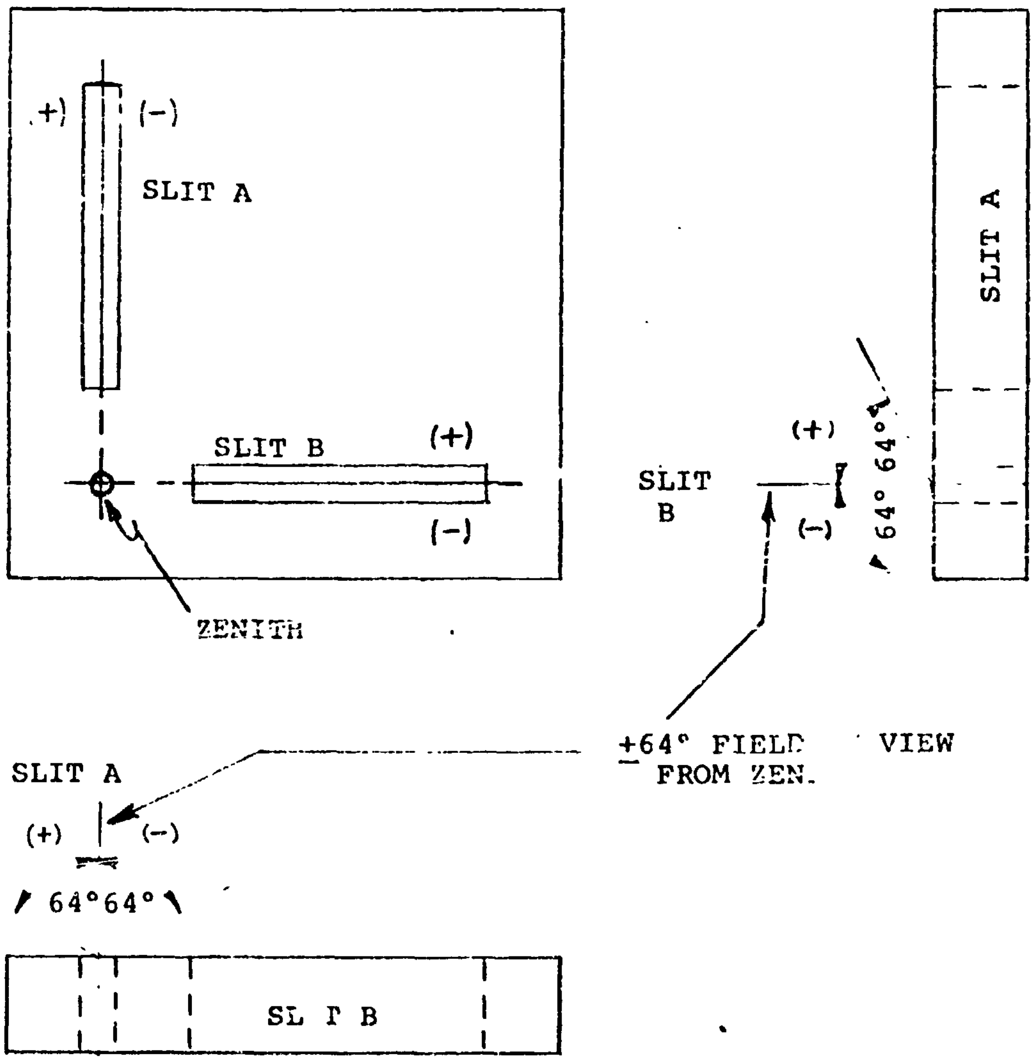
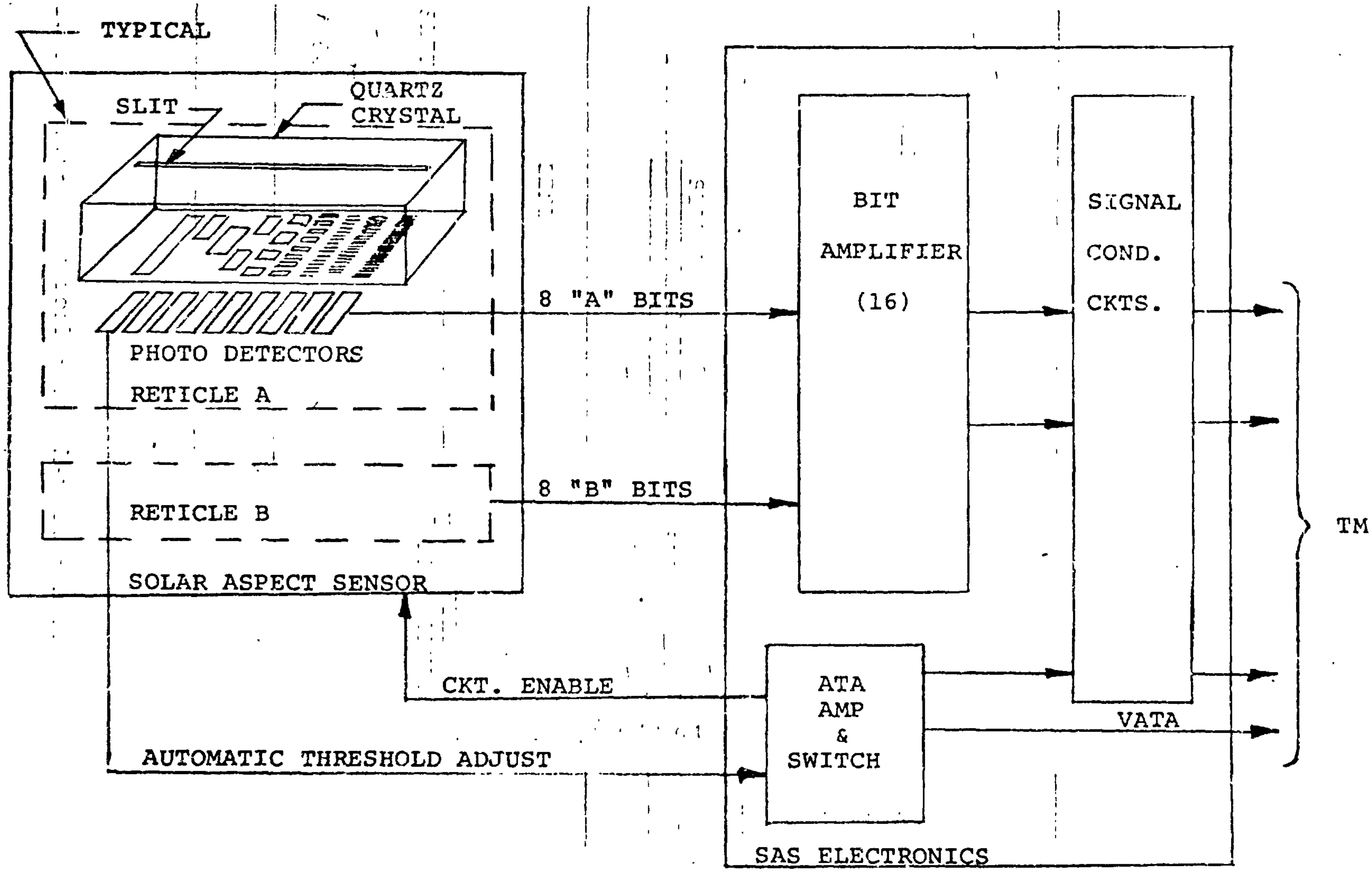


Figure 9.3.6-1 Typical Solar Aspect Sensor



BLOCK DIAGRAM

Figure 9.3.6-2 Solar Aspect Sensor Equipment (SAS)

9.3.6 OAO-C SOLAR ASPECT SENSOR

General Description

Program: OAO-C
Vendor: Adcole Corp.
Part Number: 252SCAV211-1 (GAC)

Performance Characteristics

Field of View: $\pm 64^{\circ}$
Overall Accuracy: $\leq 0.56^{\circ}$
System at steady-state: Within 2 seconds after turn-on.
With an uniform earth albedo of 0.4 and with the sun in the field of view, the solar aspect sensor will always select the sun.

Physical Characteristics

Size:
Weight:
Input Power: 1 watt, +27.0 to +28.56 VDC
Cooling Method: Radiation

References

"Solar Aspect Sensor Equipment Stabilization and Control Subsystem Orbiting Astronomical Observatory Specification for AV-252CS-92", 6 June 1967, with Amendment 1.

"Orbiting Astronomical Observatory Functional Operations Manual", "Stabilization and Control Subsystem", GSFC FO-G-0127-C, August 1972, Volume D.

"Environmental Specification", ET-252CS-39, EMC252CS-83, AV-252CS-30.

Comments

Operating life: >12000 Hours

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.3.8 SKYLAB ATM ACQUISITION SUN SENSOR

The acquisition sun sensor (see Figure 9.3.8-1) provides attitude information for the X and Y control axes of the vehicle. The sensor is an array of detectors which generate solar pointing error signals. Each detector is an electro-optical device which converts solar radiant energy into an electrical current. It consists of a photovoltaic cell, a lens, and baffles which control the geometry of the input-output characteristics. The system (sensor plus electronics) provides electrical analog signals proportional to the solar pointing error of the orbiting assembly (OA) and a sun presence signal to indicate when the sun is in the field-of-view of the error sensors.

The sensor assembly contains two pairs of solar pointing detectors and a target detector (Figure 9.3.8-2). Each detector pair produces an output current which is proportional to the offset error between the sensor block pointing axis and the solar vector. The two pairs of detectors are used as an energy balance null sensor. Using photovoltaic cells, radiant energy from the sun is gathered optically, divided into four paths (two per axis), and then converted into electrical signals. The optical elements of the cells are designed such that when the sensor assembly is aligned to the sun equal amounts of energy are shunted along each optical path resulting in equal electrical signals being generated. The signals associated with each axis are differenced to produce a null indication. The spectral and spherical aberrations of the detector optics produce error signals which vary linearly with sensor-sun misalignment over a limited range.

The target detector is aligned such that it produces an output signal when the sun is within the field-of-view of both pairs of solar pointing detectors. Externally, it is identical to the energy balance detectors but functionally, it is quite different. An internal optical baffle limits the unit's field-of-view in such a way that its output is essentially binary: high when the solar pointing detectors are operational and low otherwise.

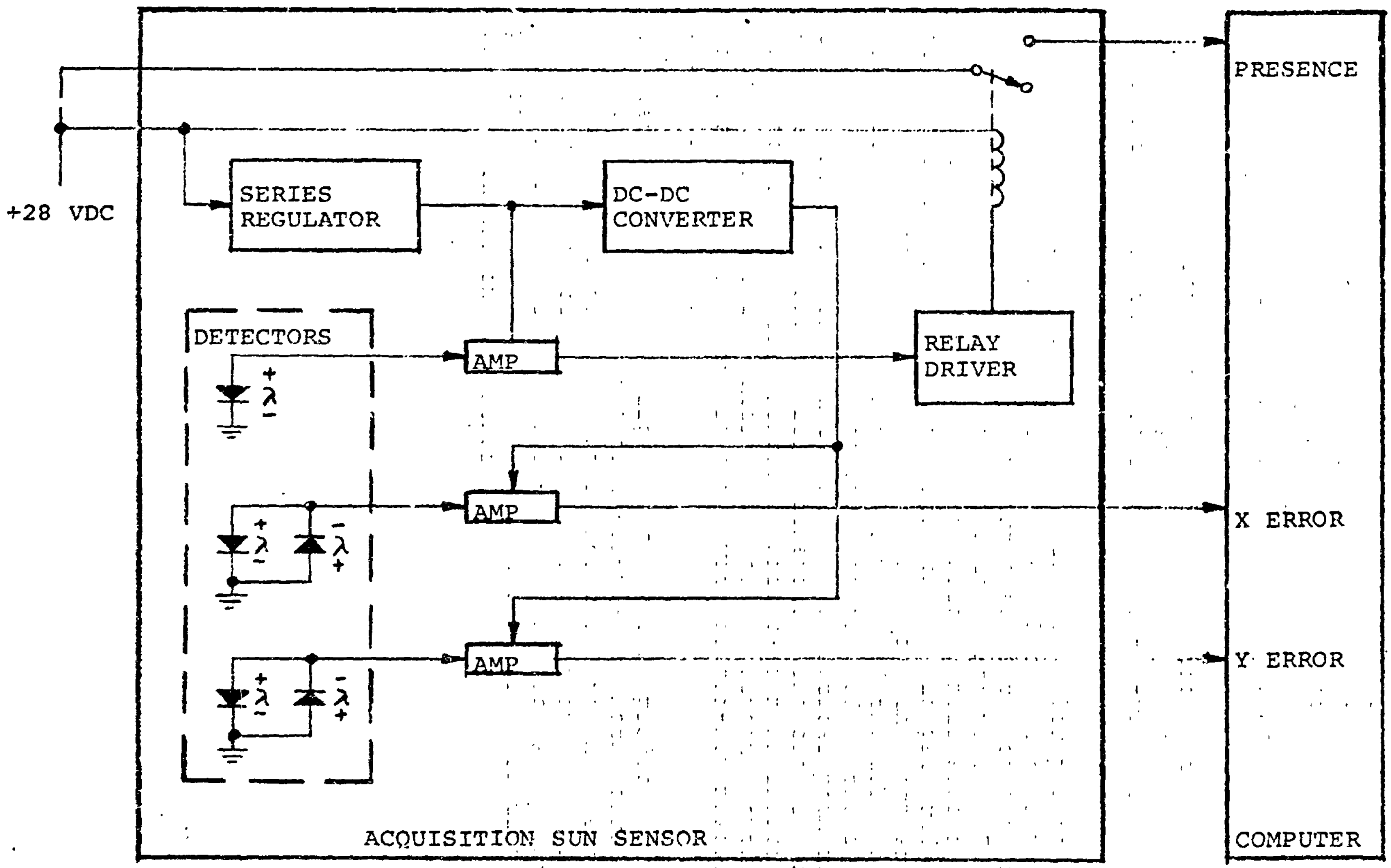


Figure 9.3.8-1 Acquisition Sun Sensor Functional Diagram

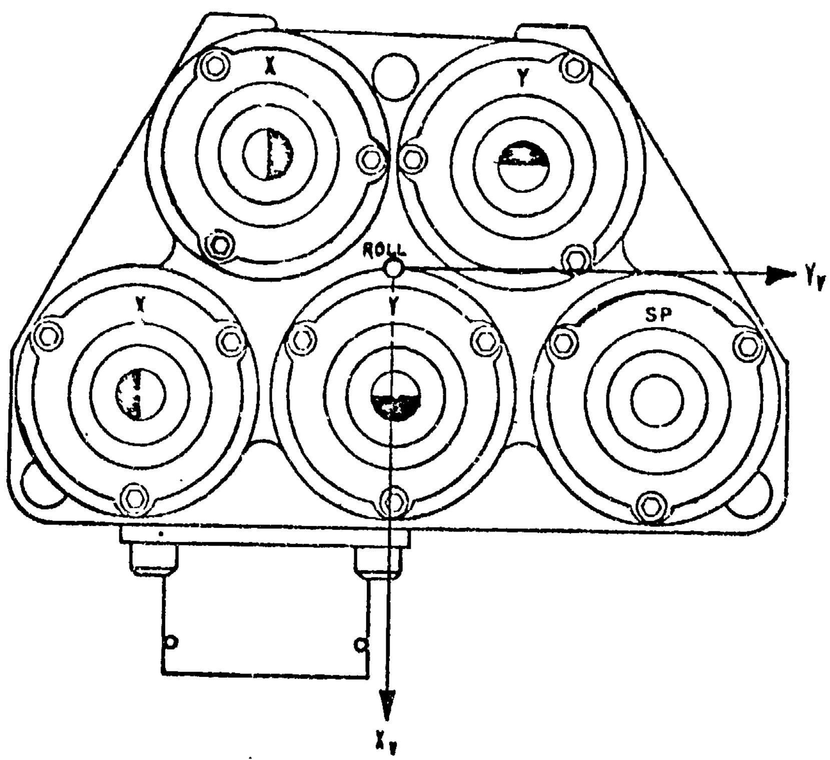


Figure 9.3.8-2 Acquisition Sun Sensor Detector Block

9.3.8 SKYLAB ATM ACQUISITION SUN SENSOR

General Description

Program: Skylab ATM
Vendor: Ball Brothers Research Corporation
Part Number: 50M22140 (Detector Block)
50M22141 (Sensor Electronics)

Performance Characteristics

Fine Null Sensor (both axes):
Field of View: ±10 degrees minimum (full output)
±20 degrees (20 percent output)
Linear Range: ±5 degrees minimum (±6 degrees typical)
Null Accuracy: 2 arc minutes or better at 0 ±5 mv null error
Target Sensor:
Field-of-View: ±9 degrees (circular) ±1 degree

Physical Characteristics

Size: 18 cm (6.9 in) by 17 cm (6.5 in) by 8 cm (3 in) sensor
13 cm (5 in) by 13 cm (5 in) by 5 cm (2 in) electronics
Weight: 2.4 kg (5.2 lbs.)
Input Power: 1.2 w maximum (+28 Vdc)
Cooling Method: Radiation to deep space

References

Skylab Operations Handbook (ATM) - MSFC dated July 19, 1971.

Design Status

This component was flown as part of the 1973 Skylab Mission.

9.3.8-2 SKYLAB ATM FINE SUN SENSOR

The Fine Sun Sensor (FSS) provides the highly accurate position information for the X-and Y-axes of the Experiment Pointing Control (EPC) system. The FSS comprises four separate packages: an Optical-Mechanical Assembly (OM), Preamplifier Electronics Assembly (PEA), Control Electronics Assembly (CE) and the FSS Signal Conditioner (SC) interconnected as shown in Figure 9.3.8-3.

Four single-axis trackers are used, two of which are aligned along the X-axis of the canister and two along the Y-axis. The redundant tracker per axis is provided to improve system reliability.

The primary and redundant systems of the OM are housed in one package. Similarly, the redundant electronics for each of the other three packages is housed with its respective primary system. If a malfunction occurs in the primary system, the complete redundant system is switched in, as there is no facility for interconnecting assemblies of primary and redundant systems.

Optical-Mechanical Assembly - A single optical channel of the FSS comprises three subassemblies: a spectral filter and alignment wedge, a deviation wedge assembly, and a critical angle prism (CAP) sensor assembly.

The spectral filter and alignment wedge filters the incoming sunlight to allow the FSS to operate in its designed spectrum of 0.8 micron to 1.0 micron. This operating spectrum was chosen to minimize errors caused by solar disturbances. The spectral regions of most intense flare activity are in the hydrogen (Hα) lines at approximately 0.6563 micron and are rejected by the filter. The long wavelength is set at one micron primarily due to the selecting of silic[illegible] as the detector. The deviation wedge assembly refracts the sunlight a fixed angle in a controllable direction.

The pitch and yaw wedges define a ± 24 arc minute scanning field. Scanning and offset pointing are accomplished by rotating the assembly in the optical path of the CAP, thus providing an optical gimbal to geometrically shift the null axis. To maintain accurate position and rate control of the wedge, a digital code wheel is attached to and rotates with the wedge.

The wheel contains three tracks. Each track comprises alternate opaque and transparent segments deposited radially. Sunlight passing through the wheel is detected by a bank of silicon cells. One track provides the direction of rotation, the second provides the wedge position, and the third carries a zero position reference mark. Total angular motion is determined by counting the transitions from light to dark on the encoder. These incremental sine-cosine signals will be amplified in the PEA and then converted in the SC into binary words. These words will go to the ATMDC when an interrogate pulse and 12 clo[illegible] pulses are received from the ATMDC. This information is use[illegible] display purposes and when certain experiments are perf[illegible]

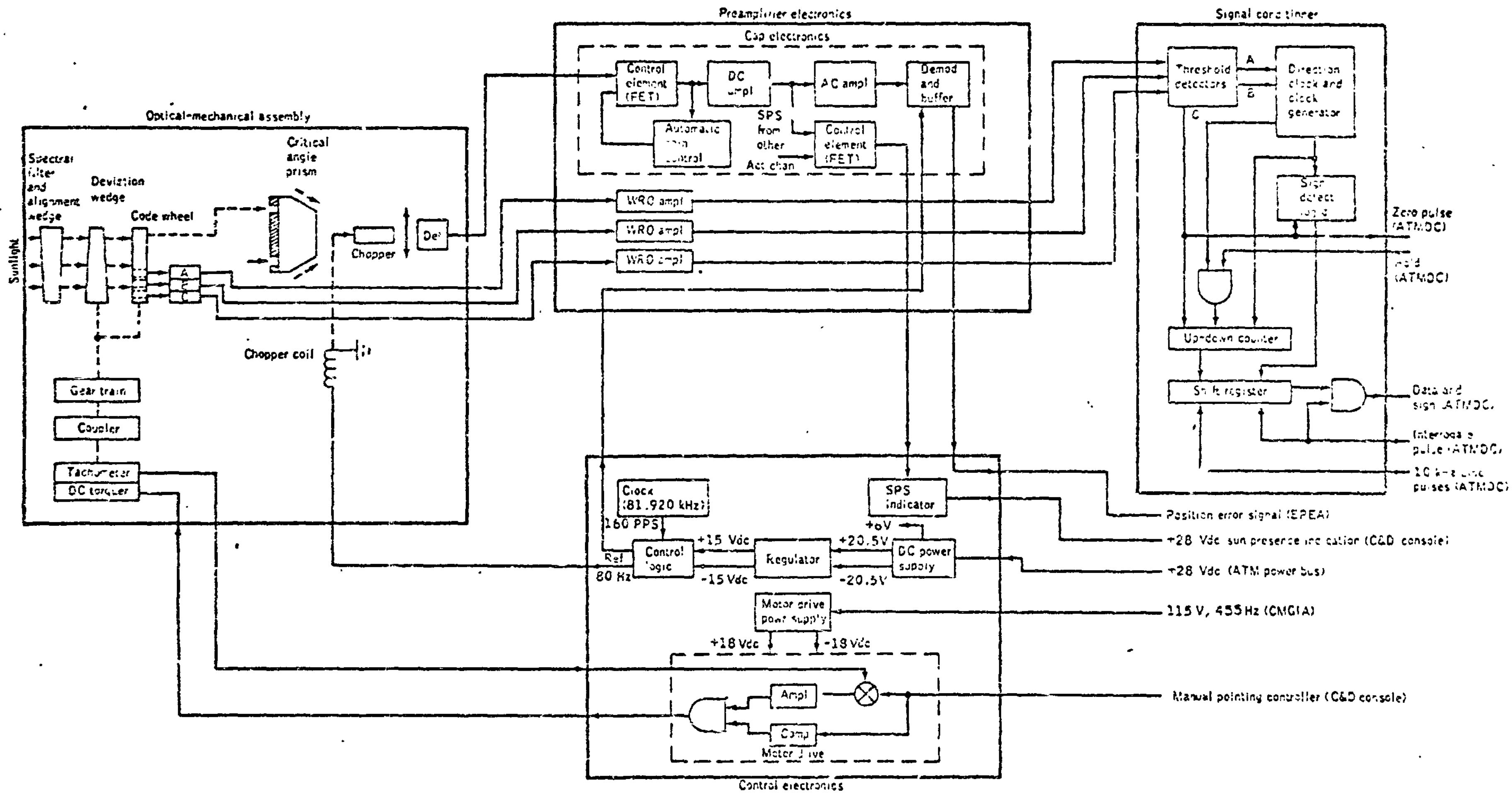


Figure 9.3.8-3 SKYLAB ATM FINE SUN SENSOR FUNCTIONAL DIAGRAM

The CAP comprises a quartz "critical angle prism" used for position error detection. The prism has isosceles angles cut close to the critical internal reflectance angle for radiation entering perpendicularly to the front face (i.e., if the sunlight enters perpendicularly to the prism base, it is almost totally reflected). A small equal amount of light passes through each of the two isosceles faces of the prism. A vibrating reed chopper allows a single silicon detector to alternately look at the radiation from each face. If the incident light is not perpendicular to the front face, more light passes through one isosceles face than the other because the critical angle was exceeded.

Preamplifier Electronics Assembly - The resultant electrical signal from the CAP silicon detector is a modulated dc voltage where the ac component is directly proportional to the attitude error. The dc component of the detector is compared with a dc reference. The difference is amplified and used to control the impedance of a field effect transistor (FET) which in turn diverts current from the input of the signal channel to restore a predetermined dc voltage level to the silicon detector output. Whenever the input solar energy drops below the range of the automatic gain control (AGC) loop authority, the dc signal on the AGC output begins to decrease and a threshold detector at the output of the AGC circuit provides an occult signal which deenergizes the sun presence relay.

The detector signal is amplified in two stages, one dc and one ac and then demodulated with reference to the chopper position to provide a dc voltage proportional to the position error.

The PEA also houses the wedge readout amplifier that amplifies the incremental sine-cosine encoder signal.

Control Electronics Assembly - The CE functions are to provide: power conversion from the spacecraft to that required for FSS operation; process manual pointing controller signals from the spacecraft to drive the FSS optical "gimbals" for pointing control; and provide motor drive electronics and a demodulator reference signal for processing the FSS error signal. The control electronics comprises the FSS power supply, the clock and control logic, and the motor and drive circuitry that includes a power supply.

The FSS power supply is a dc-to-dc converter and regulator, that provides $\pm$15 vdc for all the operational amplifiers and +6 vdc for the logic circuits.

The clock and control logic provides the 80 Hz and 160 Hz keying signals required to drive the chopper and CAP demodulator. These frequencies are derived by counting down an 81.920 kHz signal obtained from a crystal oscillator.

The motor and drive circuitry, which positions the wedges for offset pointing, is a direct current servo loop that is closed through the spacecraft and crewman by means of the MPC. An electronic switch, which closes only when a drive command is given, is included in the motor drive circuit. This is done primarily to eliminate the possibility of wedge readout error. When emerging from SI night, the wedge should be cycled through C (zero) track to reset the wedge readout signal. The motor drive power supply converts the 130 V, 455 Hz power from the CMGIA to ±18 vdc unregulated power. One power supply provides the power for two motor drive amplifiers.

FSS Signal Conditioner - The FSS SC houses the logic circuitry to convert the amplified wedge readout signal to a binary word. The three tracks from the code wheel are conditioned with three threshold detectors (operational amplifiers). The conditioned signals A and B are used to develop a one millisecond clock pulse for each negative and positive transition. Each generated clock pulse is used to either add or subtract one count in the up-down counter, depending on the state of the "up" control signal from the direction logic.

The digitized position of the wedge encoder in the up-down counter is loaded into an independent shift register to be shifted out whenever an interrogate pulse and 12 clock pulses are received from the ATMDC. The zero pulse (C channel) output to the ATMDC will be greater than 15 milliseconds wide at maximum drive rate. The zero pulse is also used to reset the counter and control logic. The "hold" pulse discrete will lock the counter and retain the position data. When the discrete is removed, the up-down counter will resume operation.

A direct-on (sign) bit is furnished to the ATMDC as the most significant bit (MSB) of the serial binary number representing the encoder position. Buffer logic is provided on input-output signals to reduce impedance and prevent loading.

9.3.8-2 SKYLAB ATM FINE SUN SENSOR

General Description

Program: Skylab ATM
Vendor: Honeywell
Part Number: 21001918

Performance Characteristics

Sensitivity	176 ±32 mV/arc-sec
Output Voltage Range	±12.5 ±2.5 V, (±14 volts nominal)
Pointing Accuracy	±2.25 arc sec (max)
Pointing Resolution	±0.625 arc sec (nominal)
Pointing Stability, Short Term	±0.1 arc sec (nominal)
Pointing Stability, Long Term	±0.5 arc sec (goal)
Temperature Stability	±0.2 arc sec/deg [illegible] (nominal)
Field-of-View	5.0 ±.5 deg ≥0 θ ≥80 arc min
Spectral Response	0.8 microns to 1.0 microns
Frequency Response	10 Hz
Offset Pointing Capacity	±24 arc min (each axis)
Offset Point Drive Rate ($\dot{\theta}$)	2 arcsec/sec minimum, 80 arcsec/sec average

Physical Characteristics

Size Optical Mechanical -33 cm (13 in) by 20.3 cm (8 in) by 19.1 cm (7.5 in)
Preamplifier Electronics -23.2 cm (9.1 in) by 15 cm (5.9 in) by 11.5 cm (4.5 in)
Control Electronics -27.9 cm (11 in) by 25.4 cm (10 in) by 11.5 cm (4.5 in)
Weight: 21.1 kg (46.6 lbs) total
Input Power: 23.5 w (peak) 10.25 w (average) (+28 Vdc)
Cooling Method: Radiation to deep space

References

Apollo Telescope Mount Fine Sun Sensor Signal Conditioner 1968 Operational Requirements and Description - MSFC 50M22144, dated August 15, 1968.

System Description Apollo Telescope Mount Fine Sun Sensor - MSFC 50M22156 dated December 9, 1970.

Comments

High density welded discrete component electronics.

Design Status

This component was flown as part of the 1973 Skylab Mission.

9.3.11 MARINER MARS SUN SENSOR

The Mariner Mars sun sensor consists of cruise and acquisition sensors. These are electro-optical devices, designed primarily to provide two single axis error signals, measuring the angular deviation of the sun line from the negative roll axis of the spacecraft. These measurements are made in the form of two component rotations about the pitch and yaw axis respectively (Figure 9.3.11-1). These two signals control the attitude of the spacecraft about the pitch and yaw axes. (Figure 9.3.11-2). The total combined field-of-view of the cruise and acquisition (pitch and yaw) sun sensors is 4 Pi steradians. The cruise sensors are defined as those covering the null region and have a conical field-of-view of 4.50 degrees total angle. The acquisition sensors cover the region outside of the null region.

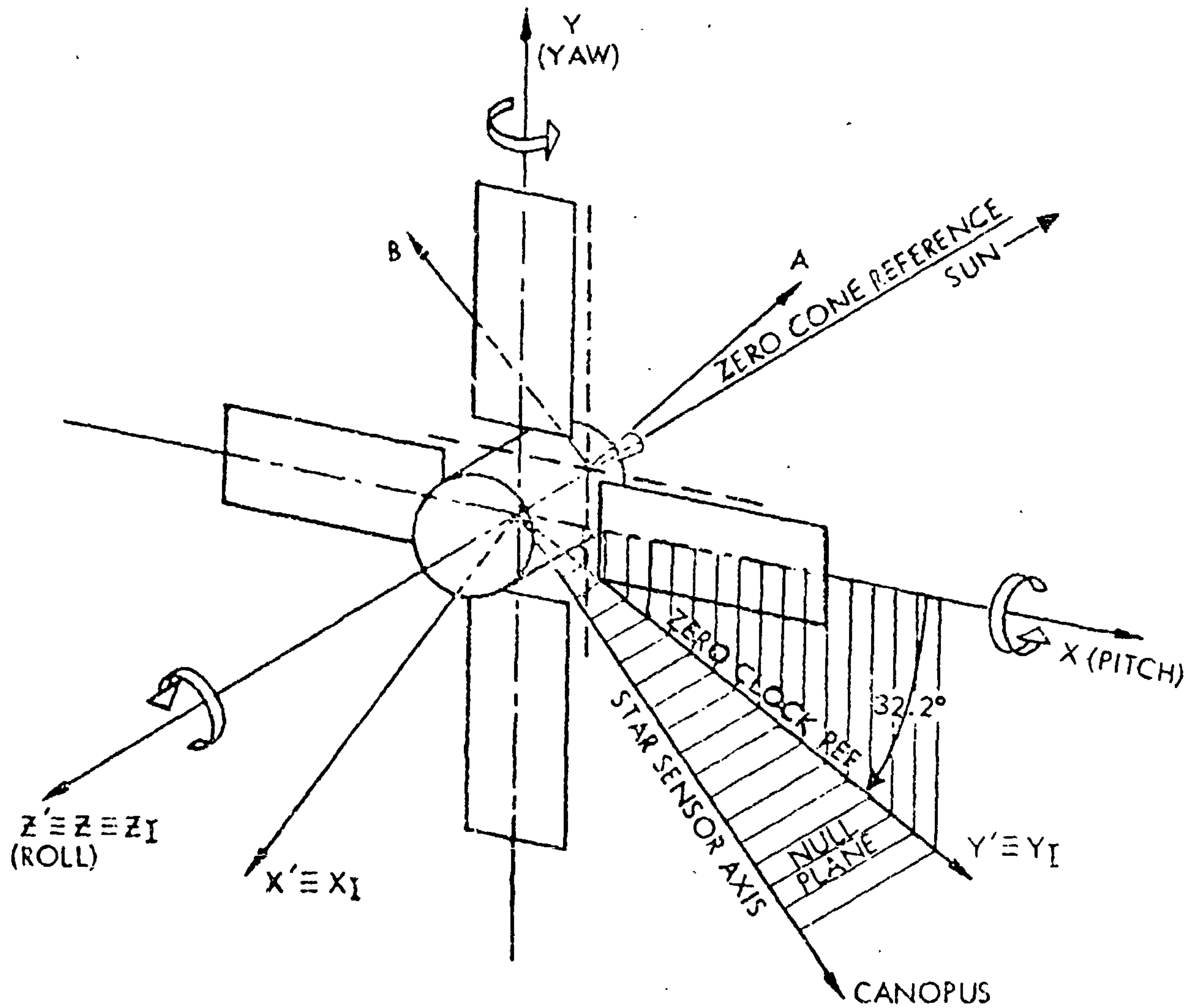


Figure 9.3.11-1 Mariner Mars Configuration and Coordinates After Celestial Reference Acquisition

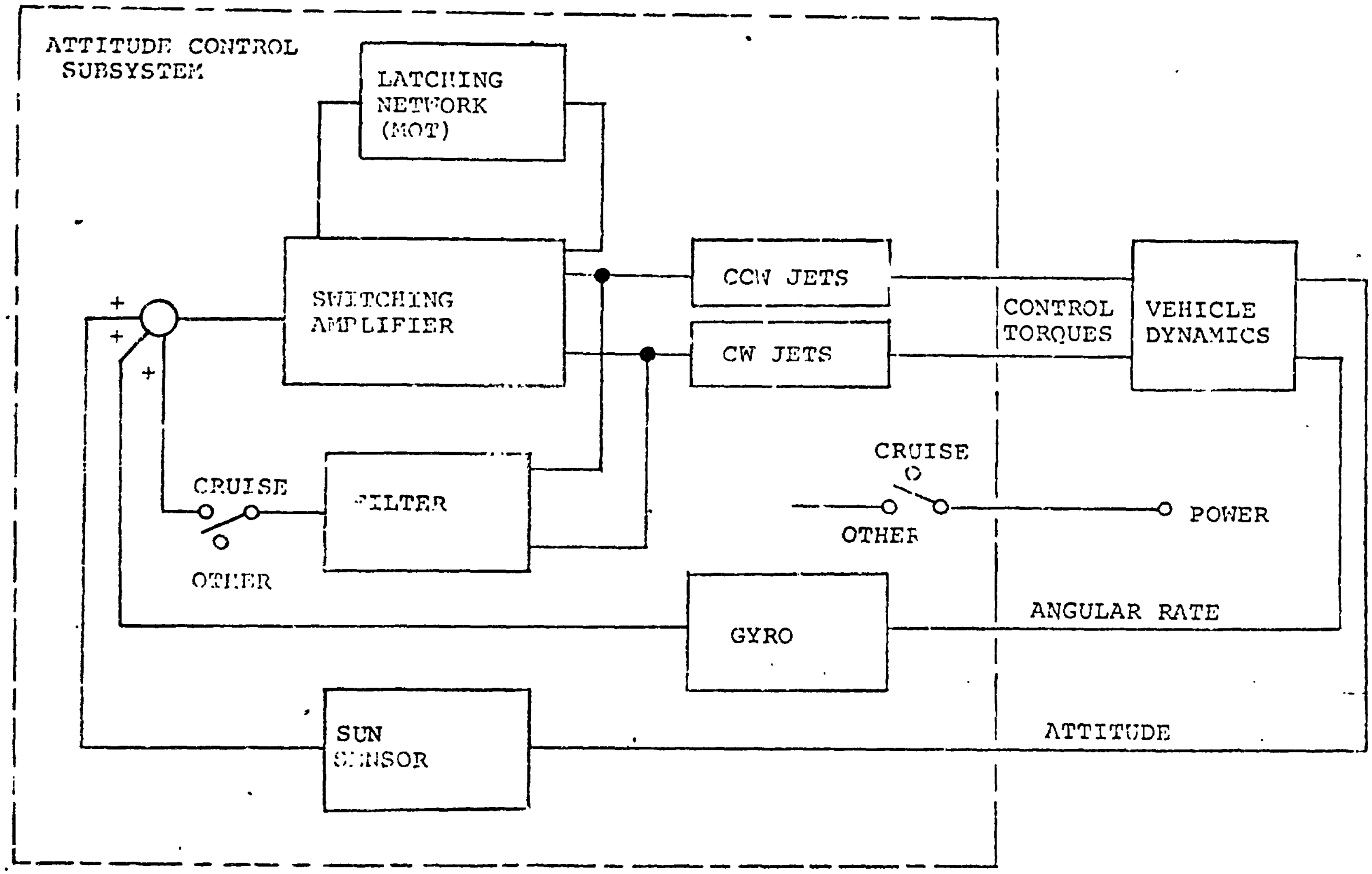


Figure 9.3.11-2 Pitch and Yaw Control in Acquisition and Cruise Conceptual Block Diagram

9.3.11 MARINER MARS SUN SENSOR

General Description

Program: Mariner Mars 71
Vendor: JPL
Part Number:

Performance Characteristics

Acquisition sensors have an unobstructed view of 4 Pi steradians in x and y. The cruise sensors have an unobstructed field-of-view of $\pm 22^{\circ}$.

Input: The sun sensors measure angular rotations about the pitch and yaw axes.

Output:

- o The output is monotonic between $\pm 1.0^{\circ}$.
- o The scale factor at $\pm 0.25^{\circ}$ is -4./5 Vdc per degree $\pm 10\%$ into a 54.8×10^{3} ohm load.
- o The combined mechanical and electrical null offset is less than 0.08° relative to the mounting plane.
- o The rise time to a step input is 0.01 second, maximum.
- o The output noise level is less than 0.05 volt, peak-to-peak.

Physical Characteristics

Size:
Weight:

References

Mariner Mars 1971 Flight Equipment Attitude Control System, M71-2007-1, Rev. A 19 May 1970.

Design Status

This component was flown as part of the 1971 Mariner Mars Mission.

9.4.4 OSO-I GYRO REFERENCE ASSEMBLY

The OSO-I gyro reference assembly (OGRA) mounted on the despun sail section, is used in conjunction with procession control electronics (see Figure 9.4.4-1) to provide the following spacecraft functional capabilities:

a) Precision inertial azimuth reference for command pointing of the sail-mounted instruments during orbit night operation.

b) Backup orbit day azimuth reference in the event of failure of the primary sun sensor.

c) Inertial azimuth rate sensing (in the caged mode) for initial sclar acquisition maneuvers.

During normal, sun-pointing, orbit day operation the OGRA is caged. A drift compensation bias voltage is supplied to the OGRA and is periodically updated. Upon entry into orbit night the OGRA is uncaged and used by the sail azimuth servo for inertial pointing reference in any of the following operating modes.

a) Sun hold - The sail holds the sun-pointing azimuth attitude through orbit night.

b) Offset Pointing - The OGRA can be precessed at $+2^{\circ}$/sec or -2°/sec, for a precise commanded time, by a torquing command supplied to the OGRA, to enable sail pointing to an arbitrary azimuth direction.

c) Sail Scan - The OGRA can be precessed at $+0.25^{\circ}$/sec, by a torquing command supplied to the OGRA, to enable one complete sail revolution during an orbit night period.

Upon re-entry into orbit day, the OGRA will be caged, and the sail azimuth control servo will be transferred to primary sun sensor reference.

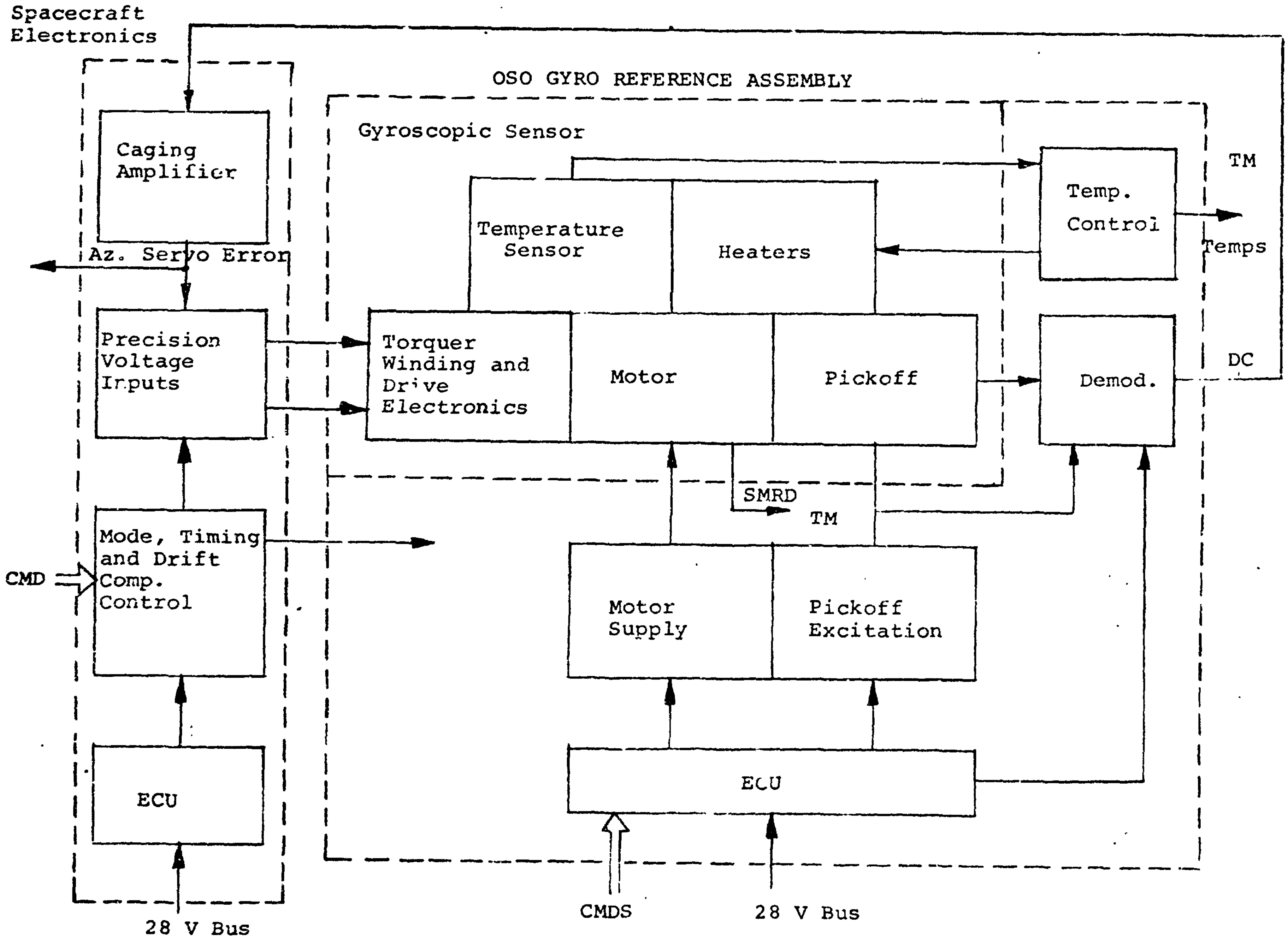


Figure 9.4.4-1 OSO-I Gyro Reference Assembly

9.4.4 OSO-I GYRO REFERENCE ASSEMBLY

General Description

Program: OSO-I
Vendor: Hughes Aircraft
Part Number: 3280405-100

Performance Characteristics

Gyro Drift Rate

Constant: 1°/hr at delivery after one cool down to 32° F; 0.5°/hr delta between operation prior to and after 32°F cool down.

0.12°/hr delta between operation prior to and after each room ambient cool down.

Random: $\pm .06^{\circ}$/hr max over 4 hr period
$\pm .01^{\circ}/hr/^{\circ}F$ max

Elastic Restraint: 0.5°/hr max

Hysteresis: 0.02°/hr max

Torque Rate

Bias Command: $\pm 20^{\circ}$ hr max

Slew Command: $\pm 2^{\circ}$/sec max

Physical Characteristics

Size: 1148 cu. cm (70 cu. in.)
Weight: 1.35 kg (3 lbs)
Input Power: 14w max starting, 2.6 w max running
Cooling Method: Heater control

References

Hughes Aircraft Co., Procurement Specification OSO Gyro Reference Assembly, PS 31331-145 Rev. D, dated April 3, 1972

Comments

Gyro warmup heater requires 50 w max.

Design Status

This component is scheduled to be flown as part of the planned 1974 OSO-I Mission.

9.4.6-1 OAO-C RATE GYRO (PART OF GYRO MOMENTUM PACKAGE)

There are three identical Gyro Momentum Packages in the spacecraft, one aligned along each of the three control axes. Each package contains a coarse inertia wheel, a fine inertia wheel and a JRT rate gyro. (Figure 9.4.6-1).

The JRTs are single degree-of-freedom spring restrained gyros. Three individual gyros are oriented to provide independent outputs of pitch, yaw, and roll rates. The output is a voltage proportional to the component of angular rate ($\dot{\theta}$) which is processed by the sensor signal processor (SSP) and the fine wheel jet controller's high thrust jets. Threshold sensitivity is approximately $\pm 0.01^{\circ}$/sec. Saturation rates are approximately $\pm 5^{\circ}$/sec. Output is 0.4 V rms/degree/sec for either direction of the rate gyro.

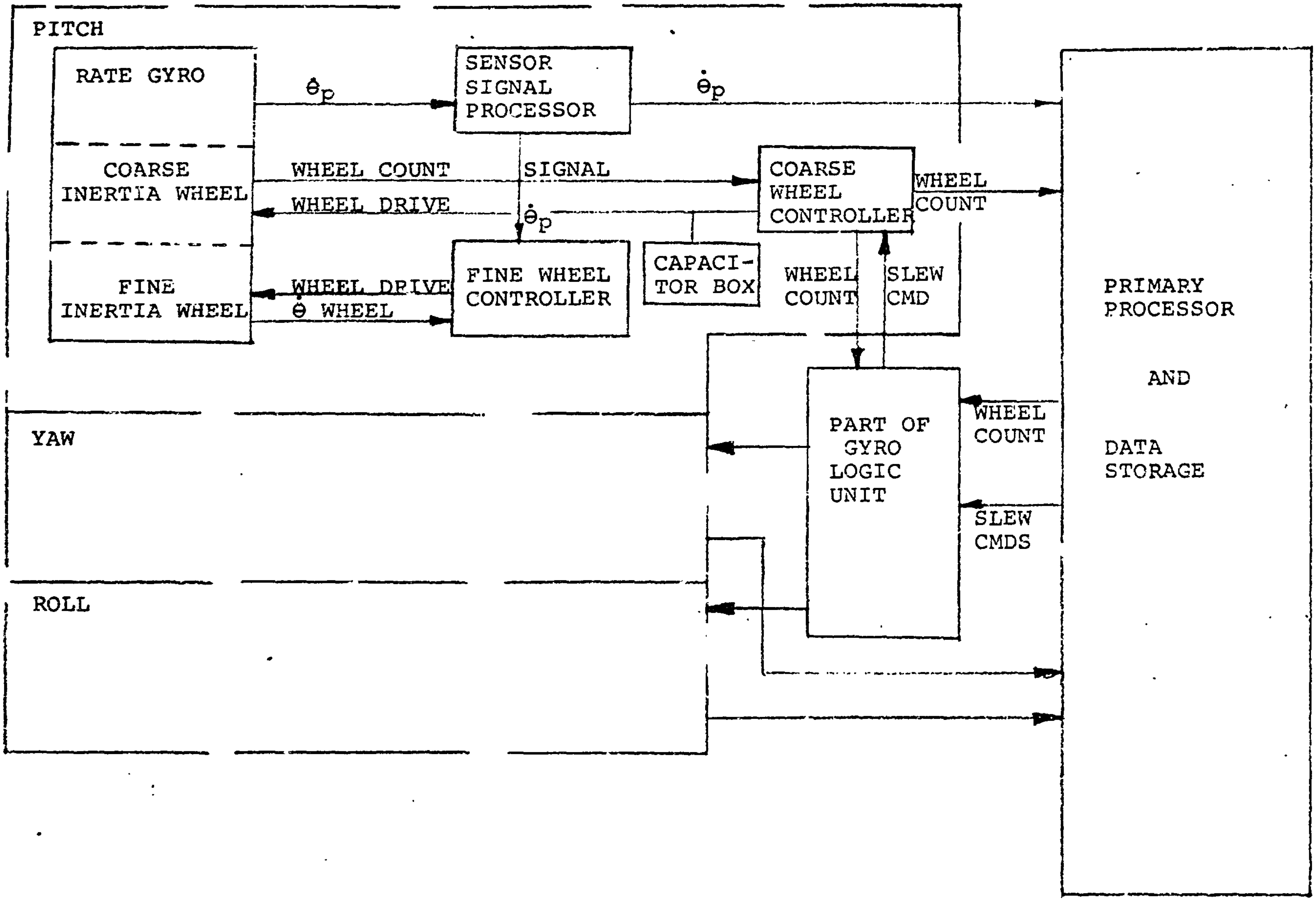


Figure 9.4.6-1 Block Diagram of OAO Gyro-Momentum Interface

9.4.6-2 OAO-C RATE GYRO UNIT

The rate and position sensor equipment (RAPS) is designed to operate independently of the stabilization and control equipment except for the secondary high-thrust pneumatic system. Part of the RAPS is a gyro unit which is a three-axis strap down gyro assembly containing three Kearfott 2564 Alpha rate integrating gyros. The RAPS electronics package contains the gyro electronics comprised of the gyro output amplifier, demodulator, torquer amplifier and output filter, one set for each gyro. Figure 9.4.6-2 is a functional block diagram on the Rate Gyro Unit.

The RAPS uses the rate gyro in either of two modes: rate only or sum of rate and attitude. In the sun-hold mode the gyros and their associated electronics are switched to the rate mode during the daylight portion of the orbit and to their attitude-plus-rate mode during the dark portion of the orbit. In the star-hold mode the gyros are switched to their attitude mode to sense changes in the spacecraft attitude.

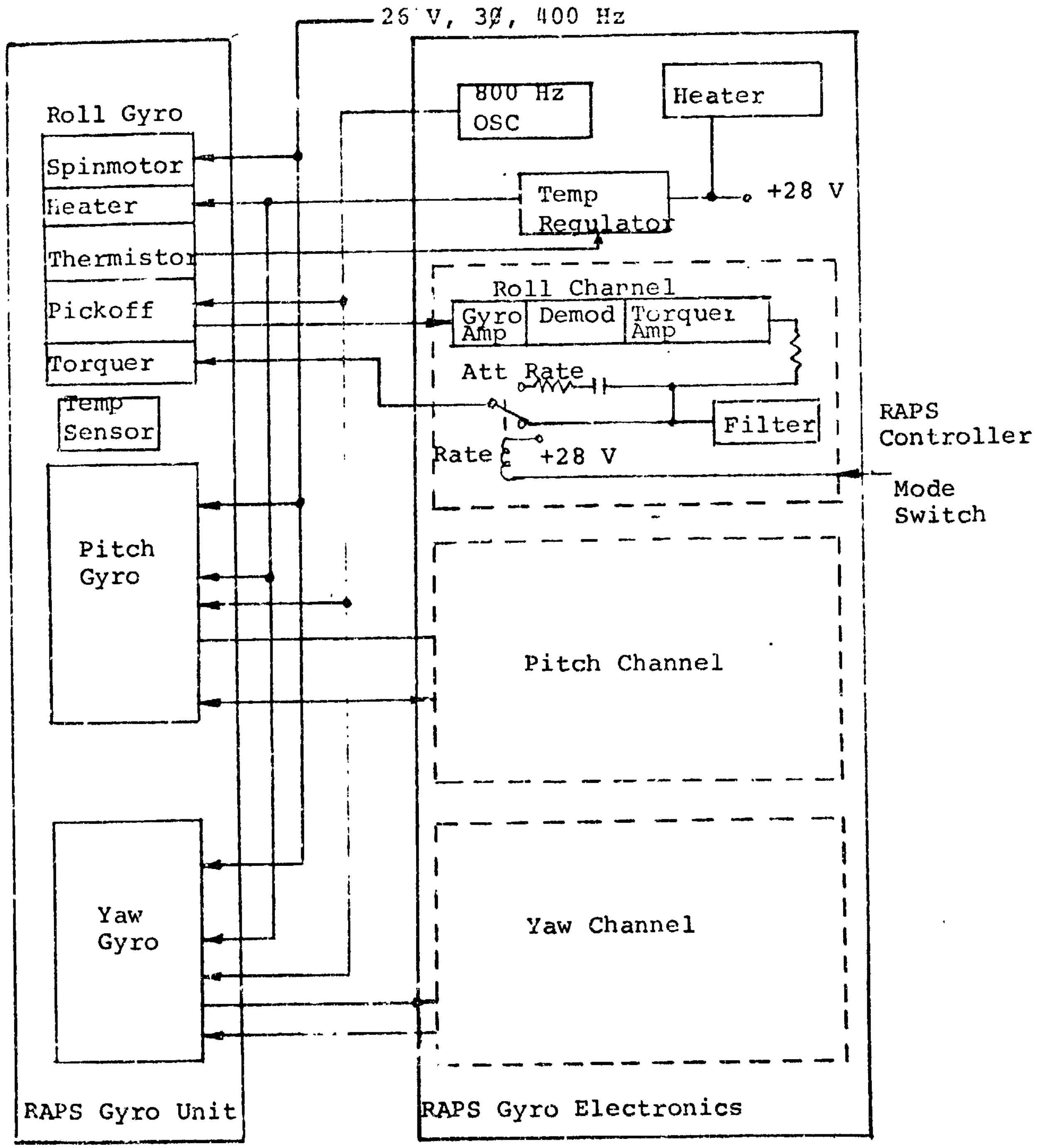


Figure 9.4.6-2 RAPS Gyro and Electronics Block Diagram

9.4.6-2 OAO-C RATE GYRO UNIT

General Description

Program: OAO-C
Vendor:
Part Number: Grumman 252SCAV19C

Performance Characteristics

Gyro Drift:
- Constant 0.25°/hr max
- Day to Day 0.05°/hr max
- Random
  - OAV 0.005°/hr (1 σ)
  - IAV 0.02°/hr (1 σ)
- G-sensitive 0.3°/hr/g max
- Day to day 0.15°/hr/g max
- Anisoelastic 0.02°/h/g^2

Torque Rate: 2°/sec max

Physical Characteristics

Size:
Weight: 3.3 kg (7.3 lbs) (gyro package only)
Input Power: 10 w max(gyro package only)
Cooling Method: Heat sink (Radiate to spacecraft skin)

References

Rate and Position Sensor Equipment, Stabilization and Control Subsystem, OAO, Specification for. Grumman #AV-252CS-73A, dated January 4, 1965, Amendment #1 dated January 31, 1968.

Comments

Gyro operating life > 3000 hrs
Electronics > 12000 hrs

Design Status

This component was flown as part of the 1972 OAO-C Mission.

9.4.8 SKYLAB ATM RATE GYRO

The Rate Gyro Package (RGP) provides rate damping for the control axes of the attitude and pointing control subsystem (APCS) nested control system and the experiment pointing subsystem. The RGPs for the nested control system are rack-mounted; the ones for the experiment pointing subsystem are spar-mounted. The rack-mounted gyro(s) outputs are integrated in the ATMDC (strapdown computation) to provide attitude information.

The ATM rate gyro package consists of a single Kearfott rate integrating gyro (King Series, C702519013) and associated electronics which allow it to operate in a rate mode. It is designed to operate over two selectable ranges of rate inputs. The low range, called fine mode, has a scale factor of 450 volts per degree per second for inputs up to 0.1 degrees per second. The coarse mode has a scale factor of 45 volts per degree per second for inputs up to 1.0 degree per second. These scale factors result in rate output commands of 0 ± 45 VDC to be supplied from each package.

Self test provisions permit the gyro to be torqued electrically by a test signal with no rate input. The resulting gyro output checks the servo loop including the range change operation.

The electronics include a 4.8 kHz generator, AC amplifier, 3-phase inverter, demodulator, power supply, torquer driver, and heater control interconnected as in Figure 9.4.8-1.

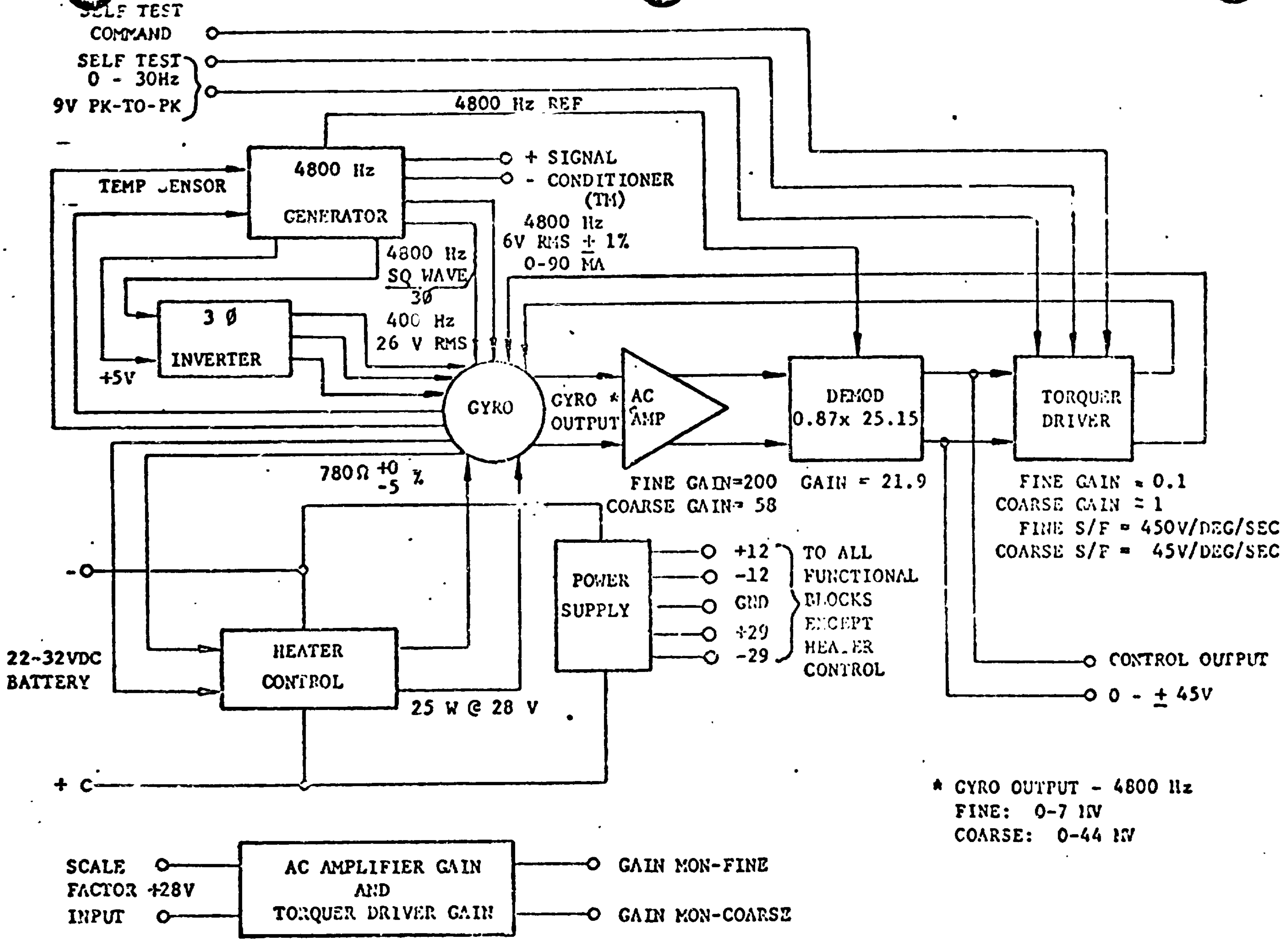


Figure 9.4.8-1 SKYLAB GYRO PROCESSOR BLOCK DIAGRAM

9.4.8 SKYLAB ATM RATE GYRO

General Characteristics

Program: Skylab ATM
Vendor: Martin-Marietta
Part Number: MSFC #50M37700

Performance Characteristics

Maximum Rate: $\pm$1.0 deg/sec coarse, $\pm$0.1 deg/sec fine
Maximum Output: $\pm$45 Vdc
Null Drift Rate: 0.1 deg/hr

Physical Characteristics

Size: 31 cm (12.1 in) by 22 cm (8.8 in) by 15 cm (5.8 in)
Weight: 5.7 kg (12.3 lbs.)
Input Power: 12 watts
Cooling Method: Radiation to deep space

References

Performance Specification, ATM Rate Gyro, MSFC #50M37705, Rev. E, dated April 27, 1970.

Comments

Uses KEARFOTT King Series gyro #2519; each package single axis, non-redundant.

Design Status

This component was flown as part of the 1973 Skylab mission.

9.5.3 HEAO-A DIGITAL PROCESSOR ASSEMBLY (DPA)

The design reference DPA for HEAO-A is a modular internally redundant version of the Control Data Corporation (CDC) 469 computer.

Basic machine characteristics are as follows:

<u>Central Processor</u> (LSI, 14 devices total)
Type: Binary, parallel, general purpose single address, plus file address
Repertoire: 42 instructions (some double precision)
Word Length: 16 bits
Register Files: 16 addressable 16-bit word files
Interrupts: 3 external levels, plus 1 direct execute
Arithmetic: Fractional, fixed point, two's complement. Hardware multiply and divide.
Typical execution times
- Add: 2.4 microseconds
- Double precision add: 3.6 microseconds
- Multiply: 10.4 microseconds
- Divide: 30.4 microseconds

<u>Memory</u>
Type: Random access, word organized, NDRO (electrically alterable) plated wire memory (PWM)
Word Length: 16 bits
Capacity: 8K words NDRO expandable to 65K in 8K word increments
Read Cycle Time: 1.6 microseconds
Write Cycle Time: 2.4 microseconds
Access Time: 500 nanoseconds

<u>Input/Output</u>
1-16 bit parallel, party line, bus input
1-16 bit parallel, party line, bus output
4-bit address control lines
1-serial input channel
1-serial output channel
External clock input
69-90 kHz - parallel continuous word rate I/O
400 kHz - parallel burst word rate I/O
130 kHz - serial bit rate I/O

Optional I/O: Solid state keyboard and display suitable for navigational, checkout and general purpose use can be integral to the computer. Also available are multiplexed A/D input channel(s), buffered and non-buffered peripheral device channel(s) on a Quote Special Equipment (QSE) basis.

An input/output block diagram of a basic single central processor unit is shown in Figure 9.5.3-1. A general block diagram of the DPA is shown in Figure 9.5.3-2 complete with the cross strapping to interconnect the DPA central processors with either redundant Transfer Assembly (TA).

The operating configuration of the DPA is controlled and selected by the online TA. Each module of the DPA receives a separate bilevel power enable signal from the TA that places it either in an operating or standby mode. The DPA is then reset and initialized by a control signal sent to the operating CPU. This causes the program counter to be initialized to a known starting location from which the first program instruction is fetched.

As part of the normal executive program function, the CPU transmits a discrete signal to the TA each time the program completes a major program cycle. This signal causes the fault timer in the TA to be zeroed. In the event that either the selected CPU or memory module have failed to the point at which the operating configuration cannot proceed through a normal program cycle, the TA fault timer will overflow, indicating an inoperative system. At this point, the DPA is shut down until commanded into a new configuration by the ground station.

A provision is incorporated to bootstrap a new program into the memory units via the uplink command system. This is accomplished by loading a few selected memory locations with a "bootstrap loader" routine via the externally forced direct execute instruction. The processor is then initialized by the TA and proceeds to software bootstrap the remaining program into memory.

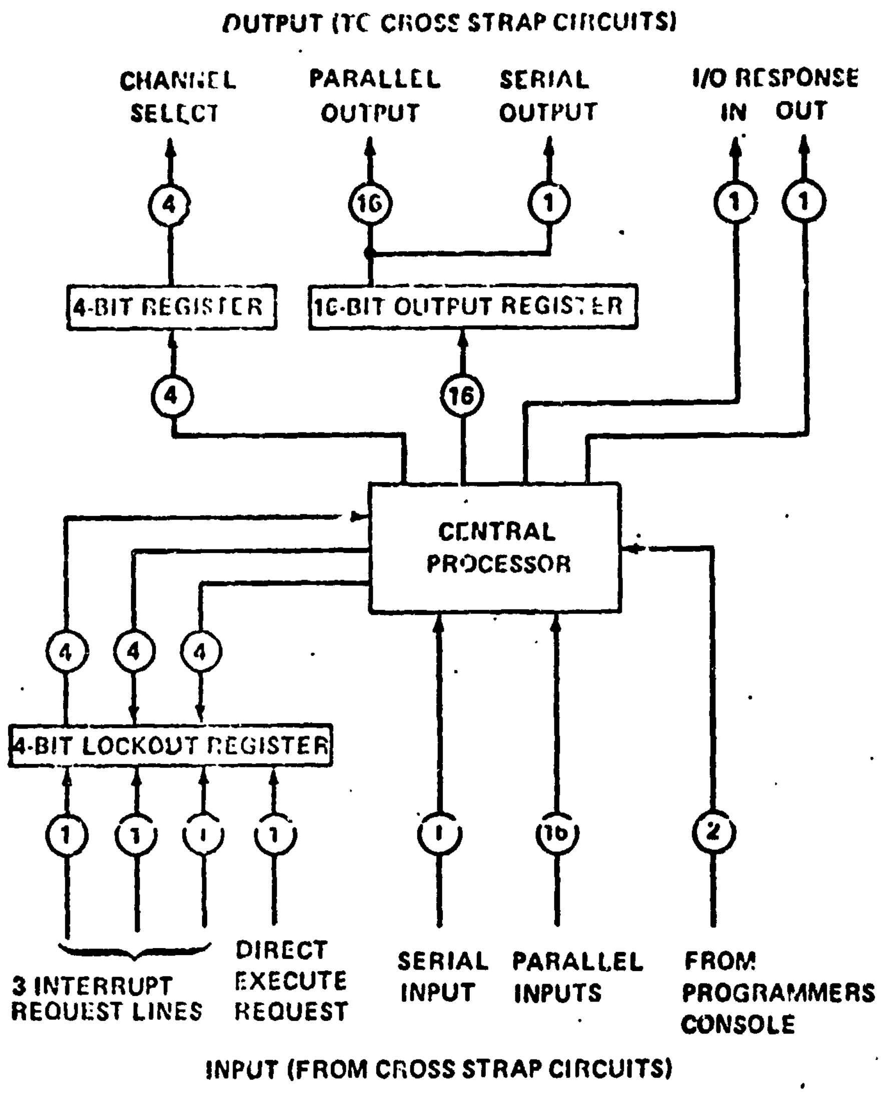


Figure 9.5.3-1. Basic Single Central Processor Unit Input/Output Block Diagram

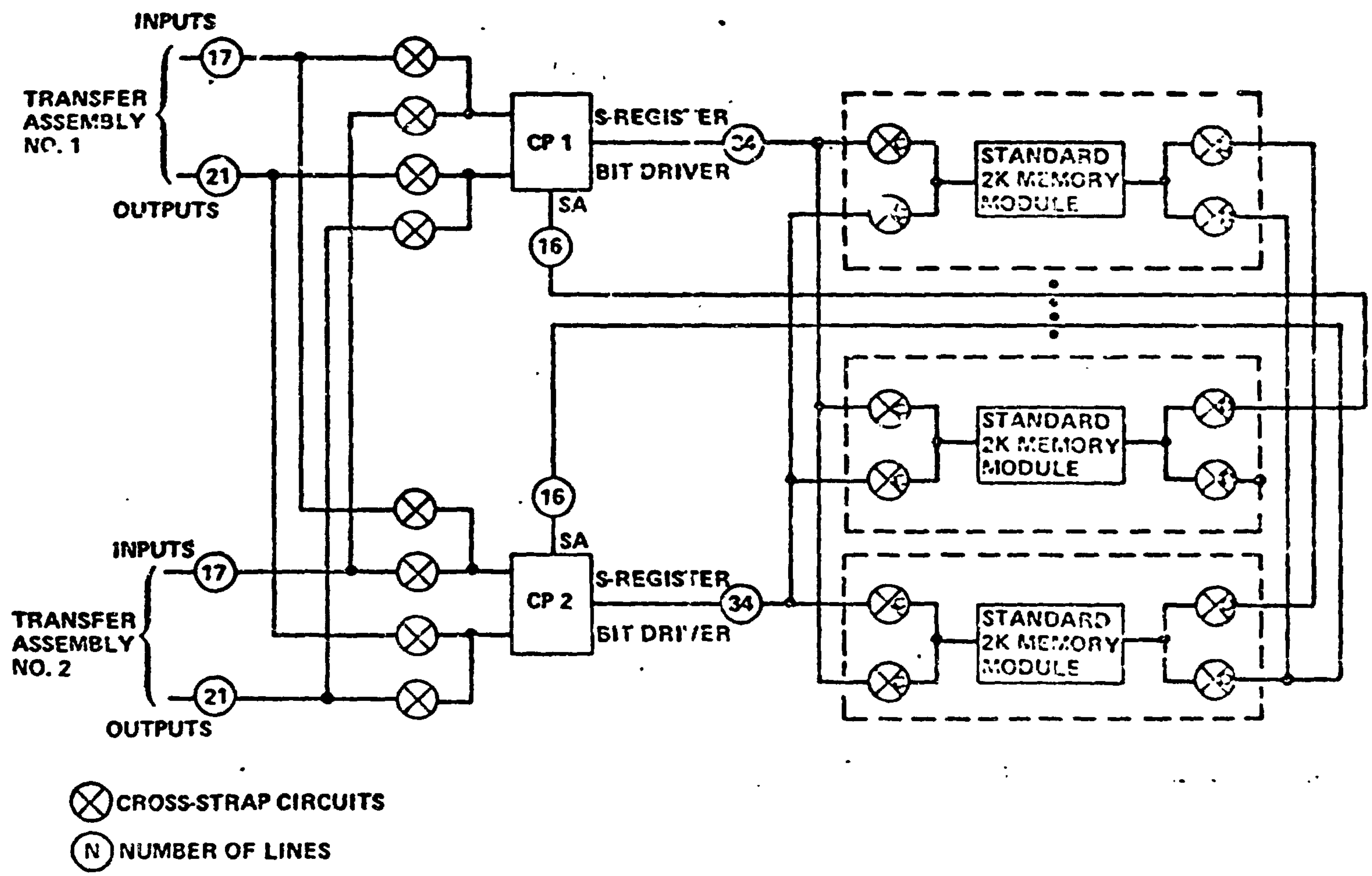


Figure 9.5.3-2. DPA Functional Block Diagram

9.5.3 HEAO-A DIGITAL PROCESSOR ASSEMBLY

General Description

Program: HEAO-A
Vendor: Control Data Corporation
Part Number: CDC 469 (Internally redundant version)

Performance Characteristics

General purpose, binary, parallel microminiature aerospace machine.

CPU
- Add time: 2.4 microseconds
- Multiply time: 10.4 microseconds
- Divide time: 30.4 microseconds
- Word length: 16 bits

Memory
- NDRO, plated wire
- Read cycle time: 1.6 microseconds
- Access time: 0.5 microseconds
- Write cycle time: 2.4 microseconds
- Capacity: 8K to 64K words, 16 bits/word

Input/Output
- Direct memory access: 90 kHz
- Multiplexer rate: 400 kHz burst
- Serial bit rate: 130 kHz

Physical Characteristics

Size: 10.7 cm (4.2 in) by 13 cm (5.1 in) by 36.6 cm (14.4 in)
Weight: 6.2 kg (13.7 lbs)
Input Power: 16 watts

References

Preliminary Requirements Review, "Attitude and Determination Subsystem," Volume III-C, HEAO Document DR No. CM-05, TRW under MSFC Contract NAS8-28300, 24 July 1972.

Large Space Telescope Phase A Final Report, Volume V-Support Systems Module, MSFC NASA TM X-64726, VI-135-137, December 15, 1973.

CDC 469 Brochures

Design Status

The HEAO-A program is currently in a state of redefinition. This component is a viable candidate for application in the 1977 HEAO-A mission.

9.5.6 OAO-C ONBOARD PROCESSOR

The primary functions of the OAO-C Onboard Processor (OBP) are auxiliary command storage, spacecraft monitoring and malfunction reporting, data compression and status summary, and possible performance of emergency corrective action for certain anomalous situations. The OBP consists of central processor, memory, input/output and power converter units and interconnection cabling. Figure 9.5.6-1 illustrates the relationships between the central processor unit, input/output controller and special interface equipment described above.

Central Processing Unit

Type: Medium scale, parallel
Number System: Binary, two's complement
Operation: Fixed point with automatic scaling for multiply and divide.
Instruction and Data Word Length: 18 bits
Instruction Set: 50 instructions, 30 requiring fetch operand
Execution Times: Add-10 microseconds
Multiply -68 microseconds
Divide -140 microseconds

CPU Registers:

Accumulator	-18 bits
Address	-18 bits
Subscript	-18 bits
Scale	- 6 bits
Storage Limit	-18 bits
Page Register	- 4 bits
Instruction	-11 bits
Carry	- 1 bit
Overflow	- 1 bit
Instruction counter	-16 bits
Memory Operand	-18 bits
Or/And	- 1 bit
Decision	- 1 bit
Operation Counter	- 6 bits

Input/Output Unit

Interrupts: 8, with program controlled priority levels
Data Channels: Redundant 18 bit parallel data bus in and out of memory

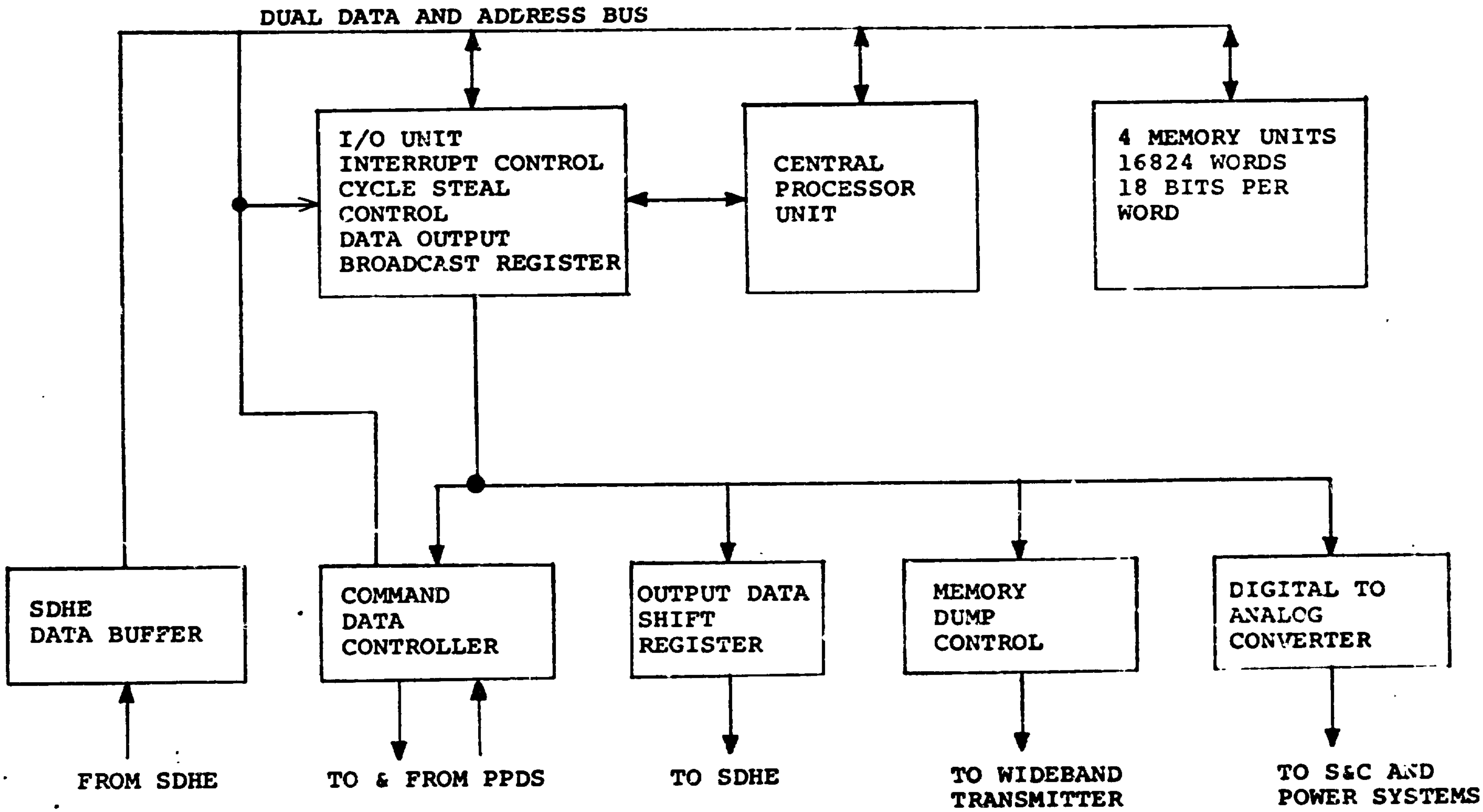


Figure 9.5.6-1. OAO-C On Board Processor (OBP) Block Diagram

The I/O unit includes five unique interfaces. These are:

Spacecraft Data Handling Equipment (SDHE) Serial Data Buffer-buffers twenty-six 18-bit words of serial SDHE data at a 1042 bps rate.

Command Data Controller - accepts 30-bit serial command words from the Primary Processor and Data Storage Unit (PPDS) and provides a 128 bit serial command format to the PPDS.

Output Data Shift Register - 30 bit output data register which serves as a parallel-to-serial converter for placing data into the SDHE bit stream.

Memory Dump Control - 32-bit shift register which serves as a parallel-to-serial converter to output data at a 50 kHz rate to the wideband transmitters.

Stabilization and Control Digital-to-Analog - three 6-bit (including sign) digital-to-analog converters, output range -5 to +5 volts.

Control Channels: Two cycle-steal, direct memory access
Data Transfer Rates: 1042 bps telemetry, 50 kbps memory dump
Discrete Inputs: 6 commands for command receiver equipment to control OBP enable functions
Discrete Outputs: 11 bilevels status data to telemetry

Main Store

Type: Random access core memory with clear/write and read/restore modes
Capacity: 4 units, 4096 words per unit, 18 bits per word (16,384 words total)
Cycle Time: 2 microseconds

9.5.6 OAO-C ONBOARD PROCESSOR

General Description

Program: OAO-C
Vendor: Micro-Technology, Inc. CPU and I/O; Electronic Memories, Inc. - Memory Units
Part Number: OBP-1

Performance Characteristics

CPU: Medium scale, parallel, binary, two's complement, fixed point with automatic scaling for multiply and divide.
Execution Times:
- Add: 10 microseconds
- Multiply: 68 microseconds
- Divide: 140 microseconds

Input/Output: parallel, redundant channels
- Word length: 18 bits
- Data Rates: 1042 bps telemetry, 50 kbps dump

Memory: Random access, core
- Capacity: 4 units, 4096 words per unit, 18 bits per word
- Cycle Time: 2 microseconds

Physical Characteristics

	CPU	I/O	MEMORY	POWER CONVERTER	CABLES
Size (cu cm)	4428 ($270 in^3$)	4920 ($300 in^3$)	8200 ($500 in^3$)	3116 ($190 in^3$)	--
Weight (kg)	6.8 (14.8lbs)	7.4 (16.3lbs)	10.6 (23.5lbs)	3.6 (3lbs)	0.7 (1.5lbs)

Power: 17.7 watts standby, 30.0 watts average, 511 watts peak from +28V unregulated supply.

0.2 watts continuous from +18 V regulated supply.

Cooling Method: Mounted on spacecraft heat sink.

References

W. N. Stewart, R. Hartenstein and C. Trevathan, "Applications of an Onboard Processor to the OAO C Spacecraft," NASA TM X 65937, June, 1972.

Design Status

This component was flown as part of the 1972 OAO-C mission.

9.5.8-1 SKYLAB ATM DIGITAL COMPUTER

The ATM Digital Computer (ATMDC) is a general purpose computer characterized by high internal computing speed, versatile operation code set, large random access storage, and a multipurpose flexible input/output capability. The ATMDC is capable of controlling data flow to and from its system interfaces. It performs arithmetic and local manipulation, routing, conversion, storage, timing and formatting on data as required. Data from the input channels are made available to the random access storage, and data from the storage are made available to the output channels during execution of certain internal commands. The ATMDC includes an isolated mission elapsed time clock. The ATMDC interfaces with a workshop computer interface unit (WCIU), test equipment, experiments, telemetry, and uplink command system equipment. Characteristics of the ATMDC are summarized below.

Central Processing Unit

Type: General purpose, byte parallel
Number System: Binary, 2's compliment, fractional
Operation: Fixed point
Instruction and Data Word Length: 16 bits, 32 bits for double precision, products and dividends.
Instruction Set: Three formats: 8-bit short arithmetic, 16-bit long arithmetic/logical, and 24-bit immediate format. The instruction set includes:

- 27 arithmetic instructions
- 4 logical instructions
- 2 shift instructions
- 7 branching instructions
- 1 input/output control instruction

Execution Times: Add: 9 to 24 microseconds
Multiply: 48 or 54 microseconds
Divide: 48 or 54 microseconds

Input/Output

Interrupts: 6 discrete inputs, inhibitable, plus internal interval timer
Data Channels: The ATMDC accepts the following input signals:

- 5 Momentary discretes
- 65 Continuous discretes
- 12 DC analogs, ±2.5 Vdc., 0.3 percent full scale, 8 bit quantization
- 4 AC analog pairs of synchronized resolver processors
- 6 Discrete inputs
- 1 Parallel 8-bit digital data channel from experiment

1 Parallel 12-bit digital data channel from command system input
4 Digital serial data channels, 10.417 KHz rate, 2 12-bit and 2 15-bit words, NRZ
1 Parallel 16-bit digital data channel to WCIU digital inputs and controls from test equipment

The ATMDC provides the following output signals:

9 Momentary discrete outputs
21 Continuous discrete outputs
12 DC analog, ±5 Vdc, 0.3 percent full scale, [illegible] bits quantization
4 Parallel 16-bit BCD digital channels data
1 Parallel 13-bit digital channel
1 Parallel 50-bit digital data channel to telemetry
1 Parallel 42-bit digital data channel, subdivided into 15, 14, 12 and 1 bit channels to experiment
1 Parallel 6-bit address channel to WCIU
1 Parallel 16-bit data channel to WCIU to control discretes and timing
1 Parallel 29-bit digital clock, mission elapsed time parallel digital data to test equipment

Main Store

Type: Random access ferrite core
Capacity: 16,384 words, 16 bits each word
Cycle Time: 2.5 microseconds

The ATMDC is divided into the following major electronic sections (Figure 9.5.8-1).

a. Digital Computer Assembly (DCA) with its memory and arithmetic unit.

b. Interface Control

c. Input/Output Assembly (IOA) with its signal conditioners and converters.

d. Power Supply Assembly (PSA) with its RFI filters.

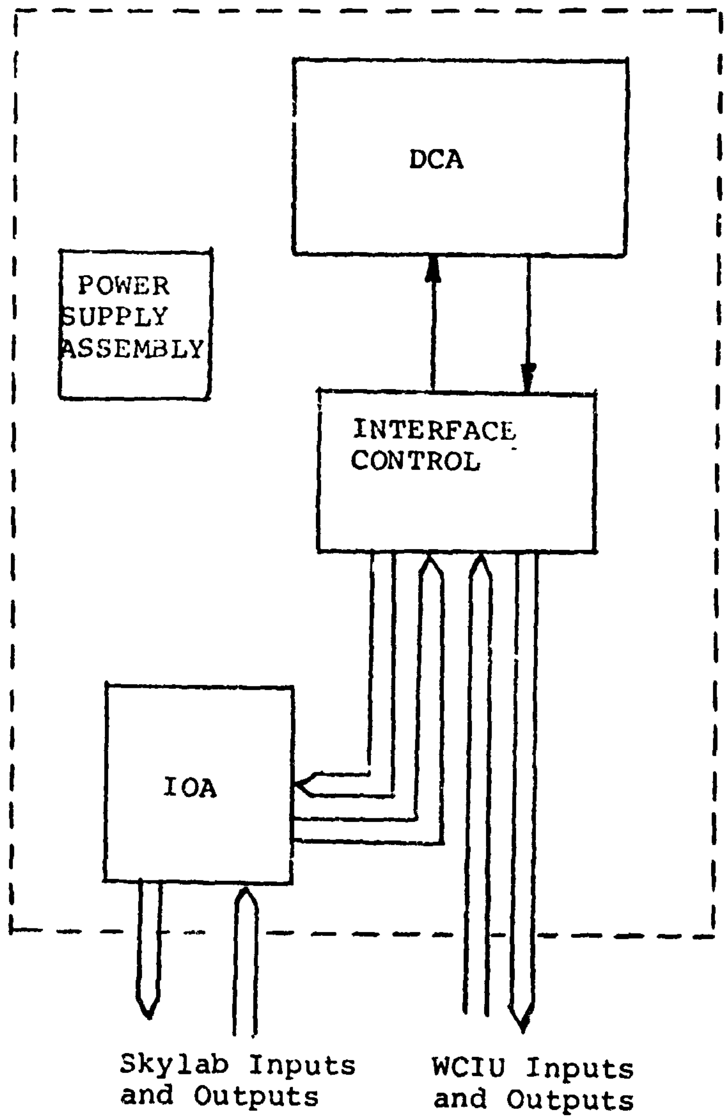


Figure 9.5.8-1 ATMDC BLOCK DIAGRAM

The DCA (Figure 9.5.8-2) is a general purpose, stored program digital computer, and features a 16-bit or 32-bit data format. Data transfer within the DCA are by 8 bit words, or bytes. By using an 8-bit (byte) data transfer format, internal registers can be utilized for more than one function.

All signals entering or leaving the DCA are digital and pass through the Interface Control Unit. Input dc power is supplied by the PSA.

Loading and verification of programs and verification of DCA operation are controlled by a Memory Load Verify Unit (MLVU), which is located in the ATMDC checkout equipment.

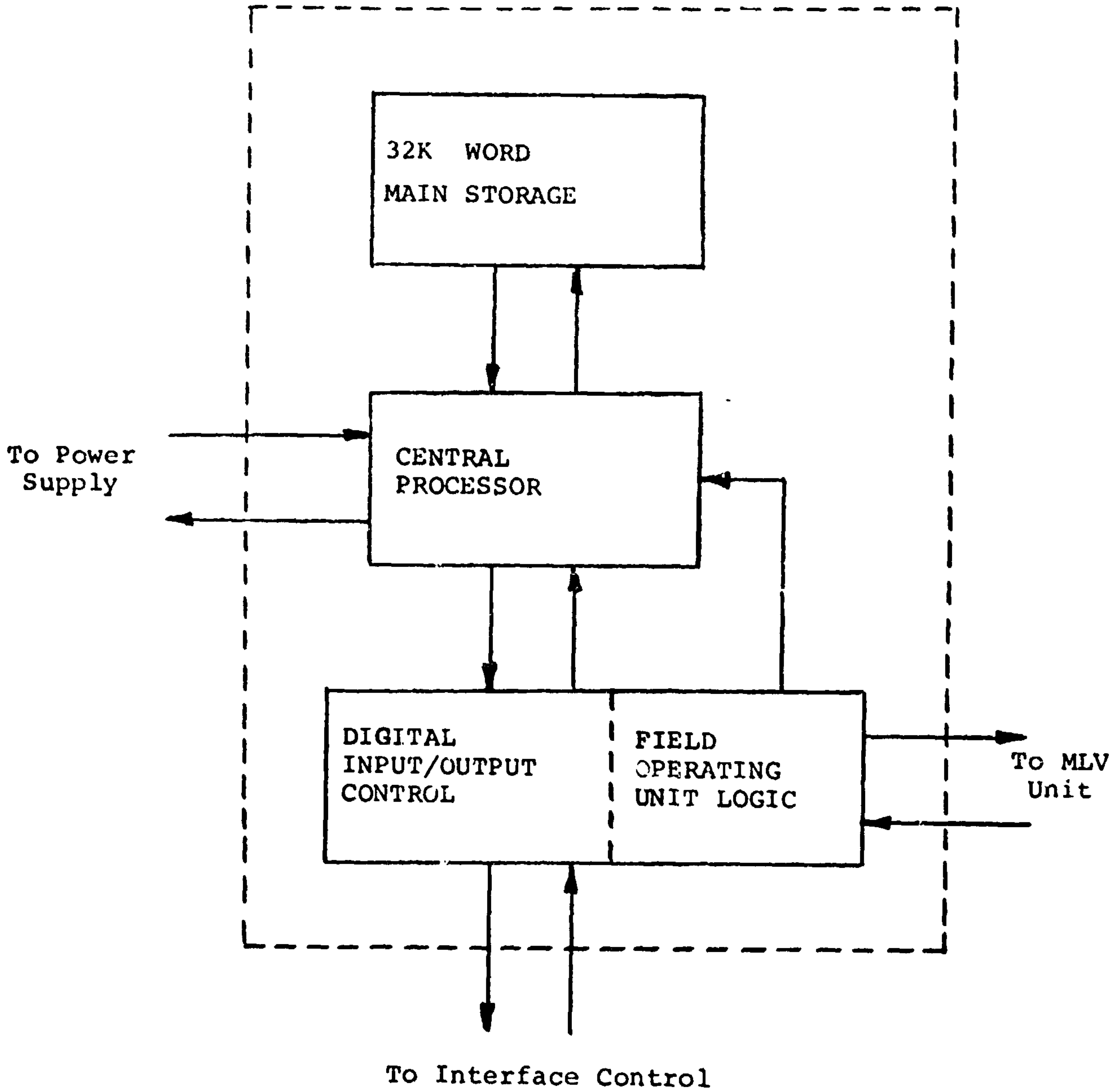


Figure 9.5.8-2 DCA BLOCK DIAGRAM

9.5.8-1 SKYLAB ATM DIGITAL COMPUTER

General Description

Program: Skylab ATM
Vendor: International Business Machines Corp., Owego, New York
Part Number: ATMDC (Modified 4 Pi TC-1)
IBM Specification 7916430, 69-520-0001
Unit Designation 702A5463 (Primary Unit)
702A241 (Secondary Unit)

Performance Characteristics

CPU
Type: General purpose, byte parallel, binary , 2's compliment, fractional, fixed point.
Speed: Add: 9 to 24 microseconds
Multiply: 48 or 54 microseconds
Divide: 48 or 54 microseconds
Input/Output: Multiple analog, digital and discrete channels
Storage: Random access core, 16,384 words, 16 bits/word, cycle time 2.5 microseconds

Physical Characteristics

Size: 48.3 cm (19 in) by 18.6 cm (7.3 in) by 80.7 cm (31.8 in)
Weight: 44.5 kg (98 lbs)
Power: 98 watts
Cooling Method: Radiation, conduction to structure

References

Contract End Item Specification, Apollo Telescope Digital Computer, IBM Drawing No. 7916430.

Operating Maintenance Handbook, "Digital Computer for ATM," IBM No. 70-218-0002, Oct. 13, 1970.

Design Status

This component was flown as part of the 1973 Skylab mission.

9.5.8-2 SKYLAB ATM WORKSHOP COMPUTER INTERFACE UNIT (WCIU)

The WCIU provides the capability to interface either of two redundant ATMDC's with the workshop attitude control system. It provides the necessary buffering of ATMDC input/output lines to provide this interface. In conjunction with an active ATMDC, it provides a means of sensing a failure of an ATMDC itself.

The WCIU accepts the following types of input signals for extending the input capability of the ATMDC.

- 8 continuous discrete input signals
- 24 momentary discrete input signals
- 35 dc analog signals ± 5 Vdc, 0.1 percent
- 3 dc analogs, ± 5 Vdc, rescaled to ± 2.5 Vdc for output to control and display panel
- 22 ac analog signals, 4.8 kHz, 0.1 percent
- 2 digital serial data, 10.417 kHz bit rate, 16 bit words, digital inputs from test equipment

The WCIU generates the following types of output signals under program control of an ATMDC:

- 32 momentary discrete outputs
- 37 continuous discrete outputs
- 9 dc analog signals, ± 5 Vdc., 0.3 percent full scale 10 bits plus sign
- 4 dc analog signals, ± 12 Vdc, 5 percent of full scale, control and display panel digital outputs to test equipment 29-bit digital clock channels (mission timer B)

The WCIU receives the following types of signals from the ATMDC to control the extended input/output capability:

- 6 parallel address lines
- 1 333 kHz clock
- 1 10.417 kHz clock
- 1 1 MHz clock
- Timing pulses
- I/O read and write commands
- Power controls
- 16 parallel data lines

The WCIU sends the following types of signals to the ATMDC to control the extended input/output capability:

- Interrupt
- 16 parallel data lines

The WCIU (Figure 9.5.8-3) has 64 bits of computer software data redundantly stored for initialization of the ATMDC. It provides the capability to interface the one-on/one-off (1 + 1 off) ATMDC's with the Workshop Attitude Control System; provides the necessary buffering of ATMDC input/output (I/O) lines to accomplish this function; and, in conjunction with an active ATMDC, provides a means of sensing a failure of an ATMDC and the committed WCIU I/O pages. The WCIU consists of four main functional sections:

a. IOA extension section A
b. IOA extension section B
c. WCIU common section
d. Power supply section.

Only one of the two IOA extension sections is powered on at any time. The common section has power applied at all times when the system is functional. The power supply section furnishes power to the active IOA extension section and to the common section.

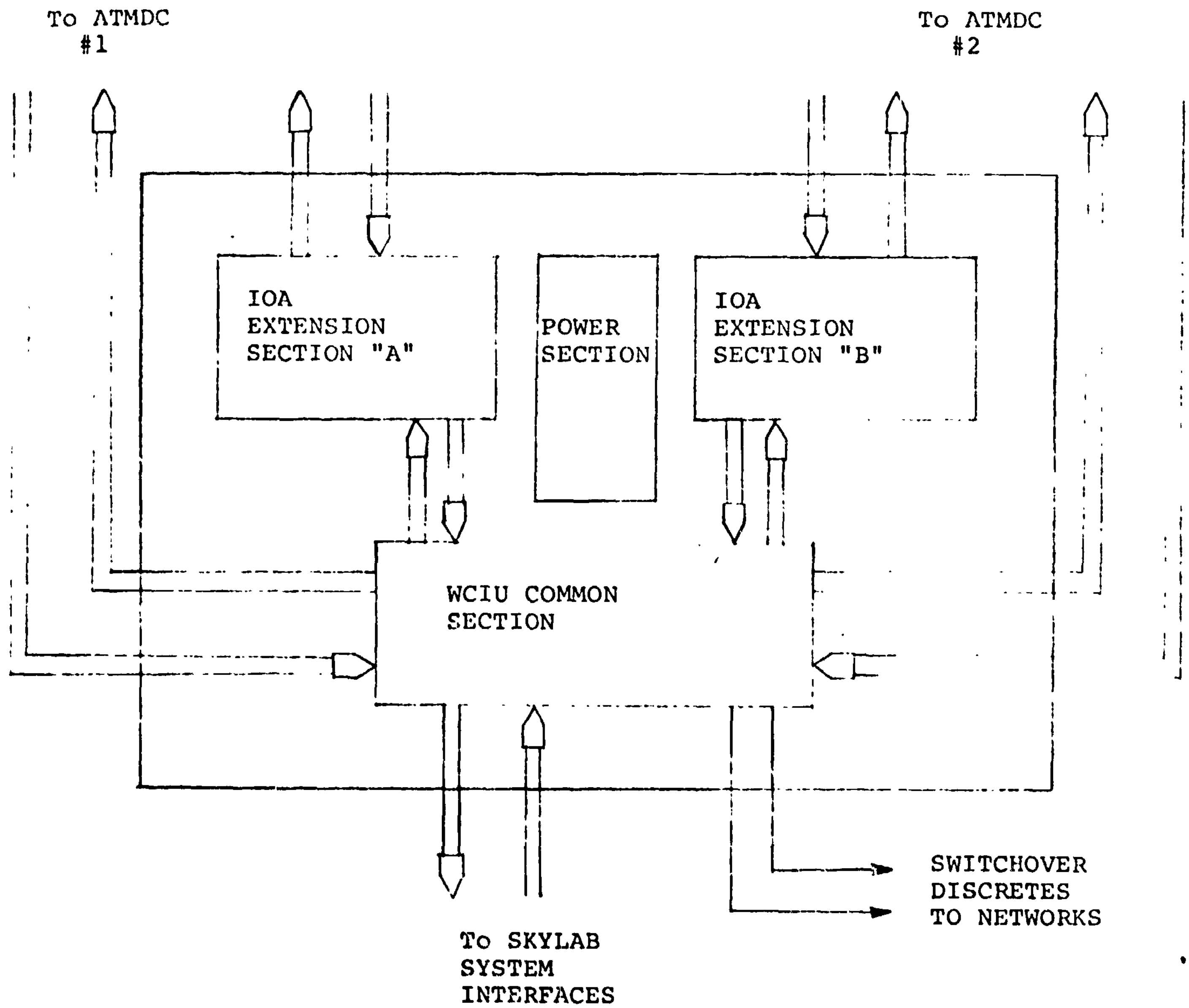


Figure 9.5.8-3. WCIU Major Sections

9.5.8-2 SKYLAB ATM WORKSHOP COMPUTER INTERFACE UNIT (WCIU)

General Description

Program: Skylab ATM
Vendor: International Business Machines Corp., Owego, New York
Part Number:

Performance Characteristics

Interfaces with either of two redundant ATMDC's
32 discrete inputs
38 dc analog inputs
22 ac analog inputs
2 serial digital data channels
69 discrete outputs
12 dc analog outputs
Other miscellaneous inputs and outputs for test and control purposes.

Physical Characteristics

Size: 50.8 cm (20 in) by 127 cm (43 in) by 13.5 cm (5.3 in)
Weight: 56.8 kg (125 lbs)
Power:
Cooling Method: Radiation, conduction to structure.

References

Contract End Item Specification, Apollo Telescope Digital Computer, IBM Drawing No. 7916430.

Operating Maintenance Handbook for Workshop Computer Interface Unit, IBM No. 70-212-0005, Oct. 13, 1970.

Design Status

This component was flown as part of the 1973 Skylab mission.

9.5.8-3 SKYLAB ATM MEMORY LOAD UNIT

The memory load unit (MLU), Figure 9.5.8-4, consists of a control and timing section, a multiplexer section, a command receiver interface section, and a power supply section. MLU operation is illustrated by a functional flow diagram, Figure 9.15.8-5.

The MLU control and timing logic performs the following functions:

- o Receive, store, and decode switch selector commands
- o Synchronization of internal and external clocks
- o Provides timing and control signals for loading ATMDC storage assembly from magnetic tape or one of two command receiver serial data input lines.
- o Select multiplexer gates as required for performing a memory load from either Apollo Checkout Equipment (ACE) in the umbilical mode or the field operating unit (FOU) in the test plate mode.

The command receiver interface logic extracts a 72 KHz clock from the manchester II biphase data received from the command receiver. Each of the two command receiver signals is transformer coupled into a comparator circuit that produces a squared output from the sinusoidal input. When the MLU is commanded to the command receiver mode, one of the two comparator outputs is selected at the AOI multiplexer stage. The output of this circuit is a negative pulse for each transition of the input signal.

The 72 KHz output extracted from the clock is synchronized with the data the first time the data has a logic 1/0 or a 0/1 sequence. From that point, the 72 KHz clock is synchronized with the data if there is no loss of data.

The multiplexer section gates signals to two ATMDC's from either of three sources. Signals to either ATMDC are gated from an associated umbilical input or the MLU control logic.

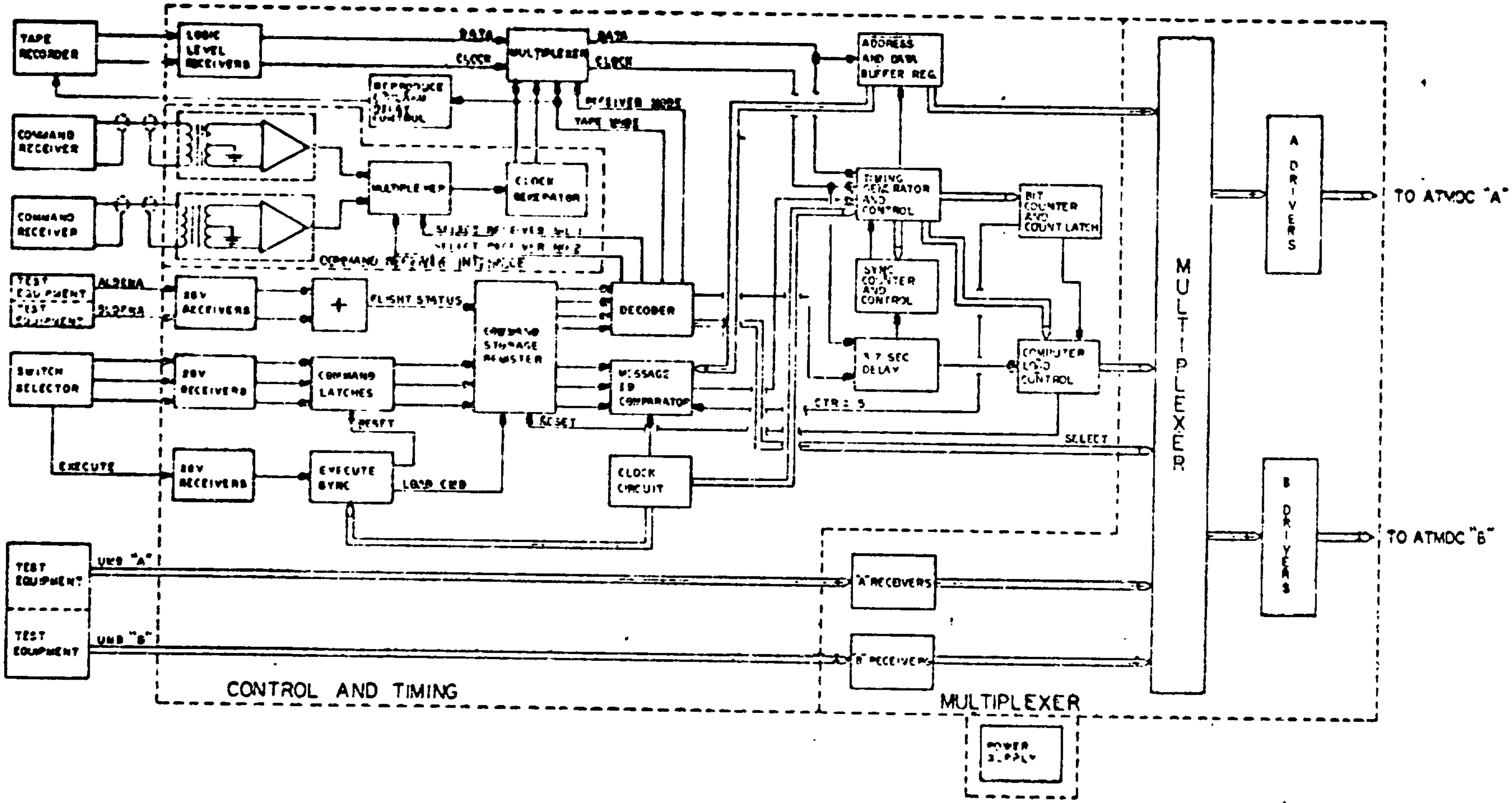


Figure 9.5.8-4 MLU BLOCK DIAGRAM

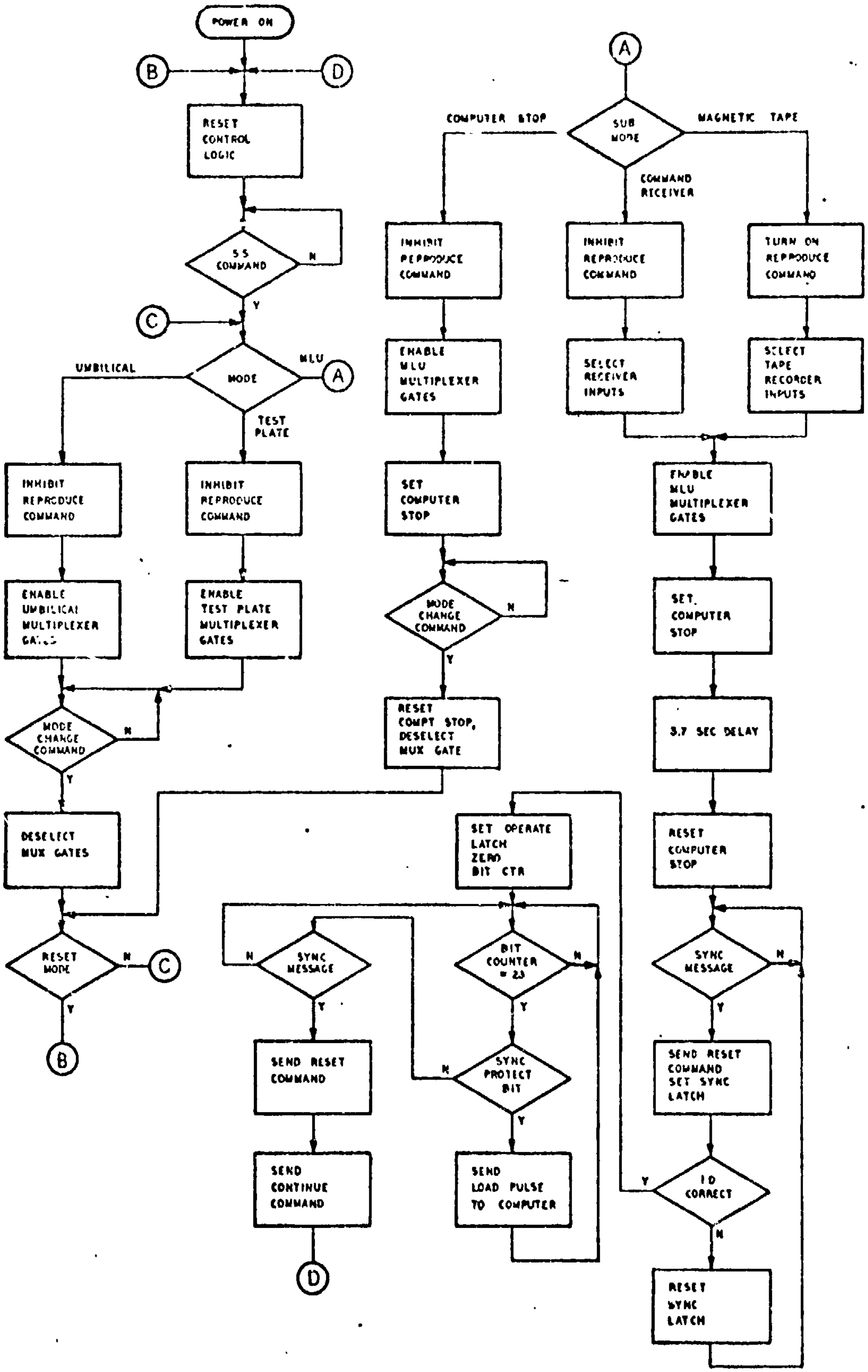


Figure 9.5.8-5 MLU FUNCTIONAL FLOW DIAGRAM

9.5.8-3 SKYLAB ATM MEMORY LOAD UNIT

General Description

Program: Skylab ATM
Vendor: IBM Corporation
Part Number:

Performance Characteristics

Inputs accepted and processed by MLU

- Manchester Type 2 biphase data from the magnetic tape unit
- Manchester Type 2 biphase address from each command receiver
- Digital serial clock from magnetic tape unit (72 KHz square wave)
- 4 momentary discretes from ATM switch selector
- High-level discrete from test panel or umbilical for each ATMDC
- 39 data, address, and control signals from test panel for each ATMDC

Outputs generated by MLU

- Parallel 8-bit digital data to each ATMDC
- Parallel 15-bit digital address to each ATMDC
- High-level discrete to each ATMDC (test connector enable)
- 15 control signals to each ATMDC
- High level discrete to the tape unit to command it to pull tape (reproduce command)

Physical Characteristics

Size: 22.9 cm (9.0 in) by 38.1 cm (15 in) by 14 cm (5.5 in)
Weight: 8.2 kg (18 lbs)
Input Power: 33.0 watts at 28 Vdc
Cooling Method: Radiation to deep space from upper surface and conduction through 4 mounting feet

References

Operating Maintenance Handbook for Memory Load Unit (MLU), IBM No. 72W-00196, October 3, 1972.

Comments

MLU page assembly is 4 Pi Medium Density Logic technology.

Design Status

This component was flown as part of the 1973 Skylab mission.

9.5.11 MARINER MARS DIGITAL COMPUTER

The Central Computer and Sequence Subsystem (CC&S) is a special purpose digital computer with fixed sequence maneuver redundancy. Figure 9.5.11-1 illustrates an organization block diagram of the CC&S. Approximately 80 percent of the total CC&S subsystem is organized into a computer which may be classified as a stored program, serial operating, special purpose digital machine. The computer consists of the following basic functional elements:

- Input decoder
- Clock
- Processor
- Memory
- Event decoder
- Volatility Protection
- Power Converter

The computer input decoder has the primary function of receiving inflight coded commands (CC's) from the Flight Command Subsystem (FCS) and routing the contained data into the computer memory. Secondary functions include controlling inflight readout of the computer memory via the Flight Telemetry Subsystem (FTS) and prelaunch memory loading and readout via the CC&S support equipment (CC&S - SE). In addition a one word readout of "pre-aim" information can be sent to attitude control. The reception and routing of CC words, for memory loading and readouts, is done in a real time mode in that normal time keeping routines are temporarily interrupted in the computer. Such command loading is thus restricted to intervals when the CC&S is in minute or hour scan modes unless time losses or subsequent reprogramming can be tolerated. This loading restriction saves considerable circuitry within the CC&S at essentially no penalty to the spacecraft since all loading cycles would normally occur during intervals when the CC&S is in a low activity mode. A timed interlock is provided in the input decoder to return the computer to real time operation should the CC word be interrupted before completion. A logic flow diagram of the input decoder is shown on Figure 9.5.11-2.

The computer clock serves as the source of reference signals required by the processor and the memory. The computer clock accepts a 2.4 kHz reference frequency and divides it down to the following:

a. Provide signals required by the processor scan control.

b. Provide the read/write digit pulse rate for the memory.

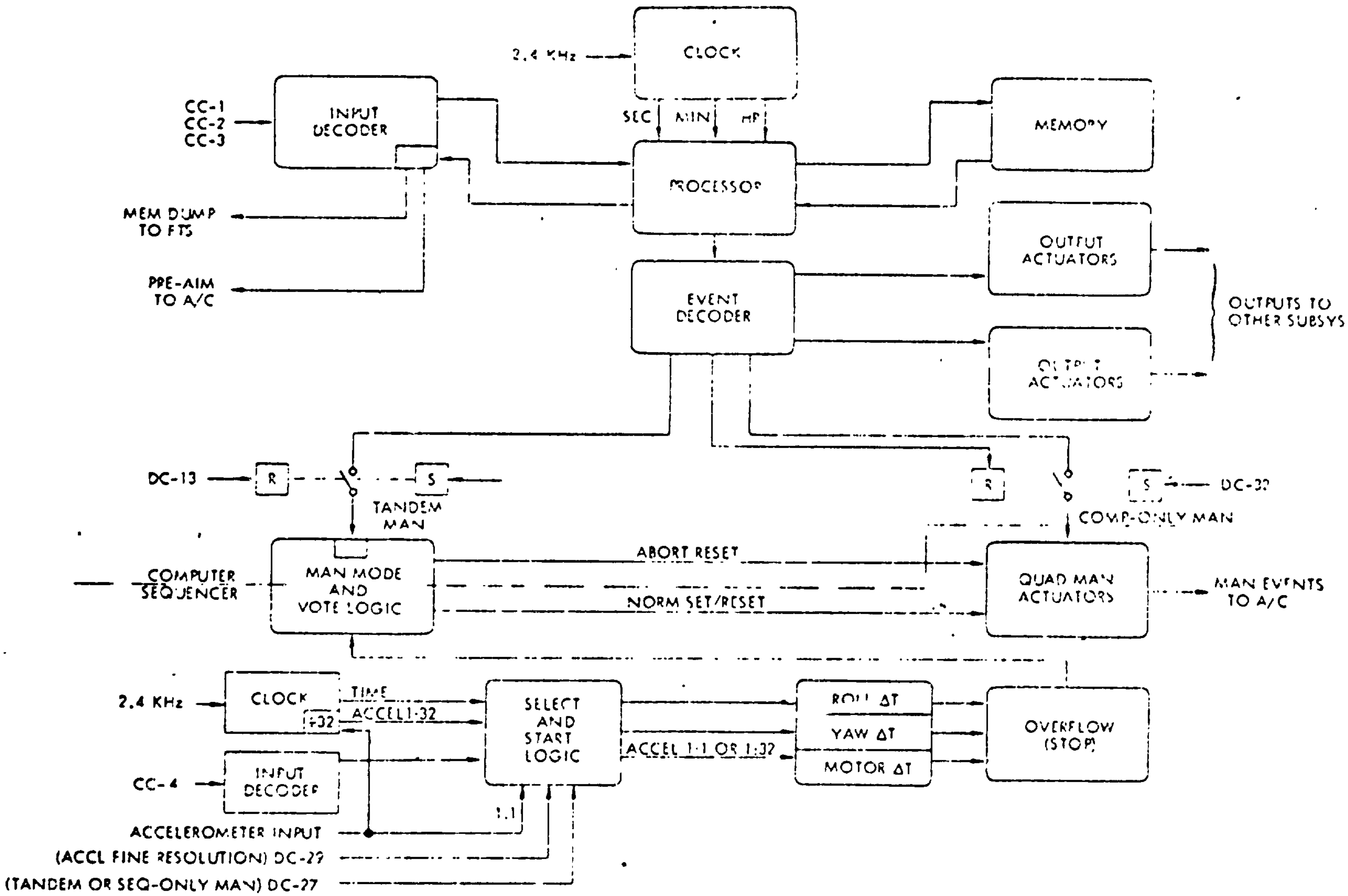


Figure 9.5.11-1 CC&S BLOCK DIAGRAM

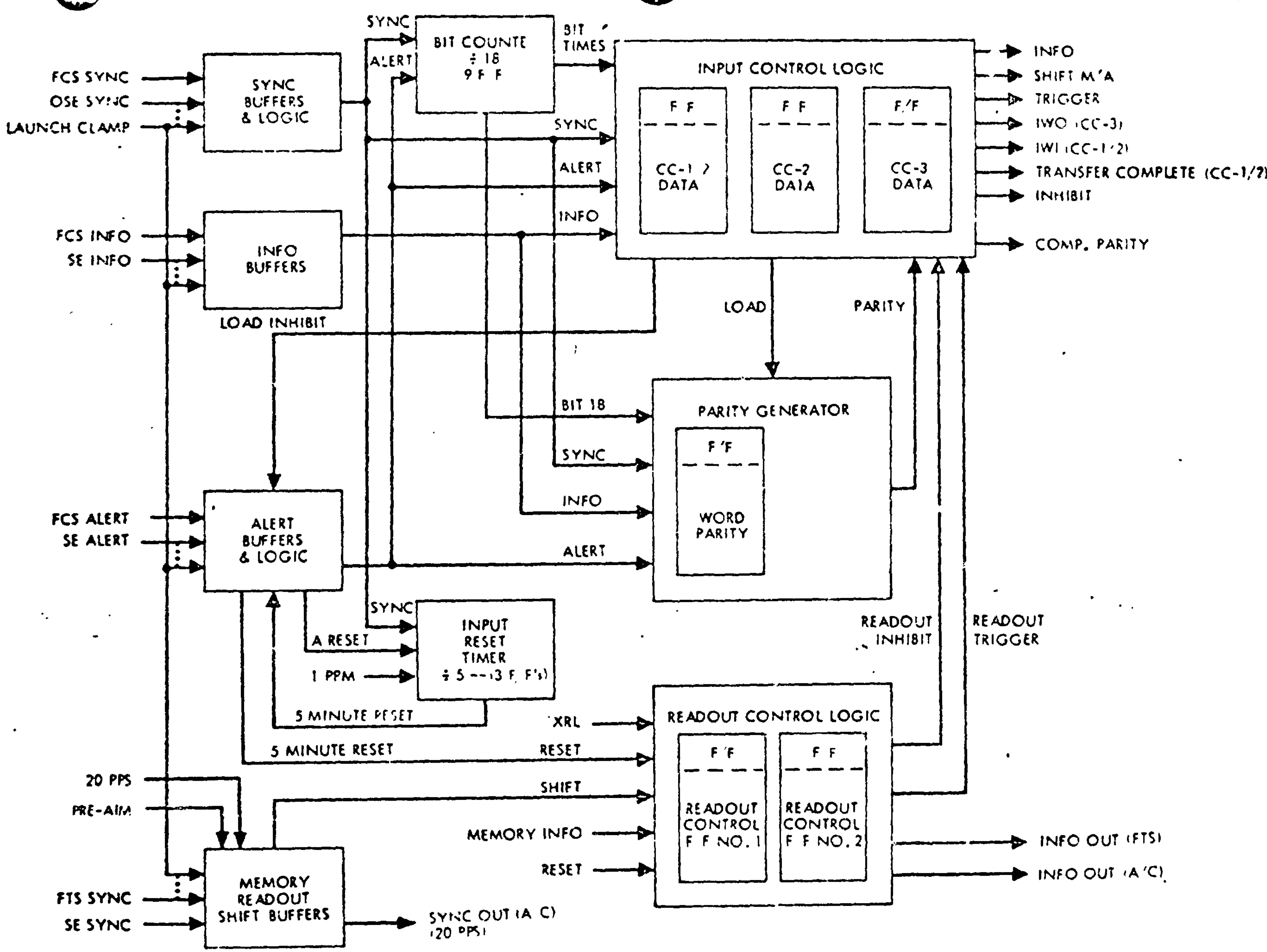


Figure 9.5.11-2 COMPUTER INPUT DECODER

c. Provide the time base for counting down the motor burn duration.

d. Provide a pulse every hour to the attitude control subsystem (referred to as the Canopus gate cyclic).

Figure 9.5.11-3 is a block diagram of the computer clock.

The computer processor contains facilities for: addressing, fetching and storing of binary information in the memory, sequencing of instructions stored in memory, arithmetic and time-dependent processing of memory information, and controlling the initiation of relay actuations. The data cycles executed by the processor are directly related to the information as fetched from the memory. The data word formats imposed on memory storage and handled by the processor are shown on Figure 9.5.11-4. A memory word could contain any one of the three formats; however, each instruction cycle shall be initiated by fetching a word of format (a) containing an operation code which controls data flow throughout the remainder of the instruction cycle. The basic operation codes and the instruction cycle they control are listed in Table 9.5.11-1.

Figure 9.5.11-5 illustrates an overall block diagram of the processor as mechanized to handle the above mentioned word formats and instruction cycles.

The memory (Figure 9.5.11-6) provides nonvolatile ferrite core storage and associated input/output circuitry for the computer. The general features of the memory included the following:

a. 512 words - - plus a one word transfer register.

b. 22 bit word length.

c. Random access word addressing.

d. Coincident current readout.

e. 30 mil lithium ferrite cores.

f. Destructive readout (requires a read/write cycle each bit time).

g. Bit serial readout.

h. Operates at 2.4 kHz bit cycles rate.

i. Power dissipation is 3.4 watts during continuous interrogation, and 0.75 watt nonoperating standby power.

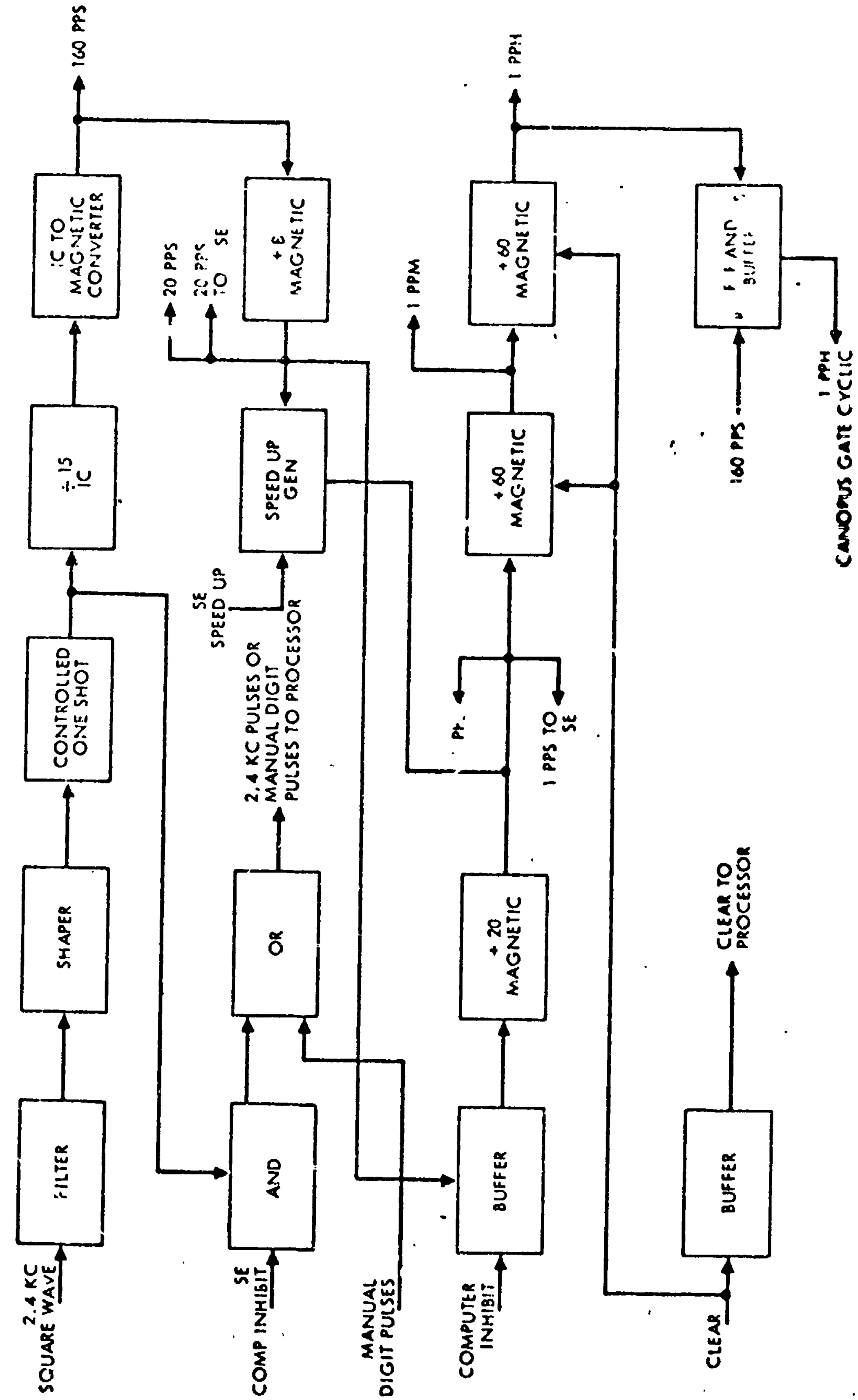
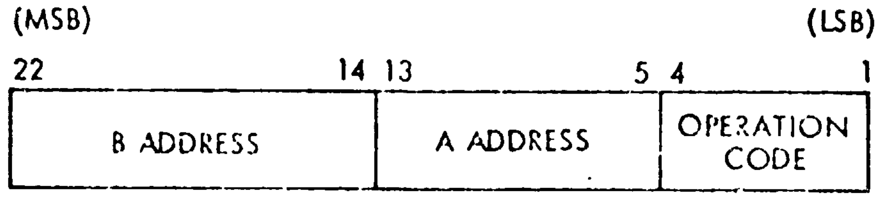
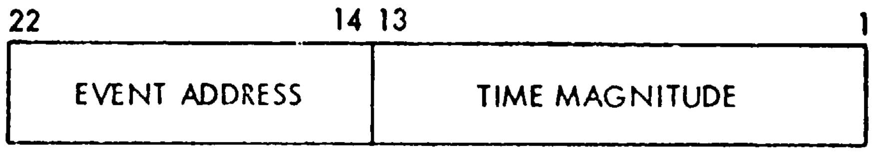


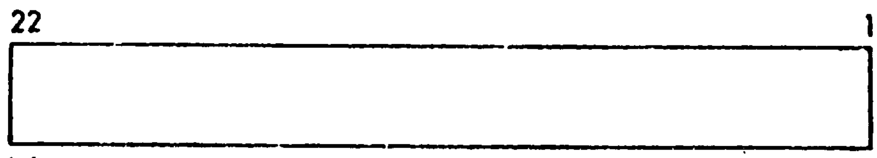
Figure 9.5.11-3 COMPUTER CLOCK



(a) INSTRUCTION FORMAT



(b) TIME/EVENT ADDRESS FORMAT



(c) MISCELLANEOUS DATA

Figure 9.5.11-4 PROCESSOR WORD FORMATS

Table 9.5.11-1 OPERATION CODE LIST-MARINER MARS 1971 CC&S

	Instruction	Mnemonic Code	Binary Code MSB LSB	Execution Time (Memory Cycle)
1	No Operation	NOP	0000	4
2	Conditional Jump	CLJ	0001	22
3	Count and Jump	CTJ	0010	5 to 22
4	Word Output and Halt	WOH	0011	44
5	Add	ADD	0100	66
6	Transfer A to B	TAB	0101	66
7	Subtract	SUB	0110	66
8	Word Input and Halt	WIH	0111	66
9	Halt	HLT	1000	4
10	Decrement Address and Jump	DAJ	1001	44
11	Unconditional Jump	UNJ	1010	22
12	Reset Operation Code and Jump	ROJ	1011	44
13	Decrement Hours and Jump	DHJ	1100	
	(Wrong Resolution)			4
	(Correct Resolution)			23 to 44
14	Decrement Variable and Jump	DVJ	1101	--
15	Decrement Minutes and Jump	DMJ	1110	
	(Wrong Resolution)			4
	(Correct Resolution)			23 to 44
16	Decrement Seconds and Jump	DSJ	1111	
	(Wrong Resolution)			4
	(Correct Resolution)			23 to 44

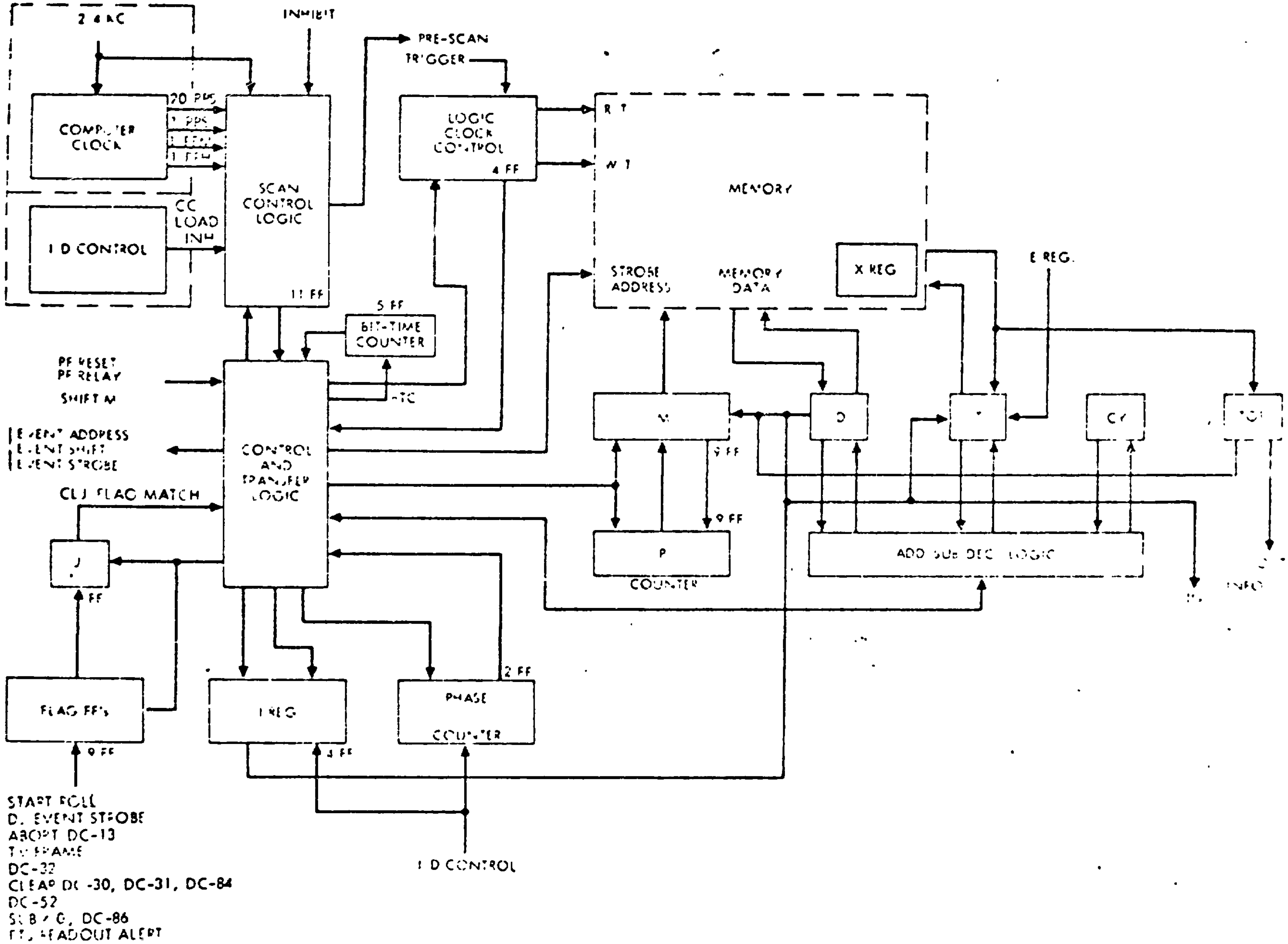


Figure 9.5.11-5 COMPUTER PROCESSOR BLOCK DIAGRAM

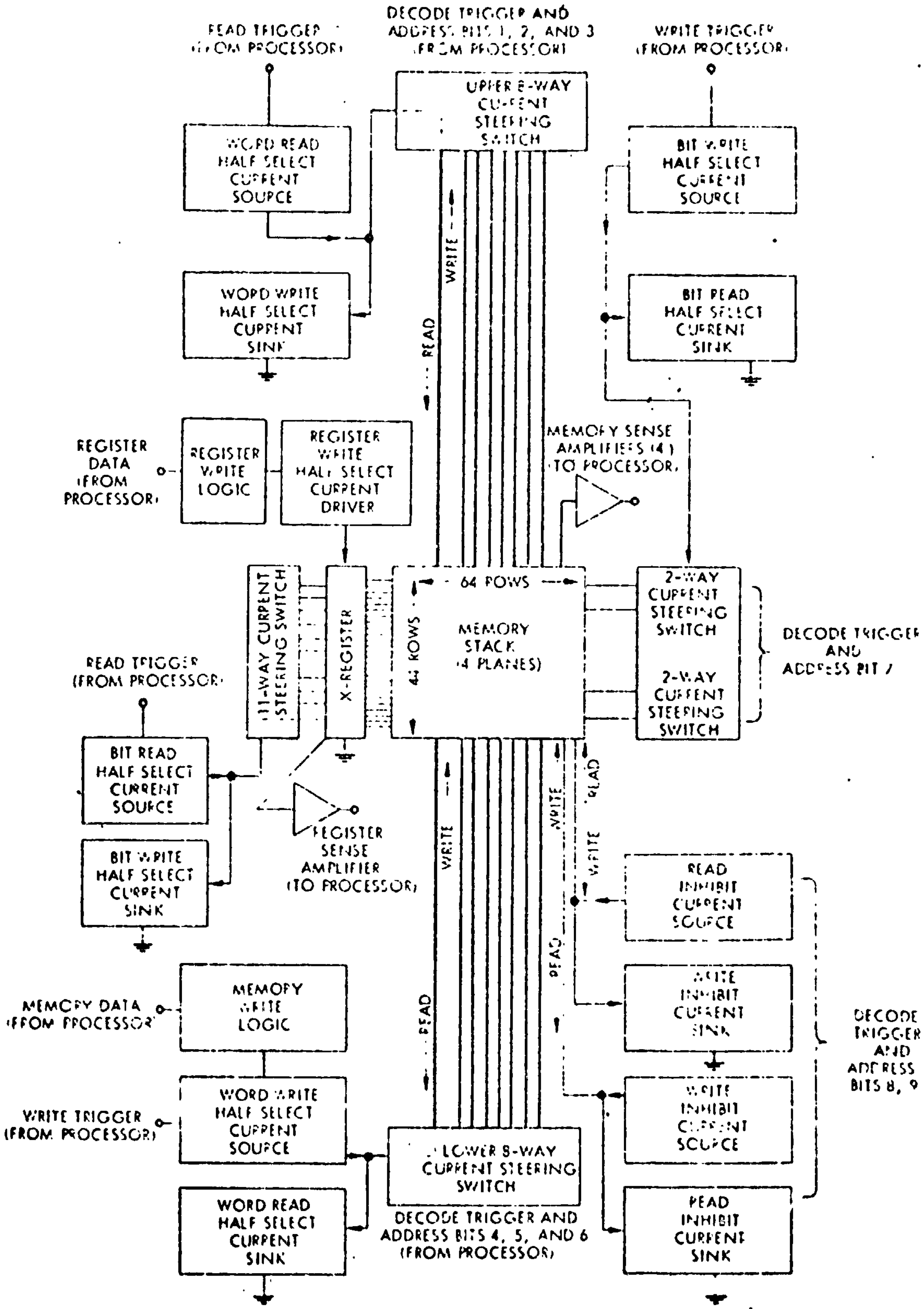


Figure 9.5.11-6 CC&S MEMORY

j. Weight is 2.8 pounds

k. Volume is 106 cubic inches.

The event decoder converts event address information furnished by the processor into relay actuations. A special decoding matrix capable of decoding 112 discrete states from a nine bit address has been mechanized so that from one to three of these discrete states in certain restricted combinations may be simultaneously selected. A block diagram of the event decoder is illustrated on Figure 9.5.11-7.

To help assure the ability of the computer to function from launch to the end of mission without reliance on ground command and to minimize the possiblity of computer interference due to abnormal performance during mission critical functions it is mandatory to protect program storage to the maximum extent practicable.

The method applied to the CC&S consists of: minimizing, by program, the intervals during which the CC&S is volatile to excessive power transients (greater than 20 percent) or power dropout, and to sense the transients so that should they occur when the program is volatile, operation will be automatically stopped until the program can be analyzed and corrected by ground command as necessary. This volatility occurs when the processor is scanning the memory storage and is executing instruction cycles dependent on the data contained in the semiconductor circuitry. It is a near certainty that excessive transients or power dropout during scan would destroy a quantity of program data which could force the computer into an uncontrolled performance mode. A power monitor (tolerance detector) is provided in the CC&S design which detects all transients in excess of 300 msec duration and minus 20 percent amplitude variance from the normal input power. Should this transient occur during scan, the computer would be inhibited by internal signals. When errors due to the transient have been determined, the program shall first be corrected by coded commands and the computer restarted.

The computer power converter consists of transformers, rectifiers, surge suppression chokes and switching circuitry, and pulse power storage capacitors. All dc power is converted from 2.4 kHz spacecraft power without secondary regulation. A block diagram of the power converter is illustrated on Figure 9.5.11-8.

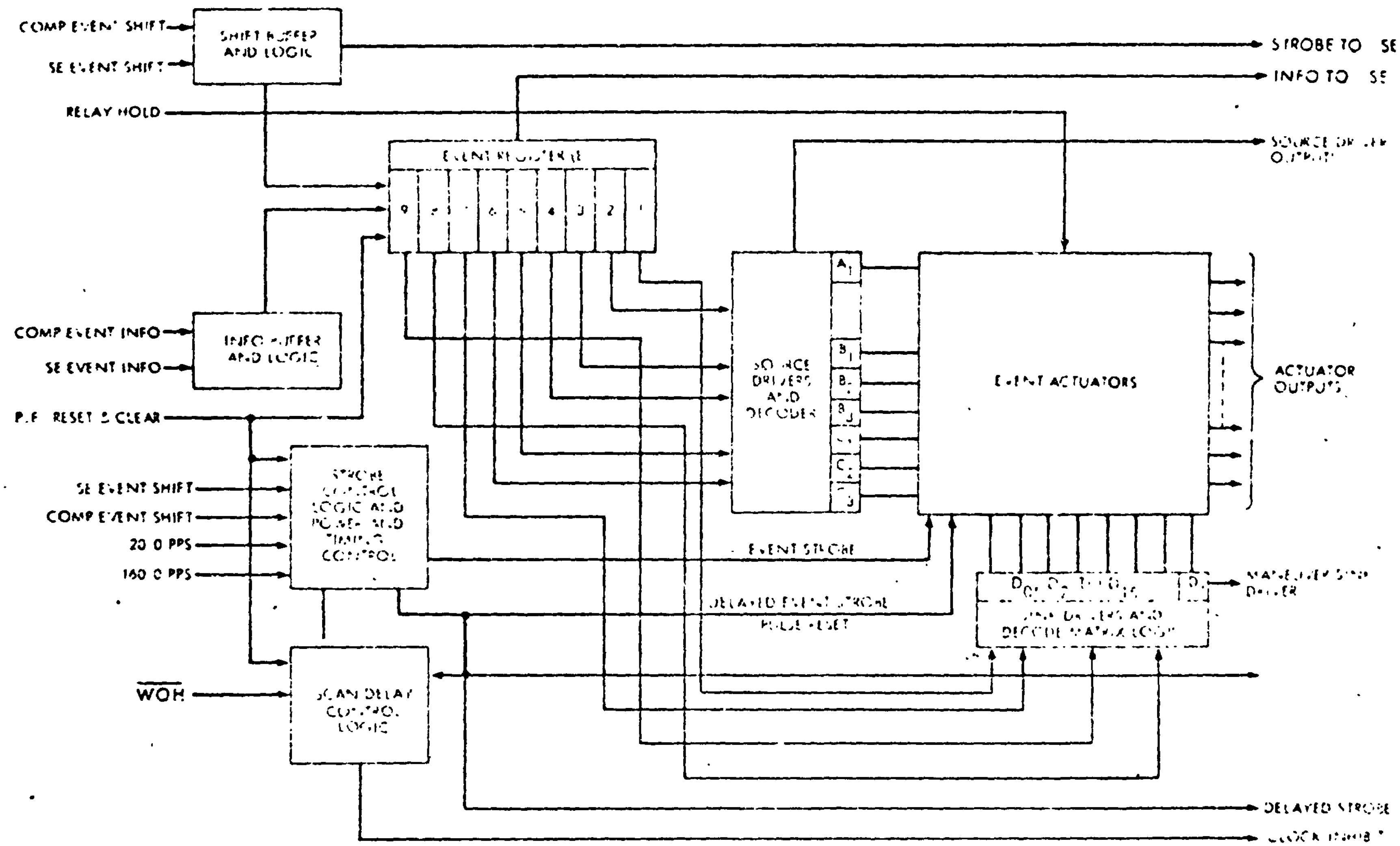


Figure 9.5.11-7 BLOCK DIAGRAM COMPUTER EVENT DECODER

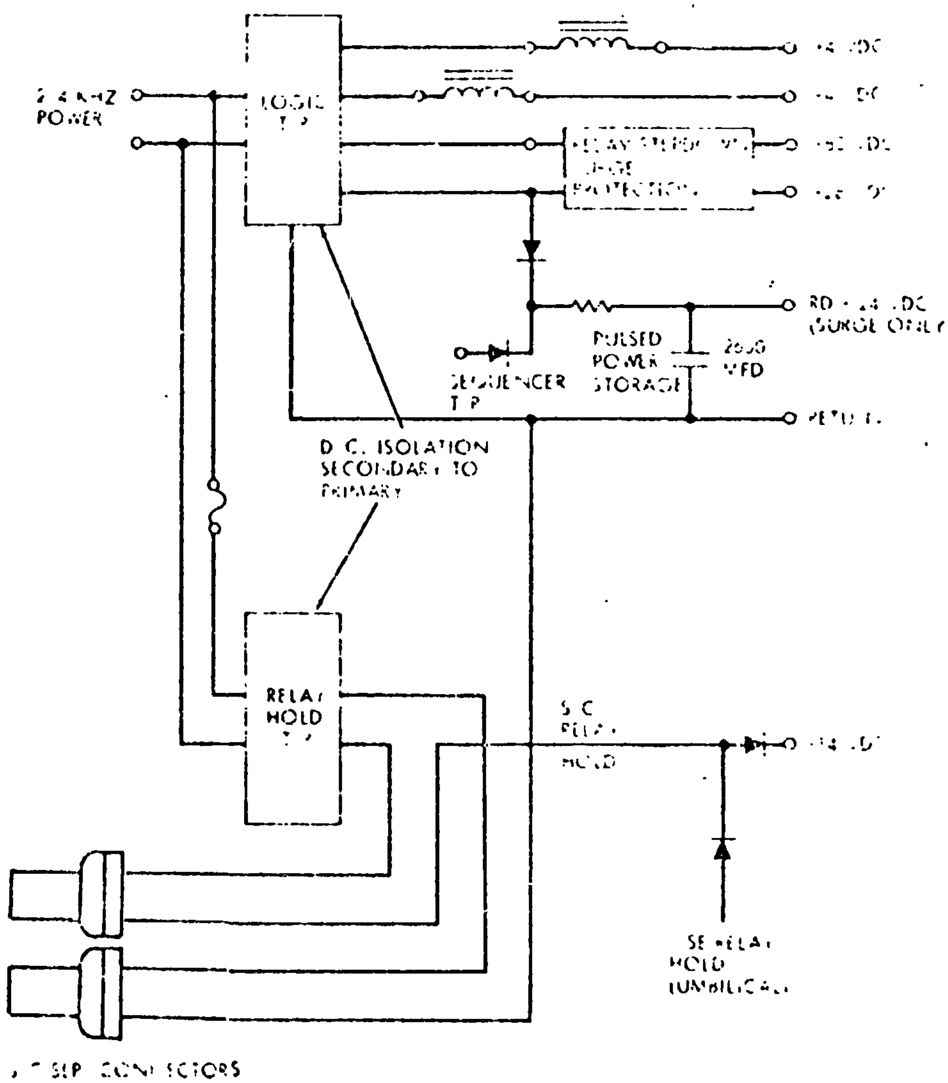


Figure 9.5.11-8 BLOCK DIAGRAM COMPUTER POWER CONVERTER

9.5.11 MARINER MARS DIGITAL COMPUTER

General Description

Program: Mariner Mars 1971
Vendor: Motorola
Part Number:

Performance Characteristics

Central Processor Unit:

Stored program, serial operating, special purpose digital machine.

Slow speed, add time estimated at 66 milliseconds.

Memory:

512 words, 22 bits per word, random access, lithium ferrite cores.

Input/Output

Special purpose output relays control space craft functions.

Physical Characteristics

Size:
Weight:

References

"Central Computer and Sequencer Subsystem," Mariner Mars 1971 Flight Equipment Design Requirement M71-2005-1, Rev. A, Jet Propulsion Laboratory, California Institute of Technology, March 8, 1971.

Design Status

This component was flown as part of the 1971 Mariner Mars mission.

9.6.6-1 OAO-C COARSE INERTIA WHEEL

The function of the coarse wheels is to provide the torques required for slewing the observatory. Three coarse momentum wheels are required, one for each of the spacecraft control axes. Each gyro-momentum package contains one of the coarse wheels. The three wheels are controlled by logic circuitry located in the Coarse Wheel Controller. A "Slew Operation" command enables the Observatory to be slewed in any one of its control axes one axis at a time. Attitude Delay Line Logic in the Primary Processor and Data Storage assembly via the Gyro Logic Unit supplies control signals for coarse wheel selection, direction of rotation, number of wheel rotations to be completed and coarse wheel braking (Figure 9.6.6-1). Maximum slew angle obtainable during one "command" maneuver is 195 degrees. Nominal rates for the OAC-C are 0.1153 deg/sec in roll, 0.0924 deg/sec in pitch and 0.0852 deg/sec in yaw. Settling time is approximately equal to slew time for small slews.

Each coarse momentum wheel consists of an AC servo motor, having essentially linear torque-speed and voltage speed characteristics, driving an inertial wheel and is contained within a hermetically sealed housing. A pulse generator (magnetic pickoff) provides a one-pulse-per revolution output. An electrical brake assembly is included.

a. Motor and Inertial Wheel - The motor and inertial wheel have the following characteristics:

   - Power - The coarse wheel 2-phase motor operates with an input of 26 Vac 400 Hz. The peak power consumed by a single coarse wheel does not exceed 70 watts stall and 80 watts operating under the worst conditions of voltage and frequency input.

   - Phasing - Phasing of the motor supply voltage establishes direction of rotation.

   - Stall Torque - The motor torque at full rated voltage is 32.7 in-oz, minimum.

   - Full Speed Momentum - The no-load speed of the rotating wheel assembly is 950 rpm minimum. The measured mements of inertia are: Roll-0.03329 slug-ft^2; Pitch-0.03341 slug-ft^2; Yaw-0.03085 slug ft^2

b. Brake - The inertial wheel brake has the following characteristics:

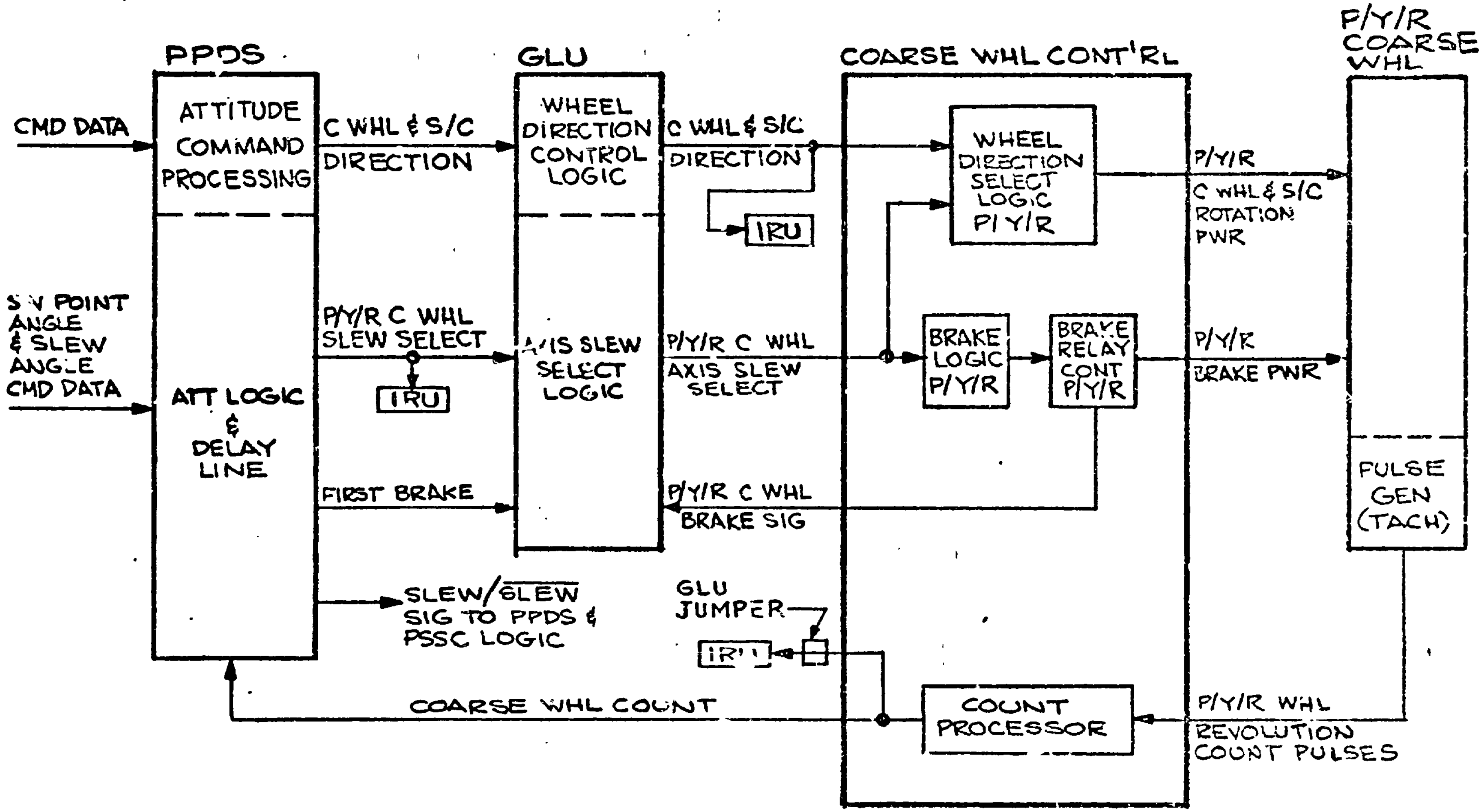


Figure 9.6.6-1 COARSE INERTIA WHEEL CONTROL

- Power - The electrical brake operates with an input of 28 Vdc. The brake is not energized during any period of application of 400 Hz power to the motor winding.

- Brake Torque - The brake is capable of stopping the rotating wheel assembly from 100 rpm in less than one complete wheel revolution after application of brake power, and of stopping the rotating wheel assembly from no-load speed when applied for $12.0 \begin{smallmatrix}+2.5\\-2.0\end{smallmatrix}$ seconds.

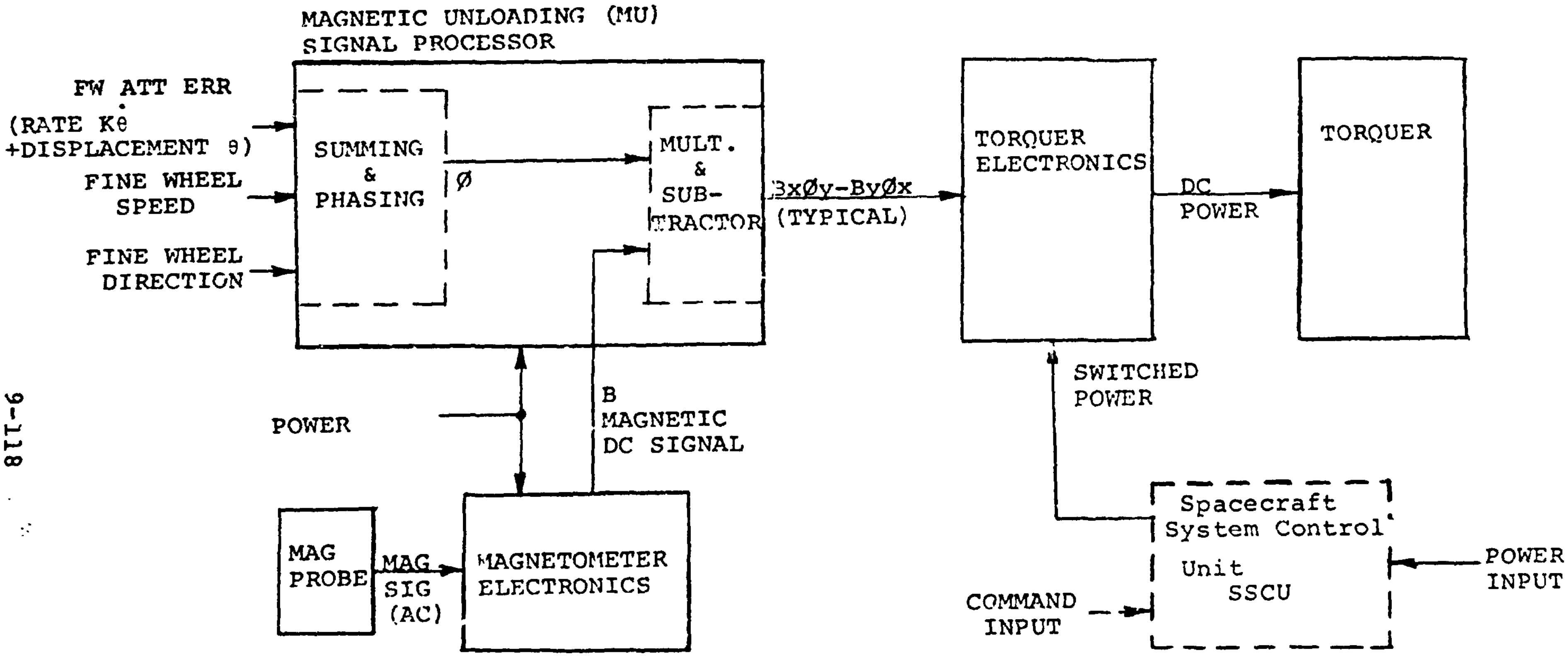
c. Pulse Generator - Each coarse momentum wheel contains an electromagnetic pickup for the purpose of generating one voltage pulse per revolution of the wheel assembly. Wheel assembly oscillation due to braking does not result in the further generation of count pulses in the particular coarse momentum wheel involved.

d. Power Factor Correction Unit - The function of each of the three (pitch, yaw and roll) power factor correction units is to improve the power factor of the electrical load of each of the three coarse inertial wheels as seen by the Coarse Wheel Controller to at least 0.90 lagging under all conditions of wheel operation.

9.6.6-2 OAO-C MAGNETOMETER

The magnetometers sense the components of the earth's magnetic field about each control axis. One each of the three probes is mounted along each vehicle axis. Outputs from the probes are conditioned in the magnetometer electronics and fed to the magnetic unloading signal processor (MU) (Figure 9.6.6-2). The MU computes the vector cross-product of the spacecraft angular momentum vector and the earth's magnetic field. Currents proportional to the cross product vector components are applied to three torque coils. Magnetic reaction between the torque coils and the earth's magnetic field produces a torque on the spacecraft which tends to unload the fine inertia wheels by reducing spacecraft momentum.

The PYR magnetometer outputs may also be used, via telemetry, for attitude determinations without imparting additional momentum to the spacecraft. This is accomplished by commanding the torquer electronics off via the spacecraft system control unit. Attitude determination may thereby be derived in either spacecraft daylight or dark periods.



MAGNETOMETER SYSTEM INTERFACE

TYPICAL FOR P, Y, & R AXES

Figure 9.6.6-2

9.6.6 -2 OAO-C MAGNETOMETER

General Description

Program: OAO-C
Vendor:
Part Number: 252SCAV120-1 (Probe), 252SCAV123-1 (Electronics)

Performance Characteristics

Range: 0 to $\pm$60000 gamma
Accuracy: Greater of $\pm$1% of known field or 300 gamma
Resolution: 100 gamma maximum
Scale Factor: 4.167×10^{-5} $\pm$1% Vdc/gamma
Ripple Content: 25 mV rms maximum
Bandwidth: 10 Hz minimum
Gain Margin: 6 $\pm$0.3 db at phase crossover
Phase Margin: 30° $\pm$3 at gain crossover
Load Impedance: 3000 Ω $\pm$10%
Operating Life: > 12000 hrs.

Physical Characteristics

Size:
Weight: .45 kg (1 lb) each probe, 2.7 kg (6 lbs) electronics
Input Power: <3 watts

Cooling Method: Radiation to skin

References

Magnetometer Stabilization and Control Subsystem, OAO, Grumman Spec AV-252CS-55B, Amend #1, dated April 25, 1968

Design Status

This component was flown as part of the 1972 OAO-C Mission.

9.6.8 SKYLAB ATM CONTROL MOMENT GYRO (CMG)

The ATM contains three double gimballed CMGs. The angular momentum of each CMG (Figure 9.6.8-1) is fixed in magnitude and the arrangement has six-degree-of-freedom.

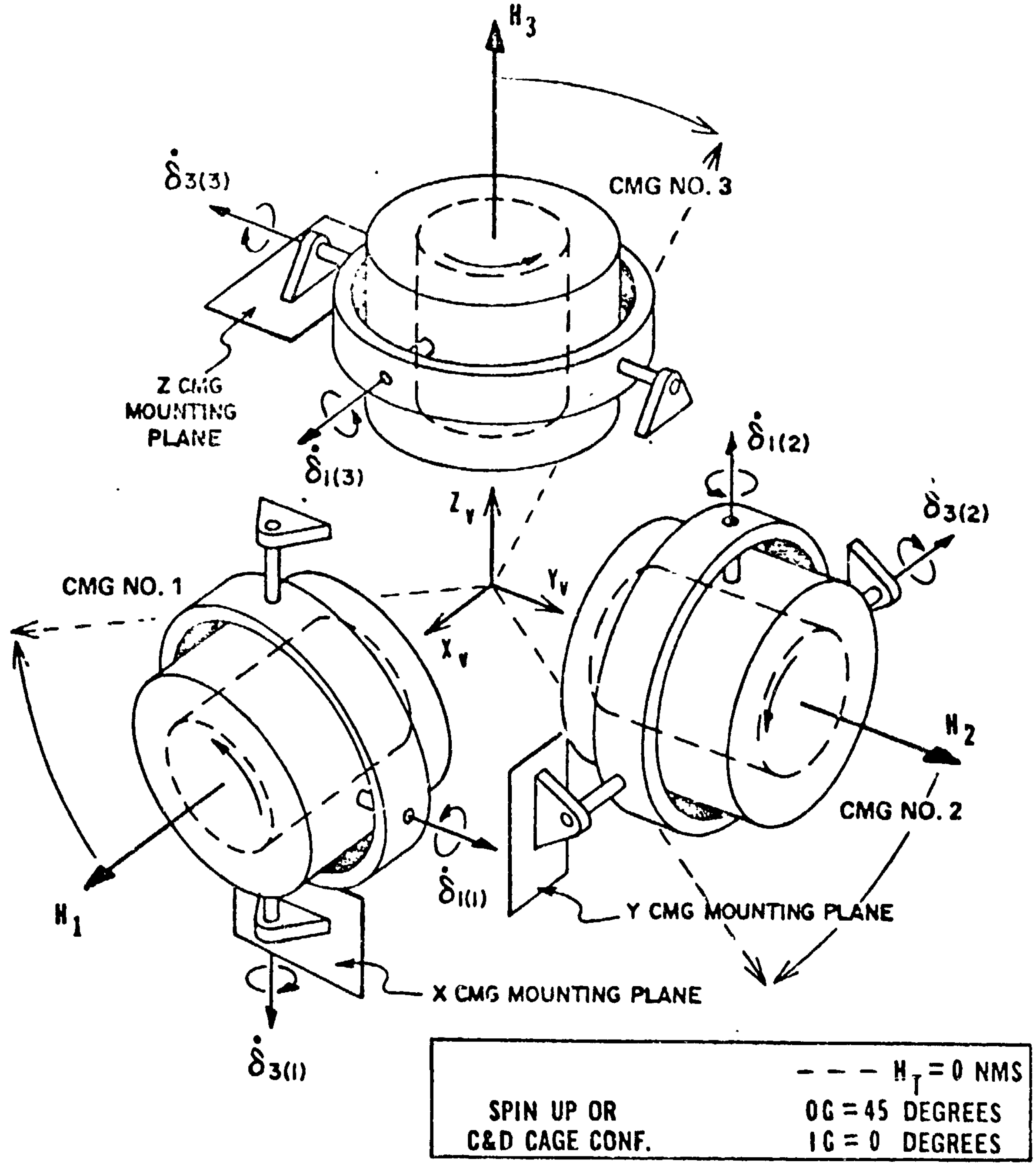
Each CMG consists of a motor driven rotor which is gimballed to provide two-degrees-of-freedom within the limits of the electrical and mechanical stops. The 22-inch diameter rotor, rotating at a speed of 9027 rpm, has a momentum storage capability of approximately 3100 nms. The instrument is capable of producing torques on a vehicle proportional to the angular rate of the gimbals. By controlling the CMG gimbal rates, attitude control of the vehicle can be achieved.

Each gimbal control system contains a rate servo that provides linear gimbal rate control.

Processing electronics for the CMG are contained in the CMG Electronics Assembly (CMGEA). The rate loop for each gimbal consists of a dc torquer, a dc tachometer, a precision speed-reduction gear train, and electronic amplifiers which provide voltage and power gain. These amplifiers are also used to implement servo compensation networks and to provide summing points for the rate command signal and the tachometer feedback signal.

Each of the three CMGs derives its gyro wheel power from a solid state inverter, designated CMG Inverter Assembly (CMGIA) (Figure 9.6.8-2). The 4.8 kHz and 455 Hz power from each CMGIA are fed to a corresponding CMGEA. The CMGIA also supplies 4.8 kHz, 800 Hz, and 455 Hz power to the remainder of the attitude and pointing control subsystem (APCS) via the control distributor. The control distributor performs the switching operations between the primary and secondary reference power. Each CMGIA is capable of supplying the following to the APCS.

- Three-phase, 455 Hz sine wave power, 130 Vrms (line-to-line).
- 800 Hz sine wave power supply, 28 vac, 12 w.
- Two 4.8 kHz sine wave power supplies, 19 Vac, 3 w each
- A $\pm$12 Vdc, 3 w bias power supply used for the conditioning electronics.



CMG GIMBAL ANGLES ARE ZERO, AND THE DIRECTION COSINES ARE (1,0,0), (0,1,0), AND (0,0,1) FOR THE MOMENTUM VECTORS OF CMG #1, CMG #2, AND CMG #3, RESPECTIVELY.

CMG MOUNTING CONFIGURATION

Figure 9.6.8-1

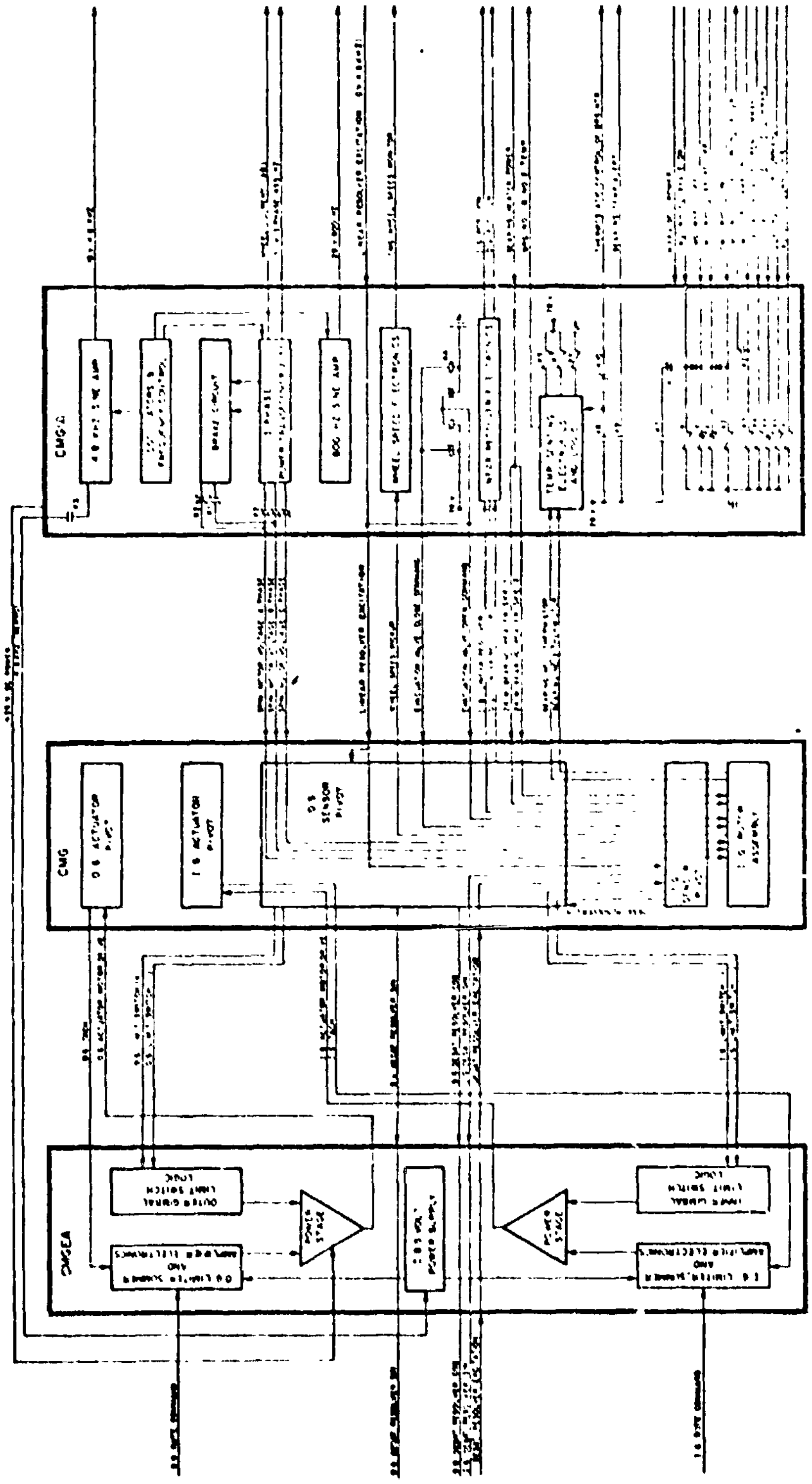


Figure 9.6.8-2 CMG FUNCTIONAL BLOCK DIAGRAM

9.6.8 SKYLAB ATM CONTROL MOMENT GYRO (CMG)

General Description

Program: Skylab ATM
Vendor: Bendix Corporation
Part Number: Bendix #2120100

Performance Characteristics

CMG: Momentum Storage - 3100 nms
Degrees of Freedom - 2
Output Torque - 160 ±25 ft/lbs, when one gimbal is held at 0 deg/sec and the other gimbal is driven from +24° to -45° at a rate of 4 deg/sec
Threshold 0.25 ft/lbs
Gimbal Rates
- Range - 0 to 7.0 degrees per second
- Threshold - 0.0057 degrees per second

Gimbal Rate Servo
- Bandwidth - 70 radians per seconds
- Phase Margin - 70 degrees

Gimbal Pivot
- Actuator - dc motor
- Torque (max) - 7.0 ft/lbs
- Tachometer Gain - 1 Vdc/rad/sec
- Tachometer Amplifier Gain - 1.013
- Gear Ratio - 56.55:1
- Resolvers Speed - Single (4.8 kHz excitation)

Gimbal Freedom
- Inner - ±80 degrees (mechanical)
- Outer - ±220 degrees, -130 degrees (mechanical)

Wheel
- Operating Speed - 9027 rpm
- Acceleration Time - 14 hours maximum; 12 hours to 90% full speed
- Deceleration Time - 4.5 hours

CMIA: Voltage Regulation (all voltages) - 5%
Frequency Regulation (all frequencies)
- Using Primary Oscillator - ±0.002%
- Using Redundant Oscillator - ±1.0%

Voltage Distortion (all voltages) - 5%
Power Output
- Three phase, 455 Hz, 130 V - 600 VA
- Single phase, 800 Hz, 28 V - 12.5 watts
- Single phase, 4.8 kHz, 10 V - 6.0 watts

Power Loads
- CMG Wheel Runup - 130 watts
- CMG Wheel at Run Speed - 55 watts

Distributor Buses
- 455 Hz - 100 VA
- 800 Hz - 12.5 watts
- 4.8 kHz - 6.0 watts

9.6.8 SKYLAB ATM CONTROL MOMENT GYRO (CMG)

Physical Characteristics

Size: 99 cm (39 in) sphere - CMG
25 cm (9.8 in) by 22 cm (8.6 in) by 8 cm (3 in) - CMEA
63 cm (25 in) by 57 cm (22.5 in) by 9 cm (3.5 in) - CMIA
Weight: 142 kg (312 lbs) - CMG
3.7 kg (8.2 lbs) - CMEA
23.7 kg (52 lbs) - CMIA
Input Power: 320 watts max
Cooling Mathod: Radiation to deep space

References

Skylab Operations Handbook Apollo Telescope Mount, Volume 1, Manned Spacecraft Center, July 19, 1971

Design Status

This component was flown as part of the 1973 Skylab Mission.

SECTION 10
COMMUNICATIONS COMPONENTS SUMMARY

COMMUNICATION SUMMARY

COMPONENT · S-BAND ANTENNAS

CHARACTERISTIC / PROGRAM	TYPE	FREQ. RANGE MHz	BANDWIDTH MHz	BEAMWIDTH	VSWR	POLARI-ZATION	GAIN DB	POWER RATING WATTS	SIZE CM	WEIGHT KG (LBS)
HEAO-A	CONICAL LOG SPIRAL	2089 to 2275	186	230°	1.5:1	RHC	-3	5	29 by 12 DIA	.45 (1)
HEAO-A	CONICAL LOG SPIRAL	2089 to 2275	186	150°	1.5:1	RHC	-3	5	8.9 by 12.7 DIA	.45 (1)
OSO-I	SLOT	2212 to 2218	6	OMNI	1.4:1		-6			6.98 (15.3)
VIKING ORBITER	PARABOLOIDAL	2110 to 2300	190	92°	1.3:1	LHC	3	60	192 by 28 DIA	1.7 (3.75)
VIKING ORBITER	PARABOLOIDAL	2110 to 2300 and 8415 ±20	190 and 40	6° and 1.6°	1.2:1	LHC	28 and 39	60	147 DIA	4.6 (10)
APOLLO-17 CSM	PARABOLIC DUAL ANTENNA	2106.4		40° WIDE		RHC	9.2	15	162 by 62 by 90.5	31.4 (69)
APOLLO-17 CSM		2272.5 to 2287.5		4.5° NARROW		RHC	26.7			
MARINER MARS '71 3 ANTENNA SYSTEM	PARABOLIC	2295	10	4.5°	1.2:1	RHC	25	25	1.02 DIA	2.4 (5.2)
MARINER MARS '71 3 ANTENNA SYSTEM	HORN	2295 and 2115	200	39°	1.3:1	RHC	14	25	53 by 23 DIA	0.9 (2)
MARINER MARS '71 3 ANTENNA SYSTEM	CIR. MONO-GUIDE	2295 and 2115	200		1.3:1	RHC	7	25	15 by 28 DIA	2.1 (2.6)

Chart 10-1 S-Band Antenna Summary

COMPONENT - UHF ANTENNAS										
CHARACTERISTIC / PROGRAM	TYPE	FREQ RANGE MHZ	BAND WIDTH MHZ	BEAM-WIDTH	VSWR	POLARI-ZATION	GAIN DB	POWER RATING WATTS	SIZE CM	WEIGHT KG (LBS)
VIKING ORBITER	CONICAL SPIRAL	381	±5		1.2:1	RHC	2.5	60	50.8 BY 76.2	1.5 (3.3)
OAO-C	FOLDED MONOPOLE	400		OMNI	1.2:1	LINEAR	-12 (2 ANT)	5	20.3 BY 8	0.2 (0.4)
SKYLAB AM/MDA	DISEONE	440-460	20	OMNI	2:1	LINEAR	-14		46 BY 38	0.68 (1.5)
SKYLAB AM/MDA	STUB	440-460	20	OMNI	1.5:1	LINEAR	-14		36 BY 3.18	0.2 (0.5)
SKYLAB ATM	DIPOLE	440-460	20	OMNI	1.25:1	LINEAR	-6		23.2 BY 25 BY 2.5	11.4 (25)
SKYLAB ATM	DIPOLE	440-460	20	OMNI	1.3:1	LINEAR	-6		3.8 BY 41 BY 2.5 BY	0.425 (1)
SKYLAB ATM	DIPOLE	440-460	20	OMNI	1.25:1	LINEAR	-6		28 BY 26 BY 2.5	0.425 (1)

Chart 10-2 UHF Antenna Summary

COMPONENT - VHF ANTENNAS										
CHARACTERISTIC / PROGRAM	TYPE	FREQ RANGE MHZ	BAND-WIDTH MHZ	BEAM-WIDTH	VSWR	POLARI-ZATION	GAIN DB	POWER RATING WATTS	SIZE CM	WEIGHT KG (LBS)
ATS-F	MONOPOLE	136-154	18	OMNI	2:1	RHC	-6 (2 ANT)		56 BY 42 BY 2.54	0.215 (0.5)
ATS-F	CROSSED MONOPOLE	136-155	19	OMNI	2:1	RHC	-6 (2 ANT)	10 (AVG)	56 BY 42 BY 2.54	28 (0.6)
OSO-I	ARRAY OF 8 WHIPS	136 TO 150	13.23	OMNI	1.4	RHC	-6 TO -13		50 BY 0.3 (DIA) (PER WHIP)	0.794 (1.75)
OAO-C	SLOT	136 TO 148	12	OMNI		LINEAR	-12	1.6	32 CM SLOT	
SKYLAB AM/MDA	HELIX	250 TO 300	50	50°	2:1	RHC	8 TO 9		81.3 BY 34.5 BY 140	6.8 (15)
SKYLAB ATM	DIPOLE	220 TO 245	25	OMNI	1.25:1	LINEAR	-6 (2 ANT)	30	53 BY 38 BY 2.5	1.13 (2.5)
SKYLAB ATM	DIPOLE	225 TO 245	20	OMNI	1.25:1	LINEAR	-6 (2 ANT)	30	55 BY 40 BY 2.5	0.54 (1.2)

Chart 10-3 VHF Antenna Summary

COMPONENT - TRANSMITTERS							
CHARACTERISTIC / PROGRAM	POWER INPUT WATTS	POWER OUTPUT WATTS	CARRIER FREQ MHZ	MODULATION	BAND-WIDTH	SIZE CM	WEIGHT KG (LBS)
OSO-I	8 @ 23 V 11.5 @ 33 V	1 MIN 2 MIN	2212.5	SPLIT PHASE PCM/PSK	1 MHZ @ 10 DB POINTS	24.4 BY 14.6 BY 7.24	1.5 (3.3)
OAO-C WIDEBAND		8	400.550	FM	62.5 KHZ	26.0 BY 10.1 BY 5.5	1.59 (3.5)
OAO-C NARROWBAND	9 @ 28 V	1.6	136.260	PCM/PSK			1.59 (3.5)
SKYLAB AM/MDA	18.9 @ 27 V	2 MIN 3 MAX	230.4	PCM/FM VOICE		5.7 BY 17 BY 7.0	1.16 (2.56)
SKYLAB AM/MDA	81 @ 2.7 V	10 MIN 13 MAX	235.0	PCM/FM VOICE		11.7 BY 11.7 BY 3.4	0.74 (1.6)
SKYLAB ATM VHF		10 MIN	231.9 237.9	PCM/FM			

Chart 10-4 Transmitter Summary

SECTION 11

DATA MANAGEMENT COMPONENTS SUMMARY

DATA MANAGEMENT SUMMARY

COMPONENT - MULTIPLEXER							
CHARACTERISTIC / PROGRAM	TYPE	MAXIMUM NUMBER OF DATA CHANNELS	MAXIMUM CHANNELS OF ANY TYPE	SAMPLE RATE	OUTPUT	SIZE CM	WEIGHT KG (LBS)
HEAO-A		250			25.6 KBPS MANCHESTER (8 BIT WORDS)	2.8 BY 5.2 BY 20.3	.45 (1.0)
OSO-I		32	*A-16 PD-32 SD-1		32 KBPS 8-BIT WORDS NRZ-L PAM	10.2 BY 8.89 BY 3.18	.23 (0.5)
SKYLAB AM/MDA	HIGH LEVEL	32	*A-32 PD-24	A 1.25 SPS PD 10 SPS	PAM DATA AT SAMPLE RATE	NA	NA
SKYLAB AM/MDA	LOW LEVEL		*A-8	1.25 0.416 SPS		NA	NA
SKYLAB ATM	DIGITAL	100	*PD-100	12,120 SPS	10 10-BIT WORDS	15.2 BY 27.96 BY 25.4	7.3 (16)
SKYLAB ATM	ANALOG	234	*A-234	12,120 SPS	PAM	15.2 BY 27.9 BY 33	9.5 (21)
SKYLAB ATM	ANALOG SUBMUX	60	*A-60	12 SPS	PAM		
*PD-PARALLEL BILEVEL DIGITAL INPUTS SD-SERIAL DIGITAL INPUTS A-ANALOG INPUT							

Chart 11-1 Multiplexer Summary

COMPONENT - TAPE RECORDER

CHARACTERISTIC / PROGRAM	TYPE	STORAGE CAPACITY (BITS)	RECORD TAPE SPEED (IPS)	PLAYBACK TAPE SPEED (IPS)	NUMBER TRACKS/ BIT PACKING DENSITY (BITS/ IN/TRACK)	TAPE WIDTH (IN) AND LENGTH (FEET)	SIZE CM	WEIGHT KG (LBS)
HEAO-A	DIGITAL	4.5×10^8	1.54	28	3/5940	¼ x 2100	22.9 by 30.5 by 12.7	7 (15.5)
HEAO-A	DIGITAL	4.5×10^8	1.10	20.5	3/8330	¼ x 1500	35.8 DIA by 15	9.4 (20.7)
OSO-I	DIGITAL	84.5×10^6	220 Minutes				32.5 by 25.9 by 13.2	7 (15.3)
OAO-C	DIGITAL	48.5×10^6	0.297	18.297	9/375	¼ x 1150	30.5 by 19.1 DIA	7.7 (17)
SKYLAB ATM	DIGITAL AND ANALOG	21.4×10^9	89 MINUTES	290 to 300 Sec.			22.9 by 22.9 by 12	5.4 (12)
MARINER MARS '71	DIGITAL	1.8×10^8	19.44	2.4 to 0.15	9/756	550 ft	22.9 by 15.5 by 17.8	5 (11)

Chart 11-2 Tape Recorder Summary

COMPONENT - CLOCK/TIMER

CHARACTERISTIC / PROGRAM	TYPE	OUTPUT	STABILITY	RESOLUTION	DISPLAY CAPACITY	TIMING OUTPUT	SIZE CM	WEIGHT KG (LBS)
HEAO-A	CLOCK	1.024 MHz 512 KHz	± 1 IN 10^9 over 24 Hrs.	4×10^{-5} Sec for TM 1×10^{-6} Sec for TIME TAGS			4 by 15.2 20.3	.9 (2.0)
OSO-I	WHEEL CLOCK	TABLE 7.3.4-1 PAGE 7-59	± 1 IN 10^9 PER DAY			5.120 MHz	41.9 by 14.7 by 8.4	3.4 (7.5)
OSO-I	SAIL CLOCK	TABLE 7.3.4-2 PAGE 7-63				64 KHz	19.6 by 14.7 by 3.5	0.6 (1.2)
SKYLAB AM/MDA	ELECTRONIC TIMER	8 PPS	0.125 Sec PER DAY	0.125 Sec				
SKYLAB AM/MDA	EVENT CLOCK		0.125 Sec PER DAY		999 Hrs. 59 Min. 59 Sec.	8 PPS		
SKYLAB AM/MDA	PORTABLE TIMER	800 Hz AUDIO TONE	0.6 Sec PER DAY					
MARINER MARS '71	TIMER	TABLE 7.3.11-1 PAGE 7-88	$\pm 0.01\%$			480 KHz		

Chart 11-3 Clock/Timer Summary

COMPONENT - PROGRAMMERS/FORMATTERS/ENCODER/DATA HANDLING EQUIPMENT										
CHARACTERISTIC / PROGRAM	DESCRIPTION	INPUT ANALOG TOTAL	INPUT DISCRETES	INPUT COMMANDS	OUTPUT FORMAT	OUTPUT CAPACITY	OUTPUT WORD RATE	OUTPUT MAJOR FRAME	OUTPUT MINOR FRAME	SIZE CM / WEIGHT KG
ATS-F	DATA ACQUISITION AND CONTROL UNIT	276 CHANNELS	79-9 BIT WORDS 1-72 BIT WORDS	12 COMMANDS	FIXED MAN. II + 180°	368 9 BIT WORDS	43.40/SEC	16 MINOR FRAMES	128 WORDS	31 BY 21.6 BY 21.6 93 KG
HEAO-A	CENTRAL PROGRAM UNIT		25.6 KBPS			256 8 BIT WORDS	32 WORDS/SEC	128 MINOR FRAMES	256 WORDS	3.3 BY 15.2 BY 20.3 2.7 KG
OSO-I	ENCODER				NRZ-L/PCM MAN.	32 KBPS DATA STREAM				33.5 BY 4.7 BY 3.6 1.54 KG
OSO-I	FORMAT GENERATOR					6400 BITS PER SEC		128 MINOR FRAMES (20.48 SEC)	128 WORDS (160 MILLSEC)	37 BY 14.7 BY 3.6 1.33 KG
OAO-C	DATA HANDLING EQUIP.	264 CHANNELS			NRZ		65 WORDS PER SUB FRAME	520 26 BIT WORDS	65 WORDS	15 BY 32 BY 24 13.6 KG
OAO-C	EXPERIMENT DATA HANDLING EQUIP.	30 CHANNELS	200 BI-LEVEL CHANNELS					21 26 BIT WORDS		15 BY 24 BY 24

Chart 11-4 Programmer/Formatter Summary

[redacted]

COMPONENT - PROGRAMMERS/FORMATTERS/ENCODER/DATA HANDLING EQUIPMENT (CONTINUED)

CHARACTERISTIC / PROGRAM	DESCRIPTION	INPUT ANALOG TOTAL	INPUT DISCRETES	INPUT COMMANDS	OUTPUT FORMAT	OUTPUT CAPACITY	OUTPUT WORD RATE	OUTPUT MAJOR FRAME	OUTPUT MINOR FRAME	SIZE CM / WEIGHT KG
SKYLAB AM/MDA	PROGRAMMER	32 MAX @ HI LEVEL 9 MAX @ LO LEVEL			NRZ-C	51.2 KBPS	640 8 BIT WORDS/ SEC	25.6 K WORDS PER 2.4 SEC DATA CYCLE	160 8 BIT WORDS/ .25 SEC	
SKYLAB ATM	MODEL 301 PCM/DDAS	30 CHAN- NELS @ 120 SPS	100 BITS		NRZ	72 KBPS	7200/ SEC	7200 10 BIT WORDS PER SEC	1800 10 BIT WORDS PER .25 SEC	23 BY 23 BY 33 12.2 KG

Chart 11-4 Programmer/Formatter Summary (Continued)

SECTION 12

ELECTRICAL POWER COMPONENTS SUMMARY

ELECTRICAL POWER SUMMARY

COMPONENT - SOLAR ARRAY										
CHARACTERISTIC / PROGRAM	ARRAY TYPE	CELL TYPE	ARRAY AREA M^2 (FT^2)	PANELS PER ARRAY	POWER OUTPUT (ARRAY) WATTS	VOLTAGE (VOLTS)	CELL EFFI-CIENCY %	NUMBER OF CELLS	SIZE CM	WEIGHT KG (LBS)
ATS-F	TWO SEMI-CYLINDRICAL ASSEMBLIES	N ON P SILICON	16 (186)	32	600		11.0	21,600		
ERTS-B	TWO PADDLES	N ON P SILICON	23 (256.7) PER PADDLE		550	38	11.4			
OSO-I	ONE PANEL	N ON P	4.5 (50)	1	400	32				7.6 (17)
OAO-C	EIGHT PADDLES	N ON P SILICON	11.1 (119.5)	8	610	23 to 34	13.0	55,544		
SKYLAB AM/MDA	TWO WINGS	N ON P SILICON	120.3 (1340)	60	10497		11.1	147,840		1841 (4056)
SKYLAB ATM	FOUR WINGS	N ON P SILICON	108 (1200)	18	11700		10.0	82,080 246,240		1725 (3800)
MARINER MARS '71	FOUR PANELS	N ON P SILICON	7.5 (83)	4	47.5	38 to 50		17,472		22.7 (50)

Chart 12-1 Solar Array Summary

COMPONENT - BATTERY						
CHARACTERISTIC / PROGRAM	TYPE	NUMBER OF CELLS	BATTERY VOLTAGE (VOLTS)	A/HRS	NUMBER OF BATTERIES	WEIGHT PER BATTERY KG (LBS)
ATS-F	NI-CD	19	22.8-28.5	15	2	
ERTS-B	NI-CD	23	28.0	4.5	8	
HEAO-A	NI-CD	22	24.2	30	4	30.9 (68)
OSO-I	NI-CD	21	28	12	2	15.0 (33)
OAO-C	NI-CD	22	26-32	20	3	37.2 (82)
SKYLAB AM/MDA	NI-CD	30	30-42	33	8	54.5 (120)
SKYLAB ATM	NI-CD	24	30-35	20	18	22.7 (48)
APOLLO-17 CSM	SILVER OXIDE ZINC	20	34-27.8	40	5	
APOLLO-17 CSM	SILVER OXIDE ZINC	20	30-32.5	296	2	59.5 (131)
Mariner Mars '71	NI-CD	26	27-36	20	1	28.1 (62)

Chart 12-2 Battery Summary

COMPONENT - REGULATORS

CHARACTERISTIC / PROGRAM	INPUT CURRENT (AMPS)	INPUT VOLTAGE D.C.	OUTPUT CURRENT (AMPS)	OUTPUT VOLTAGE D.C.	EFFICIENCY %	POWER OUTPUT WATTS	SIZE	WEIGHT KG (LBS)
ERTS-B		38 MAX.	22 MAX			550		
ERTS-B		38 MAX.	26 MAX.			650		
OSO-I		31 to 35		28				0.6 (1.3)
OAO-C	10 to 15	25 to 32		+28 -28 +10 -10 +18		135.7 11.5 48.7 17.6 81		
SKYLAB AM/MDA		33 to 125	0 to 50	24 to 30	92	1600	27.7 by 25.4 by 11	6.4 (14)
SKYLAB ATM		25.5 to 80	20 MAX.	30 ± 0.9	89	235 AVE. 415 MAX.		
MARINER MARS 71		25 to 50		56	85 to 90	295 MAX. 50 MIN.		2.8 (6.2)
MARINER MARS 71		23.5 to 50		30		150 MAX. 6 MIN.		2.7 (6)

Chart 12-3 Regulator Summary

COMPONENT - INVERTERS

CHARACTERISTIC / PROGRAM	INPUT VOLTAGE (DC)	OUTPUT VOLTAGE (AC)	OUTPUT CURRENT (AMPS)	OUTPUT FREQUENCY Hz	EFFICIENCY %	SIZE CM	WEIGHT KG (LBS)
OAO-C	25 to 35	26, 1Ø 26, 3Ø	6.75	400	62 to 68		16.7 (37)
APOLLO-17 LM	28	115	?	400			
MARINER MARS 71	55 VDC 50 V, 2.4 KHz	28, 1Ø 27.2, 3Ø			83 to 85		2.3 (5)
MARINER MARS 71	56	50		2.4K	89		1.8 (4)

Chart 12-4 Inverter Summary

COMPONENT - BATTERY CHARGER

CHARACTERISTIC / PROGRAM	INPUT VOLTAGE VDC	INPUT CURRENT AMPS	OUTPUT VOLTAGE VDC	OUTPUT CURRENT AMPS	EFFICIENCY %	SIZE CM	WEIGHT KG (LBS)
OSO-I	23 to 33		23 to 32.6				1.4 (3)
SKYLAB AM/MDA	0 to 125	56.4	39 to 48	52.5	93	28.9 by 25.4 by 18.6	12.3 (27)
SKYLAB ATM	85	15.2	33.85 to 35.65	14	92		
APOLLO-17 CSM	25 to 30 VDC; 115 VAC		39	2.7			
MARINER MARS 71	25 to 50		26 to 39	2 (HIGH) 0.65 (LOW)			1 (2.3)

Chart 12-5 Battery Charger Summary

SECTION 13

ATTITUDE AND CONTROL COMPONENTS SUMMARY

COMPONENT - STAR TRACKER

CHARACTERISTIC / PROGRAM	TYPE	FIELD OF VIEW	SENSITIVITY	ACCURACY	SIZE CM.	WEIGHT Kg (lbs)
HEAO-A		$\pm 32^\circ$ (2 AXES)	BRIGHTEST STAR IN FOV		15.2 by 15.2 by 30.5	6.6 (14.5)
OSO-I		4° IN. AZIMUTH		3 ARC MINUTES		6.6 (14.5)
OAO-C	GIMBALLED	1° by 1° SQUARE	CLASS Ao +2 MAG.	2 ARC MINUTES		11.3 (25)
OAO-C	BORESIGHTED	10 ARC MINUTES	CLASS Bo +6 MAG.	2 ARC SECONDS		
OAO-C	FIXED	8° by 8° SQUARE	+3 MAG.	10 ARC SECONDS	15.2 by 20.3 by 35.6	9.1 (20)
SKYLAB ATM		1° ACQ. 10 ARC MIN. TRACK		10 ARC SECONDS		32.7 (72)
MARINER MARS		3° by 11.8° SCANNED				2.3 (5)

Chart 13-1 Star Tracker Summary

COMPONENT - DIGITAL COMPUTER

CHARACTERISTIC / PROGRAM	TYPE	ADD TIME (USEC)	MULT TIME (USEC)	MEMORY TYPE	MEMORY CAPACITY	MEMORY CYCLE TIME (USEC)	INPUT OUTPUT	SIZE CM	WEIGHT KG (LBS)
HEAO-A	GENERAL PURPOSE, BINARY, PARALLEL	2.4	10.4	[illegible]DRO PLATED WIRE	8K to 64K [illegible]RDS, 16 BITS/WORD	2.4	90 KHz DIRECT MEMORY ACCESS; 400 KHz MUX; 130 KHz SERIAL BIT	10.7 by 13 by 36.6	6.2 (13.7)
OAO-C	GENERAL PURPOSE, BINARY, PARALLEL	10	68	CORE	4K WORDS, 18 BITS/WORD	2	PARALLEL CHANNELS; 1 KHz TELE-METRY; 50 KHz MEMORY DUMP	1260 cm^3	28.4 (62.6)
SKYLAB ATM	GENERAL PURPOSE, BINARY, PARALLEL	9 to 24	48 or 54	CORE	16K WORDS, 16 BITS/WORD	2.5	ANALOG, DIGI-TAL, DISCRETE CHANNELS	48.3 by 18.0 by 80.7	44.5 (98)
MARINER MARS	SPECIAL PURPOSE, BINARY, SERIAL	66 Milli-seconds		CORE	512 WORDS, 22 BITS/WORD		SPECIAL PUR-POSE OUTPUT RELAYS		

Chart 13-2 Digital Computer Summary