

APOLLO OPERATIONS HANDBOOK

LUNAR MODULE 4

VOLUME I SUBSYSTEMS DATA

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TOTAL NUMBER OF PAGES IN THIS PUBLICATION IS 866 CONSISTING OF THE FOLLOWING:

Page No.	Issue	Page	Issue
Title Page	15 September 1968	2.3-16 thru 2.3-23	Original
A thru C	15 September 1968	2.3-24 thru 2.3-25	15 September 1968
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1-1 thru 1-5	Original	2.3-30	15 September 1968
1-6	15 September 1968	2.3-31	Original
1-6A thru 1-6B	15 September 1968	2.3-32 thru 2.3-33	15 September 1968
1-7 thru 1-10	Original	2.3-34 thru 2.3-54	Original
1-11	15 September 1968	2.4-1 thru 2.4-14	Original
1-12 thru 1-22	Original	2.4-15	15 September 1968
2.1-1 thru 2.1-2	Original	2.4-16 thru 2.4-32	Original
2.1-3 thru 2.1-4	15 September 1968	2.5-1	15 September 1968
2.2-1	Original	2.5-2 thru 2.5-3	Original
2.2-2 thru 2.2-8	15 September 1968	2.5-4 thru 2.5-8	15 September 1968
2.2-8A thru 2.2-8B	15 September 1968	2.5-9 thru 2.5-19	Original
2.2-9 thru 2.2-13	15 September 1968	2.5-20	15 September 1968
2.2-14 thru 2.2-20	15 September 1968	2.5-20A thru 2.5-20B	15 September 1968
2.2-21 thru 2.2-22	15 September 1968	2.5-21	Original
2.2-22A thru 2.2-22B	15 September 1968	2.5-22	15 September 1968
2.2-23 thru 2.2-24	Original	2.5-23 thru 2.5-24	Original
2.2-25 thru 2.2-26	15 September 1968	2.5-25	15 September 1968
2.2-27 thru 2.2-31	Original	2.5-26	Original
2.2-32	15 September 1968	2.5-27	15 September 1968
2.2-33 thru 2.2-52	Original	2.5-28	Original
2.2-53 thru 2.2-80	15 September 1968	2.5-29 thru 2.5-38	15 September 1968
2.2-80A thru 2.2-80N	15 September 1968	2.6-1 thru 2.6-2	15 September 1968
2.2-81 thru 2.2-104	15 September 1968	2.6-3	Original
2.2-104A thru 2.2-104B	15 September 1968	2.6-4	15 September 1968
2.2-105 thru 2.2-110	Original	2.6-5 thru 2.6-6	Original
2.2-110A thru 2.2-110B	15 September 1968	2.6-7 thru 2.6-17	15 September 1968
2.2-111 thru 2.2-114	Original	2.6-18	Original
2.2-115	15 September 1968	2.6-19 thru 2.6-27	15 September 1968
2.2-116 thru 2.2-122	Original	2.6-28	Original
2.2-123 thru 2.2-124	15 September 1968	2.6-29 thru 2.6-34	15 September 1968
2.2-125 thru 2.2-146	Original	2.6-35 thru 2.6-36	Original
2.2-147	15 September 1968	2.6-37 thru 2.6-40	15 September 1968
2.2-148 thru 2.2-156	Original	2.6-41 thru 2.6-42	Original
2.2-157	15 September 1968	2.6-43 thru 2.6-52	15 September 1968
2.2-158	Original	2.6-53 thru 2.6-54	Original
2.2-159 thru 2.2-162	15 September 1968	2.7-1 thru 2.7-4	15 September 1968
2.2-162A thru 2.2-162J	15 September 1968	2.7-5 thru 2.7-6	Original
2.2-163 thru 2.2-175	15 September 1968	2.7-7	15 September 1968
2.2-176 thru 2.2-177	Original	2.7-8 thru 2.7-10	Original
2.2-178	15 September 1968	2.7-11	15 September 1968
2.2-179 thru 2.2-180	Original	2.7-12 thru 2.7-14	Original
2.2-180A thru 2.2-180B	15 September 1968	2.7-15	15 September 1968
2.2-181 thru 2.2-182	15 September 1968	2.7-16 thru 2.7-17	Original
2.2-183 thru 2.2-186	Original	2.7-18	15 September 1968
2.2-187 thru 2.2-190	15 September 1968	2.7-19	Original
2.2-190A thru 2.2-190D	15 September 1968	2.7-20 thru 2.7-21	15 September 1968
2.2-191 thru 2.2-222	Original	2.7-22	Original
2.3-1	Original	2.7-23	15 September 1968
2.3-2	15 September 1968	2.7-24 thru 2.7-28	Original
2.3-3 thru 2.3-8	Original	2.7-29 thru 2.7-30	15 September 1968
2.3-9 thru 2.3-10	15 September 1968	2.7-31 thru 2.7-34	Original
2.3-11 thru 2.3-12	Original	2.7-35	15 September 1968
2.3-13	15 September 1968	2.7-36	Original
2.3-14	Original	2.7-37	15 September 1968
2.3-15	15 September 1968	2.7-38	Original

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LMA790-3-LM-4
APOLLO OPERATIONS HANDBOOK

Page No.	Issue	Page No.	Issue
2.7-38A thru 2.7-38B . . .	15 September 1968	2.11-16	Original
2.7-39	Original	2.11-17 thru 2.11-20	15 September 1968
2.7-40	15 September 1968	2.11-21	Original
2.7-40A thru 2.7-40B . . .	15 September 1968	2.11-22	15 September 1968
2.7-41	Original	2.11-23	Original
2.7-42 thru 2.7-44 . . .	15 September 1968	2.11-24 thru 2.11-28	15 September 1968
2.8-1	15 September 1968	3-1 thru 3-30	Original
2.8-2	Original	3-31	15 September 1968
2.8-2A thru 2.8-2B . . .	15 September 1968	3-32 thru 3-38	Original
2.8-3 thru 2.8-5 . . .	15 September 1968	3-39 thru 3-41	15 September 1968
2.8-6 thru 2.8-8 . . .	Original	3-42 thru 3-43	Original
2.8-9	15 September 1968	3-44 thru 3-47	15 September 1968
2.8-10 thru 2.8-17 . . .	Original	3-48 thru 3-84	Original
2.8-18	15 September 1968	3-85	15 September 1968
2.8-19 thru 2.8-24 . . .	Original	3-86 thru 3-150	Original
2.9-1 thru 2.9-88 . . .	Original	3-151 thru 3-152	15 September 1968
2.10-1 thru 2.10-14 . . .	15 September 1968	3-153 thru 3-154	Original
2.11-1	15 September 1968	3-155 thru 3-156	15 September 1968
2.11-2 thru 2.11-4 . . .	Original	3-157 thru 3-158	Original
2.11-5 thru 2.11-7 . . .	15 September 1968	A-1 thru A-4	15 September 1968
2.11-8 thru 2.11-14 . . .	Original	B-1 thru B-2	Original
2.11-15	15 September 1968	C-1 thru C-16	15 September 1968

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TABLE OF CONTENTS

Section		Page
1	SPACECRAFT	
	Introduction	1-1
1.1	LM 4 Configuration	1-1
1.2	Ascent Stage	1-1
1.2.1	General Description	1-1
1.2.2	Structure	1-1
1.3	Descent Stage	1-15
1.3.1	General Description	1-15
1.3.2	Structure	1-15
1.4	LM - SLA - S-IVB Connections	1-19
1.5	LM-CSM Interfaces	1-19
1.5.1	Crew Transfer Tunnel	1-19
1.6	Stowage Provisions	1-21
2	SUBSYSTEMS DATA	
	Introduction	2-1
2.1	Subsystem Interface	2.1-1
2.2	Guidance, Navigation, and Control Subsystem	2.2-1
2.2.1	Introduction	2.2-1
2.2.2	Subsystem Interfaces	2.2-6
2.2.3	Functional Description	2.2-8
2.2.4	Major Component/Functional Description	2.2-130
2.2.5	Failure Modes	2.2-191
2.2.6	Performance and Design Data	2.2-194
2.2.7	Operational Limitations and Restrictions	2.2-203
2.2.8	Telemetry Measurements	2.2-205
2.3	Main Propulsion Subsystem	2.3-1
2.3.1	Introduction	2.3-1
2.3.2	Descent Propulsion Section Interfaces	2.3-1
2.3.3	Descent Propulsion Section Functional Description	2.3-2
2.3.4	Descent Propulsion Section Major Component/Functional Description	2.3-16
2.3.5	Descent Propulsion Section Failure Modes	2.3-26

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK

TABLE OF CONTENTS (cont)

Section		Page
2.3	Main Propulsion Subsystem (cont)	
2.3.6	Descent Propulsion Section Performance and Design Data	2.3-28
2.3.7	Descent Propulsion Section Operational Limitations and Restrictions	2.3-30
2.3.8	Descent Propulsion Section Telemetry Measurements	2.3-31
2.3.9	Ascent Propulsion Section Interfaces	2.3-34
2.3.10	Ascent Propulsion Section Functional Description	2.3-34
2.3.11	Ascent Propulsion Section Major Component/Functional Description	2.3-42
2.3.12	Ascent Propulsion Section Failure Modes	2.3-48
2.3.13	Ascent Propulsion Section Performance and Design Data	2.3-48
2.3.14	Ascent Propulsion Section Operational Limitations and Restrictions	2.3-51
2.3.15	Ascent Propulsion Section Telemetry Measurements	2.3-51
2.4	Reaction Control Subsystem	2.4-1
2.4.1	Introduction	2.4-1
2.4.2	Subsystem Interfaces	2.4-2
2.4.3	Functional Description	2.4-5
2.4.4	Major Component/Functional Description	2.4-16
2.4.5	Failure Modes	2.4-19
2.4.6	Performance and Design Data	2.4-22
2.4.7	Operation Limitations and Restrictions	2.4-24
2.4.8	Telemetry Measurements	2.4-25
2.5	Electrical Power Subsystem	2.5-1
2.5.1	Introduction	2.5-1
2.5.2	Subsystem Interfaces	2.5-1
2.5.3	Functional Description	2.5-3
2.5.4	Major Component/Functional Description	2.5-23
2.5.5	Failure Modes	2.5-33
2.5.6	Performance and Design Data	2.5-35
2.5.7	Operational Limitations and Restrictions	2.5-36
2.5.8	Telemetry Measurements	2.5-37
2.6	Environmental Control Subsystem	2.6-1
2.6.1	Introduction	2.6-1
2.6.2	Subsystem Interfaces	2.6-9
2.6.3	Functional Description	2.6-10
2.6.4	Major Component/Functional Description	2.6-23
2.6.5	Failure Modes	2.6-39
2.6.6	Performance and Design Data	2.6-42
2.6.7	Operational Limitations and Restrictions	2.6-48
2.6.8	Telemetry Measurements	2.6-48
2.7	Communications Subsystem	2.7-1
2.7.1	Introduction	2.7-1
2.7.2	Subsystem Interfaces	2.7-1
2.7.3	Functional Description	2.7-2
2.7.4	Major Component/Functional Description	2.7-7
2.7.5	Failure Modes	2.7-38A
2.7.6	Performance and Design Data	2.7-38A
2.7.7	Operational Limitations and Restrictions	2.7-38A
2.7.8	Telemetry Measurements	2.7-44

TABLE OF CONTENTS (cont)

Section		Page
2.8	Explosive Devices Subsystem	2.8-1
2.8.1	Introduction	2.8-1
2.8.1A	Subsystem Interfaces	2.8-2A
2.8.2	Functional Description	2.8-3
2.8.3	Major Component/Functional Description	2.8-15
2.8.4	Failure Modes	2.8-19
2.8.5	Performance and Design Data	2.8-21
2.8.6	Operational Limitations and Restrictions	2.8-22
2.8.7	Telemetry Measurements	2.8-22
2.9	Instrumentation Subsystem	2.9-1
2.9.1	Introduction	2.9-1
2.9.2	Subsystem Interfaces	2.9-1
2.9.3	Functional Description	2.9-2
2.9.4	Major Component/Functional Description	2.9-8
2.9.5	Failure Modes	2.9-75
2.9.6	Performance and Design Data	2.9-78
2.9.7	Operational Limitations and Restrictions	2.9-87
2.9.8	Telemetry Measurements	2.9-87
2.10	Lighting	2.10-1
2.10.1	Introduction	2.10-1
2.10.2	Functional Description	2.10-1
2.10.3	Failure Modes	2.10-12
2.10.4	Performance and Design Data	2.10-12
2.10.5	Operational Limitations and Restrictions	2.10-13
2.11	Crew Personal Equipment	2.11-1
2.11.1	Introduction	2.11-1
2.11.2	Extravehicular Mobility Unit	2.11-1
2.11.3	Umbilical Assembly	2.11-17
2.11.4	Crew Life Support	2.11-17
2.11.5	Crew Support and Restraint Equipment	2.11-19
2.11.6	Docking Aids and Tunnel Hardware	2.11-21
2.11.7	Crew Miscellaneous Equipment	2.11-23
2.11.8	Stowage Locations	2.11-24
3	CONTROLS AND DISPLAYS	
	ABBREVIATION LIST	A-1
	LIST OF SYMBOLS	B-1
	ALPHABETICAL INDEX	C-1

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

LIST OF ILLUSTRATIONS

Figure No.		Page
1-1	Apollo-Saturn Space Vehicle	1-2
1-2	LM 4 Configuration	1-3
1-3	LM 4 Overall Dimensions	1-4
1-4	Station Reference Measurements	1-5
1-5	Ascent Stage Configuration (3 Sheets)	1-7
1-6	Flight Stations	1-10
1-7	Forward Hatch	1-11
1-8	Front Window	1-12
1-9	Docking Window	1-12
1-10	Overhead Hatch	1-14
1-11	Descent Stage Configuration	1-16
1-12	Landing Gear	1-18
1-13	LM-CSM Reference Axis	1-20
2.2-1	Guidance, Navigation, and Control Subsystem - Equipment Location	2.2-1
2.2-1A	GN&CS Simplified Block Diagram and Subsystem Interfaces	2.2-3
2.2-2	LM Rendezvous Radar and CSM Target Orientation	2.2-5
2.2-3	Primary Guidance Path - Simplified Block Diagram	2.2-8A
2.2-4	Abort Guidance Path - Simplified Block Diagram	2.2-8B
2.2-5	LM Vehicle and GN&CS Axes	2.2-10
2.2-6	Alignment Optical Telescope Axes	2.2-11
2.2-7	Primary Guidance and Navigation Section - Block Diagram	2.2-13
2.2-8	Inertial Subsection - Functional Diagram	2.2-15
2.2-9	Inertial Subsection - Functional Loops	2.2-17
2.2-10	Inertial Subsection - Modes of Operation	2.2-21
2.2-11	Alignment Optical Telescope - Reticle Pattern	2.2-25
2.2-12	Display and Keyboard - Pictorial	2.2-26
2.2-13	Primary Guidance and Navigation Section - Functional Flow Diagram	2.2-33
2.2-14	Abort Guidance Section - Block Diagram	2.2-55
2.2-14A	Abort Sensor Assembly - Simplified Block Diagram	2.2-56
2.2-14B	Abort Electronics Assembly - Simplified Block Diagram	2.2-57
2.2-15	Data Entry and Display Assembly - Pictorial	2.2-62
2.2-16	Euler Angles	2.2-80B
2.2-17	Canted Thrust Selection Logic	2.2-80
2.2-17A	Abort Guidance Section - Functional Flow Diagram	2.2-80
2.2-18	Rendezvous Radar - Block Diagram	2.2-83
2.2-19	Rendezvous Radar - Antenna Assembly	2.2-85
2.2-20	Rendezvous Radar - Functional Flow Diagram	2.2-87
2.2-21	Rendezvous Radar - Antenna Orientation No. I	2.2-91
2.2-22	Rendezvous Radar - Antenna Orientation No. II	2.2-92
2.2-23	Transponder Electronic Assembly - Block Diagram	2.2-93
2.2-24	Transponder - Functional Flow Diagram	2.2-95
2.2-25	Deleted	2.2-96
2.2-26	Landing Radar - Block Diagram	2.2-97
2.2-27	Landing Radar - Functional Flow Diagram	2.2-101
2.2-28	Landing Radar Antenna Beam Configuration	2.2-104
2.2-29	Control Electronics Section - Block Diagram	2.2-105
2.2-30	Attitude Controller Assembly Manipulations	2.2-108
2.2-31	TTCA Manipulations	2.2-109
2.2-32	Attitude and Translation Control Assembly - Simplified Block Diagram	2.2-110
2.2-33	Attitude and Translation Control Schematic	2.2-115

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

LIST OF ILLUSTRATIONS (cont)

Figure No.		Page
2.2-34	Descent Engine Control Schematic	2.2-117
2.2-35	Ascent Engine Control Schematic	2.2-125
2.2-36	GN&CS Power Distribution	2.2-127
2.2-37	Navigation Base	2.2-130
2.2-38	IMU Gimbal Assembly	2.2-131
2.2-39	IMU Temperature Control	2.2-133
2.2-40	IMU - Forced Convection Cooling	2.2-134
2.2-41	IMU Failure Detection Logic	2.2-135
2.2-42	CDU Digital-to-Analog and Analog-to-Digital Conversion - Functional Diagram	2.2-136
2.2-43	CDU Failure Detection Logic	2.2-138
2.2-44	Alignment Optical Telescope	2.2-141
2.2-45	Alignment Optical Telescope - Optical Schematic	2.2-142
2.2-46	LM Guidance Computer - Functional Flow Diagram	2.2-145
2.2-47	Display and Keyboard - Block Diagram	2.2-158
2.2-48	Abort Sensor Assembly - Detailed Block Diagram	2.2-160
2.2-49	Abort Guidance Section - Gyro Assembly	2.2-161
2.2-49A	Abort Electronics Assembly - Detailed Block Diagram	2.2-162C
2.2-49B	Data Entry Display Assembly - Detailed Block Diagram	2.2-162J
2.2-50	Rendezvous Radar Digital Data Transfer - Functional Diagram	2.2-168
2.2-51	Transponder Antenna Assembly	2.2-169
2.2-52	Landing Radar Antenna Assembly	2.2-172
2.2-53	Deleted	2.2-174
2.2-54	Landing Radar Digital Data Transfer - Functional Diagram	2.2-175
2.2-55	Attitude Controller Assembly	2.2-176
2.2-56	Attitude Controller Assembly - Functional Diagram	2.2-177
2.2-57	Thrust/Translation Controller Assembly	2.2-178
2.2-58	Thrust/Translation Controller - Functional Diagram	2.2-179
2.2-58A	Attitude and Translation Control Assembly - Detailed Block Diagram	2.2-180A
2.2-59	Attitude and Translation Control Assembly - Jet Select Logic	2.2-181
2.2-60	Rate Gyro Assembly - Schematic and Vector Diagrams	2.2-182
2.2-61	Descent Engine Control Assembly - Block Diagram	2.2-183
2.2-62	Descent Engine Control Assembly - Throttle Control Diagram	2.2-184
2.2-63	Descent Engine Control Assembly - Trim Control Diagram	2.2-185
2.2-64	Descent Engine Control Assembly - Engine Control Diagram	2.2-186
2.2-65	Throttle Control vs TTCA Displacement	2.2-187
2.2-66	Gimbal Drive Actuator - Cutaway	2.2-188
2.2-67	Control Electronics Section - 800-cps Synchronization Loop	2.2-189
2.2-68	Gimbal Angle Sequencing Transformation Assembly - Schematic Diagram	2.2-190A
2.2-69	Orbital Rate Display - Earth and Lunar - Schematic Diagram	2.2-190C
2.3-1	Descent Propulsion Section - Interface Diagram	2.3-3
2.3-2	Descent Propulsion Section - Component Location	2.3-5
2.3-3	Descent Propulsion Section - Simplified Functional Flow Diagram	2.3-7
2.3-4	Propellant Quantity Gaging System - Simplified Functional Block Diagram	2.3-11
2.3-5	Descent Engine Assembly - Flow Diagram	2.3-14
2.3-6	Descent Engine and Head End Assembly	2.3-15
2.3-7	Descent Propulsion Section - Explosive Valves, Simplified Functional Diagram	2.3-17
2.3-8	Helium Isolation Solenoid Valve, Latched-Open Position	2.3-18
2.3-9	Descent Propulsion Section - Helium Pressure Regulators	2.3-19
2.3-10	Descent Propulsion Section - Relief Valve Assembly	2.3-20
2.3-11	Throttle Valve Actuator - Block Diagram	2.3-22
2.3-12	Flow Control Valve	2.3-23
2.3-13	Variable-Area Injector - Quarter Section	2.3-25

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

LIST OF ILLUSTRATIONS (cont)

Figure No.		Page
2.3-14	Descent Engine Combustion Chamber and Nozzle Extension	2.3-26
2.3-15	Ascent Propulsion Section - Interface Diagram	2.3-35
2.3-16	Ascent Propulsion Section - Component Location	2.3-38
2.3-17	Ascent Propulsion Section - Simplified Functional Flow Diagram	2.3-39
2.3-18	Ascent Engine Assembly - Flow Diagram	2.3-42
2.3-19	Ascent Propulsion Section - Explosive Valves, Simplified Functional Diagram	2.3-43
2.3-20	Ascent Propulsion Section - Helium Pressure Regulator Assembly	2.3-44
2.3-21	Ascent Propulsion Section - Relief Valve Assembly	2.3-45
2.3-22	Ascent Engine Assembly	2.3-47
2.4-1	Reaction Control Subsystem - Component Location	2.4-1
2.4-2	Reaction Control Subsystem - Interface Block Diagram	2.4-3
2.4-3	Reaction Control Subsystem - Simplified Functional Flow Diagram	2.4-7
2.4-4	RCS System A - Ascent Feed Interconnect Valve Arrangement	2.4-9
2.4-5	Propellant Quantity Measuring Device	2.4-10
2.4-6	Thrust Chamber Assembly Cluster	2.4-11
2.4-7	Thrust Chamber Assembly - Operation and Failure Detection System	2.4-12
2.4-8	TCA Firing - Thrust Versus Time	2.4-13
2.4-9	Thrust Chamber Assembly	2.4-14
2.4-10	Reaction Control Subsystem - Thruster-On Failure Correction	2.4-15
2.4-11	Helium Pressure Regulator Assembly	2.4-17
2.4-12	Relief Valve Assembly	2.4-18
2.4-13	Propellant Tank Assembly	2.4-18
2.4-14	Reaction Control Subsystem - Telemetry Points	2.4-31
2.5-1	Electrical Power Subsystem - Signal Interface Block Diagram	2.5-2
2.5-2	Electrical Power Subsystem - Power Interface Diagram (2 Sheets)	2.5-5
2.5-3	Electrical Power Subsystem - Functional Flow Diagram	2.5-9
2.5-4	Descent Power Distribution - Functional Block Diagram	2.5-11
2.5-5	Ascent Power Distribution - Functional Block Diagram	2.5-13
2.5-6	Battery Status-Monitoring Circuits - Simplified Schematic Diagram	2.5-14
2.5-7	Battery Voltage-and Current-Monitoring-Circuits - Simplified Schematic Diagram	2.5-16
2.5-8	Primary Bus Voltage-Monitoring Circuits - Simplified Schematic Diagram	2.5-18
2.5-9	Malfunctioning-Battery Isolation Circuitry - Simplified Schematic Diagram	2.5-19
2.5-10	Relay Junction Box - Functional Block Diagram	2.5-21
2.5-11	CSM/LM Primary Power Transfer Logic - Simplified Schematic Diagram	2.5-22
2.5-12	Deadface Relay and Staging Logic - Simplified Schematic Diagram	2.5-24
2.5-13	Descent Stage Electrical Control Assembly - Simplified Schematic Diagram	2.5-26
2.5-14	Ascent Stage Electrical Control Assembly - Simplified Schematic Diagram	2.5-28
2.5-15	Inverter No. 1 - Block Diagram	2.5-29
2.5-16	Sensor Power Fuse Assemblies - Simplified Power Distribution Diagram	2.5-30
2.6-1	Environmental Control Subsystem - Component Location (3 Sheets)	2.6-1
2.6-2	Environmental Control Subsystem - Aft Bulkhead Controls	2.6-4
2.6-3	Environmental Control Subsystem - Power Distribution Diagram	2.6-5
2.6-4	Environmental Control Subsystem - Functional Flow Diagram	2.6-6
2.6-5	Environmental Control Subsystem - Electrical Schematic Diagram	2.6-11
2.6-6	Atmosphere Revitalization Section - Simplified Functional Flow Diagram	2.6-13
2.6-7	Oxygen Supply and Cabin Pressure Control Section - Simplified Functional Diagram	2.6-15

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

LIST OF ILLUSTRATIONS (cont)

Figure No.		Page
2.6-8	Water Management Section - Simplified Functional Flow Diagram	2.6-18
2.6-9	Heat Transport Section - Simplified Functional Flow Diagram	2.6-21
2.6-10	Suit Gas Diverter Valve - Functional Schematic	2.6-23
2.6-11	Water Separator - Functional Schematic	2.6-25
2.6-12	Suit Isolation Valve - Functional Schematic	2.6-26
2.6-13	Carbon Dioxide Partial Pressure Sensor - Functional Schematic	2.6-27
2.6-14	High-Pressure Oxygen Control Assembly - Functional Schematic	2.6-28
2.6-15	High-Pressure Oxygen Regulator - Functional Schematic	2.6-29
2.6-16	Bypass Oxygen Relief Valve - Functional Schematic	2.6-29
2.6-17	Overboard Relief Valve - Functional Schematic	2.6-30
2.6-18	Burst Diaphragm Assembly - Functional Schematic	2.6-31
2.6-19	Oxygen Demand Regulator - Functional Schematic	2.6-32
2.6-20	Cabin Repressurization and Emergency Oxygen Valve - Functional Schematic	2.6-33
2.6-21	Cabin Relief and Dump Valve - Functional Schematic	2.6-34
2.6-22	Water Tank Selector Valve - Functional Schematic	2.6-36
2.6-23	Water Pressure Regulator - Functional Schematic	2.6-37
2.6-24	Cabin Temperature Control Valve - Functional Schematic	2.6-38
2.7-1	Communications Subsystem, Interface Diagram	2.7-2
2.7-2	Communications Subsystem - Simplified Block Diagram	2.7-3
2.7-3	Communications Subsystem, Power Distribution Diagram	2.7-5
2.7-4	In-Flight Communications	2.7-6
2.7-5	S-Band Transmitter Receiver - Block Diagram	2.7-16
2.7-6	S-Band Transmitter - Block Diagram	2.7-17
2.7-7	S-Band Receiver - Block Diagram	2.7-19
2.7-8	S-Band Power Amplifier - Block Diagram	2.7-20
2.7-9	VHF Transmitter-Receiver Assembly - Block Diagram	2.7-21
2.7-10	VHF Transmitter - Block Diagram	2.7-22
2.7-11	VHF Receiver - Block Diagram	2.7-24
2.7-12	Signal-Processing Assembly - Block Diagram	2.7-25
2.7-13	Premodulation Processor - Block Diagram	2.7-27
2.7-14	Audio Center No. 1 - Block Diagram	2.7-31
2.7-15	Audio Control Circuits - Simplified Block Diagram	2.7-33
2.7-16	Digital Uplink Assembly - Block Diagram	2.7-35
2.7-17	Communication Subsystem - Antenna Location	2.7-36
2.7-18	S-Band Steerable Antenna - Block Diagram	2.7-38
2.7-19	S-Band Steerable Antenna Gimbaling	2.7-39
2.7-20	S-Band Steerable Antenna Coverage Restrictions	2.7-43
2.8-1	Explosive Devices Subsystem - Overall Block Diagram	2.8-1
2.8-2	Explosive Devices - Component Location	2.8-2
2.8-3	Explosive Devices Subsystem - Overall Functional Diagram	2.8-4
2.8-4	Stage Sequence (Relays K2 through K6) Monitoring - Simplified Schematic Diagram	2.8-5
2.8-5	Bus - Arming Control - Simplified Schematic Diagram	2.8-6
2.8-6	Landing Gear Deployment - Simplified Schematic Diagram	2.8-7
2.8-7	Landing Gear Switches - Simplified Schematic Diagram	2.8-8
2.8-8	Descent Propellant Tank Prepressurization - Simplified Schematic Diagram	2.8-9
2.8-9	Stage Sequence - Simplified Schematic Diagram	2.8-11
2.8-10	Ascent Propellant Tank Pressurization - Simplified Schematic Diagram . . .	2.8-13
2.8-11	RCS Propellant Tank Pressurization	2.8-14
2.8-12	Explosive Devices	2.8-16
2.8-13	Electrical Circuit Interrupter - Functional Block Diagram	2.8-18
2.9-1	Instrumentation Power Distribution - Overall Functional Block Diagram	2.9-3

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

LIST OF ILLUSTRATIONS (cont)

Figure No.		Page
2.9-2	Instrumentation Subsystem - Overall Functional Block Diagram	2.9-5
2.9-3	Data Format No. 1 (Normal Mode)	2.9-7
2.9-4	Data Format No. 2 (Reduced Mode)	2.9-7
2.9-5	Typical Analog Multiplexer Card - Simplified Logic Diagram	2.9-34
2.9-6	High-Level Analog Multiplexer Card - Simplified Logic Diagram	2.9-35
2.9-7	High-Speed Gates - Logic Diagram	2.9-36
2.9-8	Analog-to-Digital Converter (Coder) - Simplified Logic Diagram	2.9-37
2.9-9	Programmer - Block Diagram	2.9-39
2.9-10	Digital Multiplexer - Logic Diagram	2.9-41
2.9-11	Output Register	2.9-43
2.9-12	Timing Electronics - Logic Diagram	2.9-45
2.9-13	CWEA - Functional Block Diagram	2.9-47
2.9-14	MASTER ALARM Pushbutton/Lights and C/W PWR Caution Light - Simplified Schematic Diagram	2.9-49
2.9-15	CWEA Failure Detection Circuits - Logic Diagram (3 Sheets)	2.9-57
2.9-16	DSEA - Functional Block Diagram	2.9-76
2.10-1	Lighting Control Equipment	2.10-3
2.10-2	Lighting Circuits	2.10-5
2.10-3	Exterior Lighting	2.10-8
2.10-4	Floodlighting and Utility Lights - Simplified Diagram	2.10-9
2.10-5	COAS Lighting	2.10-10
2.10-6	Lamp and Tone Test Circuits	2.10-11
2.11-1	Extravehicular Mobility Unit	2.11-2
2.11-2	Pressure Garment Assembly	2.11-3
2.11-3	Integral Thermal Micrometeroroid Garment, PLSS, OPS, and PLSS Control Box	2.11-4
2.11-4	Extravehicular Visor Assembly	2.11-7
2.11-5	PLSS Controls	2.11-8
2.11-6	Portable Life Support Subsystem - Simplified Functional Schematic	2.11-11
2.11-7	EMU Electrical Connectors	2.11-14
2.11-8	Waste Management System	2.11-18
2.11-9	Restraint Equipment	2.11-20
2.11-10	Crewman Optical Alignment Sight	2.11-21
2.11-11	COAS Functional Schematic and Docking Target	2.11-22
2.11-12	Tool B	2.11-23
2.11-13	Stowage Locations, and Crew Personal Equipment	2.11-25
2.11-14	Stowage Locations, and Crew Personal Equipment	2.11-26
3-1	Main Display Console	3-155
3-2	Environmental Control Subsystem Aft Bulkhead Controls	3-157

LIST OF TABLES

Table No.		Page
2.2-1	DSKY Verb List	2.2-35
2.2-2	DSKY Noun List	2.2-39
2.2-3	Checklist Reference Codes Used With Verb 50 Noun 25	2.2-46
2.2-4	DSKY Option Codes (FL VO4 NO6)	2.2-46
2.2-5	DSKY Alarm Codes	2.2-47
2.2-6	Planet and Star Codes	2.2-48
2.2-7	LGC Programs	2.2-50
2.2-8	LGC Routines	2.2-52
2.2-9	Abort Electronics Assembly - Input Signal Characteristics	2.2-60
2.2-10	Abort Electronic Assembly - Output Signal Characteristics	2.2-61
2.2-11	Selector Logic List	2.2-64
2.2-12	Data Entry and Display Assembly - Input List	2.2-66
2.2-13	Data Entry and Display Assembly - Output List	2.2-69
2.2-14	Data Entry and Display Assembly - Accessible Constants	2.2-71
2.2-14A	PGNS Downlink Data Update Sequence	2.2-76
2.2-14B	Abort Guidance Section - Calibration Compensations	2.2-80C
2.2-14C	Abort Guidance Section - Guidance Routines	2.2-80E
2.2-14D	Time (Minutes) Quantities Available in CDH Mode	2.2-80H
2.2-14E	Times (Minutes) of Interest for TPI Search Routine	2.2-80H
2.2-14F	Times (Minutes) of Interest for TPI Executive Routine	2.2-80I
2.2-14G	Abort Guidance Section - Translational Thrust	2.2-80
2.2-14H	Abort Guidance Section - Thrust Directions	2.2-80
2.2-15	Control Electronics Section - Summary of Modes of Attitude Control	2.2-119
2.2-16	Signal Conditioner Assembly - Signal-Conditioning Modules	2.2-139
2.2-17	LGC Input-Output Channel Assignments	2.2-151
2.2-18	Channel 7 Radar Fixed Extension Bits	2.2-155
2.2-19	Abort Electronics Assembly - Telemetry Word List	2.2-162G
2.2-20	Deleted .	
2.2-21	Frequency Synthesizer - Output Frequencies	2.2-165
2.2-22	Guidance, Navigation, and Control Subsystem - Failure Modes	2.2-191
2.2-23	Primary Guidance and Navigation Section - Performance and Design Data . .	2.2-194
2.2-24	Abort Guidance Section - Performance and Design Data	2.2-197
2.2-25	Rendezvous Radar and Transponder - Performance and Design Data	2.2-197
2.2-26	Landing Radar - Performance and Design Data	2.2-199
2.2-27	Control Electronics Section - Performance and Design Data	2.2-201
2.2-28	Orbital Rate Display - Earth and Lunar - Performance and Design Data . . .	2.2-202
2.2-29	Guidance, Navigation, and Control Subsystem - Telemetry Measurements . .	2.2-206
2.3-1	Descent Propulsion Section - Failure Modes	2.3-26
2.3-2	Descent Propulsion Section - Performance and Design Data	2.3-28
2.3-3	Descent Propulsion Section - Telemetry Measurements	2.3-31
2.3-4	Ascent Propulsion Section - Failure Modes	2.3-48
2.3-5	Ascent Propulsion Section - Performance and Design Data	2.3-49
2.3-6	Ascent Propulsion Section - Telemetry Measurements	2.3-51
2.4-1	Reaction Control Subsystem - Failure Modes	2.4-19
2.4-2	Reaction Control Subsystem - Performance and Design Data	2.4-22
2.4-3	Reaction Control Subsystem - Telemetry Measurements	2.4-25
2.5-1	Circuit Breaker Characteristics	2.5-30
2.5-2	Sensor Power Fuses - Characteristics	2.5-31
2.5-3	Electrical Power Subsystem - Failure Modes	2.5-34

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK

LIST OF TABLES (cont)

Table No.		Page
2.5-4	Electrical Power Subsystem - Performance and Design Data	2.5-35
2.5-5	Electrical Power Subsystem - Telemetry Measurements	2.5-37
2.6-1	Environmental Control Subsystem - Failure Modes	2.6-39
2.6-2	Environmental Control Subsystem - Performance and Design Data	2.6-42
2.6-3	Environmental Control Subsystem - Telemetry Measurements	2.6-49
2.7-1	Communications Links	2.7-4
2.7-2	Communications Subsystem - General Switch and Circuit Breaker Configuration	2.7-8
2.7-3	Communications Subsystem - Equipment Groups	2.7-14
2.7-4	S-Band Communications Capabilities	2.7-15
2.7-5	Communications Subsystem - Failure Modes	2.7-40
2.7-6	Communications Subsystem - Performance and Design Data	2.7-42
2.7-7	Communications Subsystem - Telemetry Measurements	2.7-44
2.8-1	Explosive Devices Subsystem - Failure Modes	2.8-19
2.8-2	Explosive Devices Subsystem - Performance and Design Data	2.8-21
2.8-3	Explosive Devices Subsystem - Telemetry Measurements	2.8-22
2.9-1	ERA-1 Conditioned and Preconditioned Interfacing Signals	2.9-9
2.9-2	ERA-2 Conditioned and Preconditioned Interfacing Signals	2.9-19
2.9-3	Caution and Warning Electronics Assembly - Malfunction Data Summary . . .	2.9-63
2.9-4	Instrumentation Subsystem - Failure Modes	2.9-77
2.9-5	Instrumentation Subsystem - Performance and Design Data	2.9-78
2.10-1	Exterior Lights - Color, Brightness, and Lighting Method	2.10-8
2.10-2	Interior Lights - Color, Brightness, and Lighting Method	2.10-9
2.10-3	Heating Section - Failure Modes	2.10-15
2.10-4	Lighting Section - Failure Modes	2.10-17
2.10-5	Heating Section - Performance and Design Data	2.10-17
2.10-6	Exterior Lights - Performance and Design Data	2.10-18
2.10-7	Interior Lights - Performance and Design Data	2.10-18
2.10-8	Heating and Lighting Sections - Telemetry Measurements	2.10-20
2.11-1	Extravehicular Mobility Unit - Telemetry Measurements	2.11-15

LMA790-3-LM 4
 APOLLO OPERATIONS HANDBOOK
 SPACECRAFT

Note:
 X-axis design reference
 points start at + 200.000.

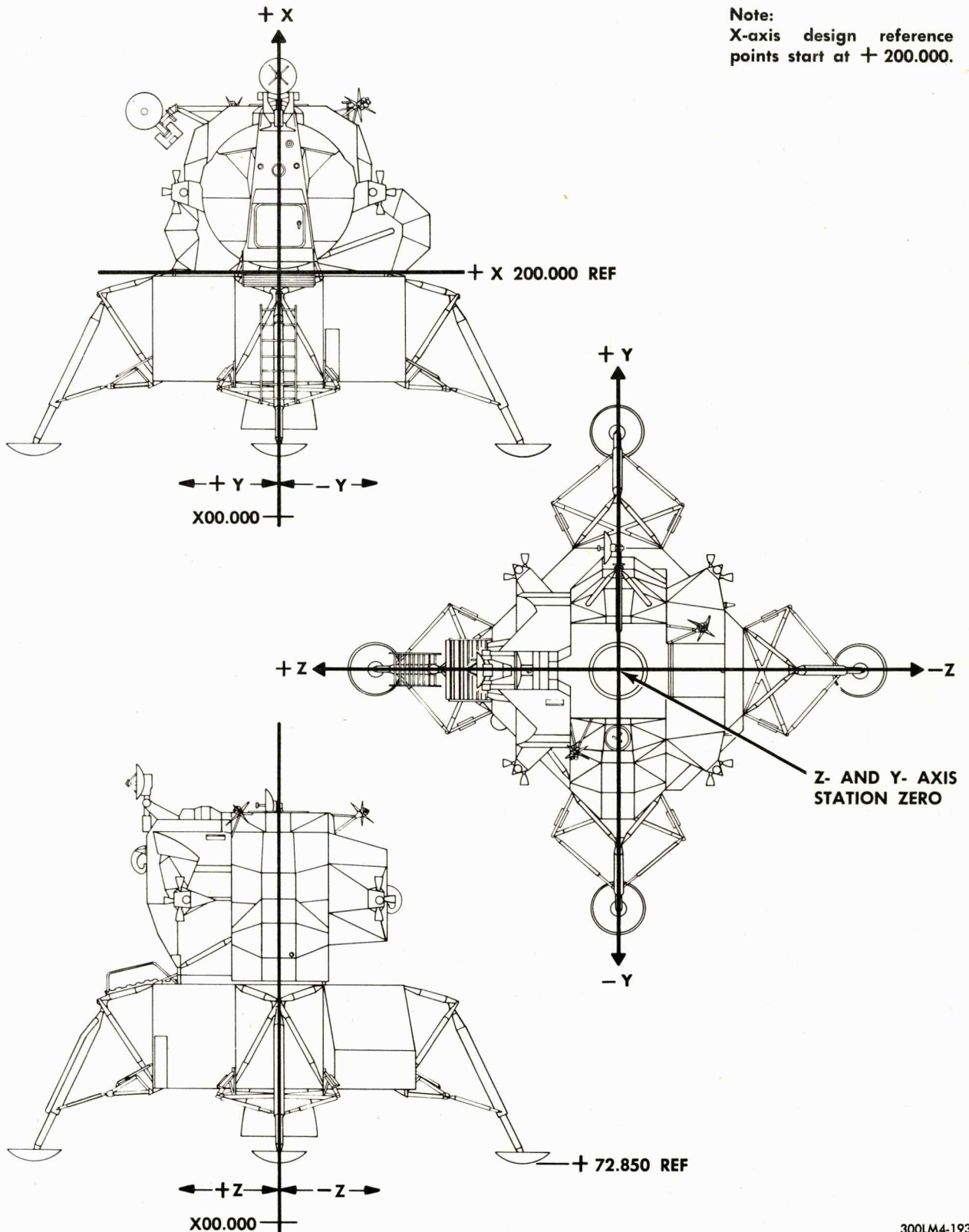


Figure 1-4. Station Reference Measurements

300LM4-193

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SPACECRAFT

1. 2. 2. 1 Crew Compartment. (See figures 1-5 and 1-6.)

The crew compartment is the frontal area of the ascent stage; it is 92 inches in diameter and 42 inches deep. This is the flight station area; it has control and display panels, armrests, body restraints, landing aids, two front windows, a docking window, and an alignment optical telescope. Flight station centerlines are 44 inches apart; each astronaut flight station has a set of controllers and armrests. A design reference point (design eye) orients flight station geometry for the required balance between external visibility and visual and physical access to controls and displays. The main control and display panels are canted and centered between the astronauts to permit sharing and easy scanning.

An optical alignment station, between the flight stations, is used in conjunction with the alignment optical telescope (AOT). A portable life support system (PLSS) donning station is also in the center aisle, slightly aft of the optical alignment station. A removable, height-adjusting PLSS supporting harness permits donning and doffing the PLSS.

The crew compartment shell is cylindrical and of semimonocoque construction. It is a fusion-welded and mechanically fastened assembly of aluminum-alloy sheet and machined longerons. The shell has an opening for the docking window, above the Commander's station.

The front face assembly of the crew compartment has two triangular windows and the forward hatch. Two large structural beams extend up the forward side of the front face assembly; they support the structural loads applied to the cabin structure. The lower ends of the beams support the two forward interstage mounts; the upper ends are secured to additional beam structure that extends across the top of the crew compartment shell and aft to the midsection structure.

The crew compartment deck measures approximately 36 by 55 inches. It is constructed of aluminum honeycomb bonded to two sheets of aluminum alloy. Handgrips, recessed in the deck, aid the astronauts during egress and ingress through the forward hatch.

1. 2. 2. 1. 1 Control and Display Panels.

The crew compartment has 12 control and display panels: two main display panels (1 and 2) that are canted forward 10°, two center panels (3 and 4) that slope down and aft 45° towards the horizontal, two bottom panels (5 and 6), two lower side panels (8 and 12), one center side panel (14), two upper side panels (11 and 16), and the orbital rate display - earth and lunar (ORDEAL) panel aft of panel 8.

Panels 1 and 2 are located on each side of the flight station centerline, approximately at eye level. The panels are constructed of two 0.015 aluminum-alloy face sheets, spaced 2 inches apart; they are reinforced and spaced by formed 0.015 channels. This forms a rigid box-like construction with a favorable strength-to-weight ratio and a relatively high natural frequency. Four shock mounts support each panel on the structure. Panel instruments are bezel mounted to the back surface of the bottom sheet of the panel. The instruments are cushioned where they protrude through the top sheet of the panel, to damp vibrations. All dial faces are flush with the forward face of the panel. Panel 1 contains warning lights, flight indicators and controls, and propellant quantity indicators. Panel 2 contains caution lights, flight indicators and controls, and Reaction Control System (RCS) and Environmental Control System (ECS) indicators and controls.

Panel 3, immediately below panels 1 and 2, spans the width of these two panels. Panel 3 contains the radar antenna temperature indicators and engine, radar, spacecraft stability, event timer, RCS and lighting controls.

Panel 4 is centered between the flight stations below panel 3. Panel 4 contains Attitude Controller Assembly (ACA) and Thrust/Translation Controller Assembly (TTCA) enabling switches, navigation system indicators, and LM Guidance Computer (LGC) indicators and controls.

Panels 5 and 6 are in front of the flight stations, at astronaut waist height. Panel 5 contains lighting and mission timer controls, engine start and stop pushbuttons, and the X-translation pushbutton. Panel 6 contains abort guidance controls.

ASCENT STAGE

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SPACECRAFT

Panel 8 is at the left of the commander's station. The panel is canted up 15° from the horizontal; it contains controls and displays for explosive devices, audio controls, and the TV camera connection.

Panel 11, directly above panel 8, has five angled surfaces that contain circuit breakers. Each row of circuit breakers is canted 15° to the line of sight so that the white band on the circuit breakers can be seen when they open.

Panel 12 is at the right of the LM Pilot's station. The panel is canted up 15° from the horizontal; it contains audio, communications, and communications antennas controls and displays.

Panel 14, directly above panel 12, is canted up 36.5° from the horizontal. It contains controls and displays for electrical power distribution and monitoring.

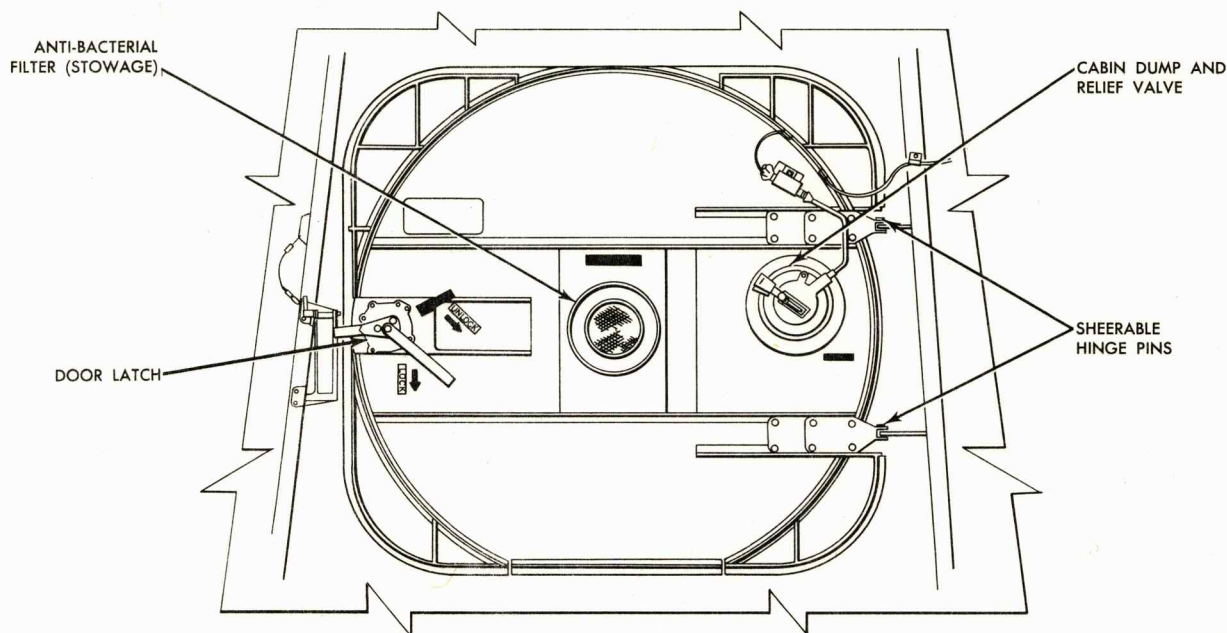
Panel 16, directly above panel 14, has four angled surfaces that contain circuit breakers. Each row of circuit breakers is canted 15° to the line of sight so that the white band on the circuit breakers can be seen when they open.

The ORDEAL panel is immediately aft of panel 8. It contains the controls for obtaining LM attitude, with respect to a local horizontal, from the LGC.

1.2.2.1.2 Forward Hatch. (See figure 1-7.)

The forward hatch is in the front face assembly, just below the lower display panels. The hatch is approximately 32 inches square; it is hinged to swing inboard when opened. A cam latch assembly holds the hatch in the closed position; the assembly forces a lip, around the outer circumference of the hatch, into a preloaded elastomeric silicone compound seal that is secured to the LM structure. Cabin pressurization forces the hatch lip further into the seal, ensuring a pressure-tight contact. A handle is provided on both sides of the hatch, for latch operation. To open the hatch, the cabin must be completely depressurized by opening the cabin relief and dump valve on the hatch. When the cabin is completely depressurized, the hatch can be opened by rotating the latch handle. The cabin relief and dump valve can also be operated from outside the LM.

ASCENT STAGE



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Figure 1-7. Forward Hatch

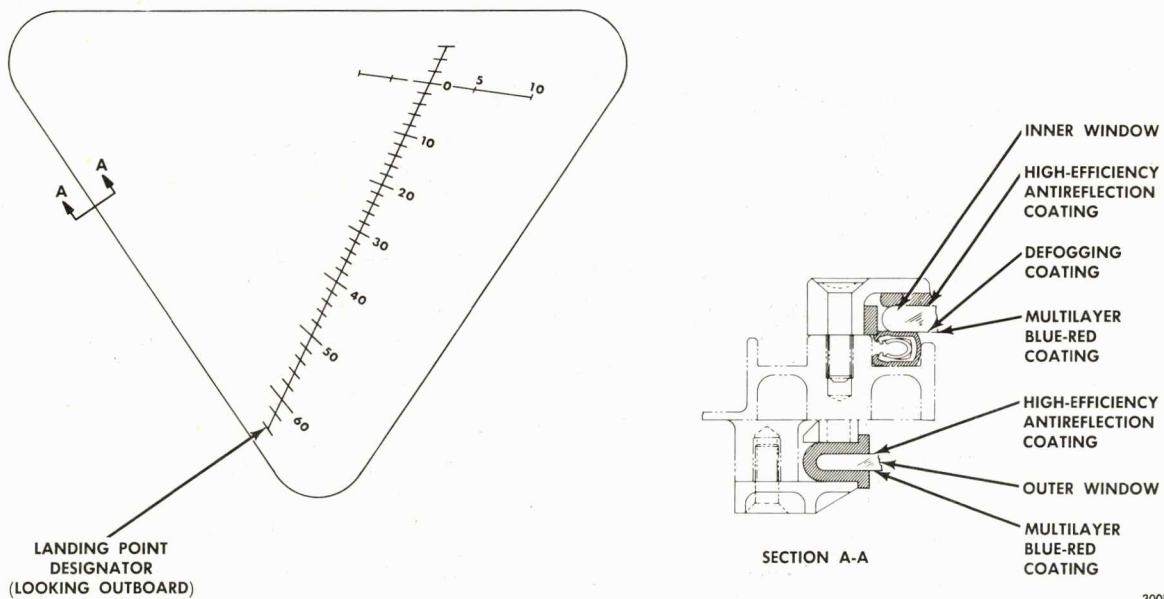
1.2.2.1.3 Windows. (See figures 1-8 and 1-9.)

Two triangular windows in the front face assembly provide visibility during descent, ascent, and rendezvous and docking phases of the mission. Both windows have approximately 2 square feet of viewing area; they are canted down to the side to permit adequate peripheral and downward visibility. A third (docking) window (figure 1-9) is in the curved overhead portion of the crew compartment shell, directly above the Commander's flight station. This window provides visibility for docking maneuvers. All three windows consist of two separated panes, vented to space environment. The outer pane is made of Vycor glass, with antireflective and thermal (multilayer blue-red) coatings. The inner pane is made of structural glass. It is sealed with a dual Racor seal and bolted to the window frame through an edge member bonded to the pane. The inner pane has antireflective, thermal (multilayer blue-red), and defogging coatings.

All three windows are electrically heated to prevent fogging. The heaters for the commander's front window and the docking window receive their power from 115-volt a-c bus A and the commander's 28-volt d-c bus, respectively. The heater for the LM Pilot's front window receives power from 115-volt a-c bus B. The heater power for the commander's front window and the docking window is routed through the AC BUS A: CDR WIND HTR. and HEATERS: DOCK WINDOW circuit breakers, respectively; for the LM Pilot's front window, through the AC BUS B: SE WIND HTR circuit breaker. These are 2-ampere circuit breakers, on panel 11. The temperature of the windows is not monitored with an indicator; proper heater operation directly affects crew visibility and is therefore visually determined by the astronauts. The window heaters are normally turned on when the astronauts enter the LM; they are left on until the astronauts leave the vehicle.

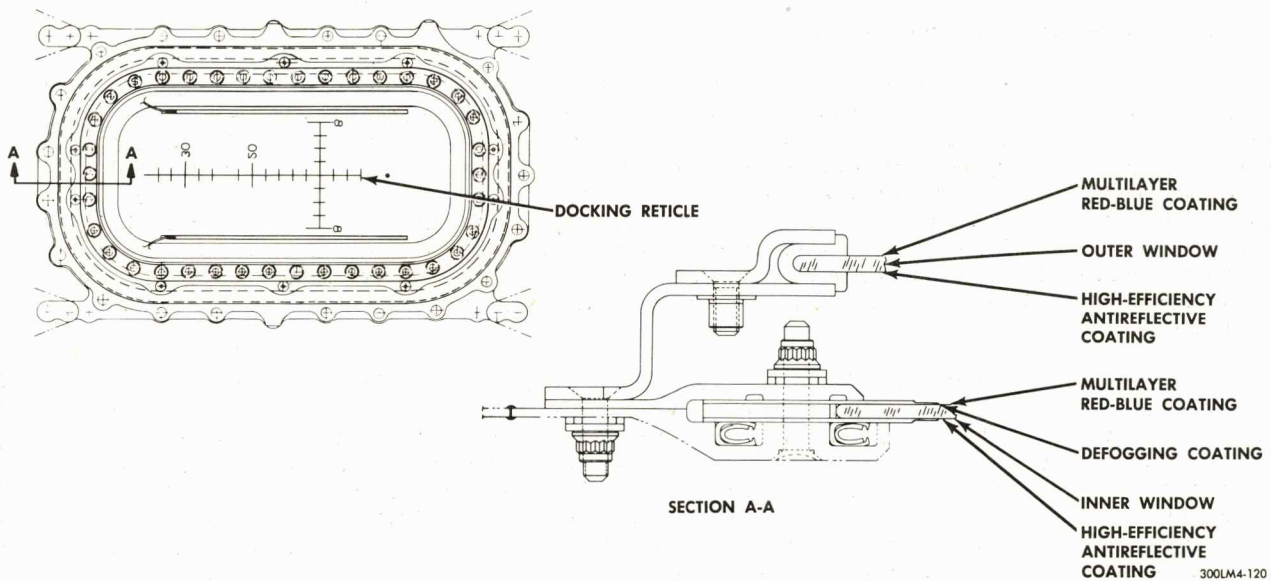
ASCENT STAGE

LMA790-3-LM 4
 APOLLO OPERATIONS HANDBOOK
 SPACECRAFT



300LM4-119

Figure 1-8. Front Window



300LM4-120

Figure 1-9. Docking Window

GUIDANCE, NAVIGATION AND CONTROL SUBSYSTEM

MAIN PROPULSION SUBSYSTEM

REACTION CONTROL SUBSYSTEM

ELECTRICAL POWER SUBSYSTEM

ENVIRONMENTAL CONTROL SUBSYSTEM

COMMUNICATIONS SUBSYSTEM

EXPLOSIVE DEVICES SUBSYSTEM

INSTRUMENTATION SUBSYSTEM

LIGHTING

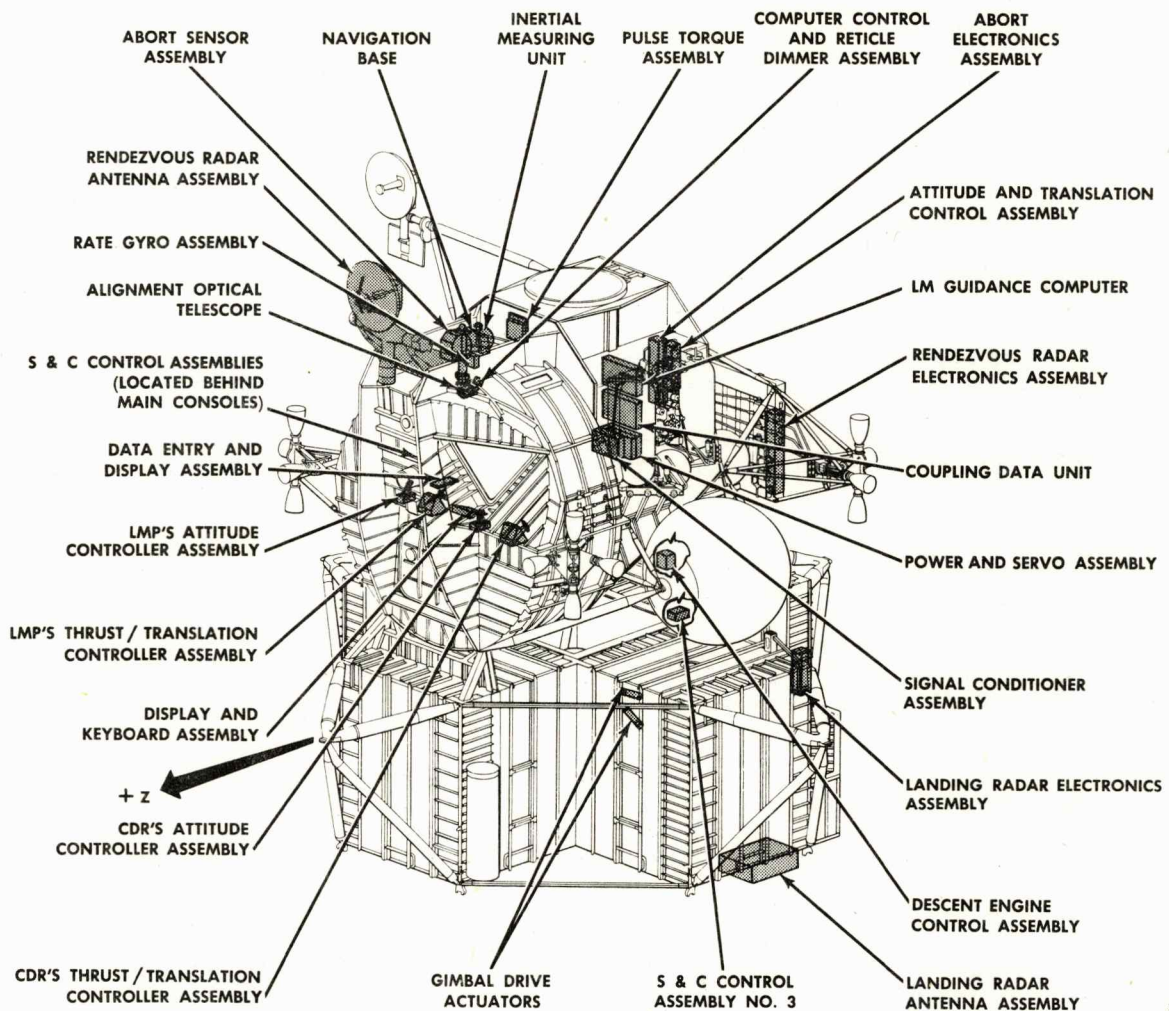
CREW PERSONAL EQUIPMENT

SUBSYSTEM INTERFACE

2.2 GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM.

2.2.1 INTRODUCTION.

The Guidance, Navigation, and Control Subsystem (GN&CS) provides vehicle guidance, navigation, and control required for successful completion of a manned LM mission. The subsystem is an aided inertial system whose principal sensors are gyroscopes, accelerometers, an optical device and radar. The GN&CS consists of the following sections: primary guidance and navigation section (PGNS), abort guidance section (AGS), and the control electronics section (CES). The GN&CS also contains a rendezvous radar (RR), a landing radar (LR), and the orbital rate display - earth and lunar (ORDEAL). Figure 2.2-1 shows the location of the LM-4 GN&CS equipment. (Refer to paragraph 2.2.4 for a detailed description of the equipment.)



300LM4-28

Figure 2.2-1. Guidance, Navigation, and Control Subsystem - Equipment Location

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

■ 2.2.1.1 Primary Guidance and Navigation Section. (See figure 2.2-1A.)

The PGNS provides the measuring and data-processing capabilities, and control functions, for the LM-4 mission. The PGNS uses inertial components for guidance, an optical device and radar for navigation, and a digital computer for data processing and generation of flight control signals.

The inertial guidance portion of the PGNS establishes an inertial reference that is used as the central coordinate system from which all measurements and computations are made. The inertial guidance portion senses instantaneous velocity and attitude provides incremental velocity and attitude data to the digital computer for the generation of attitude and translation changes. For navigation, the PGNS uses a telescope to take star sightings and radar for position and velocity calculations. These star sightings are used by the digital computer to establish an inertial reference. The position and velocity calculations enable a LM&CSM rendezvous and, during lunar missions, enable a safe landing on the lunar surface.

With the aid of the inertial guidance and navigation portions of the PGNS and the software capability of the computer, the PGNS controls the flight of the LM. The PGNS generates attitude commands to correct or control vehicle motion. During ascent engine or descent engine burns, the PGNS controls LM attitude, engine firing, descent engine thrust and direction of the descent engine thrust vector.

■ 2.2.1.2 Abort Guidance Section. (See figure 2.2-1A.)

The AGS is a strapped-down inertial system in which the inertial sensors are rigidly mounted with respect to the vehicle, rather than mounted on a stabilized platform. The AGS provides independent backup for the PGNS. It is used only if the PGNS malfunctions. If the PGNS is functioning properly when a mission must be aborted, the PGNS is used to control the LM. During the LM-4 mission, the AGS will be used even though the PGNS is functioning properly, to evaluate AGS performance.

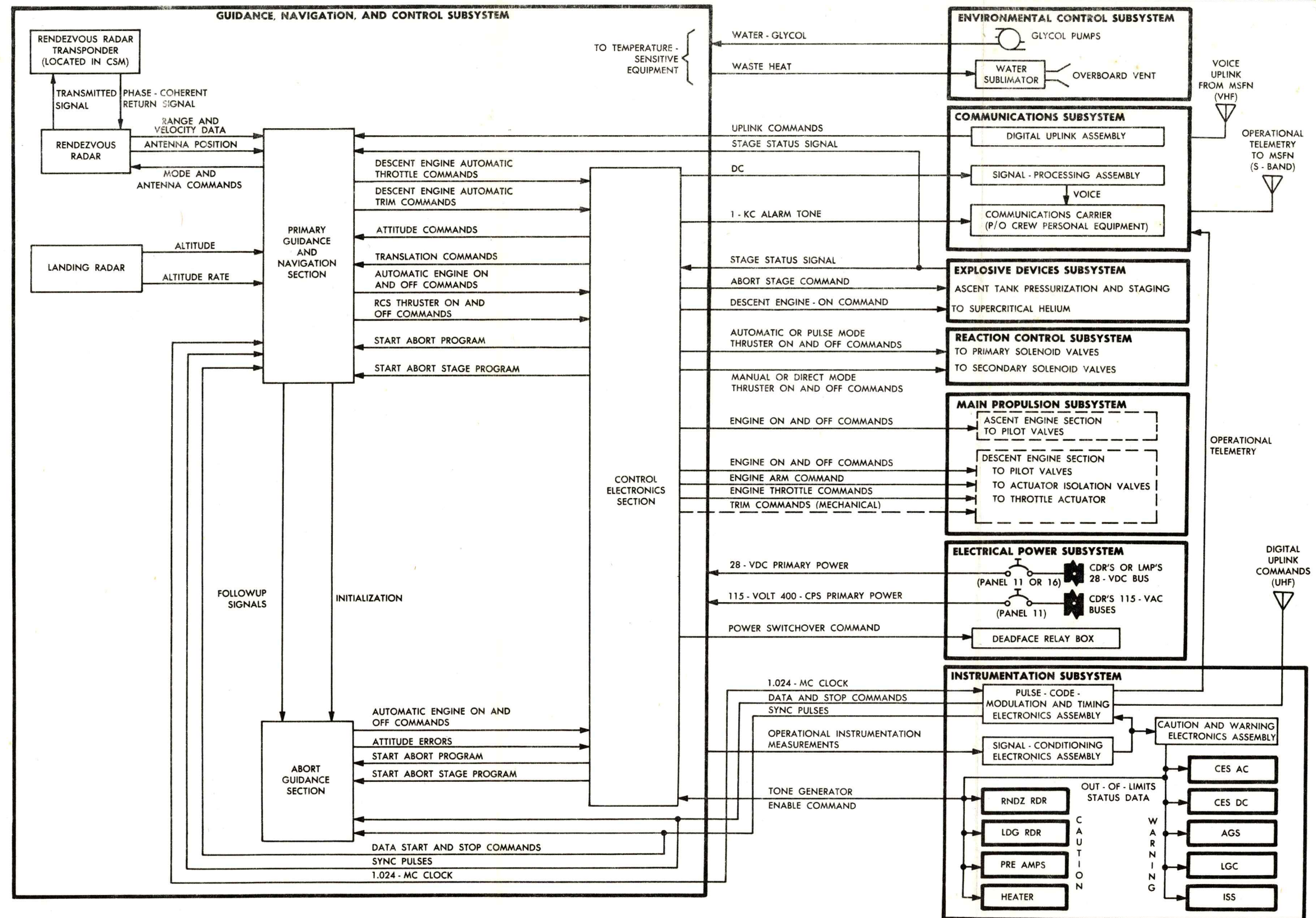
The AGS determines the LM trajectory or trajectories required for a coelliptic rendezvous sequence to place the LM in a safe parking orbit. It controls LM attitude, navigation, and guidance. Rendezvous from the abort point can be accomplished automatically under AGS control, or by the astronauts using data displayed from the AGS. When the AGS is used, the CES functions as an auto-pilot. The CSE uses AGS input signals and manual control signals to control LM attitude and translation.

2.2.1.3 Rendezvous Radar and Transponder. (See figure 2.2-1A.)

The RR, operated in conjunction with the RR transponder in the CSM, is used to acquire and track the CSM before and during rendezvous and docking. (See figure 2.2-3.) When the PGNS is in control (normal operation), the RR provides the LGC and cabin displays with LOS range and range rate data, and LOS angle and angle rate data with respect to the CSM. During AGS operation, the astronauts furnish these data to the AEA through DEDA entries. The RR line of sight is defined as the line-of-sight direction between the CSM target and the RR antenna. The RR also tracks the CSM target during the coasting and engine burn descent phases of the mission to supply tracking data for any required abort maneuver.

During the rendezvous phase, RR performance is evaluated by comparing RR range and range rate tracking values with MSFN tracking values. A large deviation from the MSFN tracking values indicates a gross error in the RR range and range rate data. Data pertaining to the RR shaft- and trunnion-axis angles and IMU gimbal angles are also transmitted to MSFN for comparison with like ground-tracking data.

The RR transponder, a radar receiving and transmitting target, increases the RR maximum-usage range capabilities by reducing the RR transmitting power losses versus range from fourth to second power as compared to skin-track radar. By providing the RR with a discrete radar transmitting signal to acquire and track, the transponder makes the CSM appear to the RR as the most illuminated space object in its field-of-view. Acquiring the RR signal illuminating the CSM, the transponder side-steps the RR three-tone, phase-modulated carrier frequency down 40.8 mc to 9792 mc and then transmits the phase-coherent 9792 mc carrier frequency for acquisition by the RR. The three tone modulation of the transponder transmitted signal is kept phase coherent with the RR transmitted signal, received by the transponder.



A-300LM4-219

Figure 2.2-1A. GN&CS Simplified Block Diagram and Subsystem Interfaces

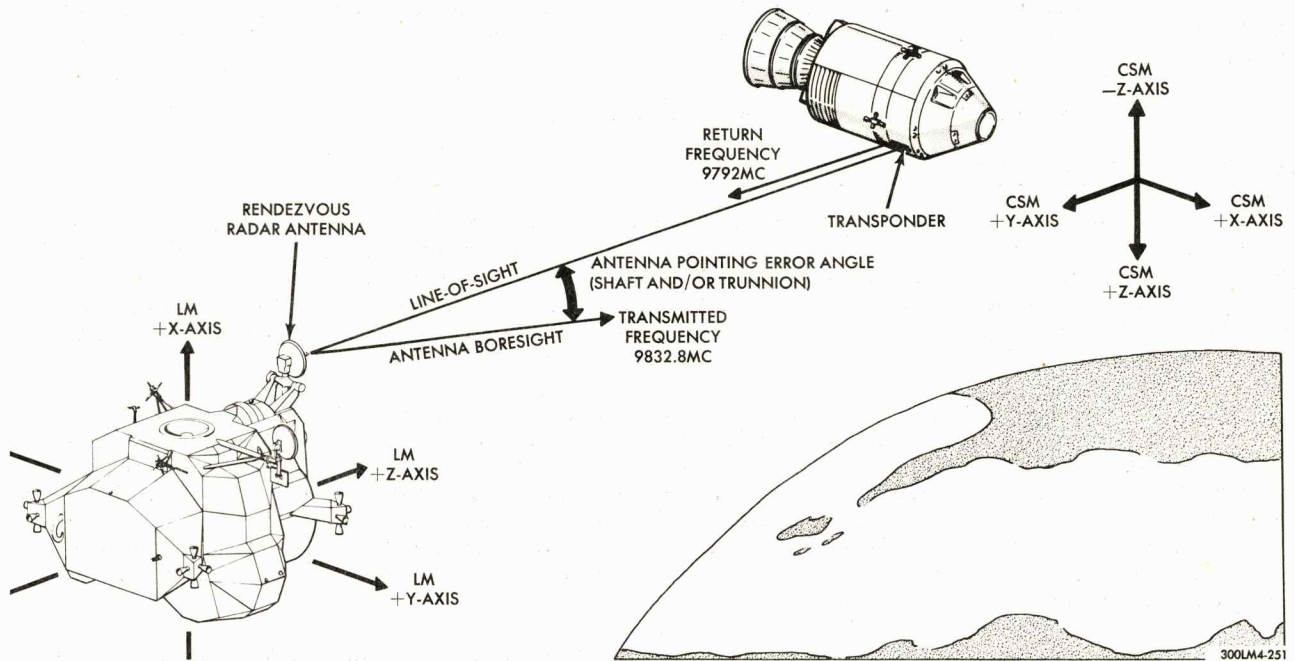


Figure 2.2-2. LM Rendezvous Radar and CSM Target Orientation

2.2.1.4 Landing Radar. (See figure 2.2-1A.)

The LR is designed to provide the LGC and astronauts with slant range and velocity data for control of descent to a lunar landing (subsequent missions). Based on these data, the LGC calculates control signals for LM rate of descent, hovering at low altitudes, and soft landing on the lunar surface. The LR is activated at approximately 50,000 feet above the lunar surface remains activated until lunar landing. Slant range data are available to the PGNS at approximately 25,000 feet; velocity data, at approximately 18,000 feet.

During LM-4 mission, the LR lunar performance cannot be evaluated because distant to the earth's surface will be beyond the LR transmitting and receiving capabilities. The LR LM-4 mission is to detect spurious signals likely to originate from reflections from the LM body or from the descent engine plumes. The LR capability in rejecting spurious signals is evaluated from the spurious signal data telemetered to MSFN ground stations.

2.2.1.5 Control Electronics Section. (See figure 2.2-1A.)

The CES controls LM attitude and translation about, and along, the three LM orthogonal axes by processing and routing command signals to fire any combination of the 16 thrust chamber assemblies (TCA's) of the Reaction Control Subsystem (RCS). The attitude and translation control signals originate automatically from the PGNS or the AGS, or are manually provided by an astronaut.

The CES also processes on and off commands for the ascent engine and the descent engine to provide major LM translations. The direction of the descent engine thrust vector is also controlled by the CES.

2.2.1.6 Orbital Rate Display - Earth and Lunar.

The ORDEAL provides an alternative to the attitude display, in pitch only. When selected, the ORDEAL produces an FDAI display of computed local vertical attitude during circular orbits around the earth.

■ 2.2.2 SUBSYSTEM INTERFACES. (See figure 2.2-1A.)

2.2.2.1 GN&CS - MPS Interfaces.

The GN&CS provides the sequence of commands to the Main Propulsion Subsystem (MPS) to control the ascent and descent engines. For ignition to occur, the engine must first be armed. Normally, this involves setting the ENG ARM switch (panel 1) to the desired position. Depending on the switch setting, a discrete is generated in the CES to enable the START pushbutton (panel 5) for ascent engine operation or to operate the actuator isolation valves of the descent engine. Under abort or emergency conditions, the MASTER ARM switch (panel 8) is used to perform the arming function.

When the PGNS is in control, on and off commands are generated automatically by the LGC under program control, or manually with the START and stop pushbuttons. With the AGS in control, on and off commands are generated automatically by the abort electronics assembly under specific routines, or manually with the START and stop pushbuttons. The on and off commands actuate pilot valves, which hydraulically open or close the fuel and oxidizer shutoff valves. Under emergency conditions, the ascent engine ignition sequence may also be completed automatically through use of the ABORT STAGE pushbutton. If the ascent engine on command from either computer is lost, a memory circuit in the CES keeps issuing the command to the ascent engine.

The descent engine receives on and off commands, throttle commands, and trim commands from the descent engine control assembly (DECA). The ignition sequence commands for the descent engine are generated automatically or manually in a manner similar to that of the ascent engine. On and off commands are routed from the LGC or AEA (dependent on the guidance mode selected), or the START and stop pushbuttons, through the DECA to actuate the descent engine pilot valves.

Throttle commands to the descent engine are generated automatically by the LGC under program control, or manually with the TTCA. The TTCA can be used to override LGC throttle commands. The AGS cannot throttle the descent engine. Throttle commands cause the throttle actuator of the descent engine to change the position of the flow control valves and vary the injector orifice of the engine. Changing the position of the flow control valves changes the amount of fuel and oxidizer metered into the engine and thus changes the magnitude of engine thrust.

The GN&CS generates trim commands to tilt the descent engine to control the direction of the thrust vector. The descent engine is tilted about the LM Y-axis and Z-axis to compensate for the offset of the center of gravity due to fuel depletion during descent engine operation. The thrust vector is controlled by the LGC with the aid of two gimbal drive actuators (GDA's). The GDA's are pinned to the descent engine and the LM structure along the Y-axis (roll) and Z-axis (pitch). When actuated, the GDA's extend or retract a screwjack-actuated arm that tilts the engine to attain the desired thrust vector. Thrust vector control for the ascent engine is achieved through firing of selected upward-firing TCA's.

2.2.2.2 GN&CS - RCS Interface.

The GN&CS provides on and off commands to the 16 TCA's to control LM attitude and translation. In the primary mode of operation (PGNS in control), the LGC generates the required commands and sends them to the proper jet drivers in the CES. The jet drivers send selected on and off commands to the RCS primary solenoids. In the secondary mode of operation (AGS in control), the AGS supplies the CES with attitude errors. The attitude and translation control assembly (ATCA) in the CES uses these inputs to select the proper TCA for attitude and translation control.

The TCA's are controlled manually with the ACA and TTCA. The ACA provides attitude commands and the TTCA provides translation commands to the LGC during the primary mode of operation and to the ATCA during the secondary mode of operation. The ACA can fire the TCA's directly during the pulse, direct, and hardover modes, bypassing the LGC or AEA, and the ATCA. The four downward-firing TCA's may be fired by pressing the +X-TRANSL pushbutton (panel 5). The on and off commands

supplied to the TCA take the form of a step function. The duration of the signal determines the firing time of the selected TCA, which ranges from a pulse (less than 1 second) to steady-state (1 second or longer).

Each TCA contains an oxidizer solenoid valve and a fuel solenoid valve which, when open, pass propellant through an injector into the combustion chamber, where ignition occurs. Each valve contains a primary (automatic) solenoid and a secondary (direct) solenoid which open the valve when energized. On and off commands from the ATCA are applied to the primary solenoids; the direct commands are applied to the secondary solenoids.

2.2.2.3 GN&CS - EPS Interface.

The Electrical Power Subsystem (EPS) supplies primary d-c and a-c power to the GN&CS. This power originates from six silver-zinc batteries (four in the descent stage and two in the ascent stage). The descent batteries feed power to the buses during all operations, before staging. Immediately before staging occurs, ascent battery power is switched on and descent battery power is terminated. A deadface relay circuit deadfaces the descent batteries when normal staging occurs. Under emergency conditions, when the ABORT STAGE pushbutton is pressed, a power switchover command, which initiates deadfacing automatically, is routed to the EPS. The 28-volt d-c battery power is routed through an inverter to provide 115-volt, 400-cps ac to the GN&CS equipment. Refer to paragraph 2.2.3.4 for a functional description of power distribution.

2.2.2.4 GN&CS - ECS Interface.

The Environmental Control Subsystem (ECS) provides thermal stability for the temperature-sensitive electronic equipment of the GN&CS. The electronic equipment (except the IMU) is mounted on cold plates and rails through which ECS coolant (ethylene glycol-water solution) is routed to remove heat. To cool the IMU, the coolant flows through its case. (Refer to paragraph 2.2.2.1.) The heat that is removed from the equipment is vented overboard by the ECS sublimators.

2.2.2.5 GN&CS - CS Interface.

The Communications Subsystem (CS) interfaces directly with the GN&CS when the astronaut uses the push-to-talk switch on his ACA. When the switch is pressed, the ACA issues a d-c signal that enables an audio center in the signal-processor assembly of the VHF/AM communications. This enabling signal allows the audio signals from the microphones to be processed by the CS. The CS interfaces indirectly with the GN&CS, using VHF/AM communications for voice uplink commands. It also interfaces with a tone generator in the CES. The generator, enabled by a command from the master alarm circuit of the IS, issues a 1-kc tone to the astronaut headsets as an indication of a subsystem malfunction.

2.2.2.6 GN&CS - EDS Interface.

The GN&CS interfaces with the Explosive Devices Subsystem (EDS) to supply a descent engine-on status signal to initiate the abort staging sequence, and to receive a stage status signal. When the normal descent engine ignition sequence is completed for the first time (when the START pushbutton is pressed), the GN&CS issues a discrete to the EDS. This signal informs the EDS to switch from ambient helium pressure to supercritical helium pressure by firing an explosive valve cartridge on the supercritical helium tank. All other normal pressurization and staging sequences are initiated from the control panels, by the astronauts.

During emergency procedures, when the ABORT STAGE pushbutton is pressed, the GN&CS generates a command to pressurize the ascent helium tanks and a command that fires appropriate explosive devices to initiate staging. Upon completion of staging, a stage status signal is routed from the EDS deadface switch to the ATCA and to the LGC. This signal automatically selects the power deadband for RCS control during ascent engine operation.

2.2.2.7 GN&CS - IS Interface.

The Instrumentation Subsystem (IS) senses GN&CS physical status data, monitors the GN&CS equipment, and performs in-flight checkout. The data signals are conditioned by the signal-conditioning electronics assembly (SCEA) and supplied to the pulse-code-modulation and timing electronics assembly (PCMTEA) and the caution and warning electronics assembly (CWEA). The PCMTEA changes the input signals to a serial digital form for transmission to MSFN. The CWEA checks the status of the GN&CS by continuously monitoring the information supplied by the SCEA. When an out-of-limits condition is detected by the CWEA, the CWEA energizes one or more of the caution and warning lights associated with the GN&CS.

The LGC interfaces directly with the IS to supply a 1.024-mc primary timing signal for the PCMTEA. This timing signal is used in generating timing and sync signals required by other subsystems. The IS supplies the LGC with telemetry data start and stop commands and sync pulses for clocking out telemetry data. It also supplies the AEA with telemetry stop commands and sync pulses.

2.2.3 FUNCTIONAL DESCRIPTION.

The GN&CS comprises two functional loops, each of which is an independent guidance and control path. The primary guidance path performs all the functions necessary to complete the LM mission. If a failure in the PGNS occurs, the abort guidance path can perform all the functions necessary to effect a safe rendezvous with the orbiting CSM.

The primary guidance path (see figure 2.2-3) comprises the PGNS, CES, LR, RR, and the selected propulsion section. The CES routes the guidance and navigation data from the PGNS and applies them to the descent or ascent engine, and selected TCA's.

The IMU, which continuously measures attitude and acceleration, is the primary inertial sensing device of the LM. The LR senses LM slant range and velocity. The RR coherently tracks the CSM to derive line-of-sight (LOS) range, range rate, and angle rate. The LGC is the central data-processing device of the LM. The LGC uses AOT sighting data to align the IMU. Using inputs from the LR, IMU, RR, TTCA's and ACA's the LGC solves guidance, navigation, steering, and stabilization equations necessary to initiate on and off commands for the descent and ascent engines, throttle commands and trim commands for the descent engine, and on and off commands for the TCA's.

Control of the LM, when using the primary guidance path, ranges from fully automatic to manual. The primary guidance path operates in the automatic mode or the attitude hold mode. In the automatic mode, all navigation, guidance, stabilization, and control functions are controlled by the LGC. When the attitude hold mode is selected, the astronaut uses his ACA to bring the LM to a desired attitude. When the ACA is moved out of the detent position, proportional attitude-rate or minimum impulse commands are routed to the LGC. The LGC then calculates steering information and generates TCA commands that correspond to the mode of operation selected via the display and keyboard assembly (DSKY). These commands are applied to the primary preamplifiers in the ATCA, which routes the commands to the proper TCA. When the astronaut releases the ACA, the LGC generates commands to hold this attitude. If the astronaut commands four-jet direct operation of the ACA by going to the hard-over position, the ACA applies the command directly to the secondary solenoids of the corresponding TCA.

In the automatic mode, the LGC generates descent engine throttling commands, which are routed to the descent engine via the descent engine control assembly (DECA). The astronaut can manually control descent engine throttling with his TTCA. The DECA sums the TTCA throttle commands with the LGC throttle and applies the resultant signal to the descent engine. The DECA also applies trim commands, generated by the LGC, to the GDA's to provide trim control of the descent engine. The LGC supplies on and off commands for the ascent and descent engines to the S&C control assemblies. The S&C control assemblies route the ascent engine on and off commands directly to the ascent engine, and the descent engine on and off commands to the descent engine via the DECA.

In the automatic mode, the LGC generates +X-axis translation commands to provide ullage. In the manual mode, manual translation commands are generated by the astronaut, using his TTCA. These commands are routed, through the LGC, to the ATCA and on to the proper TCA.

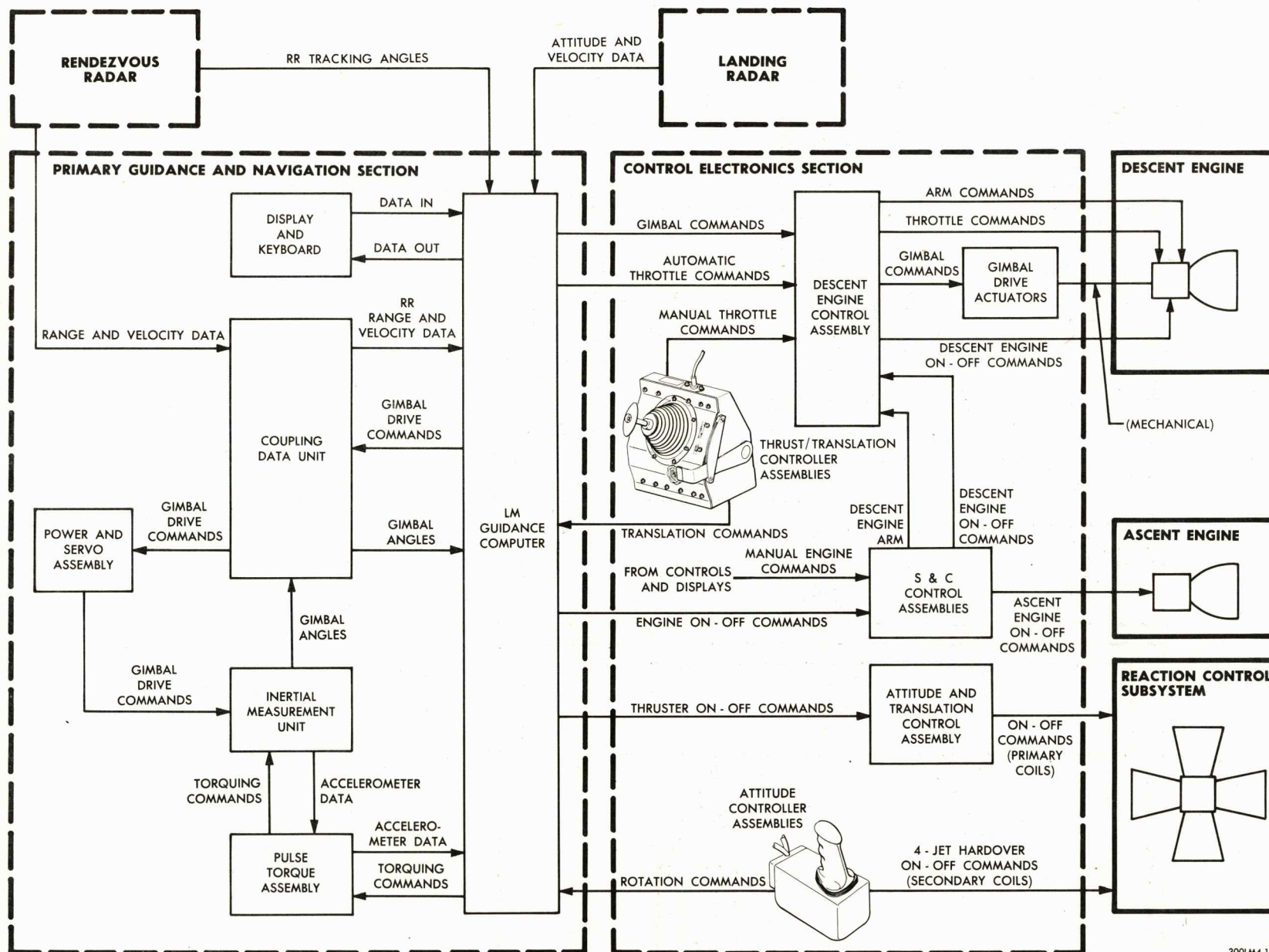


Figure 2.2-3. Primary Guidance Path - Simplified Block Diagram

300LM4-183

GN&CS

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

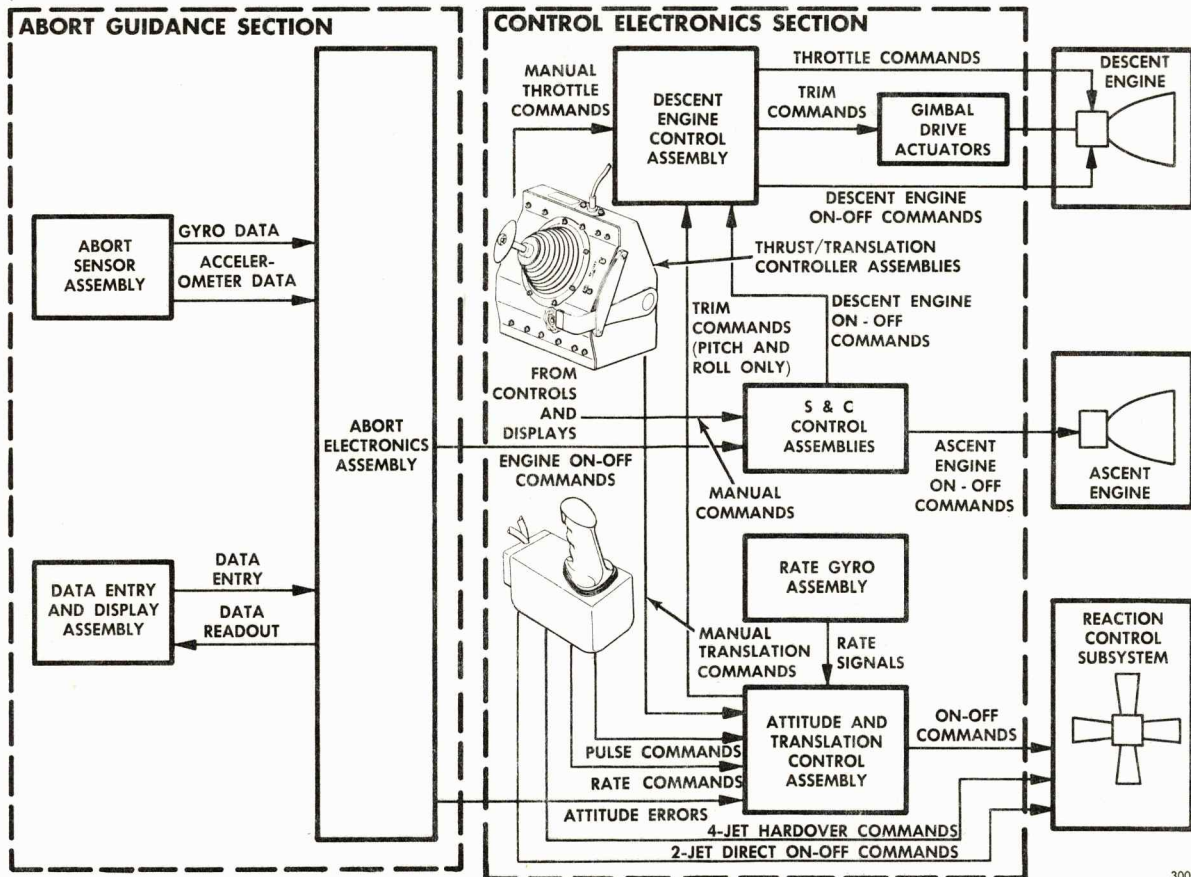


Figure 2.2-4. Abort Guidance Path - Simplified Block Diagram

The abort guidance path (see figure 2.2-4) comprises the AGS, CES, and selected propulsion section. The AGS is a backup system for the PGNS. The term "abort guidance path" is somewhat misleading in that any abort situation is normally under PGNS control; the AGS is used only if the PGNS malfunctions. The AGS performs all inertial navigation and guidance functions necessary to effect a safe orbit or rendezvous with the CSM. The stabilization and control functions are performed by analog computation techniques, in the CES.

The AGS uses a strap-down inertial sensor, rather than the stabilized, gimballed sensor used in the PGNS IMU. The abort sensor assembly (ASA) is a strap-down inertial sensor package that measures attitude and acceleration with respect to the LM body axes. The ASA sensed attitude is supplied to the abort electronics assembly (AEA). The AEA is a high-speed, general-purpose digital computer that performs the basic strap-down system computations and the abort guidance and navigation steering control calculations. The data entry and display assembly (DEDA) is a general-purpose input-output device through which the astronaut manually enters data into the AEA and commands various data readouts.

The CES functions as an autopilot when the abort guidance path is selected. It uses inputs from the AGS and from the astronauts to provide the following: on, off, and TTCA throttling commands for the descent engine; gimbal commands for the GDA's to control descent engine trim; on and off commands for the ascent engine; engine sequencer logic to ensure proper arming and staging before engine startup and shutdown; on and off commands for the TCA's for translation and stabilization, and for various maneuvers; jet-select logic to select the proper TCA's for the various maneuvers; and modes of LM control, ranging from fully automatic to manual.

The astronaut uses the TTCA to control throttling of the descent engine and translation maneuvers. The throttle commands, engine on and off commands from the S&C control assemblies, and trim commands from the ATCA are applied to the DECA. The DECA applies the throttle commands to the descent engine, the engine on and off commands to the descent engine latching device, and the trim commands to the GDA's. The S&C control assemblies receive engine on and off commands for the descent and ascent engines from the AEA. As in the primary guidance path, the S&C control assemblies route descent engine on and off commands to the DECA and apply ascent engine on and off commands directly to the ascent engine.

The abort guidance path operates in the automatic mode or the attitude hold mode. In the automatic mode, navigation and guidance functions are controlled by the AGS; stabilization and control functions, by the CES. In the attitude hold mode, the astronaut uses his ACA to control LM attitude. The ACA generates attitude-rate, pulse, direct, and hardover commands. The attitude-rate and pulse commands, AEA error signals, rate gyro assembly (RGA) rate-damping signals, and TTCA translation commands are applied to the ATCA. The ATCA processes these inputs to generate TCA on and off commands.

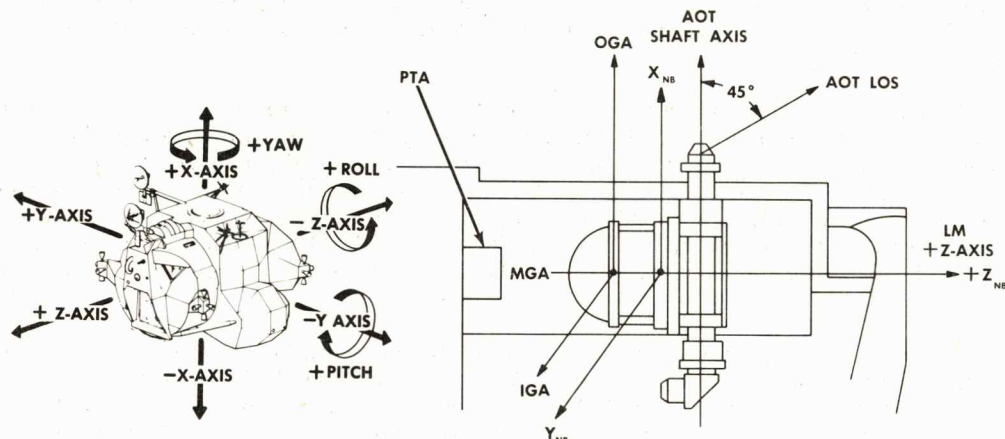
In the attitude hold mode, pulse and direct submodes are available for each axis. The pulse submode is an open-loop attitude control mode in which the ACA is used to make small attitude changes in the selected axis. The direct submode is an open-loop attitude control mode in which pairs of TCA's are directly controlled by the ACA. The astronaut can also control vehicle attitude in any axis by moving the ACA to the hardover position. In addition, the astronaut can override translation control in the +X-axis with the +X TRANSL pushbutton (panel 5). Pressing the +X TRANSL pushbutton fires all four +X-axis TCA's.

2.2.3.1 LM Vehicle, Guidance, Navigation and Control Subsystem Axes. (See figure 2.2-5.)

Several sets of axes are associated with the LM and the GN&CS. Each set is a three-axis, right-hand, orthogonal coordinate system. Figure 2.2-5 shows the relationships of various sets of axes when the IMU gimbal angles are 0°.

2.2.3.1.1 LM Vehicle Axes.

The X-axis positive direction is through the overhead hatch; the Z-axis positive direction is through the forward hatch. The Y-axis is perpendicular to the X-Z plane.



B-300LM4-150

Figure 2.2-5. LM Vehicle and GN&CS Axes

2.2.3.1.2 Navigation Base Axes.

The navigation base is mounted to the LM structure so that a coordinate system is formed by its mounting points. The Y_{NB} axis is parallel to the vehicle Y-axis. The X_{NB} axis is parallel to the vehicle X-axis. The Z_{NB} axis is perpendicular to the X_{NB} - Y_{NB} plane and parallel to the vehicle Z-axis.

2.2.3.1.3 IMU Axes.

The IMU axes are defined by the three gimbal axes. These axes are designated as outer gimbal axis (OGA), middle gimbal axis (MGA), and inner gimbal axis (IGA). The gimbal axes are defined when the gimbal angles are 0° ; they are as follows: The OGA is parallel to the X-axis, the MGA is parallel to the Z-axis, and the IGA is parallel to the Y-axis. The axes of the IMU stable member are parallel to the vehicle axes and the gimbal axes when the gimbal angles are 0° .

2.2.3.1.4 Alignment Optical Telescope Axes. (See figure 2.2-6.)

The AOT is mounted to the navigation base so that the AOT shaft axis is parallel to the X-axis. The telescope LOS is approximately 45° above the vehicle Y-Z plane. The AOT LOS is fixed in elevation and movable in azimuth to six detent positions. These detent positions are selected manually by turning a detent selector knob on the AOT; they are located at 60° intervals. Three positions (forward, left, and right), can be used for star sightings. The forward (F), or zero detent position, places the LOS in the X-Z plane, looking forward and up as one would look from inside the LM. The right (R) position places the LOS 60° to the right of the X-Z plane; the left (L) position, 60° to the left of the X-Z plane. Each of these positions maintains the LOS at 45° from the LM +X-axis. The remaining three detent positions reverse the prism on top of the AOT so that the frontal plane is protected by a shield. These positions are right-rear, closed (CL), and left-rear. The CL position (180° from the F position) is the stowage position. The right-rear and left-rear positions have minimal use.

2.2.3.1.5 Inertial Reference Integrating Gyro Axes.

The inertial reference integrating gyro (IRIG) axes, designated X_g , Y_g , and Z_g , are parallel to the LM body axes. If the attitude of the stable member is changed with respect to inertial

space, the gyro senses the change about its axis and provides an error signal to the stabilization loop of the IMU.

2.2.3.1.6 Pulse Integrating Pendulous Accelerometer Axes.

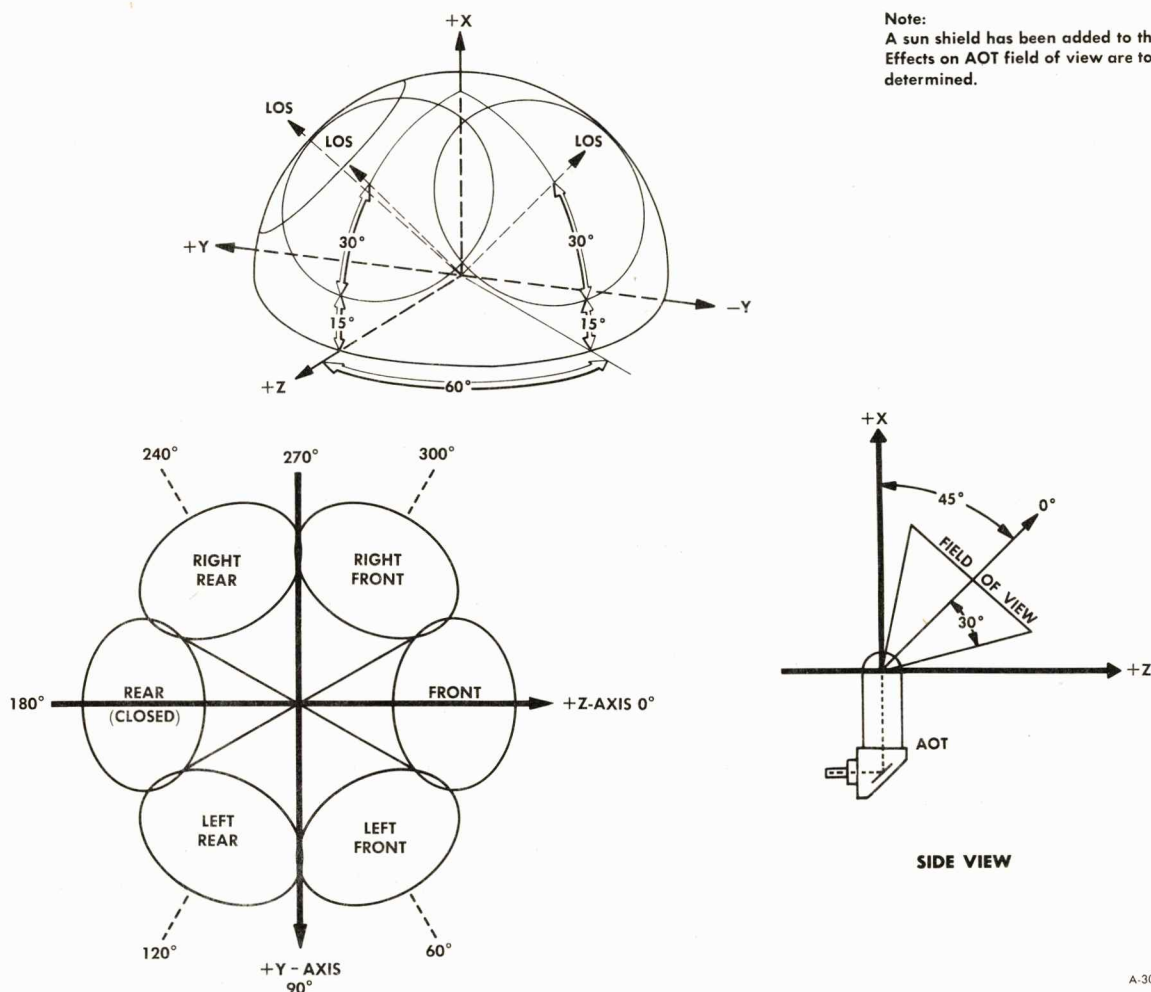
The pulse integrating pendulous accelerometer (PIPA) axes, designated X_a , Y_a , and Z_a , are parallel to the LM body axes. Velocity changes are measured along the PIPA axes.

2.2.3.1.7 Rendezvous Radar Antenna Axes.

The axes associated with the RR antenna are discussed in paragraph 2.2.3.4.

2.2.3.1.8 Rendezvous Radar Transponder Antenna Axes.

The axes associated with the RR transponder are discussed in paragraph 2.2.3.5.



A-300LM4-143

Figure 2.2-6. Alignment Optical Telescope Axes

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.2.3.1.9 Landing Radar Antenna Axes.

The axes associated with the LR antenna are discussed in paragraph 2.2.3.6.

2.2.3.2 Primary Guidance and Navigation Section. (See figure 2.2-7.)

Functionally, the PGNS is divided into three major subsections: inertial, optical, and computer. The subsections individually, or in combination, perform the following functions:

- Establish an inertial reference that is used for measurements and computations
- Enable IMU alignment through the LGC by entering optical sighting data through the DSKY
- Enable calculation of LM position and velocity by inertial techniques
- Enable LM and CSM rendezvous by radar tracking and inertial techniques
- Generate attitude-control and thrust commands to maintain the vehicle on a satisfactory trajectory
- Control throttling and gimbal trim of the descent engine
- Display pertinent flight data related to guidance status.

The inertial subsection (ISS) performs the following major functions:

- Maintains a platform (stable member) fixed with respect to a preselected inertial reference frame
- Measures the attitude of the stable member with respect to the LM
- Senses LM acceleration due to thrust along the coordinate system defined by the stable member orientation
- Assists in data conversion for display or telemetry transmission.

The optical subsection (OSS) performs the following major functions:

- Provides a means of aligning the inertial reference on the lunar surface
- Provides measurements for establishing the inertial reference in flight.

The computer subsection (CSS) performs the following major functions:

- Aligns the IMU
- Solves the guidance and navigation problems for all mission phases
- Provides control information to the PGNS and to other subsystems
- Displays pertinent information to the astronauts and MSFN when requested
- Provides a means by which the astronauts can directly communicate with the LGC
- Provides on-off control of the RCS thrusters and the ascent and descent engines
- Provides gimbaling and throttling control of the descent engine
- Provides the timing for the PGNS and other LM subsystems
- Monitors its own operation and certain other subsystem operations
- Positions the RR antenna.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

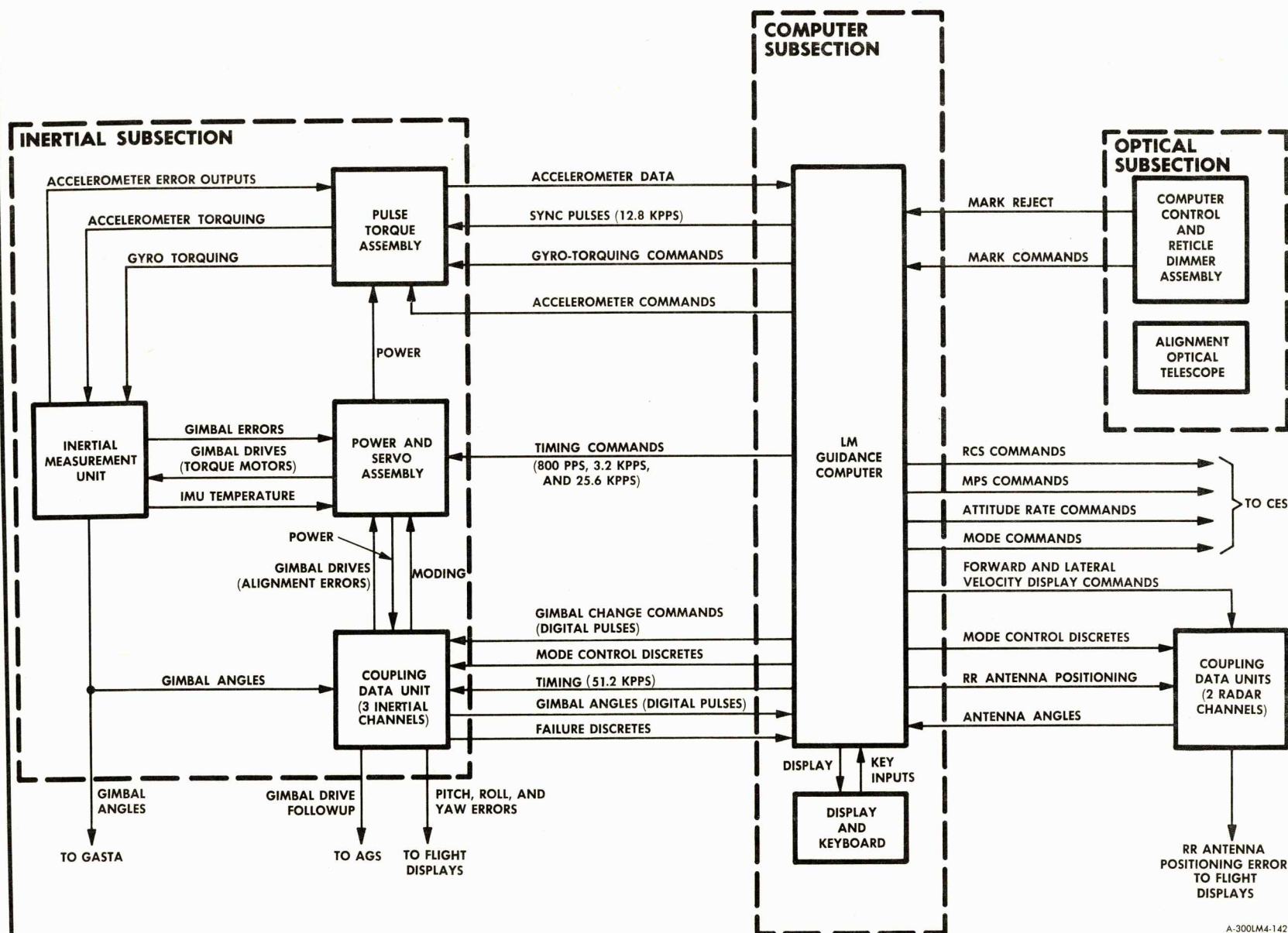


Figure 2.2-7. Primary Guidance and Navigation Section - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.2.3.2.1 Inertial Subsection. (See figure 2.2-8.)

The ISS comprises the navigation base (NB), IMU, the coupling data unit (CDU), the pulse torque assembly (PTA), the power and servo assembly (PSA), and the signal conditioner (SC).

ISS operation can be initiated automatically by the LGC or manually by the astronaut, using DSKY entries to command the LGC to select the various operating modes. The ISS status or mode of operation can be displayed on the DSKY as determined by a computer program. The IMU furnishes the inertial reference; it consists of a stable member with three degrees of freedom, stabilized by three integrating gyros. The stable member must be aligned with respect to the reference coordinate system each time the ISS is powered-up. The stable member must be realigned during flight because it may deviate from its alignment, due to gyro drift. Also, the crew may desire a new stable member orientation. The alignment orientation may be that of the CSM or that defined by the thrusting programs within the LGC. Sighting of two stars is required for in-flight fine alignment. The stable member is aligned after the LGC processes sighting data that have been combined with the known IMU angles and supplies gyro-torquing signals to the IMU.

Once the ISS is energized and aligned to an inertial reference, LM rotation is about the gimbaled stable member, which remains fixed in space. Resolvers, mounted on the gimbal axes, act as angle-sensing devices and measure LM attitude with respect to the stable member. These angular measurements are displayed to the astronaut by the flight director attitude indicator (FDAI) and angular changes of the inertial reference are sent to the LGC.

Desired LM attitude is calculated in the LGC and compared with the actual gimbal angles. A difference between the actual and calculated angles results in generation of attitude error signals, by the ISS channels of the CDU, which are sent to the FDAI for display. These error signals are used by the digital autopilot program in the LGC to activate RCS thrusters for LM attitude correction. Attitude error is displayed by the FDAI error needles. LM acceleration due to thrusting is sensed by three pendulous integrating accelerometers, which are mounted on the stable member with their input axes orthogonal. The resultant signals (velocity changes) from the accelerometer loops are supplied to the LGC, which calculates the total LM velocity.

Inertial Measurement Unit. The IMU is the primary inertial sensing device of the LM. It is a three-degree-of-freedom, gyroscopically stabilized device whose primary functions are:

- Maintaining an orthogonal, inertially referenced coordinate system for LM attitude control and measurement
- Maintaining three accelerometers in the reference coordinate system for accurate measurement of velocity changes.

Coupling Data Unit. The CDU converts angular data and transfers the data between major assemblies of the GN&CS. The CDU contains five channels for processing these data, as follows:

- One for the RR shaft axis
- One for the RR trunnion axis
- One for each of the three gimbal axes of the IMU.

The two CDU channels used with the RR provide interfaces between the antenna and the LGC. The LGC calculates digital antenna position commands before acquisition of the CSM. These signals are converted to analog form by the CDU and applied to the antenna drive mechanism to aim the antenna toward the CSM. Tracking-angle information is converted to digital form by the CDU and applied to the LGC.

The three CDU channels used with the IMU provide interfaces between the IMU and the LGC, and between the LGC and the AGS. Each IMU gimbal angle transmitter resolver provides its channel with analog gimbal-angle signals that represent LM attitude. The CDU converts these signals to digital form and applies them to the LGC. The LGC uses these signals to calculate attitude or translation commands

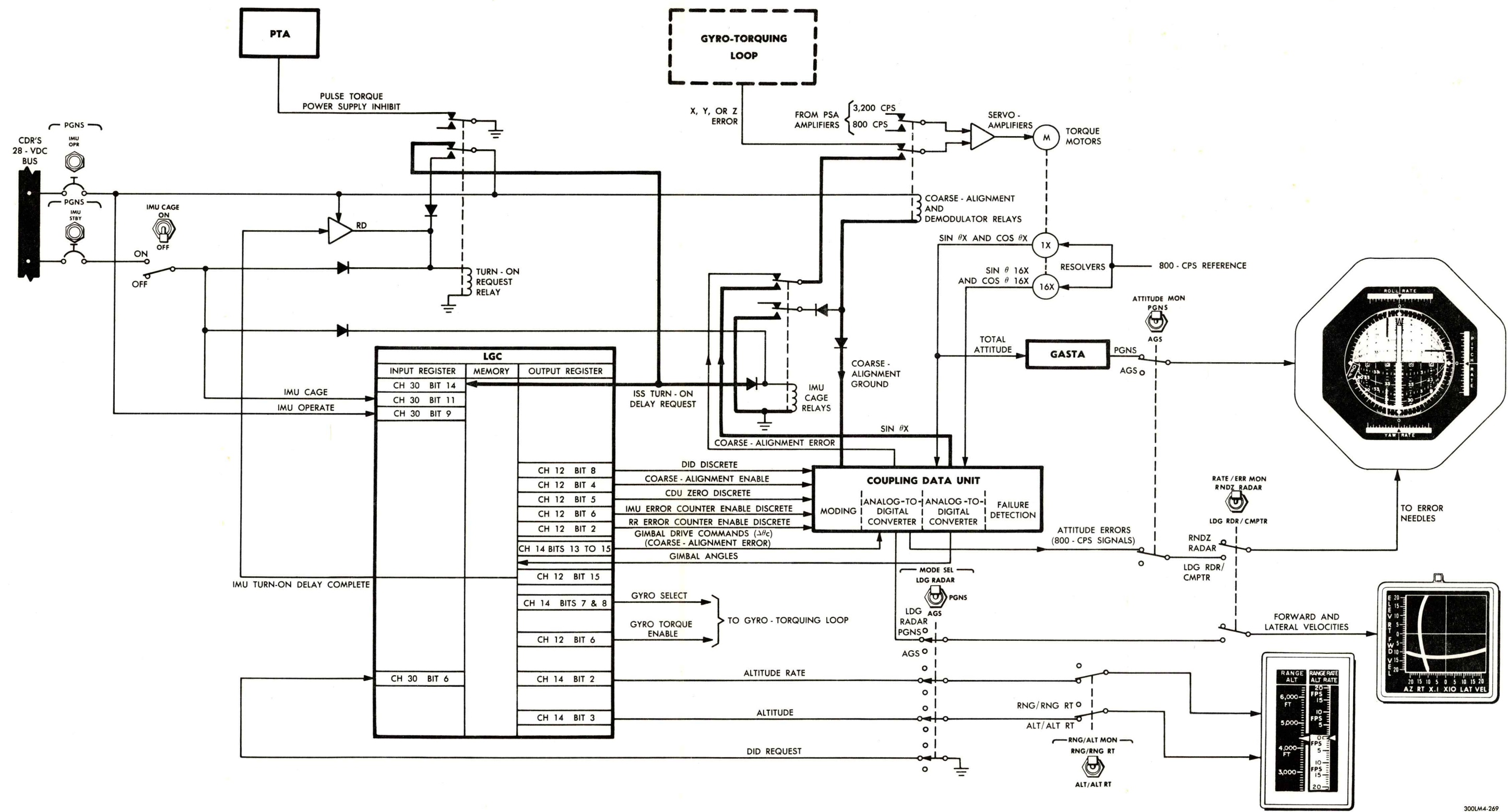


Figure 2.2-10. Inertial Subsection - Modes of Operation

After the logic circuitry in the CDU has been set up to accept commands from the LGC, the LGC begins transmitting positive or negative $\Delta\theta_c$ pulse trains at 3,200 cps. These pulses, each equivalent to a gimbal angle change of 160 arc seconds, are accumulated in the error counter. The first $\Delta\theta_c$ pulse determines the direction in which the error counter is to count and provides a (digital-to-analog converter) polarity control to the digital-to-analog converter. The polarity control provides an in-phase or an out-of-phase analog reference. An 800-cps analog signal, whose amplitude is dependent on the error counter content and the polarity of the input command ($\Delta\theta_c$), is then generated. This signal is the 800-cps coarse-alignment output from the digital-to-analog section; it is routed to the gimbal servo-amplifier, causing the gimbal to drive in the direction commanded by the LGC.

The changing gimbal angles are detected by the error-detector circuits in the CDU. These detected errors permit a pulse train, at 6,400 pps, to increase the read counter. The increasing read counter nulls the sine and cosine voltage inputs to the error-detector circuits from the IMU 1X and 16X resolvers.

As the read counter is being incremented, one output of the counter, representing a 40-arc-second-per-pulse, increase in gimbal angle, is routed to the LGC. Another output of the counter, representing 160 arc seconds per pulse, is recognized in the CDU logic as an incremental value to be entered into the error counter in a direction opposite to that of the LGC-commanded $\Delta\theta$. If $\Delta\theta$ is positive, the error counter counts up and the $\Delta\theta_g$ from the read counter decreases the counter. For each counter pulse into the error counter, the total content decreases. This decreases the digital-to-analog converter output and, therefore, the rate of drive. When the number of digital feedback pulses equals the LGC-commanded number of pulses, the error counter is empty and the digital-to-analog converter output should be zero.

The rate of drive of the gimbals during coarse alignment is limited to a maximum of 35° per second. This is due to degenerative feedback provided within the CDU mechanization.

Fine-Alignment Mode. (See figure 2.2-10.) The fine-alignment mode allows the LGC to position the IMU accurately to a predetermined gimbal angle, closer than 40 arc seconds of CDU tolerance, since each gyro-torquing pulse is equal to 0.615 arc second of displacement. The LGC does not issue any discretes to the CDU during this mode of operation; therefore, the read counter circuitry repeats the changing gimbal angles exactly as was done in the coarse-alignment mode. The LGC keeps track of the gimbal angle to within 40 arc seconds.

The commanding signals for the fine-alignment mode are issued to the time-shared fine-alignment or pulse torque electronics. The LGC first issues a torque enable discrete, which applies 28 and 120 volts dc to the binary current switch, a differential amplifier, and a precision voltage reference circuit, allowing the circuit to become operative. The current switch is reset by the no-torque pulses, allowing a dummy current, which is equal to the torquing current, to flow. This allows the current to settle to a constant value before it is used for gyro torquing. A gyro is then selected (gyro select pulses) for either (positive or negative) torquing current. After the discretes have been issued, the LGC sends torque (set) pulses or fine-alignment commands to the set side of the current switch. The pulse allows the selected torquing current (positive or negative) to flow through the gyro windings, causing the float to move. The resulting signal generator output causes the stable member to be driven through an angle equal to the command angle. The LGC receives inputs from the CDU read counter that indicate a 40-arc-second-per-pulse change in gimbal angle.

The number of torquing pulses sent from the LGC to the torquing electronics is computed on the basis of gimbal angle at an instant of time and a desired alignment angle. The difference is converted into the number of pulses necessary to drive the gimbal through the difference angle. The required number of fine-alignment pulses is computed only once; it is not recomputed on the basis of gimbal angle after the desired number of pulses have been sent. Fine-alignment loop operation is open-loop as far as the LGC is concerned; the $\Delta\theta_g$ pulses are not used for feedback.

The fine-alignment pulses generated by the LGC are issued in bursts of 3,200 pps. The fine-alignment electronics permits the torquing current to be on in the direction chosen by LGC logic, for the duration of the pulse burst. When the LGC is not issuing fine-alignment pulses or the gyro floats are not being torqued, the stable member can be considered inertially referenced.

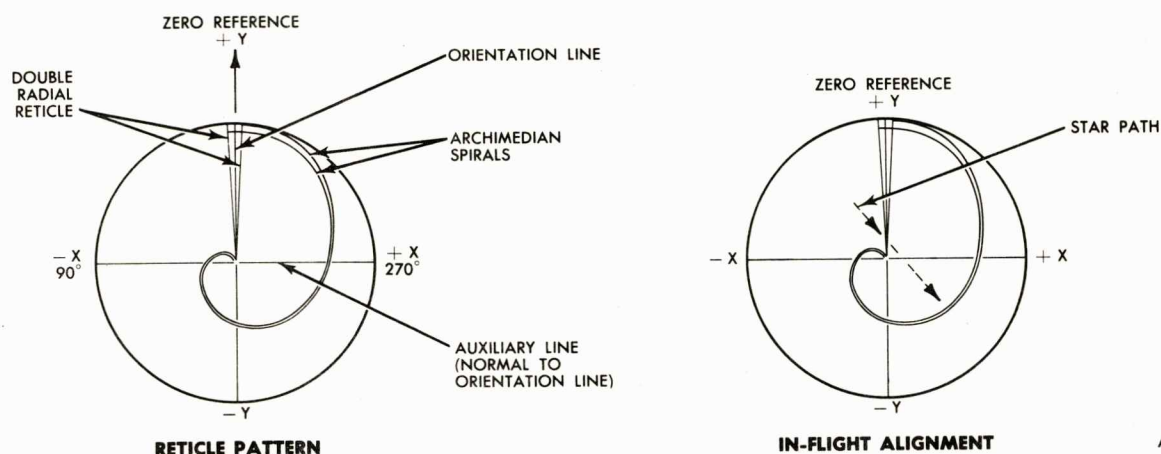


Figure 2.2-11. Alignment Optical Telescope - Reticle Pattern

and, if desired, he may erase the latest mark by pressing the REJECT pushbutton. When a MARK pushbutton is pressed, a discrete is sent to the LGC. The LGC then records the time of mark and the IMU gimbal angles at the instant of the mark.

Crossing of a reticle line by the star image defines a plane containing the star. Crossing of the other reticle line defines another plane containing the same star. The intersection of these planes form a line that defines the direction of the star. To define the inertial orientation of the stable member, sightings on at least two stars are required. Each star sighting requires the same procedure. Multiple reticle crossings and their corresponding marks can be made on either or both stars to improve the accuracy of the sightings. Upon completion of the second star sighting, the LGC calculates the orientation of the stable member with respect to a predefined reference coordinate system.

Lunar Surface Sighting. Because the LM-4 mission is an earth-orbital flight, OSS operation on the lunar surface is not discussed.

2.2.3.2.3 Computer Subsection.

The CSS is the control and processing center of the PGNS. It consists of the LGC and the DSKY. The CSS processes data and issues discrete outputs and control pulses to the PGNS, AGS, CES, and to other LM subsystems.

LM Guidance Computer. The LGC is a core memory, digital computer with two types of memory; fixed and erasable. The fixed memory permanently stores navigation tables, trajectory parameters, programs, and constants. The erasable memory stores intermediate information.

The LGC processes data and issues discrete control signals for the PGNS and the other subsystems. It is a control computer with many of the features of a general-purpose computer. As a control computer, the LGC aligns the IMU stable member and provides RR antenna drive commands to the RR channels of the CDU. The LGC provides control commands to the RR, LR, ascent engine, descent engine, RCS thrusters, and LM displays. As a general-purpose computer, the LGC solves guidance problems required for the mission. In addition, the LGC monitors the operation of the PGNS and other subsystems.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The LGC stores data pertinent to the ascent and descent flight profiles that the LM must assume to complete its mission. These data (position, velocity, and trajectory information) are used by the LGC to solve flight equations. The results of various equations can be used to determine the required magnitude and direction of thrust. The LGC establishes corrections to be made. The LM engines are turned on at the correct time, and steering signals are controlled by the LGC to orient the LM to a new trajectory, if required. The ISS senses acceleration and supplies velocity changes to the LGC for calculating total velocity. Drive signals are supplied from the LGC to the CDU and stabilization gyros in the ISS to align the gimbal angles in the IMU. IMU position signals are supplied to the LGC to indicate changes in IMU gimbal angles. The LGC provides drive signals to the RR for antenna positioning and receives, from the RR channels of the CDU, antenna angle information. This information is used in the LGC in the antenna-positioning calculations. Star-sighting information is loaded manually into the LGC via the DSKY. This information is used in the LGC to calculate IMU alignment commands.

Display and Keyboard. (See figure 2.2-12.) The DSKY provides a two-way communications link between the astronauts and the LGC to enable performance of the following functions:

- Loading of data into the LGC
- Display of data from the LGC and of data loaded into the LGC
- Monitoring of data in the LGC
- Display of LGC modes of operation
- Display of requests, by the LGC, for the astronauts to perform actions.

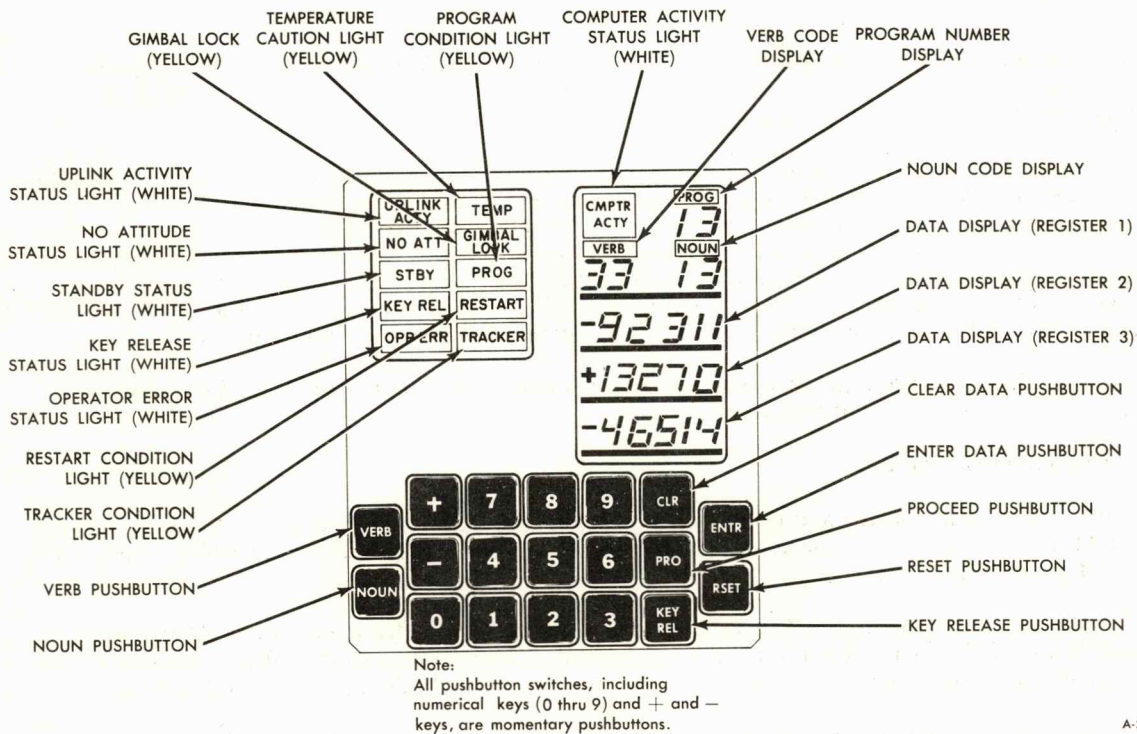


Figure 2.2-12. Display and Keyboard - Pictorial

The appropriate response to an internally initiated verb-noun combination for loading is as follows:

- Load the desired data. After the final entry, the internal sequence proceeds normally.
- Proceed, or proceed without data, by keying VERB, 33, ENTR.
- Terminate by keying VERB, 34, ENTR.

Please Perform. The operator must respond to a "please perform" request. With this request, the verb-noun combination flashes and the internal sequence is interrupted. The "please perform" verb (50) is usually used with the "checklist" noun (25) and an appropriate checklist code number (table 2.2-3) in R1. The appropriate response is as follows:

- Press the ENTR pushbutton to indicate that the requested action has been performed. (The internal sequence continues normally.) Proceed without data by keying VERB, 33, ENTR. The operator chooses not to perform the requested action, but desires the internal sequence to continue with the previous data.
- Terminate by keying VERB, 34, ENTR.

The "please perform" verb is also used with the "change of program" noun and "engine-on enable" noun. Its use in these cases is subject to the LGC program in process.

Please Mark. The "please mark" verbs are flashed when the LGC is prepared to accept optical-sighting data from the AOT.

2.2.3.2.4 Primary Guidance and Navigation Section - Modes of Operation.

The PGNS is considered to be in an operational mode upon initiation of a program by the astronauts or MSFN. When operating under one of the various programs (table 2.2-7), the LGC automatically computes required mission parameters, commands the PGNS and the other sections and subsystems, and displays pertinent data to the astronaut and MSFN (via downlink). For operational compatibility, the astronauts and/or MSFN can initiate, modify, or interrupt the automatic program sequences. The programs are initiated in certain cases, by a previous program.

The LGC is preprogrammed to display a mode number or program number on the DSKY in response to initiation of a program. This display remains on until the sequence of events for the specific mission phase, as dictated by the program, is completed.

The astronaut may also be required, or may wish, to perform specific submodes (routines) during a program. The PGNS routines (table 2.2-8) are used by most LGC programs, to perform the required input and output functions. Through these routines, the LGC can command various guidance modes, display and accept information from the DSKY and radar, provide for telemetry inputs and outputs, control positioning of the RR antenna and the IMU stable member, and remain cognizant of the PGNS and LM subsystem operations. Only the ISS of the PGNS operates under specific modes when the PGNS is used. These modes of ISS operation are listed and defined in paragraph 2.2.3.2.1.

2.2.3.2.5 General Operation of Primary Guidance and Navigation Section. (See figure 2.2-13.)

This discussion of PGNS operation is limited to astronaut interface with the PGNS, because PGNS operation is dependent on the LGC program in progress and on the mission phase. The astronaut can perform optical sightings, monitor subsystem performance, load data, select the mode of operation, and, with the aid of the PGNS, manually control the LM. The program to be performed by the LGC is selected by the astronaut or initiated by the LGC.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The DSKY enables the astronaut to communicate with the LGC and perform a variety of tasks such as testing the LGC, entering voice link data, and commanding IMU mode switching. The hand controllers permit manual changes or computer-aided manual changes in attitude or translation. The PGNS flight data that are displayed to assist the astronaut during various phases of the LM-4 mission are: total LM attitude, attitude errors, altitude and altitude rate, forward and lateral velocities, and percentage of descent engine thrust commanded by the LGC.

Total attitude is generated from the IMU gimbal angles. With the ATTITUDE MON switch (panel 1) set to PGNS, the IMU gimbal angles are routed to the GASTA. The GASTA transforms the gimbal angles into signals of the proper configuration of total attitude. The total attitude signals are applied to the FDAI sphere for direct astronaut readout. The FDAI also displays roll, pitch, and yaw rates and errors. The FDAI rate indicators monitor the rate of change of angular position. When the RATE/ERR MON switch (panel 1) is set to LDG RDR/CMPTR, the FDAI error indicators indicate the deviation between the programmed and actual attitude. The FDAI rate indicators are fed from the CES rate gyros; the pitch, yaw, and roll attitude errors are supplied from the LGC via the CDU.

The astronauts can select the LGC, PGNS, or AGS as the source for the altitude and altitude-rate parameters. When the MODE SEL switch is set to PGNS, the LGC calculates altitude and altitude rate, but issues signals for display of either altitude or altitude rate. Altitude and altitude rate are not displayed simultaneously. These signals are routed through the RNG/ALT MON switch (panel 1) to the ALT and ALT RATE indicators (panel 1).

Forward and lateral velocities are displayed on the X-pointer indicator (panels 1 and 2). The indicator receives velocity signals from the LGC via the CDU when the MODE SEL switch is set to PGNS. The LGC calculates the velocities from its stored information and from information received from the LR. The LGC feeds the calculated data to the CDU for digital-to-analog conversion before display. The X-POINTER SCALE switch (panel 3) selects the scale of the indicator. The type of velocity and the scale selected are indicated by illuminated placarding on the borders of the X-pointer indicator.

The amount of descent engine throttling, as commanded by the LGC, is routed to the CES. The CES sends this command to the THRUST indicator and to the descent engine. The THRUST indicator also displays the amount of thrust sensed at the engine thrust chamber, so that a comparison can be made.

PGNS vehicle control includes interfacing for attitude and translation control and for propulsion control (descent and ascent engine). Commands from the LGC are routed through the CES to the RCS thrusters and to the ascent and descent engines for proper flight control.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-7. LGC Programs (cont)

GN&CS

Program No.	Title	Mission Phase	Primary Purpose
35	Transfer phase midcourse (TPM)	Prethrusting	<p>Enables astronaut to insert maneuver data into LGC, for these calculations</p> <p>Calculates and displays required parameters for LM execution of possible midcourse correction during transfer phase of LM active rendezvous</p> <p>Calculates and displays information to enable astronauts to prepare for entry into final rendezvous phase</p>
40	Descent propulsion section (DPS) thrust control	Thrusting	<p>Calculates preferred IMU orientation and preferred LM attitude for LM DPS thrust maneuver</p> <p>Automatically maneuvers LM to preferred attitude and controls LM through termination of thrusting maneuver</p>
41	Reaction Control Subsystem thrust control	Thrusting	<p>Same as for program No. 40, but for RCS thrust maneuver</p> <p>Provides rapid trimming of residual steering errors left at DPS or APS cutoff</p>
42	Ascent propulsion section (APS) thrust control	Thrusting	<p>Same as for program No. 40, but for thrust maneuvers using ascent engine</p>
47	Thrust monitor	Thrusting	<p>Monitors LM acceleration during thrust maneuver that is not controlled by PGNS and displays resulting change in inertial velocity</p>
51	IMU orientation determination	Alignment	<p>Determines inertial orientation of IMU stable member with respect to celestial coordinates, for guidance and navigation computations</p>
52	IMU realign	Alignment	<p>Aligns IMU stable member to specific inertial orientation, as selected by astronaut, from previously established inertial orientation</p>
72	CSM-CSI targeting	CSM Backup	<p>Calculates and displays required parameters associated with all coelliptic rendezvous flight plan maneuvers, for CSM execution of maneuvers</p>
73	CSM-CDH targeting	CSM Backup	<p>Calculates and displays required parameters associated with all coelliptic rendezvous flight plan maneuvers, except CSI, for CSM execution of maneuvers</p>
74	CSM-TPI targeting	CSM Backup	<p>Calculates and displays required initial conditions for active CSM transfer phase initiation maneuver</p>
75	CSM-TPM targeting	CSM Backup	<p>Calculates and displays required initial conditions for active CSM midcourse correction during transfer phase of rendezvous</p>

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-8. LGC Routines

Routine No.	Title	Purpose
00	Final automatic request terminate	Provides standard exit for programs, and option to select any program desired Provides general description of LGC activity following any verb 37 entry
2	IMU status check	Checks whether IMU is on and, if on, whether it is aligned to orientation known by LGC Establishes program alarm and stores alarm code if IMU is off or not aligned to orientation known by LGC
3	Digital autopilot (DAP) data load	Loads and verifies LGC DAP data, and DPS engine gimbal trim angles Enables astronauts to select desired DAP response Performs engine gimbal trim
4	RR/LR self-test	Provides suitable DSKY displays and LGC downlink information to support RR or LR self-tests
21	RR designate	Points RR antenna at CSM until automatic acquisition of CSM is accomplished by RR
22	RR data read	Processes automatic RR mark data to update LM or CSM state vector as defined by rendezvous navigation program (P20) or lunar surface navigation program (P22) Maintains LM +Z-axis aligned along LOS from LM to CSM within impulse limit cycle of DAP when program No. 20 is in process
23	RR manual acquisition	Acquires CSM by manual operation of RR while LM is in flight
24	RR search	Acquires CSM by search pattern when RR has failed to acquire CSM in automatic tracking mode Confirms that acquisition has not attained on side lobe
25	RR monitor	Monitors RR shaft and trunnion angles Commands shaft and trunnion angles to one of two reference positions Zeroes RR channels of CDU when RR is initially placed in automatic mode
30	Orbit parameter display	Provides astronaut with pertinent orbital parameters computed by LGC, to supplement orbital information provided by MSFN
31	Rendezvous parameter display	Displays, upon astronaut request, LGC-calculated range, range rate, and theta
32	Target ΔV	Notifies LGC that CSM has changed its orbital parameters by execution of thrusting maneuver Provides LGC with ΔV applied to CSM, to enable updating of CSM state vector

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-8. LGC Routines (cont)

GN&CS

Routine No.	Title	Purposes
33	LGC - CMC clock synchronization	Synchronizes LGC clock with CMC clock
36	Rendezvous out-of-plane display	Displays LGC-calculated rendezvous out-of-plane parameters (Y, \dot{Y} , PSI)
47	AGS initialization	Provides AEA with LM and CSM state vectors in LM IMU coordinates by means of LGC digital downlink Zeroes inertial CDU channel, LGC, and AEA gimbal angle counters simultaneously, to establish common zero reference for measurement of gimbal angles, which define LM attitude with respect to IMU stable member Establishes GET of AEA clock zero if AEA clock is zeroed during this routine
50	Coarse align	Coarse-aligns IMU to desired inertial orientation
51	In-flight fine alignment	Obtains necessary star, planet, or sun sightings and then fine-aligns IMU to desired inertial orientation while LM is in flight
52	Automatic optics positioning	Aims LOS of any AOT detent, or LOS of backup optical systems at specified celestial body by maneuvering LM
53	In-flight sighting mark	Performs required optical system sighting marks on celestial bodies for use by in-flight alignment programs
54	Sighting data display	Displays accuracy of pair of celestial body sightings
55	Gyro torquing	Calculates gyro torquing angles for fine alignment of inertial platform during in-flight alignment
60	Attitude maneuver	Maneuvers LM to attitude specified by program in process
61	Preferred tracking attitude	Performs automatic trim to preferred tracking attitude if maneuver required is less than specified function of RCS DAP deadband or, if not, routine notifies crew that attitude maneuver to preferred tracking attitude is required Computes preferred LM tracking attitude, which enables RR tracking of CSM, and CSM tracking of LM beacon
62	Crew-defined maneuver	Enables astronauts to specify final vehicle attitude for use by LGC-controlled attitude maneuver
63	Final rendezvous attitude	Calculates and displays final FDAI ball angles required to aim LM +Z-axis or +X-axis at CSM
77	LR spurious test	

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Mission LM 4

Basic Date 15 June 1968

Change Date 15 September 1968 Page 2.2-53

■ 2.2.3.3 Abort Guidance Section. (See figure 2.2-14.)

The AGS consists of an abort sensor assembly (ASA), abort electronics assembly (AEA), and a data entry and display assembly (DEDA). The ASA performs the same function as the IMU; it establishes an inertial reference frame. The AEA, a high speed general purpose digital computer is the central processing and computational device for the AGS. The DEDA is the input-output device for controlling the AEA.

Navigation for the LM is performed continuously by the AGS assuming that the motion is in a spherical gravitational field. Navigation is performed by the AGS through integration of the equations of motion and substitution of instantaneous LM velocity, for the variables. The AGS decodes the PGNS downlink data to establish LM and CSM position, velocity, and associated time computations. This information is used to initialize or update the AGS navigational computations upon command from the DEDA. The AGS solves the guidance problems of fine distinct guidance routines: orbit insertion, coelliptic sequence initiate, constant delta altitude, terminal phase initiate, and change in LM velocity (external ΔV).

The AGS provides steering commands for three steering submodes: attitude hold, guidance steering, and acquisition steering. The attitude hold submode maintains the vehicle attitude that exists when the submode is entered. In the guidance steering submodes the AEA generates attitude commands to orient the LM X-axis so that it lies along the direction of the thrust vector. In the acquisition steering submode, the AEA generates attitude commands to orient the LM Z-axis along the estimated line-of-sight (LOS) between the LM and CSM.

The AGS outputs an engine-on or engine-off command during all thrusting maneuvers. If the PGNS is in control, the command is a follow-up of the signal produced by the PGNS. If the AGS is in control, the engine-on command can be routed only after the appropriate switches are set and ullage has been performed. When proper velocity-to-be-gained are achieved, an engine-off command is issued.

The AGS uses RR angle information and accepts range and range rate information from the RR for updating LM navigation so that the LM Z-axis is toward the CSM, or for midcourse correction. These data are manually inserted into the AEA by the astronaut by using the DEDA.

The AGS automatically aligns the strapped-down inertial system of the ASA by computing the direction cosines that relate the LM body axes to the desired inertial coordinate system. It also provides in-flight gyro and accelerometer calibration to compensate for fixed non-g gyro drift, and telemetry data for MSFN through the IS.

■ 2.2.3.3.1 Abort Sensor Assembly. (See figure 2.2-14A.)

The ASA senses LM angular displacements and velocity increments and generates output pulses to update the AEA. The ASA consists of three strapped-down pendulous accelerometers, three strapped-down gyros and associated electronic circuitry. The ASA is mounted on the navigation base above the astronaut's heads, between the crew compartment and the thermal and micrometeoroid shield.

The ASA operates on a current pulse torquing principle to rebalance the output from its sensors. The three gyros sense angular rates about the LM orthogonal axes and the three accelerometers sense accelerations along the same axes. These motions are converted pulses proportional to the sensed motion. These pulses are routed to the AEA for further processing.

The ASA contains three floated, pulse-rebalanced, single-degree-of-freedom, rate-integrating gyroscopes and three pendulous reference accelerometers in a strapped-down configuration. These six sensors are housed in a beryllium block, which is mounted on the navigation base. The sensors are aligned with the three LM reference axes. The assembly also includes pulse torque servo-amplifiers (one associated with each sensor), temperature control amplifiers, and a power supply.

Current generated in the accelerometer produces torque that opposes the torque caused by the force of acceleration on a pendulous mass. The average current is proportional to the acceleration. Pulses generated in the accelerometers are used by the AEA in measuring velocity increments. The output of each gyro and accelerometer is applied to a pulse torque servoamplifier. The pulse torque

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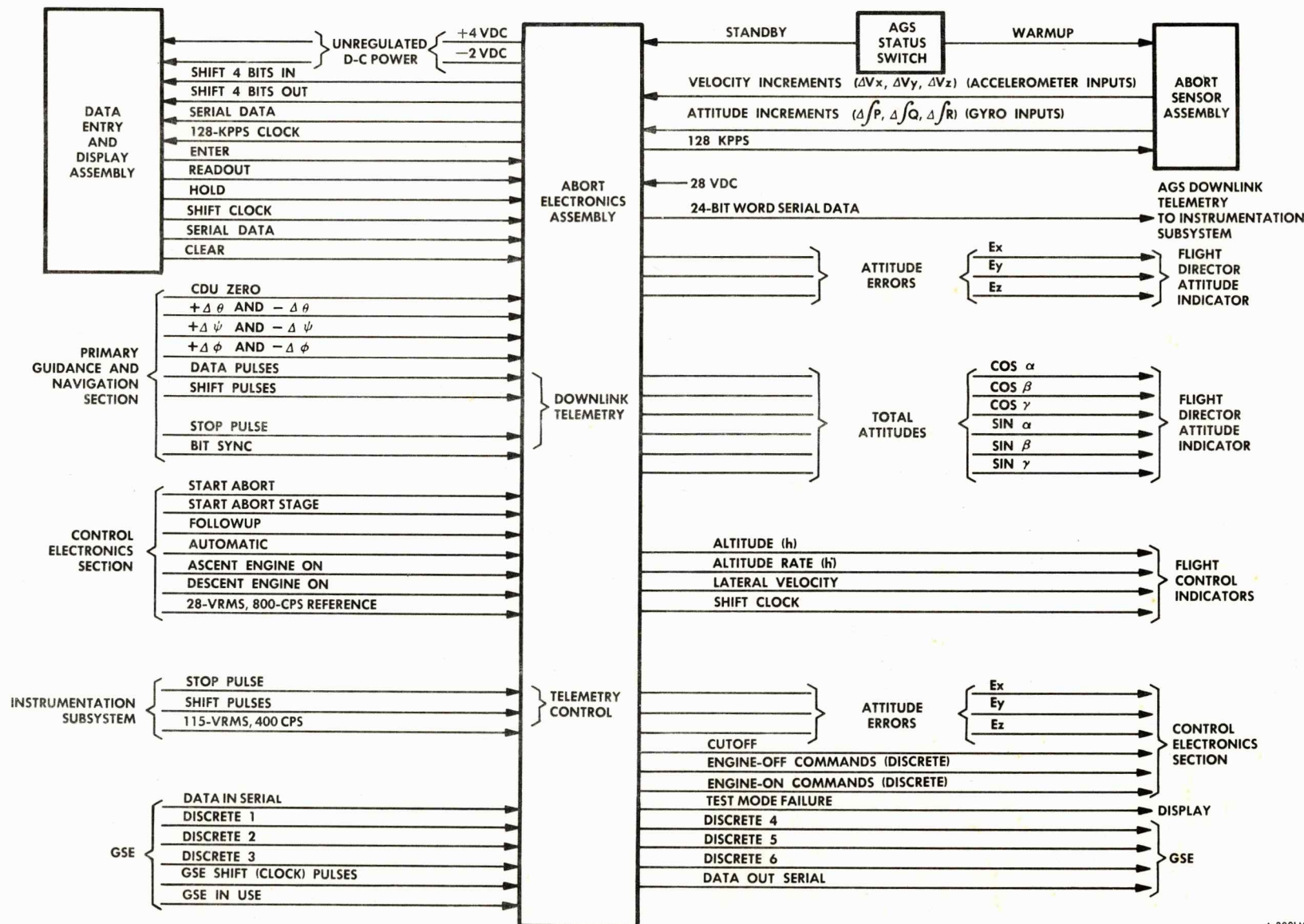
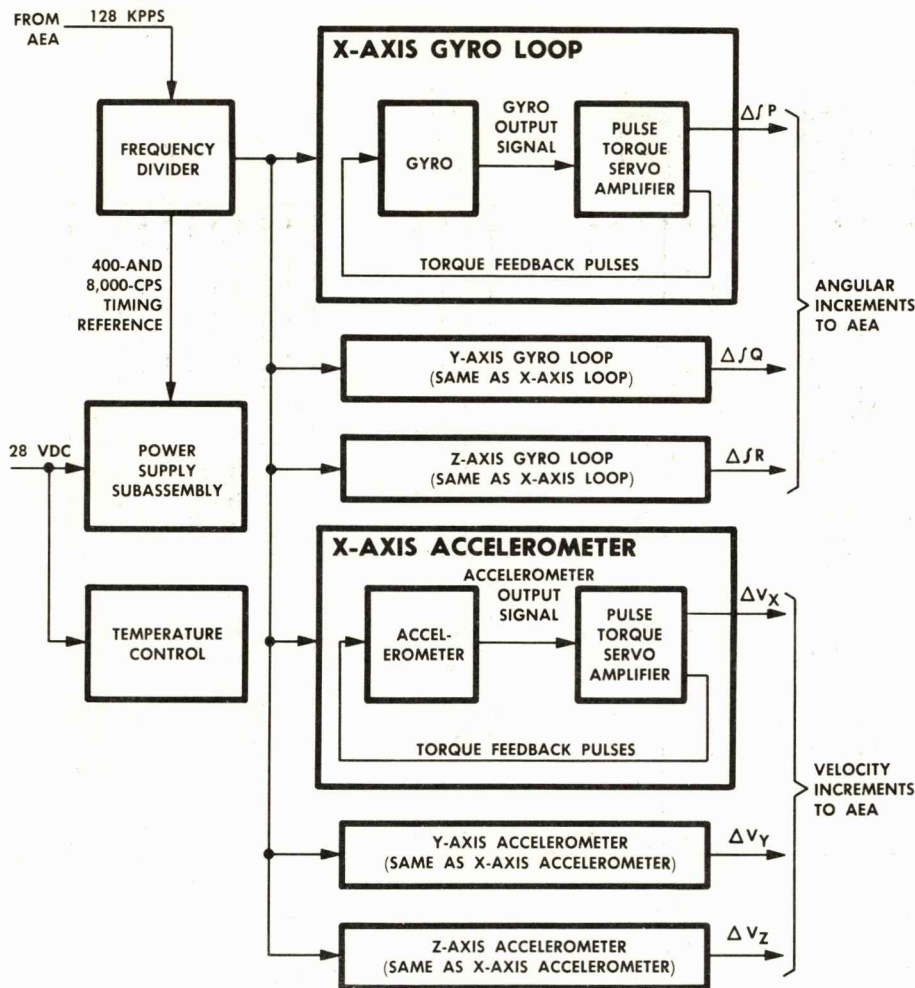


Figure 2.2-14. Abort Guidance Section - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



300LM4-141

Figure 2.2-14A. Abort Sensor Assembly - Simplified Block Diagram

servoamplifiers use quantized time-modulated feedback to torque-balance their associated gyro or accelerometer. The servoamplifiers consist of an amplifier, a quantizer bridge and driver, and a current regulator.

The strapped-down inertial guidance system has the advantage of substantial size and weight reduction over the more conventional gimballed inertial guidance system, but has the disadvantage of error buildup over sustained periods of operation. The calibration method uses the PGNS as a reference to determine the drift-compensation parameters for the ASA gyros. Calibration parameters are stored in the AEA and used to correct calculations based on the gyro inputs.

The accuracy of the ASA inertial data outputs is temperature-dependent. The ASA temperature control subassembly maintains the internal ASA temperature at +120° F, with external temperatures between -65° and +185° F. Two temperature control circuits are used, one each for fast warmup and fine temperature control. During fast warmup, ASA temperature can be raised from 0° F to approximately +116° F in 40 minutes. The ASA must warm up for at least 25 minutes (warmup mode - AGS STATUS switch on panel 6 set to STANDBY) before operation. The fast warmup circuitry consists of a thermal sensor, which detects the internal ASA temperature and provides a proportional d-c analog to heater driver electronics that maintain the required heater power constant. The constant heater power permits a uniform temperature rise within the ASA. The fine temperature control circuit controls the

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

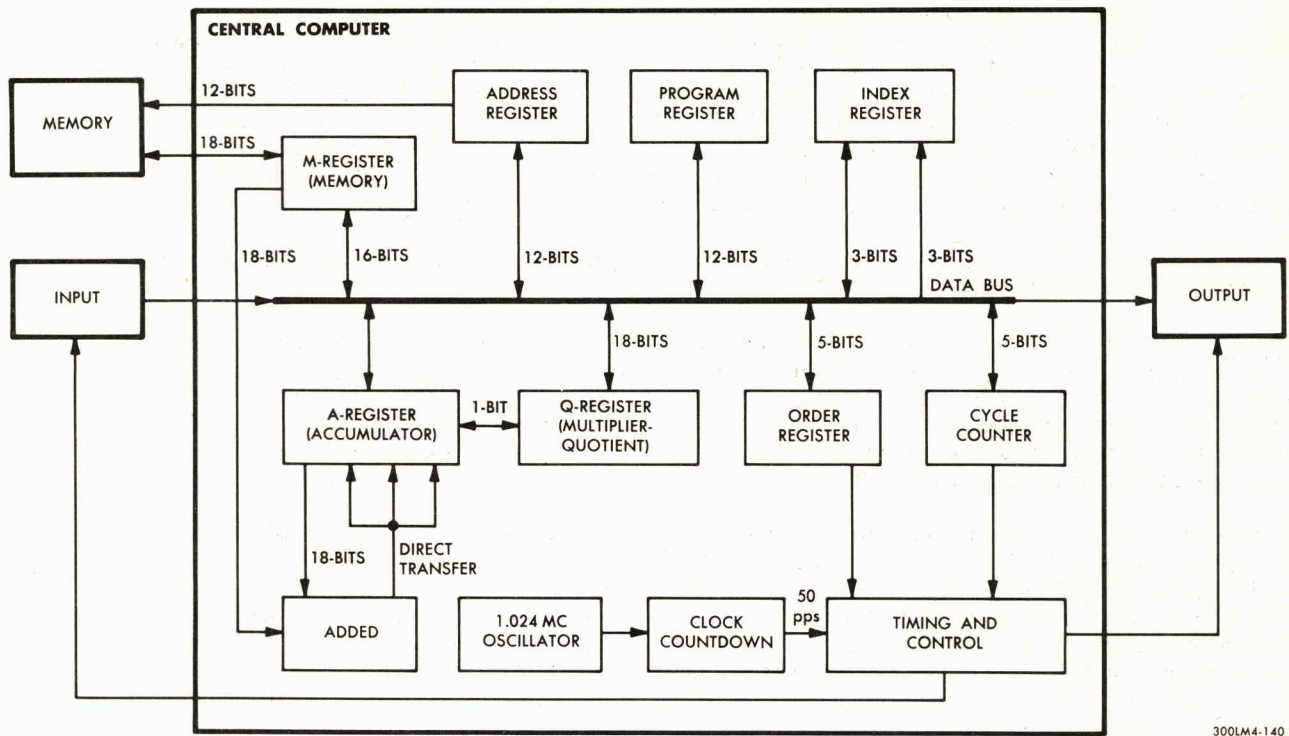


Figure 2.2-14B. Abort Electronics Assembly - Simplified Block Diagram

temperature after ASA temperature reaches $+116^{\circ}$ F. The fine temperature control circuit can raise the temperature 4° F and maintain this operating temperature within 0.2° F.

2.2.3.3.2 Abort Electronics Assembly. (See figure 2.2-14B.)

The AEA is a general-purpose, high-speed, 4,096-word digital computer that performs the basic strapped-down guidance system calculations and the abort guidance and navigation steering calculations. The computer uses a fractional two's complement, parallel arithmetic section and parallel data transfer. Instructions consist of 18 bits: a five-bit order code, an index bit, and a 12-bit operand address. Data words for the computer consist of a sign and 17 numeric bits (no parity). The AEA has three software computational sections: stabilization and alignment, navigation, and guidance.

The stabilization and alignment computational section computes stabilization and alignment on generation of mode signals by the DEDA. These mode signals (altitude hold, guidance steering, Z-axis steering, PGNS-to-AGS alignment, lunar align, gyro and accelerometer calibration, and body-axis align) determine the operation of the stabilization and alignment computational section in conjunction with the navigation and guidance computational sections. The body-referenced steering error signals and total attitude sine and cosine signals are used to control the FDAI. Direction cosine data are routed to the navigation computational section, where they are used in computing lateral velocity and inertial acceleration data.

The navigation computational section uses accelerometer inputs received from the ASA, via AEA input logic circuits, to calculate LM position and velocity in the inertial reference frame. The navigation computational section supplies total velocity, altitude, and altitude-rate data in earth-centered coordinates, and lateral velocity data in the LM reference frame, to the output logic circuits. Velocity data are routed to the DEDA, altitude-rate data are routed to the ALT RATE indicator (panel 1), and lateral velocity data are routed to the X-pointer indicator (panels 1 and 2). Velocity and position data are routed to the guidance computational section, for computing LM orbital parameters.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The guidance computational section provides trajectory computation and selection, steering computation, and midcourse-correction computation. This computational section receives data relating to the CSM state vector and the LM state vector from the LGC in other external source through the AGS input selector logic. Body-referenced steering errors are received from the stabilization and alignment computational section, for trajectory computation. The LM abort guidance problem consists of solving the equations of the selected guidance maneuver including steering, attitude, and engine control computations. Outputs of the guidance computational section, through the output select logic circuits, include engine on and off signals to the CES, and velocity to be gained (selectable by DEDA readout).

The AEA consists of a memory subassembly, central computer, an input-output subassembly, and a power subassembly.

Memory. The AEA memory is a coincident-current, parallel, random-access, ferrite-core memory with a capacity of 4,096 words (18 bits each). The cores are arranged in square planes; each core is subjected to X- and Y-coordinate selection. There is one plane for each bit in the word; the arrangement consists of 64 rows and 64 columns. The number of words in the memory is equal to the number of cores in a bit plane. An address to the computer is specified by an X-coordinate and a Y-coordinate; it refers to the core at that location in all 18-bit planes.

The memory core stack is divided into two sections: temporary memory (2,048 words) and permanent memory (2,048 words). The temporary memory stores replaceable instructions and data. Temporary results may be stored in this memory and may be updated as necessary. Checkout routines may be loaded during checkout and subsequently discarded. Each core in the temporary memory is threaded by an X-selection winding, a Y-selection winding, a sense winding, and an inhibit winding. The permanent memory stores instructions and constants that are not modified during operation of the permanent-memory section. In the permanent memory, the inhibit winding is omitted and the X-selection winding passes only through cores that represent zeros.

The cycle time of the memory is 5 microseconds. The addresses of the temporary storage locations are 0 through 3777₈; the addresses of the permanent storage locations, 4000₈ through 7777₈.

Central Computer. The central computer comprises an address register, a memory register (M register), an index register, a program counter, cycle counter, an operation code register, adder, accumulator (A-register), and a multiplier-quotient register (Q-register). The registers are interconnected by a parallel data bus. Central-computer operations are executed by appropriately timed transfer of information between these registers, between the memory and the M-register, and between the accumulator and input-output registers. The adder, M-register, Q-register, and accumulator form the basis for execution of arithmetic operations.

Address Register. The address register is a 12-bit static register that is loaded under AEA control, from the least significant 12 bits of the data bus. The address register holds the address of the memory location to which access is requested.

Memory Register. The memory register (M-register) is an 18-bit static register, which is loaded from the memory cores or the data bus. The register holds data that are to be transferred between the central computer and the memory, by the data bus. The data bus is a collection of gates through which data (up to 18 bits in parallel) may transfer from one register to another. Data transferred from the memory are held by the M-register as they are placed on the data bus. During this time, data transfer is a read-and-restore operation. Data transferred to the memory from the central computer are held by the M-register as they are written into the memory. The A-register (accumulator) holds the multiplicand during multiplication, the dividend during division, the addend during addition, and the minuend during subtraction. The A-register is loaded by clearing it, then adding the contents of the M-register to it.

Index Register. The index register is a three-bit ripple counter that is used for operand-address modification. When an indexed instruction is executed, the effective operand address is computed by a logical OR operation between the index register and the three least significant bits of the operand address. When a transfer and text index instruction is executed, the index register is decreased by one if the counter is greater than zero, and the next instruction is taken from the location specified by the address field of the transfer and text index instruction. If the index register is zero, the counter is not decreased, and the next instruction is taken in sequence. The index register is loaded under program control, from the least significant three bits of the data bus.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

Program Counter. The program counter is a 12-bit ripple counter, that is loaded from the least significant 12 bits of the data bus by execution of a transfer instruction. The program counter holds and generates instruction addresses in sequence.

Cycle Counter. The cycle counter is a five-bit counter that controls shift instructions and certain long orders. The counter is loaded from the six least significant bits of the M-register.

Operation Code Register. The operation code register is a five-bit static register that is loaded under software-control, from the five most significant bits of the data bus. The register holds the five-bit order code during its execution.

Adder. The arithmetic section is designed around an 18-bit, 3-microsecond parallel adder. Two 18-bit registers (the A-register and M-register) furnish the inputs to the adder. The sum generated by the adder is loaded into the accumulator. Shifting is implemented by displacing the sum generated by the adder one bit right or left when loading it into the accumulator.

Accumulator. The accumulator is an 18-bit static register (A-register). It communicates, in parallel, with the adder and the data bus, and serially with the Q-register for shifting operations such as multiplication, division, and double-length shifts. The accumulator holds the results of most arithmetic operations; it is used to communicate with the input or output registers.

Multiplier-Quotient Register. The multiplier-quotient register (Q-register) is an 18-bit two-way shift register that communicates, in parallel, with the data bus and serially with the accumulator. The Q-register holds the least significant half of the double-length product after multiplication and initially holds the least significant half of the double-length dividend for division. After division, it holds the unrounded quotient. For double-length shifting operations, the Q-register is logically attached to the low-order end of the A-register. After execution of a transfer and set Q-instruction, the Q-register holds a transfer instruction with an address field set to one greater than the location of the transfer and set Q-register instruction.

Timing. Timing is controlled by an eight-bit register and a three-bit register. These registers produce the timing signals required to control all operations of the central computer.

Input-Output Subassembly. There are four basic types of registers in the input-output subassembly: integrator, ripple counter, shift, and static. These registers operate independently of the central computer, except when they are accessed during execution of an input or output instruction. All transfers of data between the central computer and the input-output registers are in parallel. The PGNS Euler angles are accumulated in three 15-bit integrator registers. The integrator registers shift 15 bit positions at 512 kc upon receipt of each input pulse, serially adding or subtracting one from the previous count. The integrator registers are set to 0 by an external signal. ASA gyro and accelerometer pulses are accumulated in six 11-bit counters (in the AEA), which are set to 0 by the central computer. A four-bit DEDA register, an 18-bit input telemetry register, a 24-bit output telemetry register, and a 15-bit output register time-shared for altitude and altitude rate are all used as shift registers. The remaining registers (digital-to-analog converter registers and discrete input and output registers) are static registers. The discrete output register is modified by setting or resetting specific bits within the register with specific output instructions.

The AEA input signals and their characteristics are listed in table 2.2-9; the output signals and their characteristics, in table 2.2-10.

LMA790-3-L M 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-9. Abort Electronics Assembly - Input Signal Characteristics

Nomenclature	Routing	Signal Description	Remarks
AGS telemetry stop pulse	ISS to AEA	50-pps signal; indicates completion of telemetry shift for one digital word	
AGS telemetry shift pulses	ISS to AEA	51.2 kpps (burst of 24 pulses, 50 times per second); shifts telemetry data out of AEA.	
28-volt, 800-cps reference	CES to AEA	Sinusoidal excitation for digital-to-analog converters in AGS	± 0.5 vrms and $\pm 1\%$ frequency accuracy
Automatic	CES to AEA	Discrete used to put AEA in automatic submode	Definitions: 1 - Ground 0 - Open circuit
Abort	CES to AEA	Discrete indicating abort but that staging has not occurred	Definitions: 1 - Ground 0 - Open circuit
Follow-up	CES to AEA	Discrete used to put AEA in follow-up submode	Definitions: 1 - Ground 0 - Open circuit
Abort stage	CES to AEA	Discrete indicating abort and that staging is initiated	Definitions: 1 - Ground 0 - Open circuit
PGNS alignment signals: - $\Delta \theta$, + $\Delta \theta$, - $\Delta \psi$, + $\Delta \psi$, - $\Delta \theta$, + $\Delta \theta$,	PGNS to AEA	Digital pulse train (64 kpps maximum), denoting negative and positive angular increments of three PGNS gimbal angles	Gimbal angles are used during PGNS-to-AGS alignment, to define the inertial reference frame.
CDU zero	PGNS to AEA	51.2-kpps pulse train used to reset PGNS angle input registers	Definition: On - Application for 300 milliseconds (minimum)
PGNS telemetry downlink stop	PGNS to AEA	50-pps signal indicating termination of PGNS telemetry downlink shift operation	
PGNS telemetry bit-synchronizing pulse	PGNS to AEA	51.2-kpps pulse train used to shift in PGNS telemetry downlink data	Burst of 40 pulses with period of 20 milliseconds each
AGS initialization (downlink data)	PGNS to AEA	51.2-kpps signal comprising 40-bit words; used to initialize AGS navigation	Burst of 40 pulses with period of 20 milliseconds each
Descent engine on	Descent engine to AEA	Discrete indicating state of descent engine	Definitions: 1 - Ground, engine on 0 - Open circuit, engine off
Ascent engine on	Ascent engine to AEA	Discrete indicating state of ascent engine	Definitions: 1 - Ground, engine on 0 - Open circuit, engine off

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

Table 2.2-10. Abort Electronics Assembly - Output Signal Characteristics

Nomenclature	Routing	Signal Description	Remarks
Total attitude signals: sine α , cosine α , sine β , cosine β , sine γ , cosine γ	AEA to FDAI	0- to 15-vrms, 400-cps, sinusoidal signals for total attitude display	Scale factor: 30 milli-volts per step Computation rate: 25 times per second
Altitude signal (h)	AEA to displays	64-kpps signal with word frequency of > five times per second, used for altitude display	Resolution: 2.3 feet per bit (15-bit word)
Altitude rate signal (h)	AEA to displays	64-kpps signal with word frequency of > five times per second, used for altitude display	Resolution: least significant bit 0.5 foot per second (14 bits plus sign)
Lateral velocity (Vyb)	AEA to displays	D-C analog (-5 to +5 vdc), dynamic range -200 to +200 feet per second	Resolution: 25 mv/fps
Shift pulses for h and h	AEA to displays	64 kpps	
Attitude error signals E_X , E_Y , E_Z	AEA to CES and displays	0- to 4.5-vrms, 800-cps, sinusoidal, body-referenced, altitude error signals for steering and display	Resolution: 300 milli-volts per degree $\pm 5\%$ with -70° to $+78^\circ$ range

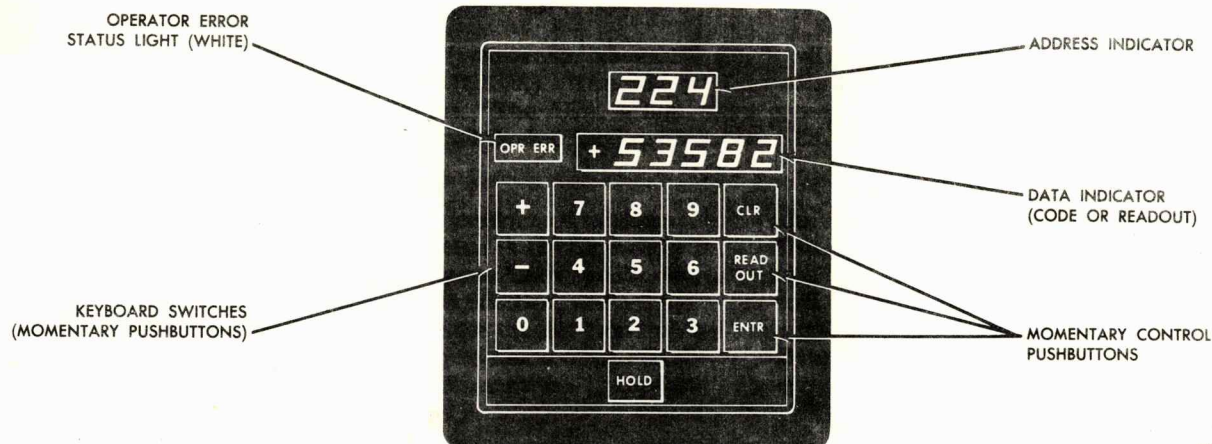
2.2.3.2.3 Data Entry and Display Assembly. (See figure 2.2-15.)

The DEDA is used to control the AGS modes of operation manually, to insert data manually into the AEA, and to command the contents of the accessible AEA memory to be displayed on numeric readouts. Only 452 of the 4,096 words stored in the AEA memory are DEDA accessible. The DEDA consists of a keyboard, logic circuits, electroluminescent displays, and driving circuits. The controls and indicators associated with the DEDA are listed in section 3.

Data input to the AEA, via the DEDA, consists of a nine-character code. The first three characters form the address entry, in octal form; the fourth is the sign (+ or -); and the remaining five characters represent the data, in decimal or octal form. DEDA inputs to the AEA are listed in tables 2.2-11 and 2.2-12. Before entering data, all nine characters must have been entered into the AEA. When requesting a readout, three numerical pushbuttons (0 through 9) are used. This three-digit code represents the address and is displayed on the upper portion of the readout. The selected information from the AEA memory is displayed on the lower portion of the readout. Table 2.2-14 contains the DEDA-accessible constants.

DEDA Operation. DEDA operation is initiated by pressing the CLR pushbutton. This enables data entry with the numerical pushbuttons (0 through 9). As each numerical pushbutton is pressed, its code is placed in a register for display in the order in which the pushbuttons are pressed. When the appropriate number of characters is entered, the ENTR or READOUT pushbutton is pressed.

The following paragraphs contain the operating procedures for the DEDA, using the front panel push-buttons and electroluminescent readouts. Before performing these procedures, the appropriate cabin switches must be set to the on position, the AC BUS B: AGS (panel 11) and STAB/CONT: AEA and ASA circuit breakers (panel 16) must be closed, and the AGS STATUS switch (panel 6) must be set to OPERATE.



300LM4-254

Figure 2.2-15. Data Entry and Display Assembly - Pictorial

Data Insertion. The sequence of operations for inserting data into the AEA via the DEDA is as follows:

- Press and release CLR pushbutton.

This operation clears the DEDA and blanks all lighted characters. The CLR pushbutton must always be pressed before any DEDA entry.

- Press and release three consecutive numerical pushbuttons.

This operation identifies the address (octal) of the AEA memory word location into which the data are to be inserted. The AEA will not process data with an octal address larger than 704.

- Press and release pushbutton that corresponds to sign (+ or -) of data to be inserted.
- Press and release desired five consecutive numerical pushbuttons that correspond to the numerical value of data to be inserted (decimal or octal, as appropriate for entry). Refer to tables 2.2-11 and 2.2-12 for DEDA inputs (addresses).
- Verify entry by comparing electroluminescent display with desired address and desired numerical value.
- Press and release ENTR pushbutton.

This operation enters the word into the AEA and blanks the address and data displays.

Data Readout. The sequence of operations for reading the contents of an AEA memory cell is as follows:

- Press and release CLR pushbutton.

This operation initializes the DEDA and blanks all lighted characters. The CLR pushbutton must always be pressed before any readout.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

- Press and release three consecutive numerical pushbuttons.

This operation identifies the address of the AEA memory core location from which data are to be extracted. The address is in octal form; it must not exceed the number 704. If a number larger than 704 is entered into the address, the DEDA readout will be extinguished when the READOUT pushbutton is pressed. Refer to table 2.2-13 and 2.2-14 for data that are DEDA accessible from the AEA.

- Verify entry by comparing electroluminescent address with desired address.
- Press and release READOUT pushbutton.

After the READOUT pushbutton has been pressed, the displayed quantity and its address are updated twice per second. The astronaut can hold a value on the display by pressing the HOLD pushbutton. This value is held on the display until the READOUT pushbutton is pressed (at which time the displayed quantity is updated twice per second) or until the CLR pushbutton is pressed (at which time the DEDA is initialized for the next command and the displays are blanked).

Decimal Point Location. There is no decimal point indication on the DEDA. To use the AGS correctly, the astronauts must have knowledge of the quantization of the decimal quantity being entered or read out. If a position variable has a quantization of 100 feet, then 42,300 feet is entered or displayed as +00423.

Operator Error. The OPR ERR light on the DEDA display panel goes on if the operator makes any of the following errors:

- Nine pushbuttons are not pressed and the ENTR pushbutton is pressed.
- Three pushbuttons are not pressed and the READOUT pushbutton is pressed.
- READOUT pushbutton is pressed after more than three digits are entered.
- More than nine pushbuttons have been pressed since the last pressing of the CLR pushbutton.
- The 8 or 9 pushbutton is pressed while loading the address portion of the word.
- The + or - pushbutton is not pressed in the fourth position of the nine-digit word.
- The + or - pushbutton is pressed in other than the fourth position.
- A quantity is being read out and the ENTR pushbutton is pressed.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The following operator errors do not cause the OPR ERR light to go on.

- If a number greater than 704 is entered into the address portion of the display along with other numbers, the DEDA exhibits an odd display when the ENTR pushbutton is pressed. The numbers shift and several entry slots are blanked.
- If a number greater than 704 is entered into the address portion of the DEDA and the READOUT pushbutton pressed, the DEDA is blanked.
- If an octal quantity is entered with a digit greater than 7 or if a number greater than its allowable range is entered via the DEDA, a computer-processing error results. If the address is read out, the display is different than what was entered.

Address values greater than 704, used for program instructions cannot be entered into the DEDA. If such entries are attempted, the symptoms described in the preceding paragraph occur. Normal operation is restored by pressing the CLR pushbutton.

Table 2.2-11. Selector Logic List

Address	Entry Code	Definition	Remarks
400	+00000	Attitude hold submode	Program automatically sets attitude hold function upon AGS engine cutoff, completion of PGNS-to-AGS alignment, or completion of calibration. If AGS is in control, LM maintains inertial attitude it had when this mode was entered.
400	+10000	Guidance steering submode	Program automatically sets guidance steering functions upon receipt of abort stage signal. If AGS is in control attitude error signals orient LM to desired thrust direction.
400	+20000	Z-body axis steering submode	If AGS is in control, attitude error signals drive LM +Z-axis in estimated direction of CSM.
400	+30000	PGNS to AGS alignment (IMU alignment) submode	AGS is in this mode for 2 seconds, during which time AGS stable member is aligned to IMU stable member. Upon completion, AEA returns to attitude hold mode.
400	+40000	Lunar alignment submode	
400	+50000	Body-axis alignment submode	
400	+60000	In-flight calibration submode	AGS selector is in gyro- and accelerometer-calibrate mode. Gyro calibration time is 300 seconds and accelerometer time is 30 seconds. Upon completion, AEA returns to attitude hold mode.
400	+70000	Accelerometer calibration only	Completion of calibration requires 30 seconds. AEA returns to attitude hold mode after 300 seconds.
407	+10000	Freezes external ΔV reference into inertial reference frame	Program automatically selects this function upon detection of ullage. This entry freezes external ΔV -to-be-gained vector of in inertial space.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2, 2-11. Selector Logic List (cont)

GN&CS

Address	Entry Code	Definition	Remarks
410	+00000	Orbit insertion maneuver	This entry guides LM to prescribed trajectory.
410	+10000	Coelliptic sequence initiate maneuver	This maneuver creates correct phasing between LM and CSM such that, following CDH, desired LOS between LM and CSM is achieved at nominal TPI time.
410	+20000	Constant Δh maneuver	This maneuver places LM in trajectory such that Δh between LM and CSM is a constant.
410	+30000	Direct transfer time increment $T\Delta$ constant (TPI search)	This entry is used to determine whether TPI should occur.
410	+40000	Direct transfer fixed time of ignition (TPI execute)	This entry is used to perform any direct transfer maneuvers.
410	+50000	External ΔV maneuver	This maneuver is used to perform maneuvers based upon externally supplied velocity-to-be-gained components.
411	+10000	RCS-DPS/APS selector	+00000 X-body axis +10000
412	+00000	Reinitiate in-flight AGS self test	If abort stage occurs during burn, it is not necessary to enter this address, because appropriate action is taken by AEA.
413	+10000	Reset lunar surface flag	Content of address 413 is only reset by AEA power turn-on sequence.
414	+00000	Navigation initialization complete	Following completion of any navigation initialization, program automatically establishes selector logic to indicate completion of navigation.
414	+10000	Initialize LM and CSM via PGNS downlink	
414	+20000	Initialize LM via DEDA.	This entry commands LM navigation initialization by DEDA. Ephemeris data are entered before this entry is made.
414	+30000	Initialize CSM via DEDA.	Same as +20000, except for CSM.
415	+00000	Program reset. Normal value of stored Z-axis direction cosines command	Reset program to zero within 2 seconds by radar filter equations.
415	+10000	Store Z-axis direction cosines.	
416	+00000	Select CDH at first crossing of LM orbit line of apsides	This entry causes CSI calculations, to compute maneuver with CDH maneuver occurring at first crossing of LM orbit line of apsides.
416	+10000	Select CDH at second crossing of LM orbit line of apsides	

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-11. Selector Logic List (cont)

Address	Entry Code	Definition	Remarks
416	+20000	Select CDH at third crossing of LM orbit line of apsides	
417	+00000	Select CDH at apside crossing	
417	+10000	Select CDH 180°, 360°, or 540° after CSI per address 416	
507	+00000	Normal position	
507	+10000	Orient +Z-axis to desired thrust axis	
623	+00000 +10000	Normal position Crew select guidance yaw steering vector	

Table 2.2-12. Data Entry and Display Assembly - Input List

Address	Definition	Range	Quantization	Remarks
047	Sine of landing azimuth angle			
053	Cosine of landing azimuth angle			
231	Launch site radial distance from center of earth	20,926,000 feet	10^3	
232	Orbit insertion targeted injection altitude	0 to 33,554,000 feet	10^3	
233	Vertical pitch steering altitude threshold	0 to 33,554,000 feet	10^3	
240	LM position, X-component	$\pm 33,554,000$ feet	10^3	
241	LM position, Y-component	$\pm 33,554,000$ feet	10^3	
242	LM position, Z-component	$\pm 33,554,000$ feet	10^3	
244	CSM position, X-component	$\pm 33,554,000$ feet	10^3	
245	CSM position, Y-component	$\pm 33,554,000$ feet	10^3	
246	CSM position, Z-component	$\pm 33,554,000$ feet	10^3	

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-12. Data Entry and Display Assembly - Input List (cont)

Address	Definition	Range	Quantization	Remarks
254	Epoch time (AGS-referenced) of LM ephemeris data	0 to 4,369 minutes	0.1	Used in LM navigation initialization
260	LM velocity, X-component	±32,767 fps	1	
261	LM velocity, Y-component	±32,767 fps	1	
262	LM velocity, Z-component	±32,767 fps	1	
264	CSM velocity, X-component	±32,767 fps	1	
265	CSM velocity, Y-component	±32,767 fps	1	
266	CSM velocity, Z-component	±32,767 fps	1	
272	Epoch time (AGS-referenced) of CSM ephemeris data	0 to 4,369 minutes	0.1	
275	AGS computer time of CSI maneuver	0 to 4,369 minutes	0.1	tigA
276	Absolute time of CDH maneuver	0 to 4,369 minutes	0.1	tigB
277	Desired TPI time for CSI computation	0 to 4,369 minutes	0.1	This time is input in AEA time.
310	Rendezvous offset time	136 minutes	0.1	
311	Time from TPI to rendezvous	0 to 136 minutes	0.1	
313	Time from present to next burn	0 to 136 minutes	0.01	T _A
314	Time of mode before rendezvous target		0.1	
316	Radar range	0 to 400 nm	0.01	
373	AGS computer time of TPI maneuver	0 to 4,369 minutes	0.1	tigC (valid only in TPI)
	CSI to CDH coast time	0 to 4,369 minutes	0.1	T _{A0}
377	AGS computer time	0 to 4,369 minutes	0.1	
450	Component of external ΔV in horizontal direction parallel to CSM orbit plane	±32,767 fps	1 fps	Plus display indicates VG in posigrade direction.
451	Component of external ΔV in horizontal direction perpendicular to CSM orbit plane	±32,767 fps	1 fps	Plus display indicates VG opposite to LM angular momentum vector.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-12. Data Entry and Display Assembly - Input List (cont)

Address	Definition	Range	Quantization	Remarks
452	Component of external ΔV in radial direction	$\pm 32,767$ fps	1 fps	Plus display indicates VG toward earth
464	Vertical pitch steering altitude rate threshold		1 fps	
465	Orbit insertion targeted injection radius rate		1 fps	
466	Orbit insertion targeted injection horizontal velocity		1 fps	
514	Guidance steering input vector (Wbx)		Octal	
515	Guidance steering input vector (Vby)		Octal	
516	Guidance steering input vector (Wbz)		Octal	
540	X-axis accelerometer bias compensation	N/A	N/A	
541	Y-axis accelerometer bias compensation	N/A	N/A	
542	Z-axis accelerometer bias compensation	N/A	N/A	
544	X-gyro drift compensation constant	$+10^\circ$ to $-10^\circ/\text{hr}$	$0.01^\circ/\text{hr}$	
545	Y-gyro drift compensation constant	$+10^\circ$ to $-10^\circ/\text{hr}$	$0.01^\circ/\text{hr}$	
546	Z-gyro drift compensation constant	$+10^\circ$ to $-10^\circ/\text{hr}$	$0.01^\circ/\text{hr}$	
547	Lunar align azimuth correction			Used during lunar missions
605	Tangent of desired LOS angle at TPI (CSI)		Octal	

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-13. Data Entry and Display Assembly - Output List

GN&CS

Address	Description	Range	Quantization	Symbol
211	Out-of-plane position	feet	10^3	y
263	Out-of-plane steering	fps	1	V_{py}
270	Present LM out-of-plane velocity	fps	1	V_{yo}
274	Predicted time from CDH to TPI	4,369 min	0.1	T_c
275	AGS time of CSI maneuver	0 to 4,369 min	0.1	t_{iga}
303	Time-to-go until rendezvous (valid in TRI only)	minutes	0.01	T_p
304	Angle between LM Z-axis and local horizon	-90° to +90°	0.1	ξ
305	CSI iteration error in LM/CSM relative central angle (Valid in CSI and CDH)	0 to 360°	0.01	C_o
306	Time-to-go until rendezvous (valid for TPI only)	0 to 136 minutes	0.01	T_r
311	Time from TPI to rendezvous	0 to 136 minutes	0.01	
312	Predicted time to LM orbit perifocus	0 to 136 minutes	0.01	T_{perg}
313	Time from present to CSI, CDH or TPI	0 to 136 minutes	0.01	T_{Δ}
315	Predicted altitude of LM apofocus			
317	Range from LM to CSM	0 to 1380 nm	0.1	R
337	LM altitude		0.1	h
340	X-component of LM position	+33,554,000 to -33,554,000 feet	10^3	r_x
341	Y-component of LM position	+33,554,000 to -33,554,000 feet	10^3	r_y
342	Z-component of LM position	+33,554,000 to -33,554,000 feet	10^3	r_z
344	X-component of CSM position	+33,554,000 to -33,554,000 feet	10^3	r_{cx}
345	Y-component of CSM position	+33,554,000 to -33,554,000 feet	10^3	r_{cy}
346	Z-component of CSM position	+33,554,000 to -33,554,000 feet	10^3	r_{cz}
347	Predicted orbit insertion burnout radius	0 to 33,554,000 feet	10^3	r_f

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-13. Data Entry and Display Assembly - Output List (cont)

Address	Description	Range	Quantization	Symbol
360	X-component of LM velocity	+32,767 to -32,767 fps	1	V_x
361	Y-component of LM velocity	+32,767 to -32,767 fps	1	V_y
362	Z-component of LM velocity	+32,767 to -32,767 fps	1	V_z
364	X-component of CSM velocity	+32,767 to -32,767 fps	1	V_{cx}
365	Y-component of CSM velocity	+32,767 to -32,767 fps	1	V_{cy}
366	Z-component of CSM velocity	+32,767 to -32,767 fps	1	V_{cz}
367	LM altitude rate	+32,767 to -32,767 fps	1	r
370	Magnitude of LM velocity-to-be-gained	0 to 32,767 fps	1	V_G
371	CDH maneuver ΔV (valid in CSI)	0 to 32,767 fps	1	V_{Po}
371	Direct transfer plus braking impulse ΔV	0 to 32,767 fps	1	V_T
373	Time from CSI to CDH	0 to 4369 minutes	0.1	T_{AO}
377	AEA time	0 to 4,369 minutes	0.1	t
402	Perigee altitude of predicted LM transfer trajectory (TPI only)	+1379.7 to -1379.7 nm	0.1	
402	Differential altitude at time of predicted CDH	+1379.7 to -1379.7 nm	0.1	Δr
403	Perigee altitude of LM trajectory	+1379.7 to -1379.7 nm	0.1	q
423	Desired final value of altitude rate	+32,767 to -32,767 fps	1	r_f
433	Magnitude of LM velocity	+32,767 to -32,767 fps	1	V
440	Range rate between LM and CSM (negative sign indicates closing rate)	+32,767 to -32,767 fps	1	\dot{R}
441	LM velocity normal to CSM orbit plane	+32,767 to -32,767 fps	1	V_{ys}

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-13. Data Entry and Display Assembly - Output List (cont)

GN&CS

Address	Description	Range	Quantization	Symbol
456	ΔV for CSI maneuver		1	V_o
463	Predicted radial velocity at CSI time		1	r_5
470	ΔV expended minus descent stage capability	+32,767 to -32,767 fps	1	V_x
500	Velocity-to-be-gained in X-direction	+32,767 to -32,767 fps	1	ΔV_{gx}
501	Velocity-to-be-gained in Y-direction	+32,767 to -32,767 fps	1	ΔV_{gy}
502	Velocity-to-be-gained in Z-direction	+32,767 to -32,767 fps	1	ΔV_{gz}
574	Descent section staging flag			
604	Lunar surface flag			
611	Predicted LM radial velocity	+32,767 to -32,767 fps	1	
612	Staging sequence counter			

Table 2.2-14. Data Entry and Display Assembly - Accessible Constants

Address	Description	Address	Description
033	Rendezvous angle sine	070	Attitude errors
034	Horizontal unit vector in transfer orbit plane	104	LM position remainders
040	Unit vector perpendicular to transfer orbit plane	107	PGNS (counts)
044	Radar null direction cosines	110	Predicted change in integrated gravity
047	Sine of azimuth angle	113	PGNS Ψ (counts)
050	Accumulated thrust velocity vector	114	Gravity time major cycle time
050	Predicted LM velocity vector at CDH burn	117	PGNS ϕ (counts)
053	Cosine of azimuth angle	120	Resolved sensed ΔV 's along inertial axes
054	Unit vector normal to CSM orbit	123	Sine of FDAI
060	Normalized LM position vector	124	Unit vector input for commanding yaw when selector 00 equals 1 (S00=1)
067	Thrust acceleration	127	Cosine of FDAI

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14. Data Entry and Display Assembly - Accessible Constants (cont)

Address	Description	Address	Description
130	XB direction cosines	241	LM ephemeris position (Y-component)
134	ZB direction cosines	242	LM ephemeris position (Z-component)
140	YB direction cosines	244	CSM ephemeris position (X-component)
160	XD direction cosines	245	CSM ephemeris position (Y-component)
163	LM semimajor axis after CDH trail	246	CSM ephemeris position (Z-component)
164	ZD direction cosines	254	LM epoch
167	CSM position magnitude	260	LM initial velocity (X-component)
170	Semilatus rectum of LM transfer orbit	261	LM initial velocity (Y-component)
170	Trials horizontal velocity increment	262	LM initial velocity (Z-component)
171	Transfer orbit semimajor axis	263	V _G component for out-of-CSM-phase steering
174	LM predicted position vector at CSI, CDH, or TPI burn time	264	CSM initial velocity (X-component)
200	CSM epoch position vector	265	CSM initial velocity (Y-component)
203	Predicted CSM position magnitude	266	CSM initial velocity (Z-component)
204	Position vector input to orbit parameter subroutine	267	Velocity-to-be-gained
207	Predicted position magnitude	270	LM out-of-CSM-plane velocity magnitude
210	LM present inertial position magnitude	272	CSM epoch
211	LM out-of-plane position	274	CDH to TPI time
216	QL set on overflow	275	Absolute CSI maneuver time
217	Initial P perturbation	276	Time of CDH
223	Nominal burnout altitude	277	ABS time of most recent radar
224	LM predicted inertial position vector at CDH burn	300	Desired TPI time for CSI computation
227	LM predicated inert POS MAG	304	Z axis/local horizontal angle
230	Δp limiter	305	Best cost saved for DEDA
231	Nominal lunar radius	306	Computed LOS
232	Orbit insertion altitude	306	LM-CSM central angle at CDH
233	Vertical pitch steering altitude threshold	310	Target offset time
234	Calculated range at time of radar range measurement	311	Desired transfer time
240	LM ephemeris position (X-component)	313	Time from present to CSI, CDH or TPI
		316	Radar range
		317	Computed range
		337	Altitude

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14. Data Entry and Display Assembly - Accessible Constants (cont)

GN&CS

Address	Description	Address	Description
340	LM position vector	427	Present LM horizontal velocity
343	Present LM perigee	430	Ellipse predictor velocity vector output
344	CSM present position vector	433	Present LM velocity
347	Predicted burnout altitude	435	Predicted time. TPI to rendezvous time since CSM epoch
360	Present LM inertial velocity vector	440	Estimated range rate between CSM and LM steering (V_y)
364	Present CSM inertial velocity vector	445	CSM period
367	Altitude rate	446	Horizontal velocity after CSI
370	Velocity-to-be-gained	447	Partial derivative protector
371	Predicted CDH ΔV (CSI coast only)	453	P-ITER convergence check
371	Total velocity required for direct transfer	454	V_G threshold
373	CSI to CDH coast time	455	Lunar/earth altitude rate scaling selector
373	Absolute TPI time	456	D-transfer or 90° mode selector for TPI
377	Absolute time	457	Horizontal velocity search increment
400	AGS function selector	463	Predicted radial velocity at CSI time
401	Discrete word one position	470	Accumulated total of sensed velocity increments updated every 2 seconds
402	LM-CSM at predicted CDH burn time	473	Descent stage ΔV capability
402	LM transfer orbit perigee altitude	474	Components of total velocity-to-be-gained during burn
403	LM present perigee altitude	500	Increments of total velocity-to-be-gained during burn
407	Reference frame selector for external ΔV	503	Range rate input (radar)
410	Guidance mode selector	504	Vertical pitch steering altitude rate threshold
411	RCS-DPS/APS selector	505	RFDOT lower limit
412	In-flight self-test status indicator	506	Orbit insertion horizontal velocity
413	Store/no-store lunar azimuth selector	507	Equal DELVH, corresponds to best cost
414	Navigation initialize	513	Rendezvous angle cosine
415	Radar gimbal null	514	X-axis DV sum for accelerometer calibration
416	CDH apsidal selector	515	Y-axis DV sum for accelerometer calibration
417	CSI elliptical/circular orbit logic	516	Z-axis DV sum for accelerometer calibration
420	CSM epoch velocity vector		
423	Desired altitude rate		
424	Velocity vector input to orbit parameter subroutine		

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14. Data Entry and Display Assembly - Accessible Constants (cont)

Address	Description	Address	Description
517	Gyro calibration counter	552	Compensation constant for attitude-rate scale factor error of Z-gyro
520	CSM epoch MS	553	High (+) low (-) angular rate scaling
521	LM epoch MS	554	Upper limit on \dot{V}_d
522	Check constant for absolute time input	555	Upper limit on \ddot{Y}_d
523	Check constant for selector S15 input	556	Lower limit on \ddot{r}_d
524	CSM epoch LS	557	Lower limit on \ddot{Y}_d
525	LM epoch LS	560	Position error variance
526	Set value of V_T	561	Position and velocity error covariance
527	Upper limit on r_f	562	Downlink word counter
530	X-axis alignment error signals	563	Downlink initialize flag (instrumentation subsystem enable)
531	Y-axis alignment error signals	564	Lower limit on Δ_6
532	Z-axis alignment error signals	566	Cam angle of engine in pitch plane
533	Discrete word one complement	567	Display flat (IS navigation update cycle)
534	Scale factor constant for X-axis accelerometer	570	Downlink ID1 (received if negative)
535	Scale factor constant for Y-axis accelerometer	571	Downlink input (complete if negative)
536	Compensation and scale factor constant for Z-axis accelerometer	572	Memory test (test if negative)
537	Compensation constant for X-gyro spin axis mass unbalanced drift	573	20-millisecond branch control
540	Compensation constant for X-axis accelerometer bias	574	Staged flag
541	Compensation constant for Y-axis accelerometer bias	575	Attitude hold flag
542	Compensation constant for Z-axis accelerometer bias	576	Calibrate I. C. flag
544	X-gyro drift compensation constant	577	Steering flag
545	Y-gyro drift compensation constant	600	Initialize CSI flag
546	Z-gyro drift compensation constant	601	Radar data flag
547	Lunar align correction	602	Cant angle of engine in yaw plane
550	Compensation constant for attitude-rate scale factor error of X-gyro	603	Logic flag for engine control
551	Compensation constant for attitude-rate scale factor error of Y-gyro	604	Lunar surfacr flat (IS lunar surface)
		606	Lunar align stop criterion
		607	DEDA data received flag
		611	Predicted LM radial velocity

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14. Data Entry and Display Assembly - Accessible Constants (cont)

GN&CS

Address	Description	Address	Description
612	Staging counter	647	Velocity-to-be-gained (V_G) threshold
613	Sign for V_1 steering	651	Initial position error variance
614	Ullage counter	652	Initial velocity error variance
615	Minor cycle counter	653	Radar error model constant
616	Ullage counter limit	654	Factor in T_B computation
617	Gyro calibration duration	655	Factor in T_B computation
620	Number of p-iterations	657	Ascent engine cutoff compensation
621	Accelerometer calibration time	660	Lower limit on thrust acceleration (a_T)
622	Staging times delay	661	Ullage threshold
623	CSM mean orbital rate	665	Maximum V_{YB} displayable
624	Constant in altitude and altitude-rate 200-millisecond readout	666	Scale factor for attitude error output
625	FDAI computation singularity threshold	667	Cycle counts 0-to-seconds factor
626	X-axis alignment gain constant	670	One second plus DEDA time bias
627	Lunar alignment constant	671	Downlink code
630	Lunar alignment constant	674	$\Delta\text{-Mu } T \text{ (ft}^3/\text{sec}^2)$
631	Lunar alignment stop criterion	675	$\Delta T/2$
632	Calibration gain constant	676	Conversion factor: $.01^\circ/\text{hr}$ to radian/20 ms scaled at -13
633	Calibration gain constant	677	Conversion factor: $\frac{100}{1000}$ feet to feet scaled at $\frac{23}{25}$
634	Accelerometer bias threshold	700	Conversion factor: 0.1 minute to second scaled at 18
635	Accelerometer calibration gain constant	701	Conversion factor: $\frac{.1}{1}$ minute to second scaled at $\frac{13}{15}$
636	Gravitational constant	702	Conversion factor: 0.01° to radians scaled at 3
637	Reciprocal of gravitational constant ($1/2K_1$)	703	Conversion factor: 0.1 minute to feet scaled at $\frac{23}{25}$
641	Filter velocity uncertainty term	704	Conversion factor: 0.01 minute to second scaled at 13.
642	Factor in r_f^* computation		
643	Coeffieicnt in evaluation of CSI cost		
644	Factor to decrease Δ_6		
645	Factor to increase Δ_6		
646	Constant error term in radar filter		

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.2.3.3.4 Abort Guidance Section - Self-Test Routine.

The self-test routine continuously checks that the computer logic and memory are functioning properly. DEDA address 412 is used, in conjunction with the test, to indicate test results to the astronauts and to allow resetting of error indicators. If a readout of address 412 on the DEDA is performed, the following outputs are possible:

- +00000 - Test not completed
- +10000 - Test successfully completed
- +30000 - Logic test failure
- +40000 - Memory test failure
- +70000 - Logic and memory test failure.

An additional output (+00000) permits the astronaut to reinitiate testing and reset the AGS warning light after failure is detected. When a failure is encountered during the test, the output in memory location 412 is displayed and the AGS warning light goes on; future tests are inhibited until testing is reinitiated. The error indicator is reset or testing reinitiated by entering +00000 in address 412. The astronauts need not monitor DEDA address 412 periodically, because the AGS warning light (panel 1) provides a visual indication of an error. However, as part of the AGS turn-on procedure, the DEDA check should be performed as a partial check of AGS warning light and computer operation.

2.2.3.3.5 Abort Guidance Section - Initialization Function.

Initialization comprises two functions: navigation initialization, and time initialization.

Navigation Initialization. The AGS accepts initialization inputs from the PGNS downlink (automatic) or from the DEDA (manual). Data are accepted when an initialized instruction has been entered through the DEDA during an unpowered flight phase of the mission. Data shall not be inserted during LM powered flight. Initialization is referenced to the AGS absolute time base and the AGS inertial reference frame at the time of initialization. Upon completion of initialization, an initialization-complete indication is generated and may be displayed via the DEDA upon request.

Ephemeris data from the LM and CSM are inputs for initialization. These inputs are initialized automatically via the PGNS downlink by entering 414, +10000, and ENTR on the DEDA. Completion of automatic initialization is indicated by the DEDA address and data indicators being set back automatically to 414 +00000. LM ephemeris data are initialized manually, using the DEDA.

The PGNS downlink data provide LM and CSM position, velocity, and epoch time, which are used to initialize LM navigation and CSM orbit parameters (state vectors). The data are provided in the AGS inertial coordinate frame. The LM and CSM epoch times are in the same time base as the AGS absolute time. The sequence in which the PGNS downlink data are used to initialize the AEA is given in table 2.2-14A.

Table 2.2-14A. PGNS Downlink Data Update Sequence

Initialization Sequence	Description	DEDA Address
1	Identification word	414
2	LM X-position, AGS coordinates	240
3	LM Y-position, AGS coordinates	241

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14A. PGNS Downlink Data Update Sequence (cont)

GN&CS

Initialization Sequence	Description	DEDA Address
4	LM Z-position, AGS coordinates	242
5	LM epoch time (most significant)	254
6	LM X-velocity, AGS coordinates	260
7	LM Y-velocity, AGS coordinates	261
8	LM Z-velocity, AGS coordinates	262
9	LM epoch time (least significant)	254
10	CSM X-position, AGS coordinates	244
11	CSM Y-position, AGS coordinates	245
12	CSM Z-position, AGS coordinates	246
13	CSM epoch time (most significant)	272
14	CSM X-velocity, AGS coordinates	264
15	CSM Y-velocity, AGS coordinates	265
16	CSM Z-velocity, AGS coordinates	266
17	CSM epoch time (least significant)	272

The sequence of the 17 words occurs at a basic rate of 50 words per second. The sequence is not interrupted. The word formats of the 17 downlink words are as follows:

Bit No.	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
ID Word	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	Y	Y
Data Word	1	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	Y	Y

X = 0 or 1, Y = Not applicable, S = Sign bit (0 or 1).

A stop pulse discrete, which is set after a new downlink word has been inserted into the downlink input register, will be available to the program. The stop pulse discrete is reset upon reading the downlink input register. The time interval between stop pulses is 20 milliseconds. The program interrogates the stop pulse discrete twice per 20-millisecond cycle, with at least a 1-millisecond separation. The input register is read only if the stop pulse discrete is set to ensure that the input register is not read while data are being shifted into the input register.

The three components of LM position are entered by keying address codes 240, 241, and 242, and the value and sign of the data, on the DEDA. The three components of LM velocity are entered by keying address codes 260, 261, and 262, and the value and sign of the data, on the DEDA. The LM epoch time (a multiple of 6 seconds) is entered by keying address code 254 on the DEDA. Keying 414 and +20000 on the DEDA completes initialization. When initialization is completed, the AEA is automatically set back to the attitude hold submode, as indicated by the DEDA display (414 + 00000). Manual initialization of CSM ephemeris data is identical with that for the LM, except for the address codes.

Time Initialization. Initialization of AGS absolute time consists of entering an absolute time of zero into memory. This is accomplished by keying 377 on the DEDA. AGS absolute time is not limited to zero, except by procedures. A time tag, referred to as epoch time, is associated with each state vector. Primary system time, referred to as ground elapsed time (GET) is referenced to the time of lift-off. Therefore, the epoch times are the values of GET at which the state vectors were determined. The LGC clock has a capacity of approximately 2.7 million seconds; the AEA clock, approximately 0.26 million seconds. This disparity requires a procedure for modifying the epoch time transmitted from the LGC to the AEA at initialization. Because the AEA does not require launch time, it is possible to modify epoch time with a subtractive time bias. This bias is the GET at which the AEA clock is started.

The procedure requires entering a verb in the DSKY and establishing an absolute time of zero on the DEDA. At the desired initialization time, the two ENTR pushbuttons are pressed simultaneously. The absolute time in the PGNS at the initialization time becomes the AGS time bias and may be read out in hours, minutes, and seconds via the DSKY. This time bias must then be subtracted from all subsequent times entered into the AGS (if the times are referenced to GET).

2.2.3.3.6 Abort Guidance Section - Modes of Operation.

The AGS has two major modes of operation: inertial reference and alignment. These modes are selected via the DEDA after the ASA has warmed up for 30 minutes, the AGS STATUS switch is set to operate, and the STAB/CONT: AEA circuit breaker is closed.

Inertial Reference Mode. When the AGS is in the inertial reference mode, it provides attitude error information and engine commands (on and off), which are used for vehicle stabilization and/or for directing vehicle abort. The inertial reference mode is always in one of the following submodes:

- Attitude hold
- Guidance steering
- Z-body axis steering

All active steering modes couple attitude errors about the LM body axes to the autopilot (CES). The magnitudes of the angular errors are limited to $+15^{\circ}$ and -15° . The autopilot attitude turning rates are limited to $+10^{\circ}/\text{second}$ and $-10^{\circ}/\text{second}$ about the Y-body axis and $+5^{\circ}/\text{second}$ and $-5^{\circ}/\text{second}$ about the X and Z-body axes, except after staging, with the DEAD BAND (panel 3) switch set to MAX, at that time, no rate limiting exists. AGS steering in no way protects the PGNS platform against gimbal lock. In the following discussions it is assumed that the GUID CONT switch is set to AGS, the LM is in orbit (lunar surface flag is not set) and the PITCH, YAW and ROLL switches are set to MODE CONT.

Attitude Hold. The attitude hold submode generates steering commands to maintain the LM in the inertial attitude that existed when the attitude hold mode was first entered. Attitude hold can be entered by setting the MODE CONTROL: AGS switch to ATT HOLD or by using address 400 on the DEDA and entering +00000. Once the attitude hold mode is established, it can be released by causing the followup signal to be issued (GUID CONT switch set to PGNS or MODE CONTROL: AGS switch set to ATT HOLD and ACA in manual control) or by entering other than +00000 into DEDA address 400. The preferred method is to set the MODE CONTROL: AGS switch to ATT HOLD and orient the LM to the desired attitude using the ACA. Reentering the attitude hold mode causes the LM to maintain the attitude that exists when the mode is reentered.

If the AGS is in the guidance steering submode (DEDA address 400 is +10000 and the MODE CONTROL: AGS switch is set to ATT HOLD), the AGS issues engine-on or engine-off commands that agree with the existing status of engine being used. If the AGS issues engine-on commands and the MODE CONTROL: AGS switch is set to AUTO, enter attitude hold by setting address 400 to +00000 via the DEDA which causes engine-off commands to be sent.

Guidance Steering. The address for ordering guidance steering is 400; the entry is +10000. The guidance steering submode orients the LM so that the LM thrust vector points in the desired direction for achieving the objective of the selected guidance routine. The LM X-body axis orientation also depends upon which engine is used. The RCS engines are assumed to thrust along the body axes. If the descent engine is

used, the thrust vector is assumed to be along the +X-body axis. If the ascent engine is used, the thrust vector is assumed to be displaced from the LM +X-body axis by an amount equal to the nominal cant of the ascent engine. Because of thrust alignment errors, the actual thrust vector will not lie exactly along the nominal direction. Small velocity errors at engine cutoff may result. These errors, sensed by the AGS, can be removed through axis-by-axis thrusting (ΔV residual removal).

When the ascent engine is used, equations that compensate for the cant of the engine should also be used. DEDA address 411 should be entered with a value of +10000 before entering AGS guidance steering, to achieve desired orientation of the thrust axis. When the ascent engine is burned the thrust direction is selected by the AEA, regardless of the value of the address 411 entry.

The AEA correction for the cant of the ascent engine is achieved by two octal constants (one for pitch and one for roll). The constants are preprogrammed into the central computer, but are DEDA-accessible so that they can be changed during the mission. The pitch constant is entered with the address 566. The value of the constant, entered in octal units of radians, should be positive if the engine causes a thrust acceleration component along the +Z-body axis. Similarly, the roll component is entered in octal form in units of radians via address 602. The sign of the value entered should be positive if the cant of the engine causes a thrust acceleration component along the +Y-axis direction.

The LM Z-body axis can be controlled as follows:

- DEDA address 623 +00000

The LM Z-body axis is driven parallel to the CSM orbit plane. If the +X-body axis points ahead of the local vertical (posigrade), the Z-axis is below the local horizontal. If the +X-body axis is behind the local vertical the Z-axis is above the local horizontal. If the X-body axis is vertically upward, the Z-body axis will be pointed horizontally forward. If the X-body axis is pointed vertically downward the Z-body axis will be directed horizontally opposite to the direction of motion.

- DEDA address 623 +10000

The LM Z-body axis is driven parallel to a plane determined by its normal unit vector W_b (DEDA entries 514, 515, or 516). This mode is normally used to maintain S-band communication during lunar orbit insertion. Unit vector W_b is entered in octal form and is nominally defined before the mission.

- X-axis override

The Z-body axis can be oriented to any desired position about the thrust axis by setting the ATTITUDE CONTROL: YAW switch to PULSE or DIR position and by means of the ACA.

The gain in the E_x channel decreases toward zero as the +X-body axis approaches a perpendicular orientation with respect to the specified plane. If +00000 is entered in address 623 and the X-body axis is perpendicular to the CSM orbit plane, the gain in the yaw steering channel is zero for any orientation of the Z-axis. This situation is typical for earth-orbit missions, during which large out-of-plane maneuvers are performed. By specifying a unit vector W_b (X axis) not normal to the CSM orbit plane and by entering +10000 into address 623 yaw steering control can be maintained.

The guidance steering submode controls three steering operations: prethrust steering, orbital rate pitch steering, and open steering channels.

Prethrust steering is guidance steering (address 400 contains +10000) that is performed before ullage is sensed and thrusting, to orient the LM for the burn. Prethrust guidance steering is implemented by setting the GUID CONT switch to AGS, the MODE CONTROL AGS switch to AUTO, and the ATTITUDE CONTROL: PITCH, YAW, and ROLL switches to MODE CONT, and entering +10000 into address 400. The AGS then orients the LM to the particular guidance mode selected (address 410). AGS steering orientation of the LM before the engine burn falls into two categories:

- CSI, CDH, external ΔV

Before initiation of these maneuvers, the desired attitude of the LM is established in the local horizontal coordinate system. Before the CSI maneuver, if the CSI computations are being performed and if the AGS is in the guidance steering submode, the LM is oriented so that the X-body axis is maintained horizontal and parallel to the CSM orbit plane.

- Direct transfer (TPI and Midcourse)

If the AGS is in a direct transfer guidance routine and the guidance steering submode, the vehicle orients itself to the desired attitude in inertial space and remains fixed in inertial space. When absolute time exceeds the TPI burn time t_{igC} , as would be the case for a midcourse maneuver when the rendezvous problem has not been retargeted, the guidance solution is in real time and LM attitude conforms to the changing solution.

Orbital rate pitch steering maintains the LM X-body axis along the local horizontal, using the external ΔV guidance routine and the guidance steering submode. The AGS can maintain the LM at any attitude in a local vertical coordinate frame during free flight. The X-inertial axis can also be maintained using the CSI guidance routine and the guidance steering submode, because the CSI maneuver is horizontal. This procedure is ideal for use between the orbit insertion phase and the CSI maneuver phase because the desired calculations (CSI) are performed simultaneously. If the calculated CSI maneuver is a retrograde maneuver, the +X-body axis points backwards and the Z-body axis points along the local vertical and not toward the lunar surface.

During open steering channels the AGS is in control, and the ATTITUDE CONTROL: PITCH, YAW, and ROLL switches are usually set to MODE CONT. Any channel can be overridden by setting the switch for that channel to PULSE or DIR. In some instances, manually orienting one LM axis more than 90° from the desired direction (AGS-computed) can cause the LM to rotate through large angles about the two AGS-controlled axes. In attitude hold and guidance steering, X-axis override can be accomplished through any desired angle. In Z-body axis steering, Z-axis override can be accomplished through any desired angle. If manual override of any other steering channel is desired, the attitude excursion about that axis should be limited to less than 90° from the desired orientation computed by the AGS.

Z-Body Axis Steering. This submode is entered via the DEDA, using the address 400 and entering +20000. No DPS or APS thrust can be accomplished in this mode. The following options are available for specifying the desired direction of the Z-body axis.

- DEDA address 507 + 00000

Orient Z-body axis toward estimated direction of CSM (used during rendezvous radar acquisition).

- DEDA address 507 + 10000

Orient Z-body axis in the desired thrust direction (used to perform small thrusting maneuvers along the Z-body axis).

In either option, with the Z-axis pointed in the desired direction, the X-body axis is oriented parallel to the CSM orbit plane. If the +Z-body axis is pointed ahead of the LM, the +X-body axis is above the local LM horizontal plane. If the Z-body axis is behind the LM, the +X-body axis is below the local horizontal plane. The gain in the roll channel (Z-axis) decreases toward zero as the +Z-axis approaches a perpendicular orientation with respect to the CSM orbit plane. When perpendicular, the gain in the roll steering channel is zero for any orientation of the +X-axis. In this submode the X-body axis can be manually controlled by setting the ATTITUDE CONTROL: ROLL switch to PULSE or DIR and operating the ACA.

Inertial Reference States. These states comprise the follow up, attitude control and complete control.

Follow Up State. In the follow up state, the AGS can perform the following functions:

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

- Maintain an inertial reference frame and provide total attitude signals for display
- Maintain the attitude error signals to the CES at zero (if automatic and followup discretes are present, error signals are computed)
- Use accelerometer inputs from the ASA to calculate LM current position and velocity in the inertial reference frame
- Solve the LM abort guidance problem, using a specific guidance scheme
- Accept initializing information from the PGNS downlink or the DEDA
- Command the engine on if the ascent engine-on signal or the descent engine-on signal is present, and command the engine off otherwise.

The flight program and the AGS are in the followup state when the followup discrete is present. If the automatic discrete is present, the attitude error signals computed in accordance with the particular submode in which the program is operating are displayed. If the automatic discrete is absent, attitude error signals are set to zero.

In the followup state, the program issues an engine-on command every 40 milliseconds if the program is in the inertial reference mode and either the descent engine-on discrete or ascent engine-on discrete is present. Otherwise, the flight program issues an engine-off command every 40 milliseconds.

Attitude Control State. The flight program and the AGS are in the attitude control state when the followup discrete, the abort discrete, and the abort stage discrete are absent. In this state, attitude error signals computed in accordance with the particular submode in which the program is operating are used for attitude control and for display. If the automatic discrete is absent, the program is in the attitude hold submode. If the automatic discrete is present any submode is permissible.

When the system is in the attitude control state, the program issues an engine-on command every 40 milliseconds if the program is in the inertial reference mode and either the descent engine-on discrete or ascent engine-on discrete is present. Otherwise, the flight program issues an engine-off command every 40 milliseconds.

Complete Control State. The flight program and the AGS are in the complete control state when the followup discrete is absent and either the abort discrete or the abort stage discrete is present. In this state, attitude error signals computed in accordance with the particular submode in which the program is operating are used for attitude control and for display. If the automatic discrete is absent, the program is in the attitude hold submode. If the automatic discrete is present, any submode is permissible except the following:

- The program shall be held in the attitude hold submode when the abort stage discrete is present, from the time the ascent engine-on discrete is first received until a preselected time thereafter. The preselected time shall have a minimum range of 0 to 10 seconds.
- At the completion of a maneuver when the engine is commanded off because the conditions for issuing engine-on commands are no longer satisfied, the program automatically returns to the attitude hold submode.

When in the complete control state and the following four conditions are simultaneously satisfied, the program issues an engine-on command every 40 milliseconds. Otherwise, the flight program issues an engine-off command every 40 milliseconds.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

- The automatic discrete is present.
- The AGS submode selector has been set to guidance steering, via the DEDA.
- Either the ullage condition has been satisfied or the ascent engine has been burned while the LM is on the lunar surface.
- Either the magnitude of velocity-to-be-gained exceeds a specified threshold or the projection of the velocity-to-be-gained on the X-body axis is in the +X-body-axis direction and exceeds the nominal ascent engine cutoff velocity impulse.

The ullage condition is satisfied when the accumulated sensed velocity increments along the X-body axis have exceeded a specified threshold in each consecutive 2-second computation cycle for three cycles. The ullage condition is reset and the count restarted if the accumulated sensed velocity increment along the X-body axis falls below the specified threshold in any one 2-second computation cycle.

Alignment Mode. Alignment consists of initializing the AEA to obtain an attitude reference to a desired inertial reference. In response to an alignment submode instruction (a DEDA entry), the inertial reference frame is aligned to a reference frame of the two alignment submodes: PGNS-to-AGS alignment and body-axis alignment.

The PGNS-to-AGS alignment submode aligns the AGS direction cosines to the attitude reference determined by PGNS Euler angles θ , ψ , and ϕ . (See figure 2.2-16.) The body-axis alignment submode uses current LM attitude as the attitude reference.

The AGS causes the attitude error signals to the CES to be reduced to zero during the alignment mode.

PGNS-to-AGS Alignment Submode. The submode is used when AGS alignment is to be accomplished, using PGNS Euler angles. This submode is entered by keying the DEDA (400, +30000, and ENTR); it should only be entered when the PGNS is operative.

Body-Axis Alignment Submode. For the body-axis alignment (orbital alignment) submode, the AGS inertial reference frame is made coincident with the LM axes. Body-axis alignment is achieved by keying the 400, +5000, and ENTR on the DEDA. To terminate this submode, the LM Pilot keys 400 and +00000 on the DEDA.

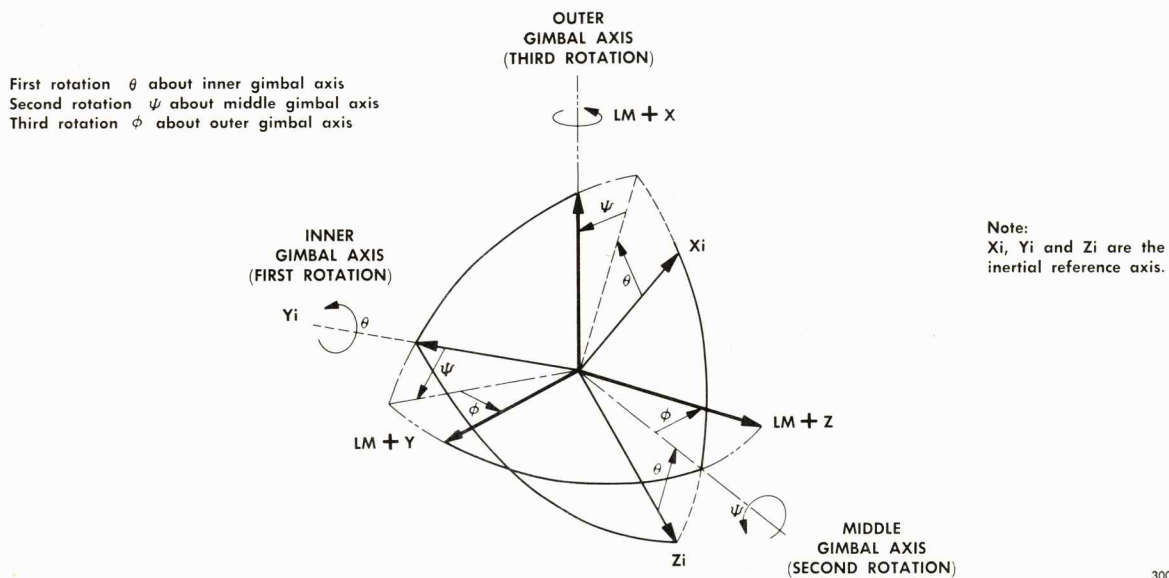


Figure 2.2-16. Euler Angles

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

Table. 2.2-14B. Abort Guidance Section - Calibration Compensations

Equation Symbol	Address	Description
1K19	540	X-accelerometer bias compensation coefficient
1K21	541	Y-accelerometer bias compensation coefficient
1K23	542	Z-accelerometer bias compensation coefficient

Calibration. Calibration of the gyros and accelerometers comprises the following:

- In-flight gyro and accelerometer calibration
- In-flight accelerometer calibration only.

In-flight Gyro and Accelerometer Calibration. This calibration is automatically selected by the flight program when calibration is selected via the DEDA, and provides in-flight calibration to update gyro fixed drift compensation. This calibration takes place during free fall with the program in the calibrate, lunar surface flag absent, and the LM in the attitude hold steering submode. The gyro drift calibration uses the PGNS IMU Euler angles as reference. The program automatically selects PGNS-to-AGS alignment during the first 2-second cycle, to provide initial alignment of the AGS to the IMU. For the remainder of the calibration period, the program computes the angular errors of the AGS indicated body axes (as defined by the transformation matrix) with respect to the PGNS indicated body axes (as defined by the IMU Euler angles). The angular errors are multiplied by an appropriate gain factor and added to the AGS compensated angular increments, causing the AGS attitude reference (transformation matrix) to remain aligned to the IMU reference. Simultaneously, the accumulated error signals are multiplied by another gain factor and added to the gyro drift compensation so that, in steady state, the compensation is equal to the gyro drift.

In-flight calibration of the accelerometer static bias compensation takes place concurrently with in-flight gyro calibration. The accelerometer velocity increments accumulated during free fall are multiplied by an appropriate gain factor and added to the accelerometer bias compensation so that the compensation is equal to the accelerometer bias. Accelerometer calibration provides updated values of 1K19, 1K21, and 1K23. (Refer to table 2.2-14B.) In-flight calibration aligns the AGS and therefore it is not necessary to follow in-flight calibration with a PGNS-to-AGS alignment. In-flight gyro and accelerometer calibration should not be performed when the PGNS is inoperative. Instead in-flight accelerometer calibration only should be performed. The procedure for in-flight gyro and accelerometer calibration when the PGNS is operative and attitude rotation rate is less than 0.1° per second is as follows:

- Set MODE CONTROL: AGS switch to ATT HOLD.
- Enter 400 +60000 via the DEDA.

Calibration requires 302 seconds and completion can be verified by reading out DEDA address 400. At completion of calibration, the readout returns to +00000 (attitude hold).

In-flight Accelerometer Calibration Only. This procedure is performed when the PGNS is inoperative. In-flight accelerometer calibration only does not affect AGS alignment. Procedure for in-flight accelerometer calibration only is as follows:

- Set AGS switch to ATT HOLD.
- Inhibit the RCS by setting the PITCH, ROLL and YAW switches to DIR. The ACA must be left in detent (no manual control.)

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

- Enter 400 +70000 via the DEDA

Calibration is completed in 32 seconds. However, it takes 302 seconds for readout of DEDA address 400 to return to +00000 (attitude hold).

2.2.3.3.7 Abort Guidance Section - Guidance Routines.

The AGS program contains the following guidance routines:

- Orbit insertion
- Coelliptic sequence initiation (CSI)
- Constant delta altitude (CDH)
- Terminal phase initiation (TPI)
- External ΔV .

The routines are selected, using the DEDA. Refer to table 2.2-14C for the address and function of each guidance routine. Guidance calculations associated with the selected routine are completed every 2 seconds of the submode selected. The LM, however, will not maneuver in accordance with the selected routine unless the AGS has control of the LM and the guidance steering submode is selected.

Before beginning guidance operations, the AGS must be aligned and initialized with LM and CSM ephemeris data. Entry of LM and CSM ephemeris data is required because every guidance routine uses these data in the guidance and steering computation. Navigation data must be updated throughout the mission, using PGNS downlink data, the DEDA, or RR data. Navigation data must also be updated before CSI and TPI maneuvers. Propellant expenditure is highly dependent on the accuracy of these maneuvers.

Table 2.2-14C. Abort Guidance Section - Guidance Routines

Address	Routine	Function
410 + 00000	Orbit insertion	Engine and steering commands are generated to drive LM to specified orbit injection conditions.
410 + 10000	CSI maneuver	Engine and steering commands are generated to establish appropriate LM-CSM phasing TPI time.
410 + 20000	CDH maneuver	Engine and steering commands are generated to drive LM into orbit that is coelliptic with CSM.
410 + 30000	TPI search	AGS evaluates various direct transfer rendezvous solutions. Astronauts select desired solution. Thrusting is not performed with this routine.
410 + 40000	TPI execute	Engine and steering commands are generated to cause LM to be on CSM intercept trajectory.
410 + 50000	External ΔV	Engine and steering commands are generated to perform externally specified thrust maneuver.

Orbit Insertion Routine. The orbit insertion routine provides attitude commands and engine-off commands to obtain predetermined cutoff conditions. These conditions are defined by the following targeting parameters entered into the AEA via the DEDA:

- Insertion altitude; DEDA address 347
- Magnitude of the horizontal component of inertial velocity at burnout; DEDA address 507
- Altitude rate at burnout; DEDA address 423.

The attitude commands provide yaw steering of the LM into the CSM orbit plane by the time the cutoff conditions are attained. The capability of steering out a wedge angle of up to 2° is provided (assuming sufficient ΔV). When the velocity to go to cutoff (ΔV_g) reaches a preselected threshold, updating of the command attitude is terminated. The engine is commanded off when ΔV_g reaches the nominal ascent engine tailoff velocity threshold and the total velocity-to-go is less than 100 fps.

Coelliptic Flight Plan Rendezvous Guidance. When the principal foci of two orbits are coincident and when the products of the length of the semimajor axis and eccentricity of each orbit are equal, the orbits are considered to be coelliptic. This implies that when the two orbits are coplanar, they have coincident centers and vacant foci. The guidance routines are based on the assumption that deviations from coplanarity are small. Coelliptic orbits have an essentially constant differential altitude (constant Δh).

The flight program contains a set of guidance routines for calculation of powered maneuvers required for the LM to execute a coelliptic flight plan for rendezvous with the CSM. This flight plan nominally calls for three powered maneuvers:

- CSI (coelliptic sequence initiate) maneuver
- CDH (constant Δh) maneuver
- TPI (terminal phase initiate) maneuver.

In the nominal coelliptic flight plan, the LM is placed in an orbit coelliptic with that of the CSM by executing the CSI maneuver, then the CDH maneuver. In both of these maneuvers, the horizontal component of thrust is maintained parallel to the estimated CSM orbit plane. The CSI and CDH routines compute and control these maneuvers. An additional capability is provided for removing out-of-plane velocity in the CSI and CDH maneuvers, by using the external ΔV routine for control. To enable this, the required incremental velocities computed by both the CSI and CDH routines are automatically furnished to the external ΔV routine to target, in advance, the in-plane components of the burn.

The final maneuver of the coelliptic flight plan sequence is the TPI maneuver, nominally a burn that places the LM on a direct intercept trajectory with the CSM. The maneuver is initiated when the angle of the LM-CSM line of sight with respect to the local horizontal is at a preferred value. The TPI routine computes and controls this maneuver; it may also be used for implementing subsequent midcourse correction maneuvers.

Coelliptic Sequence Initiation Routine. Coelliptic rendezvous is initiated by the CSI burn, followed by a CDH maneuver. The magnitude of the CSI burn is computed so that, after the CSI and CDH maneuvers, the LM arrives at the desired TPI time with the desired LOS angle to the CSM. The CSI and CDH maneuvers are computed in the AGS X-Z inertial plane. The CSI routine computes the magnitude of the horizontal velocity increment (parallel to the CSM orbit plane) required to satisfy targeting conditions.

The AGS time of the CSI routine is one targeting condition for the CSI maneuver. This entry is made by using DEDA address 275. Time-to-go until the CSI maneuver can be obtained by reading out address 313 on the DEDA.

After the CSI maneuver has converged to a solution, the time from the CSI to CDH can be obtained by using address 373 on the DEDA. Time until CDH (when less than 136 minutes) can also be obtained by using address 313. Other times available in the CSI routine are listed in table 2.2-13 (addresses 274, 275, and 377).

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The angle that the projection of the LM-CSM LOS on the X-Z inertial plane makes with the LM local horizontal at the desired TPI time (positive, measured clockwise from the local horizontal forward of the LM when looking in the direction of the +Y inertial axis) is another targeting condition.

The CDH burn is performed at the time determined by the CSI routine. This time is the predicted time of LM arrival at the first or second crossing of the LM orbit line of apsides, following the CSI burn. A DEDA input (416 + 10000) specifies whether the CDH maneuver will be performed at the second crossing. Otherwise, it is assumed that the CDH maneuver will be performed at the first crossing. If the predicted radial rate (r_f) is nearly zero at CSI coast, it is possible that the subsequent coast orbit would be circular. Selection of first or second apsidal crossing becomes meaningless for a circular orbit.

The CSI routine is repeated three times every 2 seconds. The routine computes, and makes available for readout via the DEDA, the following parameters:

- CSI iteration error
- Time from CDH to TPI
- Predicted differential altitude in the coelliptic orbit
- Predicted coelliptic burn ΔV magnitude
- Sign and magnitude of predicted horizontal (CSI) ΔV
- Predicted time from CSI to CDH.

The CSI routine provides a real-time commanded attitude. The CSI horizontal ΔV solution is valid at CSI time (t_{igA}). When real time (t) becomes greater than t_{igA} , the CSI routine is computed in real time.

Constant Delta Altitude. The CDH routine computes, in real time, attitude commands and velocity increment required to make the LM and CSM orbits coelliptic. The CDH routine provides commanded attitude such that the horizontal component of vehicle X-axis thrusting is parallel to the CSM orbit plane.

The CDH routine is selected after completion of the CSI burn, by keying 410 and +20000 into the DEDA. When this occurs, targeting is accomplished automatically for the CSI routine, by the CSI computations. The CDH solution is computed for future time, predicted during the CSI routine, until real time exceeds the predicted burn time. The solution is thereafter computed in real time.

If the CDH maneuver does not follow the CSI maneuver, the time to go until the CDH maneuver must be entered as a targeting constant. This is accomplished by keying address 315 on the DEDA. Addresses 274, 313, and 377 are timing quantities available in this mode.

In a nominal mission the CDH maneuver is assumed to occur after the CSI maneuver, therefore, targeting is accomplished automatically for the CDH maneuver by the CSI computations. The events timer is set by using the quantity T_{Δ} (after the CDH routine is entered) which is the time to go until the CDH maneuver, if T_{Δ} is no greater than 136 minutes.

If the CDH maneuver does not follow the CSI maneuver, then the AGS absolute time of the CDH maneuver (t_{igB}) must be used as a targeting constant. This is done with DEDA address 276. The times of interest during the CDH maneuver are listed in table 2.2-14D.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14D. Time (Minutes) Quantities Available in CDH Mode

GN&CS

Address	Quantization	Equation Symbol	Description
313	0.01	T_{Δ}	Time to go until CDH maneuver
274	0.1	T_c	Time from CDH to nominal TPI maneuver predicted during CSI
276	0.1	t_{igB}	AGS predicted absolute time of CDH
277	0.1	J^1	Nominal time of TPI maneuver (targeting quantity)
312	0.01	T_{perg}	Time to perifocus of present LM orbit
377	0.1	t	AGS computer time

Terminal Phase Initiation Routine. The TPI maneuver begins the direct transfer to rendezvous. There are two direct transfer routines: one, primarily for mission planning (TPI search); the other, for performing the maneuver (TPI execute). The latter routine can also be used for mission planning if desired. In general, the routines are used together, in sequence, to perform the TPI maneuver.

During mission planning, the object is to determine when direct transfer should occur. This determination is based upon achieving the desired LOS angle between the LM and CSM. Once ignition time is determined, the AGS is switched to the alternate direct transfer routine and the maneuver is performed.

This routine is used in the coelliptic rendezvous scheme, following the CDH maneuver. The CSI and CDH maneuvers are performed so that, at the nominal TPI time, the phasing between the LM and CSM is such that the desired LOS angle is achieved.

Because of errors in the AGS (due to sensors, navigation, maneuver execution, etc), the desired LOS is not achieved at exactly the targeted TPI time. To determine when the desired LOS will be achieved and, thus, when to perform the maneuver, a direct transfer search guidance routine is available in the AGS (DEDA entry 410 + 30000). During this routine, the astronauts determine total velocity required to rendezvous, and perform the TPI maneuver based upon achieving a desirable value (TBD) of total velocity. The alternate direct transfer guidance routine (DEDA entry 410 + 40000) is used to actually perform the maneuver.

There are two modes that can be used for the TPI maneuver. Tables 2.2-14E and 2.2-14F define the time quantities of interest in each mode.

Table 2.2-14E. Time (Minutes) of Interest for TPI Search Routine

Address	Quantization	Equation Symbol	Description
313	0.01	T_{Δ}	Input quantity. TPI search assumes TPI to occur in T_{Δ} minutes.
373	0.1	t_{igC}	AGS computer time of TPI maneuver; output quantity.
311	0.01	J^6	Input quantity. Time from TPI to intercept (assuming no burns after TPI).
306	0.01	T_r	Time to go until intercept; output quantity
312	0.01	T_{perg}	Time to perifocus of present LM orbit

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14E. Time (Minutes) of Interest for TPI Search Routine (cont)

Address	Quantization	Equation Symbol	Description
377	0.1	t	AGS computer time
314	0.01	J^4	Time increment prior to rendezvous at which node is created
310	0.01	J^3	Stable orbit offset

Table 2.2-14F. Time (Minutes) of Interest for TPI Execute Routine

Address	Quantization	Equation Symbol	Description
373	0.1	t_{igC}	Either input quantity or quantity obtained from TPI search mode. This is AGS computer time of the TPI maneuver.
313	0.01	T_{Δ}	Time to go until TPI maneuver; output quantity.
311	0.01	J^6	Input quantity. Time from TPI to intercept (assuming no burns after TPI).
306	0.01	T_r	Time to go until intercept; output quantity.
312	0.01	T_{perg}	Time to perifocus of present LM orbit
377	0.1	t	AGS computer time
314	0.01	J^4	Time increment prior to rendezvous at which node is created
310	0.01	J^3	Stable orbit offset

Epoch times input to the AEA via the DEDA are in units of minutes with a quantization level of 1/10 minutes or 6 seconds. Epoch data used in the AGS for navigation updates via the DEDA must be valid at a multiple of 1/10 minute. If not, position errors of 12 miles for earth missions could result due to roundoff of epoch time via the DEDA.

External ΔV Routine. The external ΔV routine accepts components of a velocity-to-be-gained vector in local vertical coordinates entered via the DEDA. A commanded attitude maintains the LM X-axis toward the resultant of the velocity-to-be-gained vectors in real time, independent of the required burn time. When ullage is sensed, and thereafter until the AEA is switched from the external ΔV , the external ΔV components remain fixed in inertial space. If the external ΔV routine (DEDA entry 410 + 50000) is initiated early or late, the desired orbit at cutoff is not achieved. Large timing errors result in large differences between the actual orbit obtained and the orbit desired.

The AGS can maintain the LM in any attitude in a local vertical coordinate frame during free flight. The LM can be pitched at the orbital rate to maintain the X-axis along the local horizontal, using the external ΔV routine. A fictitious velocity-to-be-gained vector is entered via the DEDA to obtain the desired attitude.

2.2.3.3.8 Abort Guidance Section - Descent or Ascent Engine and Thrust Chamber Assembly Burn.

When the LM is under AGS control during coasting flight, the AGS can fire the descent or ascent engine or TCA(s). The AEA is programmed in such a manner that translational thrust is applied in certain nominal directions. (Refer to table 2.2-14G.) The nominal cant angle used by the AGS is controlled by two constants in the AEA computer (DEDA - accessible in octal):

- 4K7 - Cant angle (radians) of thrust vector in pitch plane. DEDA address 566. Positive sign indicates thrust vector is canted from the +X-body axis toward the +Z-body axis.
- 4K8 - Cant angle (radians) of thrust vector in the roll plane. DEDA address 602. Positive sign indicates thrust vector is canted from the +X-body axis toward the +Y-body axis.

Table 2.2-14G. Abort Guidance Section - Translational Thrust

Engine	Direction of Thrust Assumed by AEA
Descent	+X-body axis
Ascent	Canted. Cant angle data are used by AEA to account for canted thrust direction.
TCA	The +X, +Y, +Z thrust chamber assemblies are assumed to thrust along the +X, +Y, +Z-body axes, respectively.

Before in-orbit maneuvers, the astronauts choose which TCA(s) will be used. The AGS must be informed via the DEDA, of the decision to fire the ascent engine, so that steering transients will not occur at the start of the burn. Before AGS-controlled orientation to the desired thrust attitude, the following DEDA selection should be entered or verified:

- DPS or RCS: DEDA entry 411+00000
- APS: DEDA entry 411+10000.

Regardless of the selection, the AEA undertakes the canted thrust direction whenever the APS on discrete is received. (See figure 2.2-17.)

Descent or Ascent Engine Burn. To prepare for an engine burn under AGS control, the following conditions are required:

- Condition No. 1 -

The GUID CONT switch must be set to AGS. This removes the followup signal to the AGS

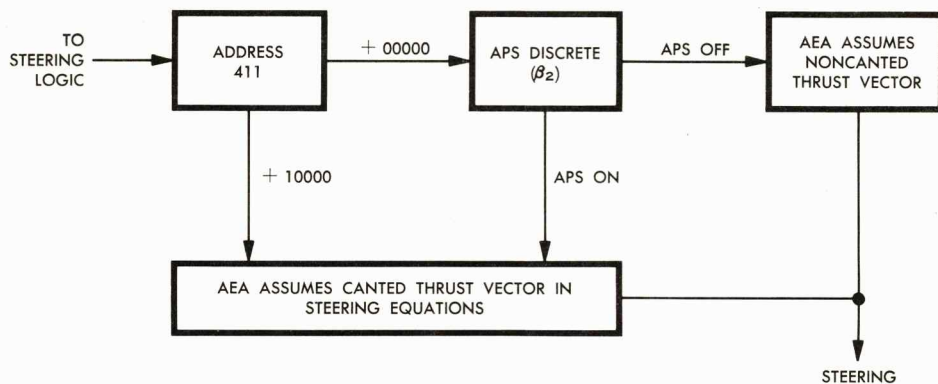
- Condition No. 2 -

The MODE CONTROL: AGS switch must be set to AUTO. This transmits an auto discrete to the AGS and removes the AGS from attitude hold.

- Condition No. 3 -

The ABORT pushbutton must be pressed. This arms the DPS.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



300LM4-258

Figure 2.2-17. Canted Thrust Selection Logic

- Condition No. 4 -

The ABORT STAGE pushbutton must be pressed. This arms the APS.

- Condition No. 5 -

+10000 must be entered into DEDA address 400. This enables the engine-on signal and guidance steering.

Conditions No. 1, 2, and 5 are sufficient for the AGS to assume control and orient the LM with the selected guidance routine. However, to fire the descent or ascent engine additional conditions (conditions No. 6 or 7, and 8) must be met.

Velocity To Be Gained. For the engine-on discrete to be issued to the CES, one of the following conditions must be met:

- Condition No. 6 -

The magnitude of the velocity-to-be-gained vector, as computed in the AEA, must exceed 100 feet per second.

- Condition No. 7 -

The component (sign and magnitude) of the velocity-to-be-gained vector must exceed the engine tail-off ΔV value (currently +2.1 feet per second).

Ullage. Before an engine can be fired, the AGS must sense ullage. Ullage must be supplied manually by pressing the +TRANSL pushbutton until the engine fires. The condition governing the AEA recognition of ullage is as follows:

- Condition No. 8 -

The AEA recognizes whether the sensed velocity change per 2-second compute cycle equals or exceeds 0.2 foot per second in the +X-body - axis direction for three consecutive compute cycles.

If the sensed velocity increment drops below 0.2 foot per second in a 2-second compute cycle, an engine-off discrete is sent to the CES. Before the engine can be fired again, the AGS must again sense ullage. During the powered-flight abort-staging sequence, the sensed velocity accumulated over any 2-second period will exceed 0.2 foot per second; therefore, ullage is maintained and the AEA generates continuous engine-on signals. The time required to generate AEA ullage signal can vary between 4 and 6 seconds,

depending upon when in the compute cycle ullage is initiated and to a lesser extent upon the thrust acceleration level.

Satisfaction of Ullage Condition Due to Angular Accelerations. The maximum distance between the ASA and the LM center of gravity is approximately 10 feet in the +X-body-axis direction and 5 feet in the +Z-body-axis direction. There is effectively no displacement in the Y-body-axis direction. Because the ASA is offset in the Z-direction, angular accelerations about the pitch axis could cause the ullage condition to be satisfied. The required angular acceleration is slightly greater than $1^\circ/\text{second}^2$ and must exist for a minimum of 6 consecutive seconds.

If all conditions for engine burn, except the ullage condition, are satisfied and the LM is to be reoriented through a large angle under AGS control, the engine STOP pushbutton must be pressed until LM reorientation has occurred, to avoid sending the engine-on discrete.

Conditions for descent or ascent engine burn are as follows:

- To fire the descent engine, conditions No. 1, 2, 3, 5, 8, and either or both 6 and 7 are sufficient.
- To fire the ascent engine, conditions No. 1, 2, 4, 5, 8, and either or both 6 and 7 are sufficient.

Manual Engine Operation. If an engine is fired manually by pressing the engine START pushbutton, it can only be shut down by pressing the engine STOP pushbutton (unless fuel depletion or staging occurs first). If the engine is fired manually, the AGS will not shut it down. When an engine is fired manually, the astronauts can monitor ΔV_{gx} (quantized at 1 foot per second) via DEDA address 500 and press the engine STOP pushbutton when ΔV_{gx} is near zero.

Thrust Chamber Assembly Burns. TCA translation can be used with AGS steering control when the following is performed:

- Conditions No. 1, 2, and 5 must be established in sufficient time for the LM to orient to the desired thrust attitude.
- DEDA address 500 must be selected for ΔV_{gx} readout.
- The +X TRANSL pushbutton must be set to the desired number of jets (2 or 4).
- The engine STOP pushbutton must be pressed. This inhibits the engine-on discrete.
- The ABORT or ABORT STAGE pushbutton must be pressed. This provides for attitude hold at the end of the burn.
- At maneuver time, use the TTCA to null the DEDA reading.
- When ΔV_{gx} equal zero, reduce thrust to zero.

2.2.3.3.9 Abort Guidance Section - Thrust Termination

When the LM is under PGNS control, the AGS outputs engine-on or engine-off discretes which are followups of the actual engine condition. These discretes are inhibited when the GUID CONT switch is in the PGNS position. When the LM is under AGS control the engine-off discrete is sent to the CES, when the velocity to be gained magnitude falls below the threshold value.

Ascent or Descent Engine. When the AGS sends an engine-off discrete to the CES, the ascent or descent engine thrust automatically terminates, provided the engine was not turned on manually (by pressing the engine START pushbutton). Under AGS control the engine-off discrete is generated if any of the following conditions occur:

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

- Velocity-to-be-gained magnitude is equal to or less than 100 feet per second and the component of the velocity-to-be-gained vector resolved along the +X-body axis is equal to or less than 2.1 feet per second.
- Ullage recognition signal (less than 0.2 foot per second sensed ΔV in a 2 second computer cycle) is last.
- Address 400 does not contain the value +10000.

Attitude Hold. If the MODE CONTROL: AGS switch is set to ATT HOLD during an AGS-controlled powered maneuver the AGS is placed in an engine follow-up mode. The existing LM attitude and the engine-on discrete to the CES is maintained; an engine-off discrete is inhibited by the MODE CONTROL switch. The astronauts can assume manual attitude control by moving the ACA out of detent. When this option is used thrust must be terminated manually. Returning the ACA to the detent position and setting the MODE CONTROL: AGS switch to AUTO allows the AGS to resume guidance control; thrust terminates when the velocity to-be-gained magnitude has decreased to the threshold established for engine shutdown.

Thrust Chamber Assemblies. The following procedure applies to each axis individually if axis-by-axis translation is being performed (LM in attitude hold) or to the X-body axis if AGS steering is used:

- Monitor via the DEDA the velocity-to-be-gained magnitude in the appropriate thrust direction. (Refer to table 2.2-144.)
- Terminate thrust manually with the TTCA when velocity to be gained (along the appropriate axis) is zero.

When in the AGS steering mode, the LM is placed in attitude hold when thrust termination occurs. If the TCA burn has continued long enough (longer than 6 seconds) for the ullage recognition signal to be generated in the AEA, the AGS switches to attitude hold when ΔV_{gx} is equal to, or less than, 2.1 feet per second. If the burn is too short for ullage recognition, or if thrusting along the Z-body axis the AGS orients the LM +X-body axis along the direction of the residual ΔV vector after thrust termination. This could produce an undesired LM rotation in an attempt to align the +X body along the residual ΔV vector.

In addition, in the AGS steering mode, when velocity to be gained is less than 15 feet per second and the sensed ΔV in the +X-body axis direction exceeds 0.2 foot per second in a 2-second computer cycle, the AEA ceases updating the desired thrust direction. To avoid undesired LM rotations, the procedure for performing TCA burns of less than 15 feet per second when AGS steering is used is as follows:

- The GUID CONT switch must be set to AGS and MODE CONTROL: AGS switch to AUTO
- DEDA entry 400+10000 (guidance steering) must be verified
- Immediately preceding the maneuver (within 10 seconds), the MODE CONTROL: AGS switch must be set to ATT HOLD
- ΔV_{gx} must be monitored via DEDA address 500.
- The maneuver must be performed with the +X TRANSL pushbutton.
- TCA thrusting must be terminated when ΔV_{gx} is zero.

If the option to perform a Z-body axis TCA burn with the LM Z-body oriented in the desired thrust direction is chosen (DEDA entries 400+20000, and 507+10000) set the MODE CONTROL: AGS switch to ATT HOLD when velocity to be gained along the Z-body axis is less than 15 feet per second.

2.2.3.3.10 General Operation of Abort Guidance Section. (See figure 2.2-17A.)

Control of the LM by the AGS depends on the settings of various cabin switches and on DEDA entries. In addition, attitude control, using the RCS, must be under mode control (ROLL, PITCH, and YAW ATTITUDE CONTROL switches on panel 3 set to MODE CONT).

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Figure 2.2-17A. Abort Guidance Section - Functional Flow Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-14H. Abort Guidance Section - Thrust Directions

Quantity*	DEDA Address	Units	DEDA (Least Significant Digit)
			Earth Mission
ΔV_{gx}	500	fps	1.0
ΔV_{gy}	501	fps	1.0
ΔV_{gz}	502	fps	1.0
*These quantities are computed every 40 milliseconds and updated on the DEDA display every 1/2 second.			

For the AGS to effect guidance steering (not merely attitude hold) and engine control, the GUID CONT switch (panel 1) must be set to AGS and the MODE CONTROL: AGS switch (panel 3) must be set to AUTO. For nominal DPS operation, the ENG ARM switch (panel 1) is set to DES and the engine START pushbutton (panel 5) is pressed. For abort DPS operation, the ABORT pushbutton (panel 1) is pressed to arm the descent engine. Ascent engine operation is similar to descent engine operation, except that the ENG ARM switch is set to ASC. For APS operation in abort situations, with the descent stage attached, the ABORT STAGE pushbutton (panel 1) is pressed to arm the ascent engine.

The MODE SEL, ATTITUDE MON, and RATE/ERR MON switches (panels 1 and 2) are used to monitor AGS maneuvers. When these switches are set as indicated in figure 2.2-17A, the FDAI's, X-pointer indicators, and the ALT and ALT RATE indicators display the information required to monitor AGS operations.

The AGS STATUS switch provides power to the AGS when the AC BUS B: AGS and STAB/CONT: ASA and AEA circuit breakers are closed. With the AGS STATUS switch set to OFF, closing the ASA circuit breaker causes the ASA to be in a temperature-controlled condition. Closing the AEA circuit breaker causes power to be applied to the AEA. Closing the AGS circuit breaker applies 15-volt (rms), 400-cps power to the AGS power supply.

Initial conditions for AGS operation require that the AGS STATUS switch be set to STANDBY, then to OPERATE. The time between closing the circuit breakers and setting the AGS STATUS switch to OPERATE should be 40 minutes; for at least the last 25 minutes, the switch should be set to STANDBY. Degraded performance is available after 10 minutes in the standby mode. When the AGS STATUS switch is set to OFF, the AEA has no functional capability. After 20 seconds in the standby mode, the AEA can accept the CDU zero signal and integrate the PGNS Euler angle changes. Complete AEA capability is afforded when the switch is set to OPERATE. In the operate mode, the AEA enters a core-priming routine that ensures that the memory is properly magnetized.

AGS operations are performed mainly through two DEDA addresses: 400 and 410. (Refer to table 2.2-11.) Address 400 is the AGS submode selector; address 410, the guidance routine selector. The selected routine is computed every 2 seconds, regardless of the submode selected. The AGS does not respond to orient the LM in accordance with the routine selected, unless DEDA address 400 (mode selector) is set to +00000 (attitude hold), +10000 (guidance steering), or +20000 (Z-axis steering).

When the LM is under full AGS control, the engine-on signal cannot be generated unless the guidance steering submode is selected. The engine-on signal is generated automatically after ullage has been sensed for three (DEDA-accessible constant) consecutive computer cycles (2 seconds per cycle). The AGS recognizes ullage to have occurred when the average acceleration in the +X-direction exceeds 0.1 fps². (The average acceleration is DEDA-accessible.) The ASA (containing the accelerometers) is located ahead of the center of gravity (in the +X-direction). Therefore, LM rotations cause sensed accelerations in the -X-direction. For this reason, LM rotations cannot cause the AGS to sense that ullage has occurred.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

When the LM is not under full AGS control (neither the ABORT nor ABORT STAGE pushbutton has been pressed, or the MODE CONTROL: AGS switch is not set to AUTO, or the GUID CONT switch is not set to AGS), the AGS issues engine commands (on or off) that duplicate actual engine operation.

Under full AGS control, the ascent or descent engine is commanded off automatically when the velocity to be gained in the +X-direction is less than the nominal ascent engine thrust decay velocity and if the total velocity to be gained is less than a prescribed threshold (a DEDA-accessible constant currently set at 100 fps). This dual check maintains the engine on if an abort occurs during powered flight with the LM incorrectly oriented for the abort maneuver and the velocity to be gained large greater than the 100-fps threshold.

When the velocity to be gained (LM under full AGS control) is less than 15 fps and the sensed thrust acceleration level in the +X-direction is greater than 0.1 fps^2 , the desired thrust direction is fixed in inertial space (a form of attitude hold). If this were not done, the LM desired attitude might go through an undesirably wide excursion in an attempt to achieve perfect velocity cutoff conditions. Large variations near the end of a maneuver are undesirable. The velocity cutoff errors incurred by fixing the desired attitude before engine cutoff are small. After the maneuver is completed, small cutoff errors can be removed (if desired) by the axis-by-axis velocity trim capability of the AGS.

The LM descent stage is staged (when the AGS is in control) by pressing the ABORT STAGE pushbutton. The staging sequence begins only when engine-on commands are issued. During a thrusting maneuver, the staging sequence begins immediately upon pressing the ABORT STAGE pushbutton (assuming that all panel controls that transfer control of the LM to the AGS have been set properly). The AGS senses sufficient average thrust acceleration throughout the staging maneuver to maintain ullage. When the AGS receives verification from the CES that the ascent engine is on, the AGS enters the attitude hold submode automatically. After a prescribed interval, between zero and 10 seconds (DEDA-controlled, presently set at 1 second), the AGS enters the normal guidance steering submode automatically.

When the PGNS controls the LM (GUID CONT switch set to PGNS), the AGS is in the followup mode. Manual control of the LM by the astronauts (MODE CONTROL: PGNS switch set to ATT HOLD, attitude controller out of detent) also causes the followup signal to be routed to the AGS. In the followup mode, the AGS follows the PGNS by routing engine commands (on or off) in accordance with ascent or descent engine operation and provides zero attitude control error signals. The AGS provides attitude error signals (corresponding to the AEA guidance solutions) for FDAI display when the PGNS is in control, the MODE CONTROL: PGNS switch is set to AUTO, the ATTITUDE MON switch is set to AGS, and the RATE/ERR MON switch is set to LDG RDR/CMPTR.

2.2.3.4 Rendezvous Radar. (See figure 2.2-18.)

The RR provides LOS range, range rate and antenna angular data to the LGC and to the LM cabin displays. By DEDA entry, the AEA is supplied with range and range rate data for a computer-controlled (LGC or AEA) rendezvous of the LM and CSM.

The RR, when search sweeping or tracking and locked on to the RR transponder in the CSM, is a continuous-wave coherent system, which uses phase-lock techniques in the radar and in the transponder. When operating with the LGC, the RR derives range, range rate, and shaft and trunnion angle data and routes these data to the LGC. The AEA state vector is updated automatically by range and range rate data from the LGC or by manually inserting this data via the DEDA. Range is determined by measuring the time delay between the transmitted signal and the received signal. Range rate is determined by measuring the two-way Doppler shift of the carrier signal received from the transponder.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Range and range rate are accepted by the LGC only upon program demand. The CDU continuously receives the shaft and trunnion angles, converts them to digital form, and sends these data to the LGC upon program demand. In addition to the RR outputs to the LGC; the range and range rates are displayed in the cabin. The shaft and trunnion angles are indicated as elevation and azimuth by the X-pointer and as pitch and yaw rates by the FDAI on panels 1 and 2 of the cabin display. The azimuth and yaw indications are derived from the trunnion angles, and the elevation and pitch indications from the shaft angles.

The RR has an unambiguous, range capability from 80 feet to 400 nm and a range rate capability from -4,900 to +4,900 fps. The RR consists of an antenna assembly and an electronic assembly.

The RR consists of an antenna assembly and an electronics assembly. The antenna assembly multiplies by the 96 the 102.425 mc carrier signal, generated in the electronics assembly, to obtain the 9832.8 mc transmitter signal. In free flight, the LGC computer the LOS angle to CSM and then points the LM+Z axis to the CSM target. The LGC designates the RR antenna angles, pointing the RR antenna to the target, so that the 9832.8-mc transmitter signal can be acquired by the transponder.

The transponder sweeps ± 104 kc about its nominal 9792-mc transmitter frequency until the RR 9832.8-mc signal is detected. Concurrent with automatic detection, the transponder halts the frequency sweep and acquires the RR transmitter signal by phase locking to the carrier frequency. The transponder side-steps the RR transmitter signal 40.8 mc to 9792 mc. The transponder transmits the 9792-mc signal for acquisition by the RR.

The RR search sweeps until it acquires the transponder transmitted signal, locks on to the 9792-mc carrier frequency. The RR goes into auto-track mode; the radar angle tracking loop is closed between the RR and the transponder. The RR modulates its transmitted 9832.8-mc signal with low and medium frequency (LF and MF) range tones (200 cps and 6.4 kc).

The transponder demodulates the received RR transmitted signal and modulates its 9792-mc signal with the LF and MF range tones, phase coherent with the range tones present on the received RR transmitted signal. The RR locks on to the LF and MF range tones. The RR drops the LF range tone upon initial range acquisitions, locks on to the LF range tone. The high frequency (HF) range tone is used when the range, distance between the LM and CSM, is less than 50.58 nm or for increased range accuracy.

The transponder signal is received by a four-port feedhorn. The feedhorn, arranged in a simultaneous lobing configuration, is located at the focus of a Cassegrainian antenna. (See figure 2.2-19.) If the transponder is directly in line with the RR antenna boresight, the transponder signal energy is equally distributed to each port of the feedhorn. If the transponder is not directly in line, the signal energy is unequally distributed among the four ports.

The RF signal passes through the polarization diplexer to a comparator which processes the RF signal to develop sum and difference signals. The sum signal represents the sum of energy received by all feedhorn ports (A+B+C+D). (See figure 2.2-19.) The difference signals, representing the difference in energy received by the feedhorn ports, are processed along two channels, a shaft-difference

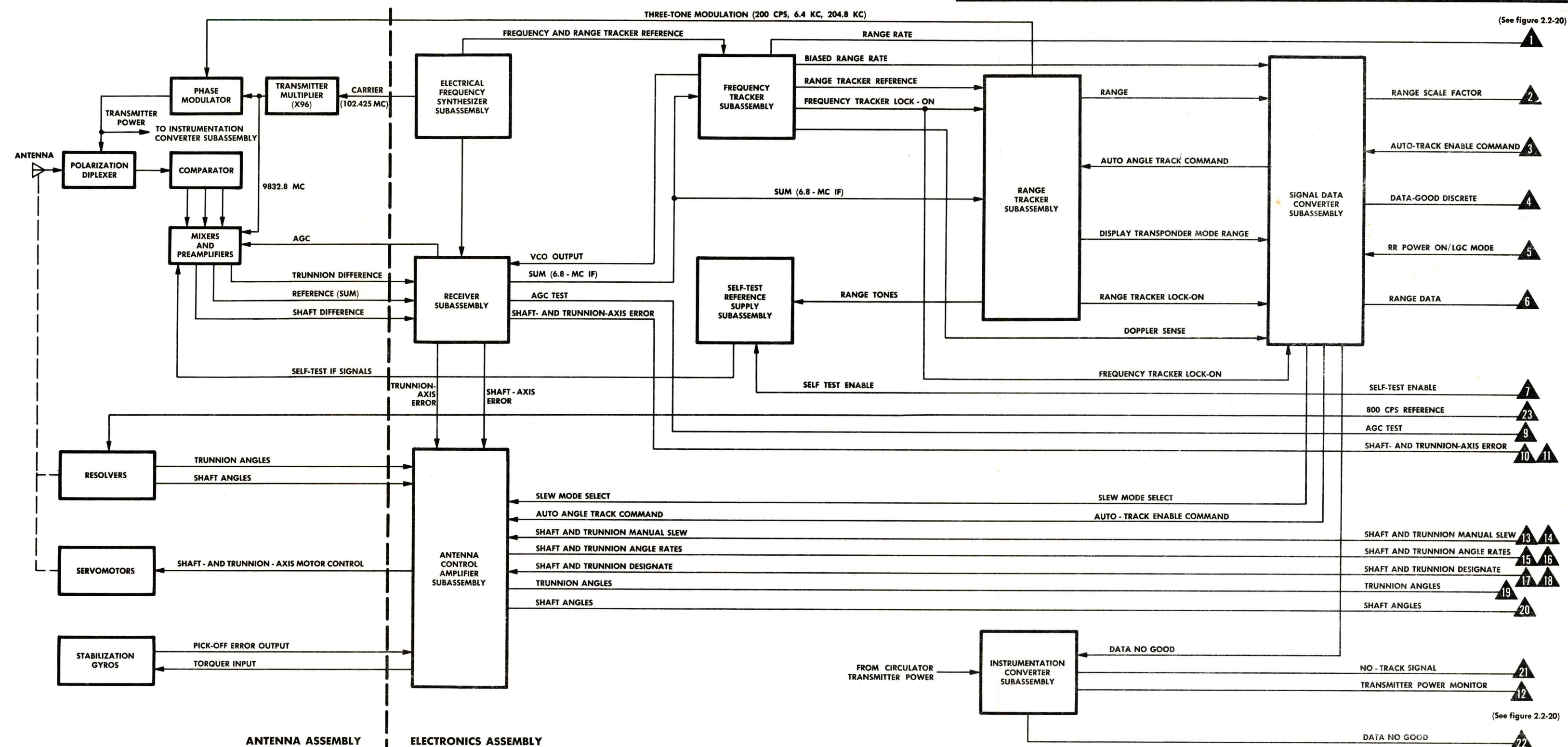


Figure 2.2-18. Rendezvous Radar - Block Diagram

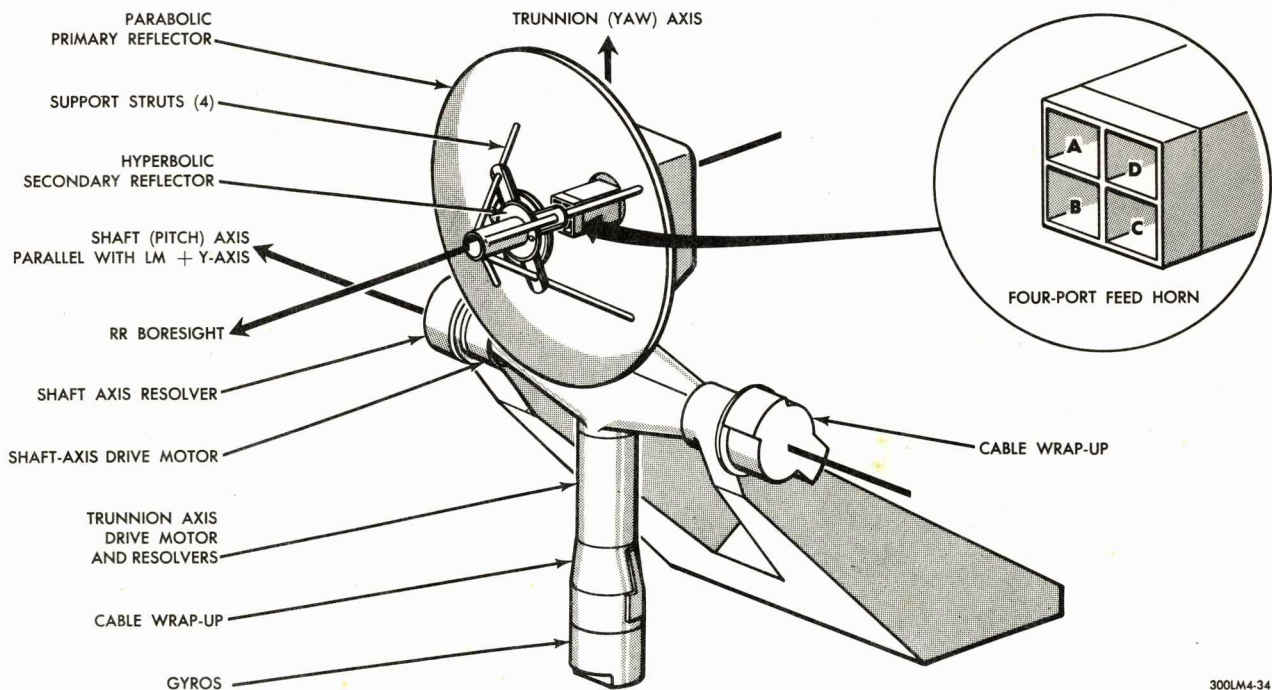


Figure 2.2-19. Rendezvous Radar Antenna Assembly

channel and a trunnion-difference channel. The shaft-difference signal represents the vectoral sum of the energy received by the vertically adjacent ports (A+D)-(B+C) of the feedhorn. The trunnion-difference signal represents the vectoral sum of the energy received by the horizontally adjacent ports (A+B)-(C+D) of the feedhorn. The comparator outputs are beat with the transmitter frequency (9832.8 mc) to obtain three 40.8-mc IF signals. The sum channel (reference), trunnion channel and shaft channel of the preamplifiers amplify their respective IF signals, which are then routed to the receiver subassembly of the electronic assembly.

The electronics assembly furnishes crystal-controlled signals, which drive the RR transmitter and provide a reference for processing the return signal. The electronics assembly also supplies servo drive signals for antenna positioning and angle tracking. The frequency synthesizer subassembly develops the carrier frequency (102.425mc) which is multiplied by 96 to 9832.8 mc (X-band) in a multiplier chain in the antenna assembly and phase modulated by three tones (200 cps, 6.4 kc, and 204.8 kc) generated by a reference generator in the range tracker. The receiver subassembly processes the three 40-mc IF preamplifier outputs in three separate channels. The three 40-mc IF signals sum, trunnion, and shaft are converted into 6.8-mc IF signals. The sum signal (6.8-mcIF) is fed to the range tracker subassembly.

The range tracker sequentially compares the phase of the 200-cps, 6.4-kc, and 204.8-kc return modulations with the phase of the reference signal from a reference counter, to provide range information to the signal data converter. This operation cannot take place until the frequency tracker has locked up and the LGC has issued an autotrack enable signal so that the range tracker can lock up. Comparison of the phase of the return and reference signals provides unambiguous ranging from 80 feet to 400 nautical miles. The range output is fed to the LGC and to the cabin displays.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The sum signal output of the receiver subassembly is also fed to the frequency tracker subassembly. The received frequency is not constant; it varies with the range rate between the LM and the CSM. The difference in frequency (Doppler frequency) is a function of the rate of change of range. By measuring the Doppler frequency, the range rate may be determined. The frequency tracker subassembly detects the Doppler frequency by continuously comparing the received frequency, after it has been converted to 6.8 mc in the receiver subassembly, with a reference frequency. A voltage-controlled oscillator (VCO) in the frequency tracker subassembly is maintained at the received frequency plus Doppler frequency by means of an automatic phase-lock loop. Any change in Doppler frequency results in a corresponding change in VCO frequency. The instantaneous frequency of the VCO is a function of range rate. Range rate data are fed to the LGC and to the cabin displays.

The Doppler frequency component must be eliminated from the ranging and antenna-positioning circuits because it would cause inaccuracies. Therefore, the loop that controls the VCO frequency encompasses part of the receiving circuits. A feedback loop removes the Doppler frequency from the ranging and antenna-positioning circuits, but maintains the VCO at the received frequency (6.8 mc + fd).

After double conversion to 1.7 mc in the receiver, the trunnion- and shaft-difference signals and the sum signal are demodulated. The trunnion and shaft-difference signals are compared with the sum (reference) signal to obtain drive signals (trunnion- and shaft-axis error) for correcting antenna-pointing errors. The position of the transponder with respect to the RR boresight determines the type of antenna pointing error signal generated. With the RR boresight facing above or below the transponder, a shaft pointing error signal is generated in the angle error detector. With the RR boresight facing to either side of the transponder, a trunnion error pointing signal is generated. Routed to the antenna control amplifier subassembly, the d-c error signal (shaft- or trunnion-axis motor control) causes the antenna servo system to provide tracking compensation which reduces the antenna-pointing errors. An AGC loop in the receiver subassembly makes the output of the angle error detector dependent on the antenna angle error, rather than on amplitude of the signal. The output of the angle error detector causes the motors of the antenna trunnion and shaft to position the antenna so that the boresight is always toward the CSM.

A gyro stabilization loop compensates for vehicle angular motion. When tracking the transponder, RR antenna attitude indicates the direction of the transponder with respect to the LM. Resolver outputs, indicating antenna-to-LM vehicle angles, are fed to the CDU and through the CDU to the LGC. When the antenna is in antenna orientation No. 1, angular data is also fed to the LM displays.

2.2.3.4.1 Modes of Operation.

The RR operates in three modes: automatic tracking, slew (manual) and LGC control. The mode of operation is selected with the RENDEZVOUS RADAR selector switch on panel 3.

Automatic Tracking Mode. The automatic tracking mode is selected by setting the RENDEZVOUS RADAR selector switch to AUTO TRACK. This mode enables the RR to track the CSM automatically after it has been acquired; tracking is independent of LGC control. When this mode is selected, tracking is maintained by comparing the received signals from the shaft and trunnion channels with the sum channel signal. The resultant error signals drive the antenna, thus maintaining track.

Slew Mode. The slew (manual) mode is selected by setting the RENDEZVOUS RADAR selector switch to SLEW. This mode enables an astronaut to slew the antenna manually to a position for acquisition of the CSM.

LGC Control Mode. The LGC control mode is selected by setting the RENDEZVOUS RADAR selector switch to LGC. In this mode, the LGC automatically controls antenna positioning, initiates automatic tracking once the CSM is acquired, and controls change in antenna orientation. When this mode is selected, the RR sends a RR power-on/LGC discrete to the LGC. Automatic control of RR search and acquisition operation is provided by the PGNS, which transmits slewing commands to position the RR antenna.

2.2.3.4.2 General Operation. (See figure 2.2-20.)

After the frequency tracker has locked up with the transponder carrier frequency (9792 mc), the RR angle tracks the transponder target. The radar antenna is designated in angle by LGC so



GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM
Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.2-87/2.2-88

that its transmitted continuous-wave radiation can be received at the transponder. The shaft and trunnion resolver outputs are routed through the SHFT/TRUN switch (panel 1) and the RATE/ERR MON switch to the FDAI error needles. With the RATE/ERR MON switch set to RNDZ RADAR, the shaft angle signals are applied to the pitch error needle and the trunnion angle signals are applied to the yaw error needle. The SHFT/TRUN switch provides the corresponding scaling factor ($\pm 5^\circ$, $\pm 50^\circ$) for the resolver outputs. With the RATE/ERR MON switch set to RNDZ RADAR, the shaft and trunnion rates, corresponding to RR boresight angular rates, are routed through the X POINTER SCALE switch for display on the X pointer indicator. At this switch position, the ELEV RT and AZ RT placarding on the X-pointer indicator are illuminated. The setting of the X-POINTER SCALE switch determines whether the X.1 or X1.0 placarding is illuminated.

When the angular position of the antenna is within 0.5° of the LGC estimated direction to the transponder reply signal (a total maximum antenna pointing error of 2.5°), the LGC provides an auto track enable signal that initiates the auto-tracking function. The antenna pointing error may deviate more than 0.5° from the LGC estimated direction because of cumulative mechanical, electrical and LGC programming errors. The enable signal causes the RR to track the target automatically, independent of LGC control. During automatic tracking, the RR supplies shaft and trunnion angles to the LGC via the CDU analog-to-digital converters.

The RR angle-tracking loop cannot be completed until the transponder is phase locked to the received radar signal and the RR receiver is phased locked to the received transponder signal. In the RR, lock up to the transponder signal (9792 mc) activates ranging modulation and the ranging tracking error is nulled within 11 seconds after RR locks on to the transponder signal or after LGC issues the auto-track enable discrete, whichever is completed last. After range and range-rate lock-up are completed, the RR issues a data-good discrete to the LGC. The NO TRACK light (panel 3) is extinguished. Range and range rate data are now available to the LGC and to cabin displays. Besides shaft and trunnion angles, angle rate data are available to cabin displays.

If the target is lost during LGC mode tracking, the PGNS takes control and positions the antenna to regain acquisition and automatic tracking. The NO TRACK light goes on, indicating that the RR is not locked on to the CSM. A no-track signal is supplied to the caution and warning electronics assembly, which turns the RNDZ RDR caution light (panel 2) on. The acquisition sweep generator in the frequency tracker begins to sweep and continues sweeping until acquisition is achieved and angle tracking is restored. When ranging circuits in the electronics assembly achieve lock-on, data good discretes are restored, allowing range and range rate from the electronics assembly to be routed through the RNG/ALT MON switch (set to RNG/RNG RT) to the RANGE and RANGE RATE indicator (panel 1).

With the RENDEZVOUS RADAR selector switch set to SLEW, the RR is operated in the slew mode using the SLEW and SLEW RATE switches (panel 3). With the SLEW RATE switch set to HI, the slew rate is a nominal 7° per second; with the switch set to LO, the slew rate is a nominal 1.3° per second. The astronaut can slew the antenna in the direction desired by holding the SLEW switch in the required positions (UP, DOWN, LEFT, OR RIGHT) and using the proper slew rates for the periods of time required. When automatic tracking is desired, the RR must be placed in the automatic tracking mode or LGC mode.

With the RNDZ RADAR selector switch set to AUTO TRACK, placing the RADAR TEST switch in RNDZ implements the RR self-test. By use of the TEST/MONITOR switch and the SIGNAL STRENGTH indicator (panel 3), the test parameters for AGC, transmitter power, and shaft- and trunnion-angle rate are read out.

Regardless of the RR mode, the range and range rate self-test parameters are monitored on the RANGE and RANGE RATE indicators (panel 1). The astronauts can monitor antenna temperature, using the TEMP MONITOR switch and the TEMP indicator on panel 3. If antenna temperature exceeds the temperature limits (15.6° to 148.8° F), the caution and warning electronics assembly turns the HEATER caution light (panel 2) on.

The antenna is automatically changed from antenna orientation No. I to antenna orientation No. II (see figure 2.2-22) in the LGC control mode by LGC programmed slewing commands. This antenna reorientation involves a specific combination of single-axis, alternate shaft and trunnion slewing to move the antenna to within the desired tracking area. Dual-axis control commences when

the antenna is at a specific position within the tracking area for positioning of the radar beam onto the transponder target.

The PGNS gains control of the RR antenna when the LGC receives the RR power on/LGC discrete. This signal is initiated with the PGNS:RNDZ circuit breaker closed and setting the RENDEZVOUS selector switch to LGC. The LGC initializes the control loops by sending the zero RR-CDU discrete to the CDU (RR channels). This discrete clears the read counters (figure 2.2-47) in the trunnion and shaft CDU section and inhibits transmission of incrementing pulses from the read counters. The error counters are also cleared and inhibited by a logic circuit within the CDU. The LGC removes the zero discrete to enable the read counters and issues a RR-CDU error counter enable discrete to enable the error counters. The LGC then begins sending pulses representative of positioning angles to the digital-to-analog converters. The resulting analog signal is routed to the radar electronics to position the antenna. The signals are applied to the antenna-mounted gyros, which are torqued, and the resulting error signals are applied to the servomotors via servoamplifiers. Positioning data are obtained from the resolver signals. These data are applied to the analog-to-digital converter in the CDU, which converts the positional angles to digital pulses and then routes these pulses to the LGC.

In positioning the antenna to LOS with respect to the CSM, the LGC commands a maximum slew rate of 7° per second until the antenna boresight is within 7° of LOS with the CSM target. Within 7° of LOS, the antenna is slewed for acquisition of target. The antenna slew rate decreases exponentially to 1.5° per second until target is acquired. Within 7° of LOS, the LOS slew rate decrease in degrees/sec is equal to the angular error decrease in degrees/sec between the antenna position and the LOS with respect to the CSM. The slew rate decreases and finally terminates when the RR acquires the CSM target.

2.2.3.4.3 Antenna Orientation. (See figures 2.2-21 and 2.2-22.)

The attitude of the antenna axes with respect to the LM axes depends on its antenna orientation. There are two types of antenna orientation: Antenna orientation No. I and antenna orientation No. II; both can be manually initiated by the astronauts by DSKY entries. The operational and mechanical limits for antenna orientation No. I and II are depicted in figures 2.2-21 and 2.2-22, respectively.

The RR antenna performs within the operational limits with optimum accuracy and minimum degradation. The operational limits are the LGC program limits and operational slewing limits. For antenna orientation No. I and II, the antenna mechanical limits are the shaft and trunnion mechanical coverages, stated in boresight angular degrees, from one hardstop to the other hardstop. The shaded areas on the vector diagrams of the figures denote the mechanically restricted areas. The actual mechanical limits are 90° counterclockwise from the boresight mechanical limits. The hardstops act against the trunnion and shaft axis. The hardstops prevent damage to the RR antenna and LM body.

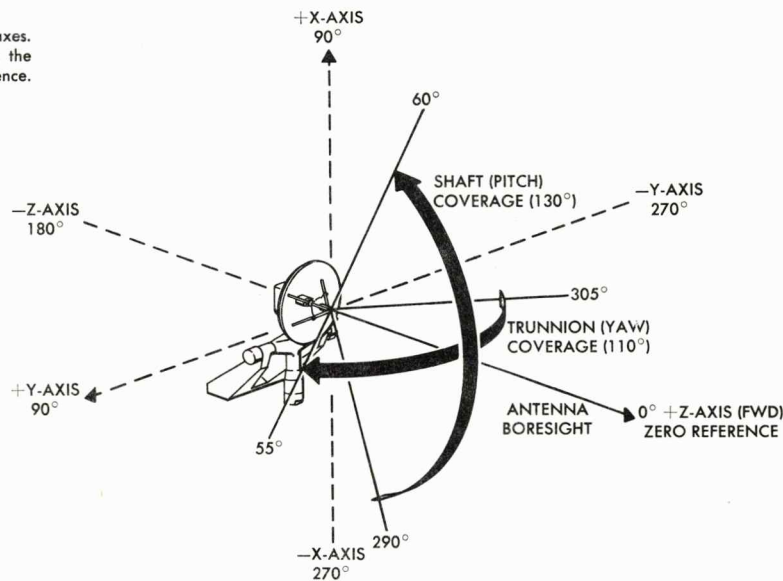
Antenna Orientation No. I. This orientation is used for transponder tracking during rendezvous operation. With reference to the +Z axis, orientation No. I provides LOS operational shaft coverage from -70° to $+60^\circ$ and LOS operational trunnion coverage from -55° to $+55^\circ$.

Antenna Orientation No. II. This is an overhead orientation, used during lunar stay (lunar mission) and subsequent rendezvous. With respect to the +Z axis, orientation No. II provides LOS operational shaft coverage from $+45^\circ$ to $+155^\circ$ LOS operational trunnion coverage from -55° to $+55^\circ$ is with respect to the X-axis. In orientation No. I, the antenna is rotated 180° about the trunnion axis to obtain the required angle coverage.

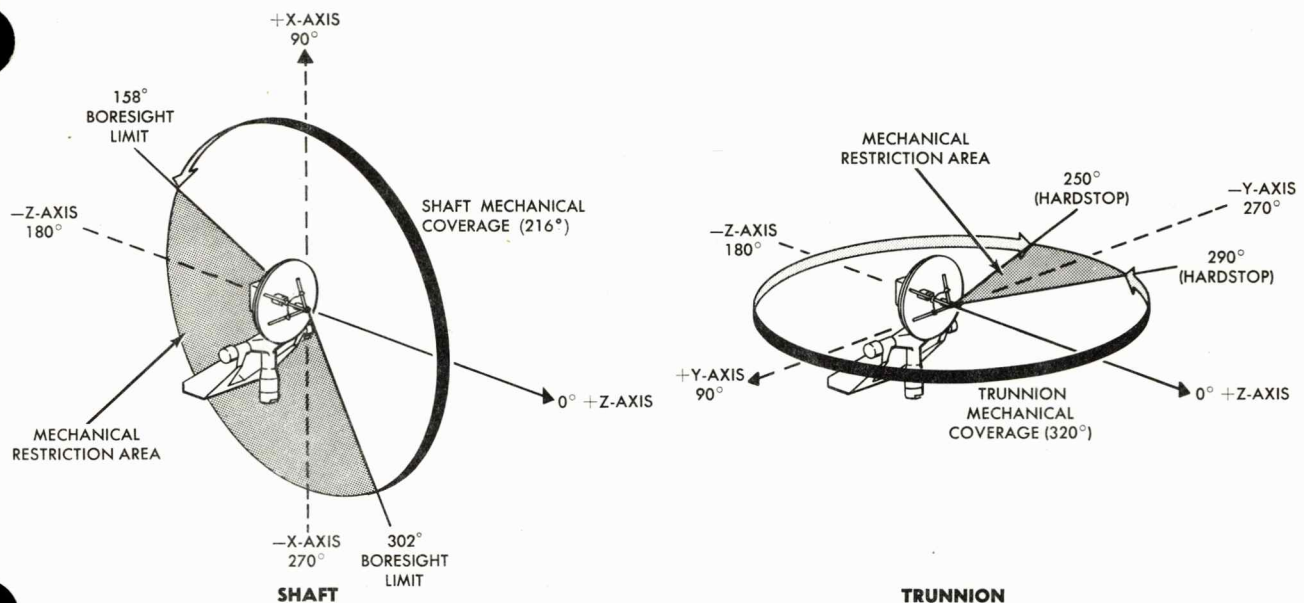
2.2.3.5 RR Transponder. (See figure 2.2-23.)

The RR transponder consists of an antenna assembly and an electronics assembly. The electronics assembly contains the following functional subassemblies: microwave, receiver, carrier lock loop, tone modulation detector, and power supply. The transponder antenna receives the 9832.8-mc phase-modulated signal transmitted from the RR. The received signal is routed through a waveguide to the microwave subassembly. A unidirectional circulator in the microwave subassembly transfers the received signal to a mixer, where it is mixed with an internally generated 9792-mc signal. The output of the mixer is the difference frequency of 40.8 mc, which is fed to the receiver subassembly.

- Notes:
 1. The X, Y and Z axes shown are the LM axes.
 2. All positions defined in degrees use the
 + Z-axis (FWD) as zero (0°) reference.



ANTENNA BORESIGHT COVERAGE (OPERATIONAL LIMITS)

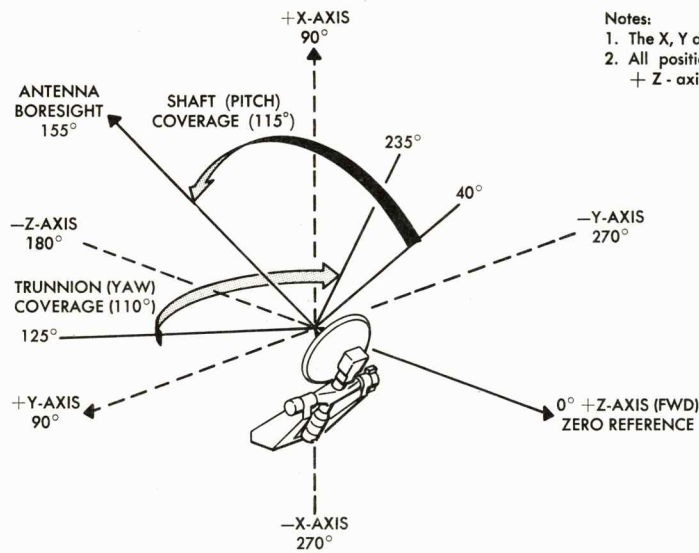


ANTENNA COVERAGE (MECHANICAL LIMITS)

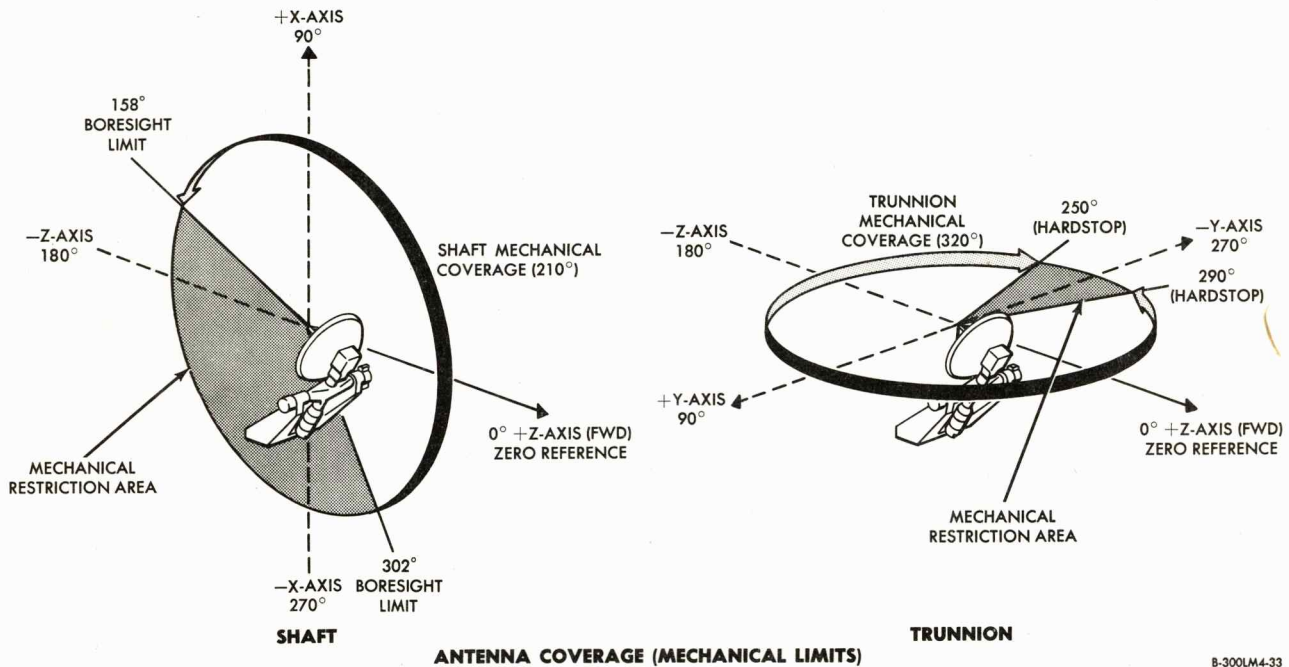
B-300LM4-32

Figure 2.2-21. Rendezvous Radar - Antenna Orientation No. 1

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



ANTENNA BORESIGHT COVERAGE (OPERATIONAL LIMITS)

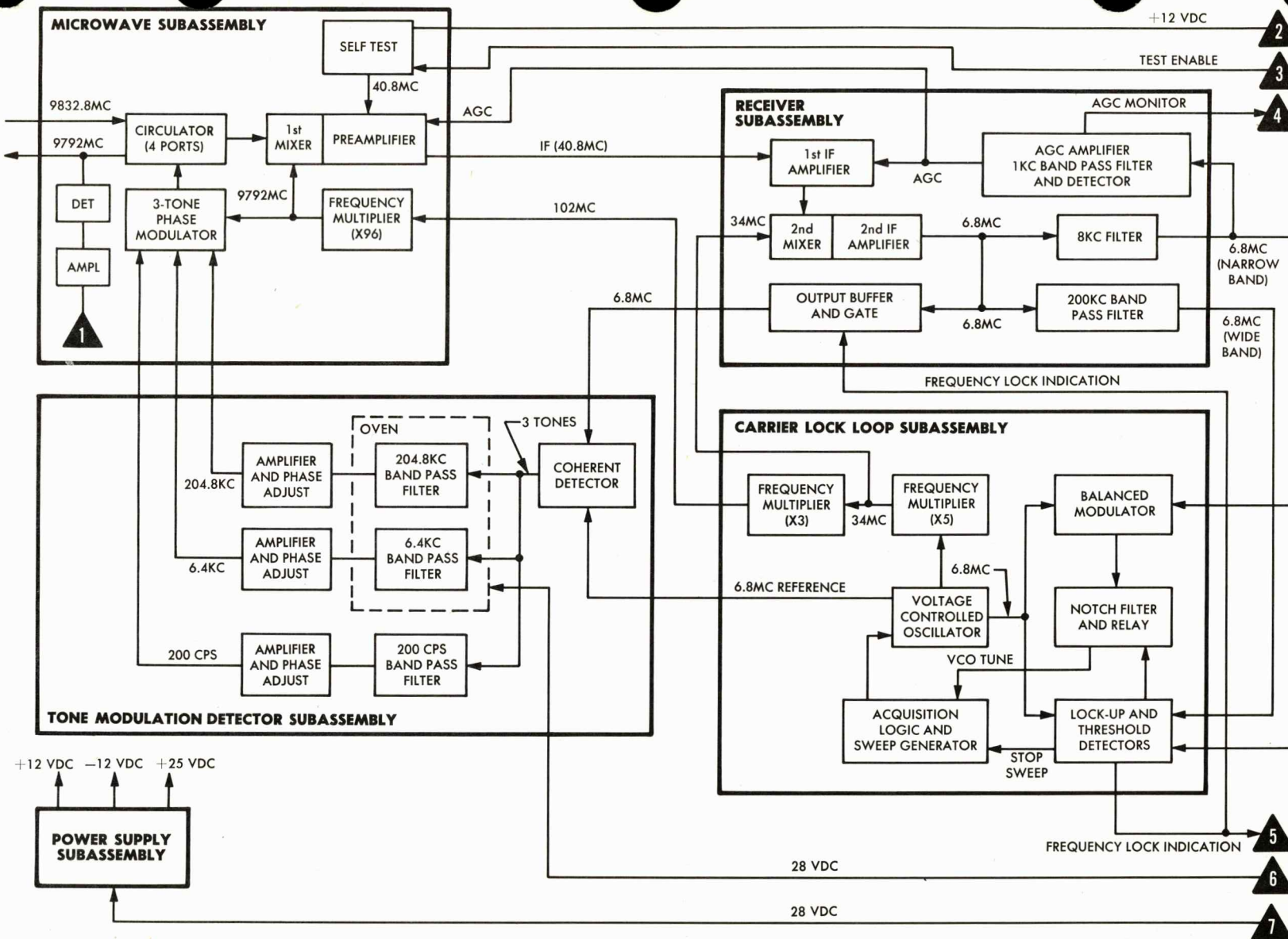


ANTENNA COVERAGE (MECHANICAL LIMITS)

B-300LM4-33

Figure 2.2-22. Rendezvous Radar Antenna Orientation No. II

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM



Note:  For all cross references see figure 2.2-24

Figure 2.2-23. Transponder Electronic Assembly - Block Diagram

A-300LM4-250

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The receiver subassembly mixes the 40.8-mc signal generated in the carrier lock loop subassembly, to provide a tone-modulated difference frequency of 6.8 mc. This signal is applied to a coherent detector in the tone modulator detector subassembly, along with a 6.8-mc reference signal from a VCO in the carrier lock loop subassembly. Only two range tones, either 200 cps and 6.4 kc or 6.4 kc and 204.8 kc, are extracted by coherent demodulation. When the RR is in the process of acquiring the transponder signal, the transponder receives a RR signal which is unmodulated. In RR tracking, the transponder receives a RR signal which is modulated by the 6.4 kc and 204.8 kc or only the 204.8 kc tone, depending on the range. After the range tones are removed, they are filtered, amplified, and sent to a phase modulator in the microwave subassembly. Simultaneously the phase modulator receives a 9792-mc unmodulated signal, derived from the 6.8-mc reference signal in the carrier lock loop subassembly by frequency multiplication.

The phase modulator of the microwave subassembly modulates the transponder transmitter 9792-mc signal with the same range tones from the tone modulation detector subassembly, that were previously removed from the received RR signal. The two range tones are kept phase coherent with the two range tones of the received RR signal. The transponder phase-modulated transmitter signal goes through a circulator to the antenna, where it is propagated for interception by the RR.

2.2.3.5.1 Modes of Operation.

The transponder operates in three basic modes: transponder, beacon and self-test.

Transponder Mode. The transponder operates in this mode during the time it is receiving RR transmitter signals. In this mode, the transponder provides continuous-wave, phase-modulated signal whose frequency (9792 mc) is 40.8 mc lower than that of the RR transmitted signal. The transponder phase-modulating tones are the same as the RR phase-modulating tones: 200 cps, 6.4 kc, and 204.8 kc.

Beacon Mode. The transponder is in this acquisition mode when it is operational and no signal is received from the RR. A sweep generator and an acquisition logic portion of the carrier lock loop subassembly cause a frequency sweep scan of ± 104 kc about the transponder nominal frequency (9792 mc). When the transponder antenna receives RR transmitter signals that exceed minimum signal strength, a threshold detector in the carrier lock loop subassembly provides a d-c level to stop the sweep generator and permit tracking to take place.

Self-Test Mode. The self-test mode enables the astronauts to check the transponder performance. The self-test mode is initiated by application of an external self-test signal. This test verifies frequency lock-on and AGC response at a 200-nm simulated range. A self-test oscillator (see figure 2.2-23) in the microwave subassembly provides a 40-mc test signal for the 1st mixer preamplifier. When the 40-mc test signal is applied, the 1st mixer stage is bypassed. When the self-test mode is not used, the self-test oscillator is deenergized.

2.2.3.5.2 General Operation. (See figure 2.2-24.)

After it is energized and placed in operation, the transponder performs automatically in conjunction with the RR. Transponder turn-on, monitoring and self-test capabilities are provided by the CSM cabin controls and indicators. Figure 2.2-24 shows the transponder signal flow through switches and indicators and the location of these switches and indicators in the CSM cabin.

To place the transponder in operation, the RNDZ XPNDR FLT BUS circuit breaker is closed, then the RNDZ XPNDR switch is set to HEATER. This action feeds 28 volts dc to the oven in the tone modulation detector subassembly. The oven brings the 204.8-kc and 6.4-kc bandpass filters to the operating temperature ($+160^{\circ}\text{F}$). A waiting time of 1,000 seconds is necessary before the RNDZ XPNDR switch should be set to PWR. This minimum waiting time prevents degradation in RR range accuracy due to temperature. With the switch set to power, 28 volts dc is routed to the power supply subassembly.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

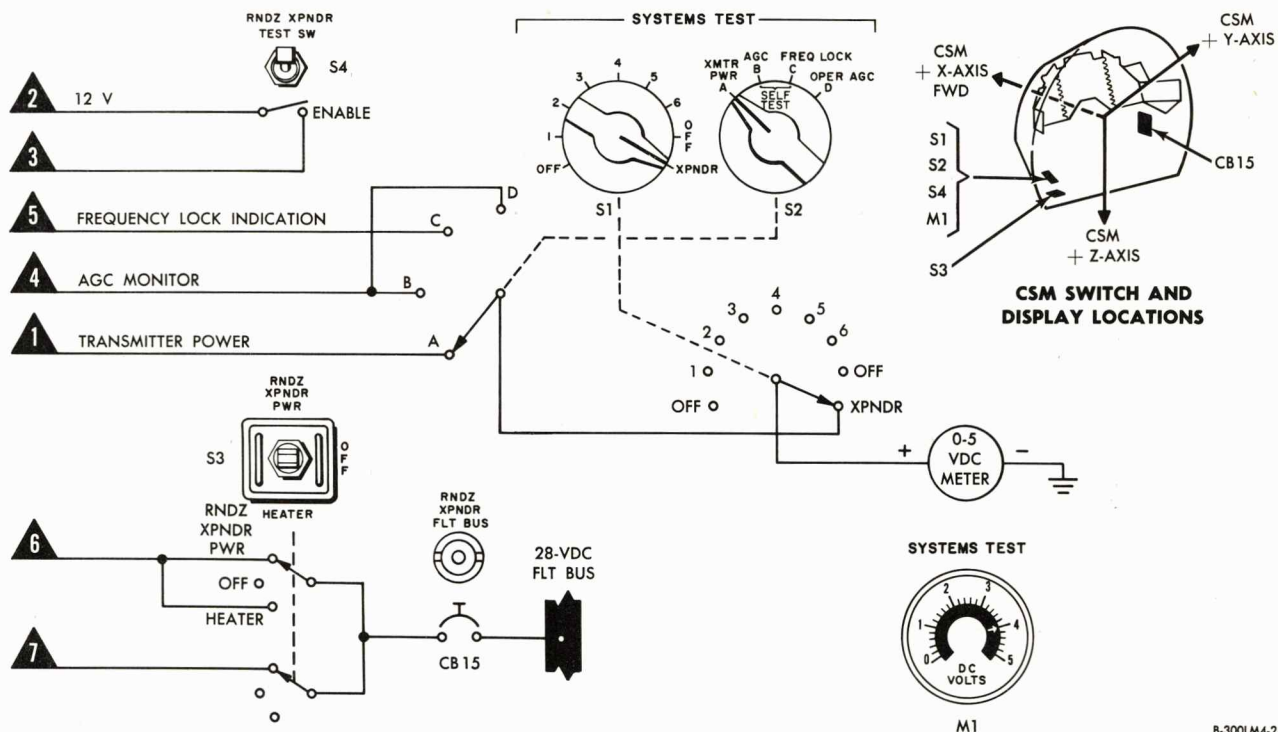


Figure 2.2-24. Transponder-Functional Flow Diagram

The SYSTEM TEST switches (S1 and S2) and the SYSTEM TEST dc voltmeter permit monitoring of transmitter power, AGC and frequency lock. The proper dc voltage readouts for those parameters are:

<u>SYSTEM TEST Switch (S2)</u>	<u>Parameters</u>	<u>DC Voltage Readout</u>
Position A	Transmitter Power	2.7 volts
Position B	AGC	2.0 to 3.5 volts
Position C	Frequency Lock	3.5 to 4.5 volts

At the output of the microwave assembly, a power monitor provides a d-c readout proportional to the transmitter output power.

The astronauts use the RNDZ XPNDR TEST SW switch to initiate the self-test enable signal for transponder self-test. This signal energizes a self-test oscillator in the microwave subassembly (see figure 2.2-23) and bypasses the 1st mixer stage. The astronauts then verify the transponder performance by using the SYSTEM TEST switches and SYSTEM TEST dc voltmeter to monitor for the proper dc voltage indications as designated in the preceding paragraph.

Figure 2.2-25. Deleted

2.2.3.6 Landing Radar. (See figure 2.2-26.)

During the LM-4 earth-orbital mission, the LR, designed for a lunar-landing mission, is checked on its ability to reject spurious signals, detected by its antenna four-separate, highly directional arrays. These spurious signals are likely to emanate, or be reflected, from the LM body or engine plumes. The four receiving arrays are structurally aligned to detect incoming spurious signals within their separate, narrow-beamwidth, RF fields-of-view.

By severing or rerouting certain internal signal-flow lines (figure 2.2-26), the LR is modified as three velocity beam channels and an altimeter beam channel for the four-separate, receiving arrays of the LR antenna. Spurious signals, not rejected by dual preamplifiers (LR antenna assembly) or dual AF amplifiers (LR electronics assembly), are processed by the LR electronics assembly into LGC computer words. By keying the DSKY, these computer words are telemetered on the LGC downlink to the MSFN.

During a lunar-landing mission, the frequency trackers enable velocity and range data-good discretes to the LGC, when they are locked on to discrete Doppler frequencies at the centers of four received microwave beams (three velocity and one altimeter). Because the frequency trackers are not likely to track anything during the LM-4 mission, the tracker lock-on discrete lines (TL1, TL2, TL3, and TLR) are severed and +4 volts dc is used to simulate tracker lock-on discretes to tracker lock-on (TL) logic gates. These enabled logic gates then send range and velocity data-good signals to the LGC to enable normal LGC acceptance of data.

During a lunar-landing mission, the frequency deviation rate of the altimeter transmitter signal (9.58 gc) is changed from $\pm 4\text{mc/sec}$ to $\pm 20\text{mc/sec}$ when the LM altitude is 2,500 feet or less with respect to the lunar surface. Because the change in frequency deviation rate would compromise the spurious signal frequencies that may be acquired and tracked by the range frequency tracker, the frequency deviation control line to the frequency modulator is severed. The altitude scale factor line, that would provide the LGC with a low-altitude indication, is also severed. The blanking-signal line from the frequency modulator to the range frequency tracker is severed, because the blanking signal is a function of the frequency-deviation control signal.

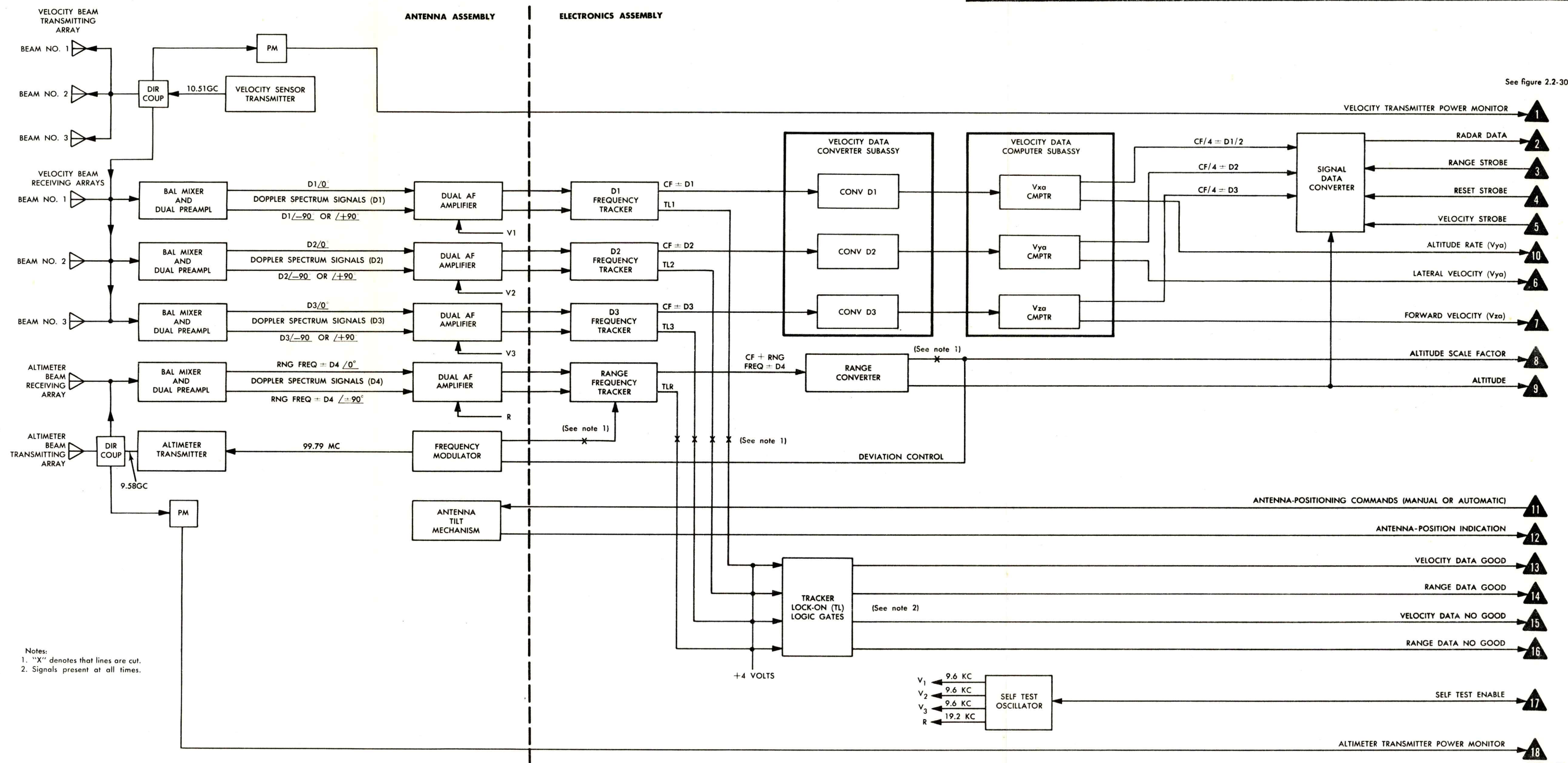


Figure 2.2-26. Landing Radar - Block Diagram

During a lunar-landing mission, several frequency tracker outputs are fed in different combinations to the three computers within the velocity data computer subassembly. From these different combinations of frequency tracker outputs, the computers derive the LM altitude rate and LR antenna coordinate velocities, forward and lateral, with respect to the LM axes. For the LM4 mission, the inputs to the three computers are wired so that each computer will derive a computer word, in PRF form, from one of the three velocity frequency tracker outputs (D1, D2, and D3). Computer word Vxa is derived from the output of the D1 frequency tracker, which scans any spurious signal on returning velocity beam No. 1; computer word Vya, from the D2 frequency tracker, which scans, velocity beam No. 2; computer word Vza, from the D3 frequency tracker, which scans velocity beam No. 3.

When a receiver channel rejects all spurious signals received by its corresponding receiver array, the frequency tracker output, fed to the computer in the same receiver channel, is a center frequency ($CF=614.4$ kc), varied across a predictable frequency range above and below the normal CF. When a receiver channel allows a frequency tracker to acquire and track a spurious signal, the computer receives, from the frequency tracker output, the CF plus offset frequencies. These offset frequencies are either the same as, or proportional to, the spurious signal frequencies. Therefore, when a receiver channel rejects all spurious signal frequencies, the frequency of the corresponding computer word varies in a predictable manner. When a frequency tracker in a receiver channel is allowed to acquire and track a spurious signal, the corresponding computer word represents, in PRF form, the spurious signal frequencies. When the contents of a computer word varies in a predictable manner, this indicates that the corresponding receiver channel has rejected all spurious signals from its receiving array. When the contents of a computer word remain relatively fixed at a constant quantity, this indicates that corresponding receiver channel has permitted a spurious signal to be tracked by its frequency tracker. This relatively quantity represents the spurious signal frequencies in PRF form.

The three velocity computer outputs (Vxa, Vya, and Vza) and the range computer output are fed to the signal data converter. The velocity computer outputs are represented by these terms: $Vza=CF/4\pm D1/2$, $Vya=CF/4\pm D2$, and $Vza=CF/4\pm D3$. The symbol D denotes frequency tracker output offset frequencies. The signal data converter puts each computer word into the LGC 15-bit serial word format. In response to commulative velocity and range strobe commands from the LGC, the signal data converter shifts out, in sequence, each computer word (Vxa, Vya, Vza, and altitude) to the LGC. The contents of computer word Vxa represent the output of the velocity beam No. 1 receiver channel; the contents of computer word Vya, the velocity beam No. 2 receiver channel; the contents of computer word Vza, the velocity beam No. 3 receiver channel; and the computer word for altitude (range), the altimeter beam receiver channel.

The astronauts can program the LR spurious test, also known as the RF view factor test, via the DSKY by keying V78E (VERB, 7, 8, and ENTR). Keying V78E sends the computer words, representing the four receiver channel outputs, to the IS. From the IS, the computer words are telemetered, on the LGC downlink, to MSFN. Velocity beams No. 1, 2, and 3, and the altimeter beam receiver channels, as computer words, are sampled at a computer word/sec rate for the duration of the LR spurious test routine. The astronauts terminate this test routine by keying V79E into the DSKY. The LM displays do not provide the astronauts with the capability of monitoring the LR spurious signal test.

In the lunar-landing configuration, LR senses the velocity and altitude of the LM relative to the lunar surface by means of a three-beam Doppler velocity sensor and a radar altimeter. Velocities are calculated by measuring the Doppler shift along three beams of microwave energy. LM altitude from the lunar surface is derived from slant range. Slant range is calculated from the frequency-deviation difference between the altimeter transmitted and received signals, resulting from frequency modulation of the transmitted signal and the time delay between the transmitted and received signals. Velocity and range data are made available to the LGC as 15-bit binary words. The velocity data are also made available to LM displays as d-c analog voltages, and the range data, to LM displays as high- or low-altitude, pulse repetition frequencies (PRF).

The LR antenna consists of an antenna assembly and an electronic assembly. During a lunar-landing mission, the antenna assembly forms, directs, transmits, and receives four microwave beams (three velocity and one altimeter), reflected from the lunar surface. Transmission frequency for the three velocity beams is 10.51 gc; for the altimeter beam, 9.58 gc. Two interlaced phased arrays (see figure 2.2-52) transmit the three velocity beams and one altimeter beam, and four broad-side arrays receive these microwave beams. Concurrently, the electronics assembly searches, tracks,

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

processes, converts, and scales the Doppler and FW/CW returns, which provide the velocity and slant range data for the LGC and LM displays.

2.2.3.6.2 General Operation. (See figure 2.2-27.)

Using the LM controls and indicators, the astronauts can monitor the LR critical voltages and temperatures, apply power to heat the antenna and power to the LR, initiate LR self-test, and place the antenna in position No. 1 (descent) or position No. 2 (hover). The LR self-test permits operational checks of the LR without radar returns from external sources.

The HEATERS: LDG RDR circuit breaker (panel 11) is closed at the launch pad and left closed for the mission duration. This circuit breaker activates an antenna temperature control which applies power to antenna heaters to protect antenna components against the spatial cold when power is not applied to the LR. Closing the PGNS: LDG RDR circuit breaker applies power to the LR voltage power supply and to the antenna pedestal motor driver, and removes heater power from the antenna. The activated LR furnishes the antenna with 30 watts minimum of dissipated power.

The astronauts can initiate a self-test of the LR four separate receiver by setting the RADAR TEST switch (panel 3) to LDG and observing two indicators and two scales on panel 1. The indicators and scales and the corresponding receiver channels are: ALT RATE indicator, for velocity beam No. 1; LAT VEL scale of the X-pointer indicator, for velocity beam No. 2; FWD VEL scale of X-pointer indicator, for velocity beam No. 3; and ALT indicator, for the altimeter beam. The normal indications for the velocity beams No. 2 and 3 are for the LAT VEL and FWD VEL needles (X-pointer indicator) to swing and peg over at predictable oscillatory rates. When the LDG ANT switch is set to DES, the readout on the ALT indicator should be approximately 8276 feet; ALT RATE indicator readout, -450 fps. When the LDG ANT switch is set to HOVER, the ALT indicator readout should be approximately 7994 feet; the ALT RATE indicator readout, -450 fps.

The LR is first turned on and self-tested for 5 minutes, during LM checkout before separation from the CSM. The self-test is accomplished by activating the LR with the PGNS; LDG RDR circuit breaker, waiting 2 minutes for adequate warmup, and initiating self-test by setting the RADAR TEST switch on panel to LDG. An initiated self-test bit command activates self-test circuits, which apply simulated Doppler signals (9.6 kc) to the LR velocity sensors and simulated lunar range signals (19.2 kc) to the altimeter sensor.

To monitor the LM simulated forward and lateral velocities with the X-pointer indicator on panel 1, switches on panel 1 are placed in these positions; MODE SEL to LDG RADAR, RATE/ERR MON to LDG RDR/CMPTR, and X-POINTER SCALE to either HI MULT or LO MULT. The HI MULT position provides a 200-fps scale and lights the X10 multiplier placard on the X-pointer indicator. The LO MULT provides a 20-fps scale and lights the X1.0 multiplier placard on the X-pointer indicator. To monitor the LM simulated altitude and altitude rate, the RNG/ALT MON switch on panel 1 is placed in ALT/ALT RT and the MODE SEL switch is left in LDG RADAR.

Altimeter transmitter and velocity transmitter power is monitored by placing the TEST/MONITOR selector switch on panel 3 to ALT XMTR and VEL XMTR and observing the SIGNAL STRENGTH meter. A half-scale reading indicates an acceptable minimum power output level. LR antenna temperature is monitored by setting the TEMP MONITOR selector switch to LDG and observing the TEMP indicator for an indication within the operating temperature range of +50° to +150°F. When the antenna temperature exceeds +148.8°F or drops below 15.6°F, the HEATER caution light on panel 2 goes on.

2.2.3.6.3 Antenna Positions. (See figure 2.2-28.)

The LR antenna is oriented to antenna position No. 1 (powered descent) or 2 (hover). Antenna position No. 1 orients the antenna boresight 24° with respect to the LM X-axis. In antenna position No. 2, the antenna boresight is parallel to the LM X-axis and perpendicular to the Z-axis; the angle between the antenna boresight and the altimeter beam is 21°.

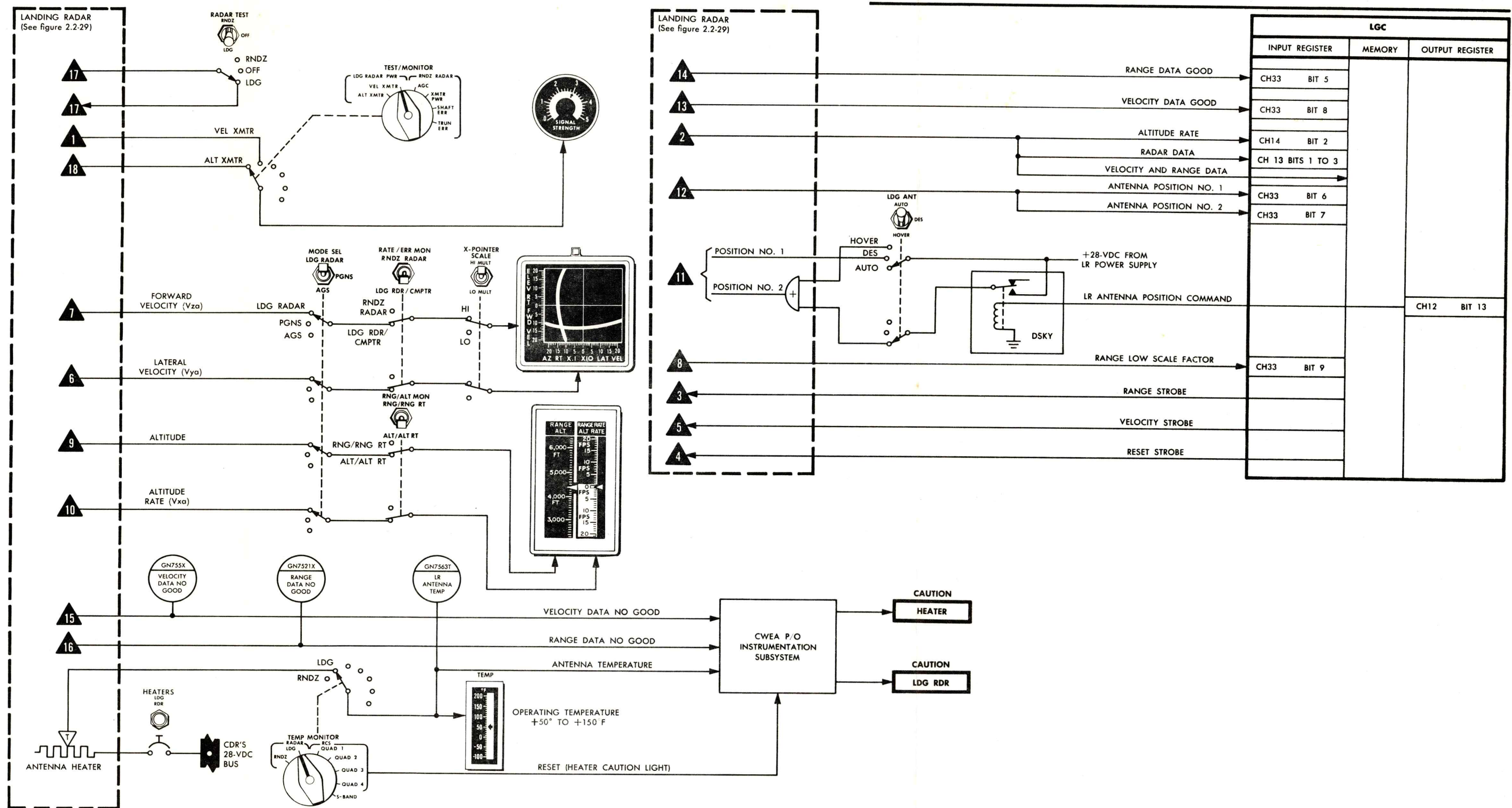


Figure 2.2-27. Landing Radar - Functional Flow Diagram

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.2-101/2.2-102

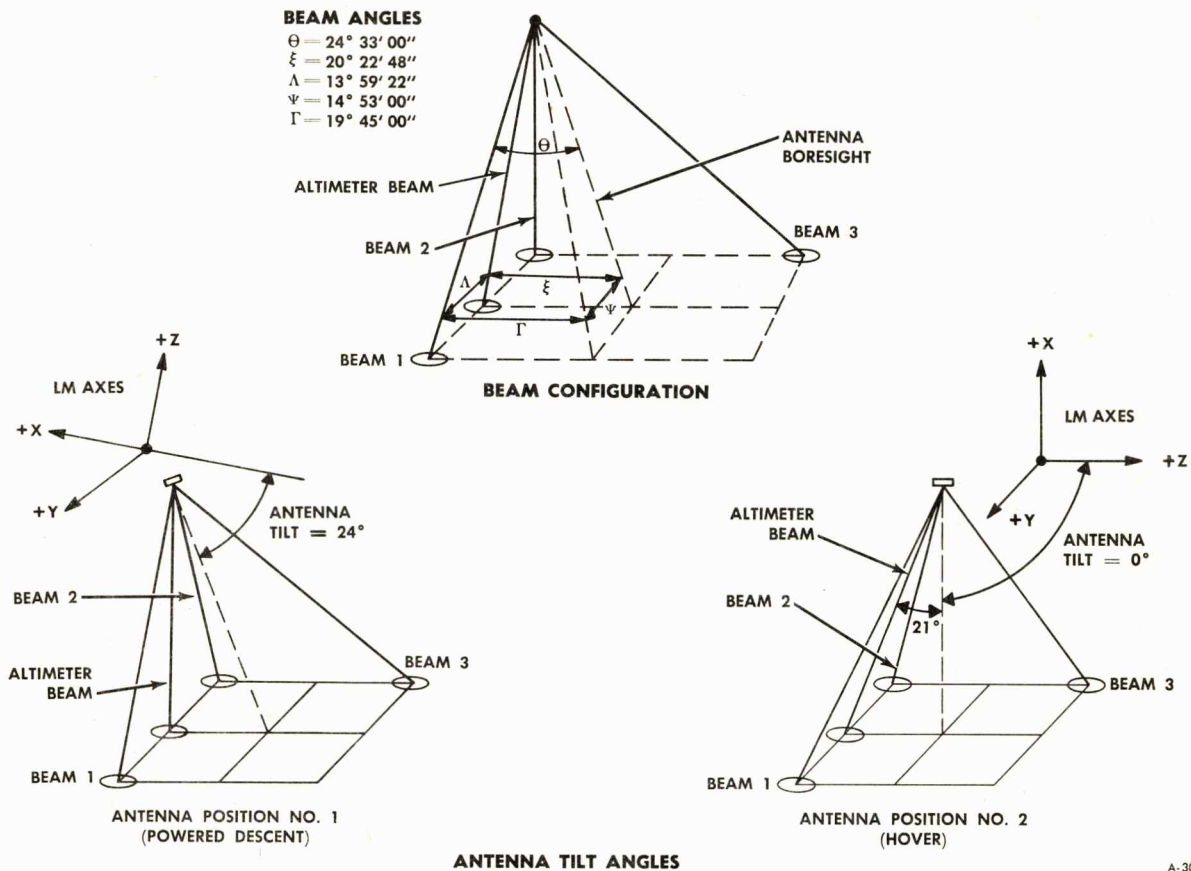


Figure 2.2-28. Landing Radar Antenna Beam Configurations

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

During the LM-4 mission, the beam configurations of antenna positions No. 1 and 2 are used to detect spurious signals that may reflect or emanate from LM body or engine plumes.

In a lunar-landing mission, the antenna beam configuration provides the LR with velocity Doppler and range information relative to LM and the lunar surface. The Doppler shift along each velocity Doppler beam (D1, D2 and D3) is directly proportional to velocity between the LM and the lunar surface. A closing velocity beam conveys positive Doppler sense; an opening velocity beam, negative Doppler sense. By linearly frequency-modulating the altimeter beam carrier frequency, the frequency-deviation difference between the transmitted and received altimeter beams becomes the range frequency with Doppler component. The Doppler component is removed later, leaving a Doppler compensated range frequency which is proportional to the beam transit time.

The Doppler velocity sensor transmitting array (figure 2.2-52), consisting of parallel waveguides with E-plane slots, forms and propagates the velocity beam microwave energy for detection by three-separate Doppler velocity sensor (DVS) receiving arrays. Each receiving array is structurally positioned for a maximum gain in the field of view coinciding with the reflected beam of microwave energy. The altimeter beam transmitting array consists of parallel waveguides with H-plane slots. Each returning velocity two-way beam width is 3.7° E-plane and 7.3° H-plane. The returning altimeter two-way beam width is 3.9° E-plane and 7.5° H-plane.

The LR antenna position is determined by the setting of the LDG ANT switch (panel 3). Antenna position No. 1 is selected with LDG ANT switch (figure 2.2-27) placed in DES position and antenna position No. 2 with the switch in HOVER position. In a lunar-landing mission, the switch is set to AUTO so that the LGC can automatically command the antenna to position No. 2 when LM executes its "High Gate" pitch-up maneuver at 12,000 to 10,000 feet above the lunar surface. The antenna must be in position No. 1 before the LGC can command it to position No. 2.

With the LDG ANT switch set to AUTO, the LGC commands the LR antenna to antenna position No. 2 when the LM pitch indicates that the LM is ready to execute the high gate pitch-up maneuver. The LR antenna position command energizes a DSKY relay solenoid thereby applying an antenna position No. 2 discrete to a pedestal motor switch which switches a pedestal motor driver to the position No. 2 activate command line. From the antenna pedestal motor driver, a current-amplified signal is routed through a closed cam-limit switch to drive the tilt-actuator (d-c reversible motor). The driven motor tilts the LR antenna to position No. 2.

When the antenna reaches antenna position No. 2, a cam mechanism cuts off the motor drive signal by opening the position No. 2 activate command limit switch. Any movement of the antenna, out of position, closes a cam-limit switch, causing the LR antenna to be driven back. In antenna position No. 2, only the position No. 1 activate command and position No. 2 indication switches are closed. A position No. 2 indication discrete is sent to the LGC which uses this discrete to transform radar data into inertial coordinates for guidance computations. Only an antenna position No. 1 command can actuate the tilt-actuator motor to drive the LR antenna to antenna position No. 1.

The LR antenna is commanded manually by setting the LDG ANT switch to DES or HOVER. In the DES position, an antenna position No. 1 antenna command causes the tilt-actuator motor to drive the antenna to position No. 1. Only the position No. 2 activate command and position No. 1 indicator cam switches are closed. A position No. 1 discrete is sent to the LGC which uses this discrete to transform radar data into inertial coordinates for guidance computations. Only an antenna position No. 2 command can actuate the tilt-actuator motor to drive the LR antenna to antenna position No. 2. In the HOVER position, the antenna position No. 2 discrete causes the tilt-actuator motor to drive the antenna to position No. 2.

2.2.3.7 Control Electronics Section. (See figure 2.2-29.)

The CES processes attitude and translation signals when operating in the primary guidance mode or the abort guidance mode.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

When operating in the primary guidance mode, the CES performs the following functions:

- Converts RCS commands to the required electrical power to operate the RCS solenoid valves
- Accepts discrete (on and off) descent engine gimbal commands and, upon receipt of an on command, causes the descent engine to move about its gimbal axes
- Accepts LGC and manual engine on and off commands and routes them to the MPS to fire or stop the descent or ascent engine
- Accepts LGC and manual thrust commands to throttle the descent engine 10% to 92.5% of its range
- Provides manual attitude and translation commands to the LGC.

When operating in the abort guidance mode, the CES performs the following functions:

- Accepts attitude error signals from the AGS, or manual attitude rate commands from the ACA's and rate-damping signals from the RGA, and fires the RCS TCA's to achieve attitude control
- Accepts manual translation commands and fires the TCA's to accelerate the LM in the desired direction
- Automatically gimbals the descent engine for trim control in accordance with signal polarity

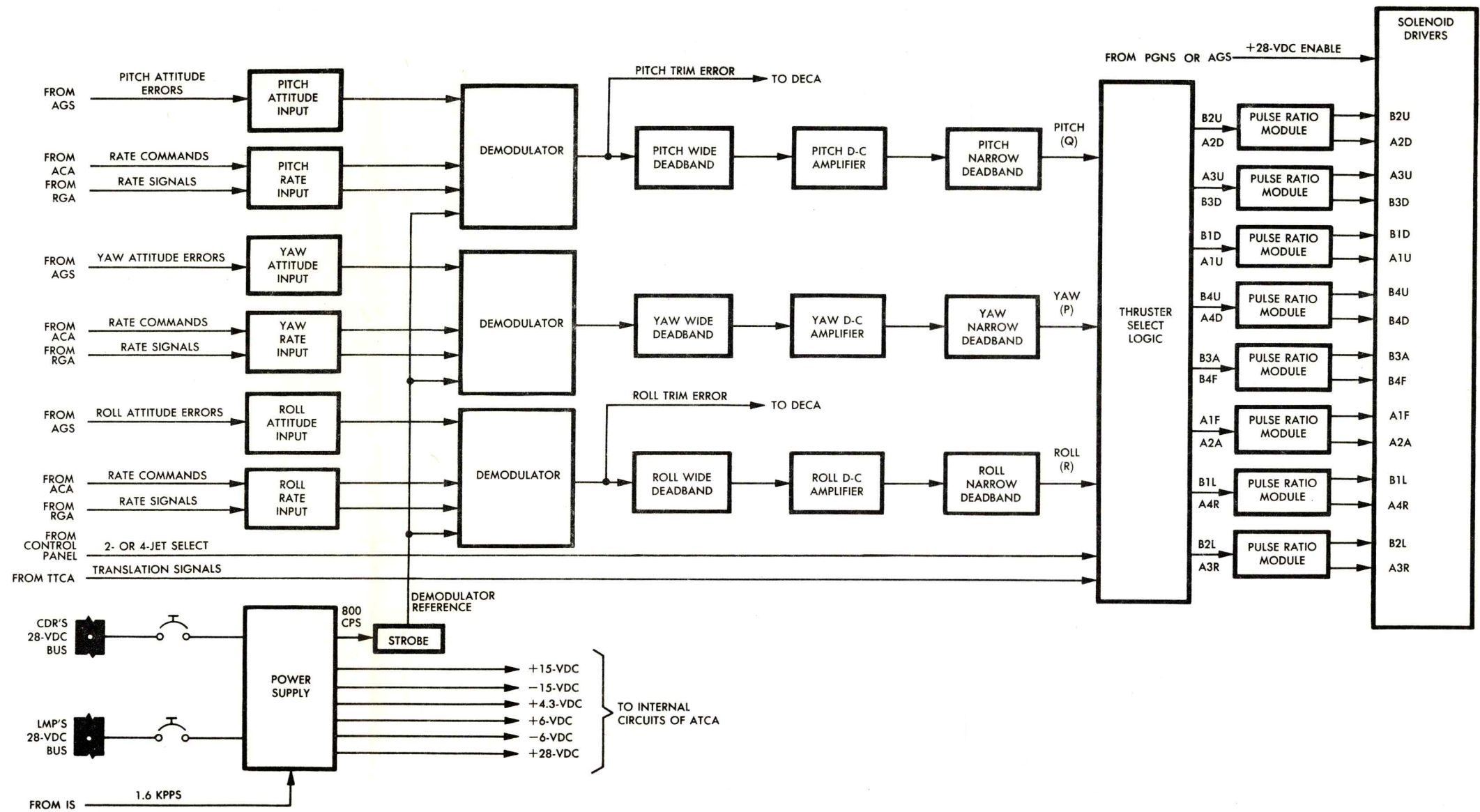


Figure 2.2-32. Attitude and Translation Control Assembly - Simplified Block Diagram

Trim Control. Descent engine trim is controlled automatically during the primary guidance and abort guidance modes, to compensate for center-of-gravity offsets during descent engine operation. In the primary guidance mode, the LGC routes trim on and off signals in two directions, for each gimbal axis, to the DECA. These signals operate power control circuitry, which drives the GDA's. In the abort guidance mode, Y- and Z-axis error signals that drive the GDA's are routed from the ATCA to the DECA. The GDA's tilt the descent engine along the Y-axis and Z-axis a maximum of +6° or -6° from the X-axis.

Ascent Engine Control. (See figure 2.2-35.) Ascent engine ignition and shutdown can be initiated by the PGNS, AGS, or the astronaut. Automatic and manual commands are routed to the S&C control assemblies. These assemblies provide logically ordered control of LM staging and engine on and off commands. The S&C control assemblies provide a positive command for fail-safe purposes if the engine-on command is interrupted. In the event of an abort stage command while the descent engine is firing, the S&C control assemblies provide a time delay before commanding LM staging and ascent engine ignition. The time delay ensures that descent engine thrusting has completely stopped before the LM is staged.

2.2.3.8 Power Distribution. (See figure 2.2-36.)

Each section of the GN&CS receives its power independently of the other sections, from the CDR's and the LMP's buses through the circuit breakers on panels 11 and 16, respectively. The flight displays associated with the GN&CS receive power from CDR's a-c and d-c buses. When power is supplied to a particular display, a power-on indicator is energized. For the X-pointer, THRUST, RANGE, and RANGE RATE indicators, the power-on indicator is a lamp; for the FDAI, a flag is used. The MISSION TIMER and the EVENT TIMER do not have power-on indicators.

2.2.3.8.1 PGNS Power Distribution.

The LGC receives 28-volt d-c primary power from the PGNS: LGC/DSKY circuit breaker. The primary power is used by power supplies within the LGC to develop +14- and +4-volt d-c power. These outputs are used for logic power within the LGC. The +14-volt d-c power supply also provides an input to the DSKY power supplies. The +4-volt d-c power supplies (2) of the LGC provide power for the standby and operate modes of LGC operation.

The standby mode of operation is initiated by a verb-noun entry on the DSKY. During standby, the LGC is put into a restart condition and the +4- and +14-volt d-c supplies are switched off. This places the LGC in a low-power mode in which only the LGC timer and a few auxiliary assemblies are operative. The DSKY power supply receives +28 and +14 volts dc and an 800-pps sync signal from the LGC. The power supply develops 275-volt, 800-cps power, which provides power for the DSKY electroluminescent displays.

The power and servo assembly (PSA) receives input power from the PGNS: LGC/DSKY, IMU STBY, and IMU OPR circuit breakers. The input voltage is 27.5 ± 2.5 volts dc, with transient limits between 24.0 and 31.8 volts dc. In addition to the d-c input from the EPS, the PSA power supplies require clock pulses (800 pps, 3.2 kpps, and 25.6 kpps) from the LGC. The PSA power supplies are as follows:

- 28-volt, 800-cps, $\pm 1\%$ power supply, 1Ø
- 28-volt, 800-cps, $\pm 5\%$ power supply, 2Ø
- -28-volt, d-c power supply
- 28-volt, 3,200-cps power supply

The 800-cps power supplies provide the PGNS with $1\% 0^\circ \emptyset$, $5\% -90^\circ \emptyset$, and $5\% 0^\circ \emptyset$ power. The 28-volt, 800-cps, 1% power supply provides the IMU resolver excitation, stabilization amplifier demodulator reference, a reference signal to the FDAI, a reference to the coupling data unit, RR resolver excitation, ACA excitation, and a reference to the SCA for demodulating 800 cps measurements. The 28-volt, 800-cps, 5% power supply provides the -90° and 0° excitation power for the gyro wheels, the IMU blowers, PIPA fixed heater power and bias heater power. The -28-volt d-c power supply provides negative d-c inputs to the d-c amplifiers used in the inertial loops and power to the three gimbal servoamplifiers in the stabilization loops and to the pulse torque assembly power supply to generate -20 volts dc for use in the accelerometer loops. The 28-volt 3,200-cps power supply provides the

IMU with 28-volt power, which is then reduced through a transformer to 2- and 4-volt levels. The power supply provides excitation voltages (2 and 4 volts) for signal ducosyn signal-generator excitation and for magnetic suspension winding excitation for the torque and signal ducosyns of the IRIG's and PIPA's. The 3,200-cps output is also used as a reference for the demodulator of the gimbal servoamplifier in all modes of operation, except the coarse-alignment mode.

The pulse torque assembly (PTA) receives input power from the PSA when the IMU OPR circuit breaker is closed. The PTA power supply is synchronized by a 12.8-kpps clock pulse from the LGC. The PTA power supply provides +28 volts d-c to the three binary current switches in the PIPA loops and to the binary current switch and d-c differential amplifier in the fine-alignment electronics associated with the stabilization loops of the IMU.

The CDU power supply receives 28-volt, 800-cps power and +28 volts dc (IMU operate signal) from the PSA, and a sync pulse from the LGC. The power supply provides a regulated output voltage for use in the CDU logic circuitry. Under full load conditions, the power supply is required to provide +4 volts dc $\pm 1\%$, at 2.5 to 3.0 amperes.

The signal conditioner assembly (SCA) receives its operating power from the PSA. The operating power includes +28 volts dc used for B+ voltage in the SCA circuits, and reference voltages consisting of 800-cps and 3,200-cps 1% feedback voltage from the IMU. Three additional reference voltages, (2.5 volts dc for bias, and 800-cps square wave, and a 3,200-cps square-wave) are generated in the SCA.

The AOT receives 115 volts ac for illumination of the reticle, from the AC BUS A and the AC BUS B: AOT LAMP circuit breakers. The heaters in the AOT receive power from the CDR's d-c bus through the HEATERS: AOT circuit breaker.

2.2.3.8.2 AGS Power Distribution.

All power (ac and dc) required by the ASA is provided by the ASA power supply, which receives 28-volt d-c power from the STAB/CONT: ASA circuit breaker (panel 16). The power supply provides regulated 28 volts dc for current regulators, +12 volts dc to bias amplifiers in the accelerometers, +4 volts dc for use in a frequency countdown subassembly and the gyros, -12 volts dc for use in the gyros, -6 volts dc for use as bias for the gyros, and -2 volts dc for use as bias in the frequency countdown subassembly. A-C voltages are provided for the accelerometer gyros and pulse torquing servo assemblies by 28 volt 800-cps inputs from the ATCA.

The AEA uses two power supplies. One operates in the standby and operates modes. It supplies power to clock countdown circuits and for the three integrating registers of the input-output subassembly. The other power supply operates in the operate mode and supplies power to the remainder of the AEA. These power supplies receive 28-volt d-c power from the STAB/CONT: AEA circuit breaker (panel 16) and 115-volt, 400-cps power from the AC BUS B: AGS circuit breaker (panel 11). They also receive 28-volt 800-cps power from the ATCA power supply for synchronization. The operate power supply provides -2, +4, +6, +13.5, +14, -13.5, and -18 volts dc.

The DEDA operating power consists of +4 and -2 volts dc supplied by the operate power supply of the AEA.

2.2.3.8.3 Radar Power Distribution.

The RR and LR receive 28-volt d-c power for the antenna heaters from the HEATERS: LDG RDR, RNDZ RDR STBY, and RNDZ RDR OPR circuit breakers (panel 11). Refer to paragraph 2.11 for a detailed description of the heaters.

The RR receives power from the CDR's d-c and a-c buses through the PGNS: RNDZ RDR and the AC BUS A: RNDZ RDR circuit breakers (panel 11), respectively. The 28-volt d-c power, applied to the RR power supply subassembly, provides regulated outputs of +26, +25, +12, -12, +6, and +4.3 volts dc. The 115-volt a-c power drives the servo electronics associated with the antenna assembly.

Incremental transmissions occur in the form of pulse bursts from the output channels to the CDU, the gyro fine alignment electronics, the RCS, and the radars. The number of pulses and the time at which they occur are controlled by the LGC program. Discrete outputs originate in the output channels under program control. These outputs are sent to the DSKY and other subsystems. A continuous pulse train at 1.024 mc originates in the timing output logic and sent as a synchronization signal to the timing electronics assembly in the IS.

The uplink work from MSFN via the digital uplink assembly (DUA) is supplied as an incremental pulse to priority control. As this word is received, priority control produces the address of the uplink counter in memory and requests the sequence generator to execute the instructions that perform the serial-to-parallel conversion of the input word. When the conversion is completed, the parallel word is transferred to a storage location in memory by the uplink priority program. The uplink priority program also retains the parallel word for subsequent downlink transmission. Another program converts the parallel word to a coded display format and transfers the display information to the DSKY.

The downlink operation of the LGC is asynchronous with respect to the IS. The IS supplies all the timing signals necessary for the downlink operation. (Refer to paragraph 2.2.2 for interface discussion.)

Through the DSKY, the astronaut can load information into the LGC, retrieve and display information contained in the LGC, and initiate any program stored in memory. A key code is assigned to each keyboard pushbutton. When a DSKY pushbutton is pressed, the key code is produced and sent to an input channel of the LGC. A signal is also sent to priority control, where the signal produces the address of a priority program stored in memory and a priority request signal. The priority request signal is sent to the sequence generator. This results in an order code and initiates an instruction for interrupting the program in progress and for executing the key interrupt No. 1 priority program stored in memory. This program transfers the key code temporarily stored in an input channel, to the central processor, where it is decoded and processed. A number of key codes are required to specify an address or a data word. The program initiated by a key code also converts the information from the DSKY keyboard to a coded display format. The coded display format is transferred by another program to an output channel of the LGC and sent to the DSKY for display. The display is a visual indication that the key code was received, decoded, and processed properly by the LGC.

Timer. The timer generates the timing signals required for operation of the LGC. It is the primary source of timing signals for all subsystems.

The master clock frequency, generated by an oscillator, is applied to a clock divider logic circuit. The clock divider logic divides the master clock frequency into gating and timing pulses at the basic clock rate of the LGC (1.024 mpps). This basic clock rate is also applied to a scaler and a time pulse generator. The scaler further divides the output of the clock divider logic into pulses and signals which are used for gating, for generating rate signal outputs, and for accumulating time. The time pulse generator produces a recurring set of time pulses which define a specific memory cycle during which access to memory and data flow take place within the LGC.

Sequence Generator. The sequence generator executes the instructions stored in memory, processes instruction codes and produces control pulses that regulate data flow of the LGC. The control pulses control the operations assigned to each instruction and the data stored in memory.

The sequence generator consists of an order code processor, a command generator, and a control pulse generator. The sequence generator receives order code signals from the central processor and priority control. These signals are coded by the order code processor and supplied to the command generator. Another set of control pulses are used for gating the order code signals into the sequence generator at the end of each instruction. The command generator decodes the input signals and produces instruction commands which are supplied to the control pulse generator.

The control pulse generator receives 12 time pulses from the timer. These pulses occur in cycles and are used for producing control pulses in conjunction with the instruction commands. There are five types of control pulses: read, write, test, direct exchange, and special purpose. Information in the central processor is transferred from one register to another by the read, write, and direct exchange control pulses. The special purpose control pulses regulate the operation of the order code

processor. The test control pulses are used within the control pulse generator. Branch test data from the central processor change the control pulse sequence of various functions.

Central Processor. The central processor performs all arithmetic operations required of the LGC, buffers all information coming from and going to memory, checks for correct parity on all words coming from memory, and generates a parity bit for all words written into memory. The central processor consists of flip-flop registers; write, clear, and read control logic; write amplifiers; a memory buffer register; a memory address register; a decoder; and parity logic.

Primarily, the central processor performs operations dictated by the basic instructions of the program stored in memory. Communication within the central processor is accomplished through write amplifiers. Data flow to or from memory to the registers, between individual registers, or into the central processor from external sources. Data are placed on the write lines and are routed to a specific register or to another part of the central processor under control of the write, clear, and read logic. This logic accepts control pulses from the sequence generator and generates signals to read the contents of a register onto the write lines and to write the contents into another register of the central processor or another area of the LGC. The particular memory location is specified by the contents of the memory address register. The address is fed from the write lines into this register, the output of which is decoded by the address decoder logic. Data are subsequently transferred from memory to the memory buffer register. The decoded address outputs are also used as gating functions within the LGC.

External inputs through the write amplifiers include the contents of the erasable and fixed memory bank registers, all interrupt addresses from priority control, control pulses associated with specific arithmetic operations, and the start address for an initial start condition. Information from the input and output channels is placed on the write lines and routed to specific destinations within or external to, the central processor.

Priority Control. The priority control establishes a processing priority for operations that are performed by the LGC. Priority control is related to the sequence generator in that it controls the instructions to the LGC. The priority control processes input-output information and issues order code and instruction signals to the sequence generator and a 12-bit addresses to the central processor.

The priority control consists of start, interrupt, and counter instruction control circuits. The start instruction control circuit initializes the LGC if the program works itself into a trap, if a transient power failure occurs, or if the interrupt instruction control is not functioning properly. The LGC is initialized with the start order code signal, which not only commands the sequence generator to execute the start instruction, but resets other circuits. When the start order code signal is being used, issued, a stop signal is sent to the timer. This signal stops the time pulse generator until all essential circuits have been reset and the start instruction has been initiated by the sequence generator.

The interrupt instruction control forces execution of the interrupt instruction by sending the interrupt order code signal to the sequence generator and the 12 bit address to the central processor. There are 10 addresses, each of which accounts for a particular function that is regulated by the interrupt instruction control. The interrupt instruction control links the DSKY, telemetry, and time counters to the program. The interrupt addresses are transferred to the central processor by read control pulses from the sequence generator. The input-output circuits are the source of the DSKY, telemetry, and time counter inputs. The interrupt instruction control has a built-in priority chain which allows sequential control of the 10 interrupt addresses. The decoded interrupt addresses from the central processor control the priority operation.

The counter instruction control is similar to the interrupt instruction control in that it unites input-output functions to the program. It also supplies 12-bit addresses to the central processor and instruction signals to the sequence generator. The instruction signals cause a delay (not an interruption) in the program by forcing the sequence generator to execute a counter instruction. The addresses are transferred to the central processor by read control pulses. The counter instruction control also has a built-in priority of the 29 addresses it can supply to the central processor. This priority is also controlled by decoded counter address signals from the central processor. The counter instruction control contains an alarm detector, which produces an alarm if an incremental pulse is not processed properly.

2.2.4.1.10 Display and Keyboard. (See figure 2.2-47.)

The DSKY consists of a keyboard, display panel, condition indicators, and a relay package. The keyboard enables the astronauts to insert data into the LGC and to initiate LGC operations. Through the keyboard, the astronauts can also control ISS moding. The DSKY display panel provides visual indications of data being loaded into the LGC, LGC condition, and LGC program. The display panel also provides the LGC with a means of displaying or requesting data. The condition indicators display PGNS status and malfunctions. The controls and displays associated with the DSKY are discussed in section 3.

Keyboard. The DSKY keyboard is used to manually insert or read out LGC data. The keyboard consists of 10 numerical pushbuttons (0 to 9), two sign pushbuttons (+ and -), and seven instruction pushbuttons (ENTR, CLR, VERB, NOUN, RSET, STBY, and KEY REL). All the pushbuttons, except the STBY pushbutton, have five-bit codes associated with them; they convey information to the LGC. The STBY pushbutton is hardwired into the LGC power supplies.

Displays and Indicators. There are two types of displays on the DSKY: control displays and data displays. Each display can display any decimal character or remain blank. The indicators on the DSKY are referred to as condition lights; they represent various PGNS operating conditions. Each control display (VERB, NOUN, and PROG) can display two decimal characters or remain blank. The VERB and NOUN displays can also flash. The data displays are three separate registers, referred to as R1, R2, and R3. Each register can display as many as five decimal characters, with or without a plus or minus sign, or remain blank. Each of the 11 condition lights on the DSKY is labeled with the PGNS condition it represents; it goes on if that condition occurs. The condition lights and the conditions they represent are described in section 3.

Relay Package. The relay package consists of a relay matrix and decoding circuits.

The inputs entered from the keyboard, as well as other information, appear on the displays after processing by program. Display of information is accomplished through the relay matrix. A unique code for the characters to be displayed is formed by 15 bits from output channel No. 10 in the LGC. Bits No. 12 through 15 are decoded by the decoding circuits, and, along with bits No. 1 through 11, energize specific relays in the matrix, causing appropriate characters to be illuminated. The information displayed is the result of a key code punched in by the astronaut, or is LGC-controlled information. The display characters are formed by electroluminescent segments, which are energized by a voltage from the power supply, routed through relay contacts. Specific inputs from the PGNS are also applied, through the LGC, to certain relays in the matrix through output channel No. 10 of the LGC. The resulting relay-controlled outputs are caution signals to the PGNS.

To be supplied

Figure 2.2-47. Display and Keyboard - Block Diagram

2.2.4.2 Abort Guidance Section.

2.2.4.2.1 Abort Sensor Assembly. (See figure 2.2-48)

The abort sensor assembly comprises three gyros, accelerometers, and pulse-torquing servoamplifiers, a power supply, a frequency countdown subassembly, and temperature control circuits. Only the X-axis gyro and accelerometer are shown in figure 2.2-51; the other gyros and accelerometers receive identical signals from the frequency countdown subassembly and power supply. The accelerometers and gyros (one each for each LM axis) sense LM body axis motion with respect to inertial space. The accelerometers sense acceleration along the LM orthogonal axes. The gyros sense motion around the orthogonal axes. The gyros and accelerometers are securely fastened to each of the LM X, Y, and Z axes so that motion along or around one or more axis is sensed by one or more gyros or accelerometers

Accelerometers. Each accelerometer is used in a closed servo loop in which the sensing mass (pendulum) is displaced by acceleration along the input axes. Fluid between the pendulum and housing provides damping for the servo loop.

An inductive pickoff device of the accelerometer detects pendulum displacement. This displacement produces an a-c error signal, which is fed to a pulse torquing servoamplifier, where it is demodulated and changed to a d-c voltage. The d-c voltage is fed to a pendulum torquer, which produces a force equal and opposite to the acceleration force. The pendulum torquer output returns the pendulum to its null position. A d-c voltage proportional to acceleration is obtained from a precision resistor in series with the pendulum torquer. This d-c voltage is a direct measure of acceleration.

Gyros. (See figure 2.2-49.) The gyros are of the single-degree-of-freedom type; they can precess in only one phase of three orthogonal axes. The spin axis is the axis of rotation for the gyro spin motor. For any force exerted at right angles to the spin axis (in the plane of the input axis), a precession occurs because the gyro flywheel resists changes in the position of the plane in which it develops its angular momentum. The developed opposing force acts in a third plane and tends to cause rotation of the gyro float. Therefore, for any rotation about the gyro input axis, force is developed to cause rotation about the gyro output axis. Each gyro comprises two major assemblies: a primary structure and a spin motor assembly.

Primary Structure. The primary structure is formed by two machined castings, which support a float and a pickoff and torquer. Two sapphire endstones in the primary structure support the float pivots. Part of the primary structure is filled with a high-density viscous fluid that supports the float and completely fills the space between the primary structure and the float. A bellows at one end of the primary structure expands and contracts with changes in gyro temperature to prevent damage to primary structure. The float provides structural support for the gyro spin motor and wheel, and a frictionless support for the output pickoff. The specific gravity of the viscous fluid in the gyro changes with changes in temperature; for this reason, the temperature of the gyros must be closely maintained. An advantage of floatation is viscous damping, which provides an integrating action that makes the gyro float less susceptible to vibration and shock along its input axis and cushioning of the float against shock and vibration along its output axis.

Each gyro has thermistors mounted on its case to sense temperature. Thermistor resistance is $5,376 \pm 100$ ohms at a nominal gyro temperature of $+120^{\circ}\text{F}$; the temperature gradient, 112 ohms/ $^{\circ}\text{F}$ at 120°F . The thermistor leads are routed to the ASA test connector.

The pickoff and torquer are installed in one end of the primary structure. The pickoff is an air-core differential transformer that has a four-pole primary and a two-pole secondary winding. The torquer is an eight-pole, D'Arsonval-type torque generator.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

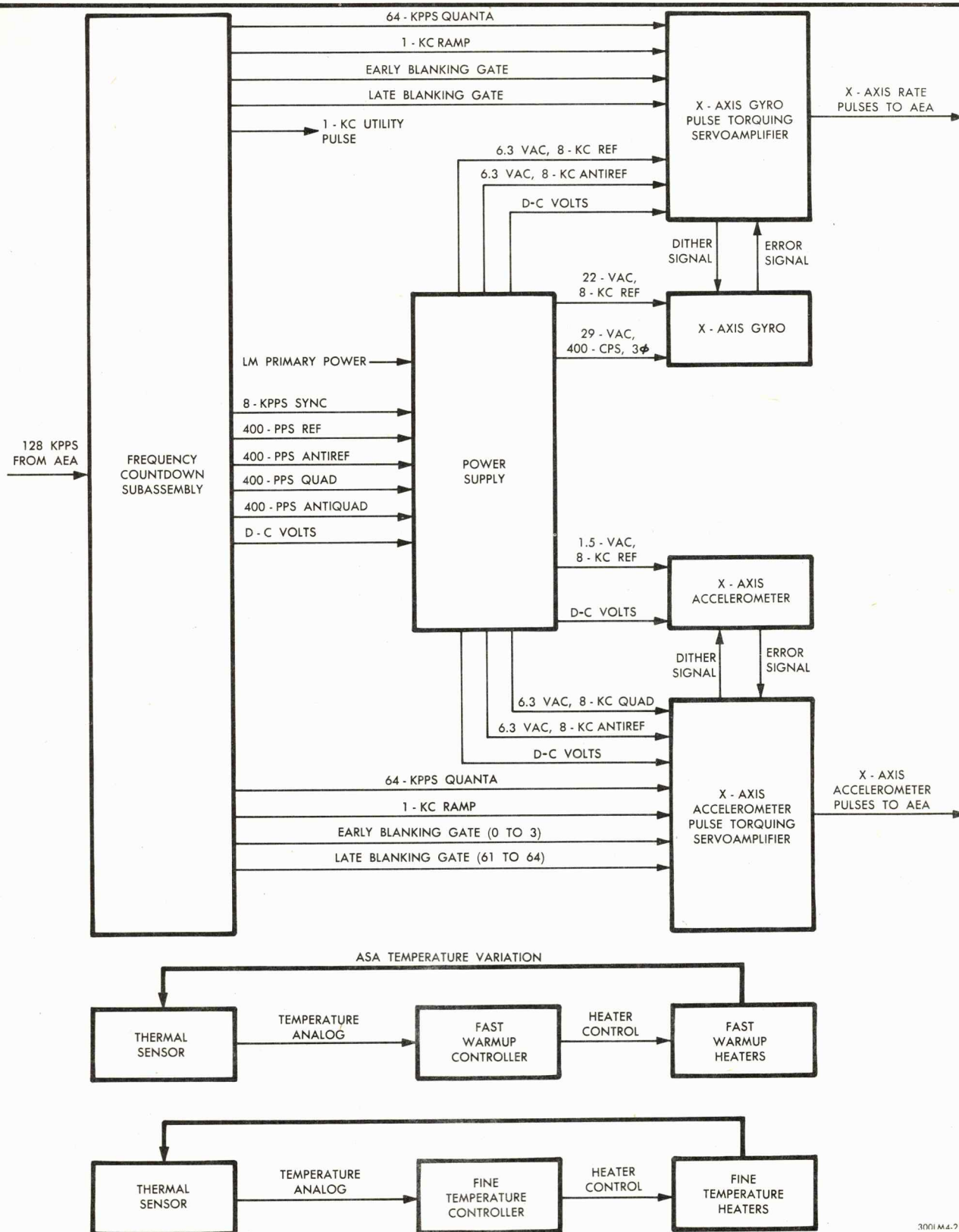
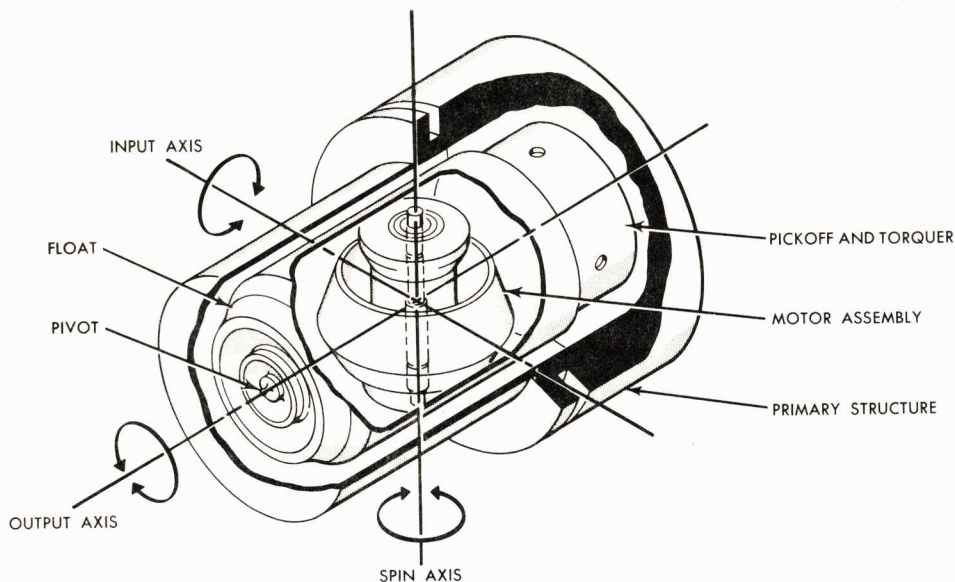


Figure 2.2-48. Abort Sensor Assembly - Detailed Block Diagram

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM



300LM4-260

Figure 2.2-49. Abort Guidance Section - Gyro Assembly

The pickoff can be considered as a transformer that has a stationary primary winding and a rotatable secondary winding. The primary, attached to the primary structure, is excited by a 22-volt, 8-kc signal from the ASA power supply. The secondary consists of an unexcited winding within the torquer core; the secondary rotates with the float. Because of its position relative to the excited primary, no voltage is induced in the secondary when the float is in its null position. When the float is rotated in either direction, the relative position of the secondary to the excited primary changes and the excited primary induces an a-c voltage in the secondary. The magnitude of this voltage is proportional to the amount of float displacement, and its phase is indicative of the direction of float displacement. This voltage is phase-demodulated by pulse torquing servoamplifier electronics, quantized, and converted to a torquing signal that is fed back to the torquer to return the float to the null position. The current flowing in the torquer, which is attached to the float and rotates with it, sets up a magnetic field that reacts with the field produced by a permanent magnet attached to the primary structure. The reaction between the magnetic field produced in the torquer coil and the magnetic field produced by the permanent magnet creates sufficient torque to return the float to its null position. Heat produced by current flow through the torquing coil adversely affects temperature-sensitive components of the gyro. The resultant errors appear in the output as displacing forces, which are minimized by the forced limit cycle mode. The forced limit cycle mode keeps the average heat around the gyro equal, which enables gyro drift to be predictable.

Spin Motor Assembly. The spin motor assembly is a four-pole, synchronous, hysteresis motor that is housed within the float. The assembly includes a wound-pole stator secured to the gyro float by stator shaft supports. Three-phase excitation of the stator produces a rotating magnetic field. The motor maintains at synchronous speed any load that it can accelerate from a dead standstill. The float chamber housing is filled with helium, which carries heat away from the motor. Each gyro has an auxiliary inductive pickoff near its motor. The pickoff is excited by a small magnet on the flywheel, each time the magnet passes the pickoff; for each revolution of the motor flywheel, a pulse is induced in the pickoff. The output from the pickoff is routed to the test connector so that, during test, motor velocity can be measured.

Frequency Countdown Subassembly. The frequency countdown subassembly controls timing functions. It receives a 1-microsecond pulse, with a repetition rate of approximately 7.8 microseconds (128 kpps), as the basic clock signal from the AEA. The subassembly processes these 1-microsecond pulses to generate signals for control of the timed functions of the other assemblies in the ASA. The following seven fundamental signals are generated by the frequency countdown subassembly:

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

- 64-kpps, 1-microsecond signal
- 1-kc ramp signal
- Early blanking gate (0 to 3 blocking gates)
- Late blanking gate (61 to 64 blocking gates)
- 8-kpps synchronizing signal
- 400-pps signals
- 1-kc utility pulse.

64-KPPS, 1-Microsecond Signal. The 64-kpps, 1-microsecond-wide signal is the fundamental digit or bit representing the quantized analog signals from the gyros or accelerometers. These pulses (hereinafter called quanta) are gated through the pulse torquing servoamplifiers by a time-modulated signal produced in the gyro or accelerometer torquing loops. The number of quanta permitted as outputs from each pulse torquing servoamplifier during each forced limit cycle is dependent on the magnitude and direction of the force displacing the gyro or accelerometer sensors and the time this force is acting. If a force providing a 50 per second displacement of the referenced axis acts for one-half second, the gyro output magnitude indicates a total change of 2.5° in the angular position of the referenced axis. Similarly, if a force providing an accelerometer displacement equivalent to 5 feet per second acts for one-half second, the magnitude of the accelerometer output indicates a change in speed of 2.5 feet per second along the referenced axis.

For the gyros and their respective pulse torquing servoamplifiers, after algebraic manipulation of the nominal quanta count of 32 pulses per limit cycle, each quantum has the weight of 0.00088049° . For the accelerometers and their respective pulse torquing servoamplifiers, each quantum has the weight of 0.003125 foot per second. Thus, for a positive output angle of 2.5° , the referenced gyro pulse torquing servoamplifier provides a total of 18,840 pulses (out of a possible maximum of 30,500) during the one-half second that the force acts on the gyro input (or reference) axis. For an output velocity of +2.5 feet per second, the referenced accelerometer pulse torquing servoamplifier provides 16,800 pulses (out of a possible maximum of 30,500) during the one-half second that the displacing force acts on the accelerometer sensitive axis. For a gyro sensed negative output angle of 2.5° , the gyro pulse torquing servoamplifier provides a total of 13,160 pulses (out of a possible minimum of 1,500). For a velocity of -2.5 feet per second, the accelerometer pulse torquing servoamplifier provides 15,200 pulses (out of a possible minimum of 1,500).

1-KC Ramp Signal. The 1-kc ramp signal provides the basic timing functions for the dither signal used to retorque the accelerometers and gyros. This signal is applied to all pulse torquing servoamplifiers.

Early Blanking Gate (0 to 3 Blocking Gate). The early blanking gate is a pulse that occurs at the start of each forced limit cycle and whose width is equal to the width of three quantized pulses (approximately 46.8 microseconds). The early blanking gate provides a quanta-limiting function when the dither signal duty cycle is three or less pulses wide, to prevent occurrence of a 0% duty cycle. A 0% duty cycle is equivalent to maximum negative torque acting on the sensors and produces scale factor nonlinearities, which are suppressed by a three-pulse limit. When the dither signal duty cycle is three or less pulses wide, the limiting function is enabled. Less than three pulses from a gyro pulse torquing servoamplifier represents an angular displacement about the gyro input axis of $(3 - 32) \times 10^3 \times 8.805 \times 10^{-4}$, or approximately -25.6° . Three or less pulses from an accelerometer pulse torquing servoamplifier represents an angular displacement about the accelerometer input axis of $(3 - 32) \times 10^3 \times 3.125 \times 10^{-3}$, or approximately -90.625 feet per second.

Late Blanking Gate (61 to 64 Blocking Gate). The late blanking gate is a pulse that occurs at the end of each forced limit cycle and whose width is equal to the width of three quanta periods (approximately 46.8 microseconds). The late blanking gate is used to prevent occurrence of a 100% duty cycle. The output from a gyro pulse torquing servoamplifier represents an angular displacement of $(61 - 32) \times 10^3 \times 8.805 \times 10^{-4}$, or approximately $+25.6^\circ$. When the duty cycle is more than 61 pulses wide, the output from

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

an accelerometer pulse torquing servoamplifier represents a velocity of $(61 - 32) \times 10^3 \times 3.125 \times 10^{-3}$, or +90.625 feet per second.

8-KPPS Synchronizing Signal. The 8-kpps synchronizing signal synchronizes the dc-to-dc converter in the ASA power supply. The converter contains a Royer oscillator, which drives the main output power stage of the converter. When power is applied, the oscillator free-runs at approximately 6 kc. When the d-c outputs come up to the proper voltage, power is supplied to the frequency countdown subassembly, which then provides an 8-kpps synchronizing signal for the oscillator. The oscillator then drives the main power stage of the converter at an 8-kc rate. Use of a separate oscillator driver and a power output stage reduces the starting power requirements.

400-PPS Signals. The 400-pps signals generated in the frequency countdown subassembly are used to develop the gyro spin motor drive. There are four of these signals, one primary and three secondary. The primary signal is developed from the basic 8-kc synchronizing signal and is used as a reference for comparing the secondary signals. The three secondary signals are: a 400-pps signal 180° out of phase with the reference (antireference), a 400-pps signal 90° out of phase with the reference (quadrature), and a 400-pps signal 180° out of phase with the quadrature signal (antiquadrature). All these signals are applied to the ASA power supply for further signal processing.

1-KC Utility Pulse. The 1-kc utility pulse has a 50% duty cycle at the forced limit cycle rate. This pulse triggers the ramp generator in the ASA, and synchronizes test equipment to the forced limit cycle period during test.

Fast Warmup Controller. The fast warmup controller provides heater power to bring the ASA up to operating temperature within the allotted warmup time. The controller comprises a thermal sensor, two flat-pack resistive networks, four transistor matched pairs, two zener diodes, and three ordinary diodes. Three heater control transistors and three heaters mounted on the controller housing are also part of the fast warmup circuitry. The thermal sensor senses the internal ASA temperature and provides a proportional d-c analog to the regulation and heater driver electronics that maintain the total fast warmup power constant.

Fine Temperature Controller. The fine temperature controller maintains ASA operating temperature at a nominal $120^\circ \pm 0.2^\circ \text{F}$. For maximum operating efficiency, the controller is supplied directly with 28-volt d-c power from the LM primary power source. The controller has two thermal sensors, which sense ASA internal temperatures and provide a proportional d-c analog to the detecting electronics. The controller provides switched, rather than continuous, power to the fire temperature heaters. It comprises four transistor matched pairs, three resistive flat-pack networks, two integrated circuit amplifiers, and a unijunction transistor operated as a ramp generator. A transistor matched pair and a single transistor used as a heater control device, mounted on the housing, are also part of the fire temperature controller.

Power Supply. All a-c and d-c power required by ASA is provided by its power supply, which receives 28-volt primary power and synchronizing signals from the frequency countdown subassembly. In addition to the d-c voltages for use by ASA subassemblies, other d-c voltages are produced by the power supply for its internal use. The a-c voltages are used by the accelerometers, gyros, and pulse torquing servoamplifiers. The subassemblies of the ASA that use a-c and d-c power are as follows:

- The +28-volt d-c power is used by the current regulators and budget drivers in each pulse torquing servoamplifier.
- The +12-volt d-c power is used to bias the amplifiers in the accelerometers and to bias integrated circuit amplifiers in the pulse torquing servoamplifiers.
- The +4-volt d-c power is used in the frequency countdown subassembly and the pulse torquing servoamplifiers to bias logic integrated circuit modules.
- The -12-volt d-c power is used as bias voltage in the pulse torquing servoamplifiers.

- The -6-volt d-c power is used as bias voltage in the pulse torquing servo-amplifiers.
- The -2-volt d-c power is used as bias in the frequency countdown sub-assembly and to bias logic integrated circuit modules in the pulse torquing servoamplifiers.
- The 6.3-volt, 8-kc power is used by the demodulators in the pulse torquing servoamplifiers.
- The 22-volt, 8-kc power is used to excite the pickoffs in the gyros.
- The 1.5-volt, 8-kc power is used to excite the pickoffs in the accelerometers.
- The 29-volt, 400-cps, three-phase power is used to drive the gyro spin motors.

■ 2.2.4.2.2 Abort Electronics Assembly. (See figure 2.2-49A.)

The AEA comprises a memory, central computer, input-output subassemblies and power subassemblies.

Memory. The memory is divided into two sections: temporary memory and permanent memory. In the temporary memory, each core is threaded by a sense winding, an inhibit winding, an X-selection winding, and a Y-selection winding. All windings, except the inhibit winding, are single turn. There are 18 sense and inhibit windings. They thread all cores in a single bit plane. Each bit phase has an associated sense amplifier and inhibit driver. In the permanent memory, each core is threaded by a sense winding and Y-selection winding. An X-selection winding is present only if the core is to contain a 0. The sense windings, Y-selection windings, and sense amplifiers are common to the temporary and permanent memories.

The binary state of a core is read by applying equal pulses to the X and Y drive lines that cross the desired bit. This applies an effective full-select current $+I_S$ to the selected core and drives it to the 1 state. All other cores threaded by the pulsed X and Y drive lines receive $+I_S/2$. If, while reading, the selected core is changing from the 0 state to the 1 state, a signal is present across the sense winding. This signal is read by the sense amplifier as a 0. If the core is in the 1 state at the start of reading, no signal appears across the sense winding. Absence of a signal is read by the sense amplifier as a 1. In the permanent memory, a selected core that is threaded by both an X-selection and a Y-selection winding switches from the 0 state to the 1 state and is, therefore, read as a 0. A selected core that is not threaded by an X-selection receives only a pulse of $+I_S/2$ and is thereby always read as a 1.

When a bit is read, the core is left in the 1 state. To achieve nondestructive readout, the word read must be written back into the memory. When writing, equal pulses ($-I_S/2$) are applied to the X- and Y-selection lines. Because the polarity of these selection currents is opposite to that used in reading, the X- and Y-drivers must be bidirectional. Applying $-I_S/2$ to an X- and Y-selection line causes the selected bits to switch to the 0 state. If a 1 is to be written in the temporary memory, $+I_S/2$ is applied to the inhibit line simultaneously with pulsing of the X and Y lines. The inhibit current pulse overlaps the X- and Y-selection currents to preclude partial switching of the selected core to the 0 state.

The system logic is such that only 0's can be written into the memory. The 1 state is brought about by inhibiting the writing of 0's. Therefore, the cores of the word selected must be set to the 1 state before information can be written into the cores. This core setting is accomplished by performing the read operation, which sets all cores to the 1 state, before the write operation. No inhibit current is required in the permanent memory because all cores that were in the 0 state can be restored to 0 by applying X and Y write currents ($-I_S/2$ each). Cores that were in the 1 state remain 1's because they were subjected to a total current of only $-I_S/2$.



GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.2-162C/2.2-162D

The memory operates in three modes: read-restore, read-clear, and clear-write. At the beginning of a memory cycle, the central computer supplies the memory with an initiate-memory-cycle signal, a mode signal, and an address. Address information is in a 12-bit address register in the central computer. Decoding is performed within the memory. The address register always contains the address location at the beginning of a memory cycle. During the read-restore mode, the contents of the location specified by the address register are read from the memory into the memory register (M-register). The read-restore mode is used for the temporary and permanent memories. The read-clear mode is similar to the read-restore mode, except that the information is not restored to the memory. This mode applies only to the temporary memory and when it is unnecessary to restore information after reading. In the clear-write mode, the contents of the M-register are written into the location specified by the address register. The word to be written is transferred into the M-register during the second microsecond of the memory cycle.

Central-Computer. The central computer consists of eight data and control registers (L, R, P, A, Q, M, X, and C), a data bus, an 18-bit parallel adder, and two timing and control registers. The data and control registers are interconnected via the parallel data bus. There are, however, some nondata-bus parallel paths through the adder. Operations are executed by appropriately timed transfer of information between these registers, between memory and the M-register, and between the A-register and a specified input or output register. The functions of the eight data and control registers are discussed in paragraph 2.2.3.3.

Data Bus. The data bus consists of gates through which data (up to 18 bits in parallel) may transfer from one register to another. A typical transfer consists of 18 bits of data from the A-register to the M-register. When this occurs, the timing and control logic generates two control signals. The first control signal gates all 18 bits of the A-register onto the data bus for 2 microseconds (some data bus transfers take 3 microseconds); at the same time, the second control signal gates the contents of the data bus into the clocked set input of the 18 M-register flip-flops. At the end of the second microsecond, the M-register is clocked and the information transfer is complete.

For several transfers from the data bus, both true and complement information is gated into the receiving register. However, for transfers into the P-register only selected true signals are gated; therefore, this register must always be cleared to 0's before receiving new information. For transfers into X-, C-, and M-registers, only selected complement signals are gated; these registers must be cleared to 1's before receiving new information. Information transfer into most registers is via the clocked set-reset inputs at the appropriate clock time; however, the three registers (C, P, and X) connected as ripple counters are loaded from the data bus via the flip-flop direct set-reset inputs.

Adder. The 18-bit parallel adder changes the contents of the A-register by adding or subtracting information (true or complement data, respectively) from the M-register. Three microseconds are required for the carry to propagate and the correct sum to be available at the adder outputs after the M-register information is gated into the adder. At the end of the 3-microsecond period, the sum is clocked into the A-register, replacing its previous contents (augend). A one-bit shift (either left or right) may be implemented simultaneously by gating the output of the adder into adjacent bits of the A-register. The data bus is not used to interconnect the adder with the A-register or M-register.

Timing and Control Registers. The timing and control registers generate 11 signals, which are logically combined to produce all the timing required for all operations of the central computer.

Input-Output Subassembly. The input-output subassembly consists of special registers, counters, amplifiers, and gates, which operate independently of the central computer, except when specifically accessed by a particular input or output order. There are basically four types of registers in the input-output section. The PNGS angles are accumulated in three integrator registers, each of which consists of a 15-flip-flop shift register, a half-adder/subtractor, and control logic. The registers shift 15 bit positions at 512 kc upon receipt of each new pulse input, to serially add or subtract one from the previous count. The counting is inhibited momentarily when an input transfer to the A-register via the data bus is required. The registers are reset to zero by a 51.2-kc pulse input on the CDU zero line. Data from the ASA gyros and accelerometers are accumulated in six 11-bit flip-flop ripple counters. The inputs to these counters are inhibited when they are accessed by the central computer; they are reset to 0 after each access. The four-bit DEDA register, the 18-bit input telemetry register, the 24-bit output telemetry register, and the 15-bit register that is time shared for altitude and altitude rate are all implemented as shift registers. The first three of these registers are asynchronous (shifted by an external

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

shift clock); the other one has associated shift controls and generates shift clocks in the AEA. In each case, shifting is inhibited during parallel access by the central computer. All the other registers are static flip-flop registers. The D/A registers are loaded in parallel, via the data bus, from the A-register. The discrete register is changed by setting or resetting a specific flip-flop within the register with a specific output order (unique address); the A-register is not involved in changing the discrete register.

Combinations of certain individual input-output addresses can be made, within an input or output instruction, to facilitate setting or resetting a group of registers or discretes. Only addresses that have the same most significant octal character can be combined. To combine one or more addresses, each address (excluding the most significant character) is added numerically. For example, combining the addresses of all six 11-bit counters would result in an address of 6177. An input instruction with the address 6177 would reset all the counters.

Integrator Registers. Three integrator registers accumulate PGNS angles θ , ϕ , and ψ . The PGNS angles are represented by asynchronous pulse train signals, with a frequency range of 0 to 6.4 kpps. Each integrator register has two input lines: one for positive pulses and one for negative pulses. These registers operate during the standby mode, when the central computer logic is off.

The integrator registers are up-down counters that serially add or subtract one count per pulse input. Each integrator register consists of an input buffer section, a shift control section, and a 15-bit shift register section with half-adder and control logic. The input buffer section synchronizes incoming PGNS pulses with the AEA clock and provides temporary storage for one pulse. The shift control section allows the register to shift the appropriate number of times for each pulse input from the input buffer.

Input Counters. Six input counters accumulate ASA gyro and accelerometer data. These data (ΔV_x , ΔV_y , ΔV_z , ΔQ , ΔR , and ΔP) are represented by pulse trains with a frequency range of 0 to 64 kpps. These pulses are asynchronous to the AEA. The input counter consists of an input buffer section and an 11-bit ripple counter with control logic.

Downlink Telemetry Register. The downlink telemetry register is an 18-bit input register. It receives a serial 40-bit PGNS downlink telemetry word. The first 18 bits are retained in the register. The 40-bit word is shifted into the register by externally controlled shift pulses at a rate of 51.2 kpps. These shift pulses are asynchronous to the AEA. A 1 in the input data is represented by a pulse (coincident with a shift pulse); a 0, by lack of a pulse. A stop pulse from the external control indicates the end of a transmitted word. The register is used by GSE to enter data into the central computer, but cannot be used by GSE and telemetry simultaneously. An 18-bit word is used by the GSE. The register consists of a shift pulse buffer section and a 18-bit shift register section with shift control logic. The shift pulse buffer section synchronizes the input shift pulses with the AEA and provides a delay between data pulses received and shift pulses received.

Altitude-Altitude Rate Register. The altitude-altitude rate register is a 15-bit output register. It is used to output, serially, altitude and altitude rate data. A 1 in the output data is represented by a pulse; a 0, by lack of a pulse. There are separate output data and output shift pulse lines for the altitude and altitude rate. An output discrete selects the data and shift pulse lines to be used. Before data are entered into the register, the output discrete corresponding to the desired output must be set.

Output Discretes. There are 11 output discrete flip-flops in the AEA. Each discrete can be used independently of the others. Except for the two DEDA discrete flip-flops, all the output discrete flip-flops are set and reset by different addresses. The DEDA output discrete flip-flops are automatically reset by the hardware; they cannot be reset by the program. The DEDA discrete flip-flops are clock-set by the appropriate output instruction, and clock-reset 1 microsecond later. These discretes output a single pulse, nominally 1-microsecond wide, each time the flip-flop is set. The test mode failure output discrete is automatically set if the 20-millisecond timing pulse occurs during any instruction other than the delay instruction. The engine-on output discrete flip-flops are automatically reset during the power turn-on sequence. The altitude and altitude rate output discretes have a common reset address. These discretes are used for internal control of the altitude-altitude rate register. They enable the desired serial data and shift pulse output lines. The GSE output discretes are used to gate out a 128-kpps pulse train. The ripple carry inhibit output discrete is used internally to test the adder carry mechanism.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS

Output Telemetry Register. The output telemetry register, a 24-bit output register, outputs telemetry serial data or GSE serial data. The telemetry data word is 24 bits long; the GSE data word, 18 bits long. Telemetry and GSE cannot use the register simultaneously. A 1 in the output data is represented by a pulse; a 0, by lack of a pulse. Each data word is shifted out with the most significant bit first. Data in the register are shifted out by asynchronous, externally controlled, shift pulses at a rate of 51.2 kpps. Twenty-four shift pulses are sent for telemetry data; 18, for GSE data. The external control sends a stop pulse to the output telemetry register when all shift pulses have been sent.

The telemetry word list comprises a 50-word block of data. (Refer to table 2.2-19.) Each output telemetry word comprises 24 bits (a six-bit identification (ID) code followed by an 18-bit central computer word).

Table 2.2-19. Abort Electronics Assembly - Telemetry Word List

Word No.	ID Code (Octal)	Equation Symbol	Earth Binary Scaling	Least Significant Bit Weight	Unit	Description
1	01	δ_{41}	-	-		DEDA readout mode flag; 1 in sign bit indicates DEDA processing is in readout mode.
2	02	DD	-	-	-	Most recent DECA data word in computer units
3	03	δ_{38}	-	-	-	DEDA clear mode flag; 1 in sign bit indicates DEDA processing is in clear mode.
4	04	a_{11}	} 1	2^{-16}	-	Row No. 1 of direction cosine matrix
5	05	a_{12}				
6	06	a_{13}				
7	07	ADST	-	-	-	Octal address associated with most recent DEDA communication
8	10	a_{31}	} 1	2^{-16}	-	Row No. 3 of direction cosine matrix
9	11	a_{32}				
10	12	a_{33}				
11	13	h	25^{-3}	2^8	feet	LM altitude above mean earth (lunar) surface
12	14	} \underline{r}	25^{-3}	2^8	feet	Components of LM inertial position
13	15					
14	16					
15	17	EONS10	-	-	-	Engine-on indicator and GUID CONT switch
16	20	} r_c	25^{-3}	2^8	feet	Components of CSM inertial position
17	21					
18	22					
19	23	r_f	25^{-3}	2^8	feet	Predicted value of LM radius at completion of current guidance routine, except in external Δv
20	24	ΔV_x	} 3	2^{-14}	fps	Compensated incremental velocity components accumulated per 20 milliseconds along X, Y, or Z-body-axis by corresponding X, Y, or Z accelerometer
21	25	ΔV_y				
22	26	ΔV_z				

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-19. (Cont.)

Word No.	ID Code (Octal)	Equation Symbol	Earth Binary Scaling	Least Significant Bit Weight	Unit	Description
23	27	t_2	1	2^{-16}	sec	Least significant half of AGS absolute time
24	30	$\Delta\alpha_x$	-6	2^{-16}	rad	Compensated incremental components per 20 milliseconds of vehicle rotation about X, Y, or Z body axis, measured by X, Y, or Z gyro.
25	31	$\Delta\alpha_y$				
26	32	$\Delta\alpha_z$				
27	33	T_B	9	2^{-8}	sec	Time to LM engine burnout
28	34	$\left\{ \begin{array}{l} V \\ - \end{array} \right.$	15	2^{-2}	fps	Components of present LM inertial velocity vector
29	35					
30	36					
31	37	μ_{8T}	17	1	-	Ullage counter for telemetry
32	40	$\left\{ \begin{array}{l} V \\ -c \end{array} \right.$	15	2^{-2}	fps	Components of CSM inertial velocity vector
33	41					
34	42					
35	43	\dot{h}	15	2^{-2}	fps	LM altitude rate
36	44	V_G	15	2^{-2}	fps	Magnitude of velocity to be gained
37	45	$\left\{ \begin{array}{l} V_{P0} \\ V_T \end{array} \right.$	$\left\{ \begin{array}{l} 15 \\ 15 \end{array} \right.$	$\left\{ \begin{array}{l} 2^{-2} \\ 2^{-2} \end{array} \right.$	$\left\{ \begin{array}{l} \text{fps} \\ \text{fps} \end{array} \right.$	$\left\{ \begin{array}{l} \text{Predicted velocity to be gained in CDH burn} \\ \text{Total velocity to rendezvous in TPI} \end{array} \right.$
38	46	S_{12T}	3	-	-	S_{12} for telemetry
39	47	$\left\{ \begin{array}{l} T_{A0} \\ t_{igC} \end{array} \right.$	$\left\{ \begin{array}{l} 18 \\ 18 \end{array} \right.$	$\left\{ \begin{array}{l} 2 \\ 2 \end{array} \right.$	$\left\{ \begin{array}{l} \text{sec} \\ \text{sec} \end{array} \right.$	$\left\{ \begin{array}{l} \text{Time from nominal CSI burn to CDH burn corresponding to best cost function for this computing cycle} \\ \text{Time from present to TPI maneuver} \end{array} \right.$
40	50	$\left\{ \begin{array}{l} X_{bD} \\ \end{array} \right.$	1	2^{-16}	-	Components along X, Y, and Z inertial axes of unit vector commanding desired pointing direction for LM X body axis
41	51					
42	52					
43	53	t_1	18	2^1	sec	Most significant half of AGS absolute time
44	54	S_{00}	-	-	-	Function selector by which submode logic is selected via DEDA
45	55	$\beta's$	-	-	-	Discretes that make up discrete word one
46	56	$\left\{ \begin{array}{l} \Delta r \\ q_{ld} \end{array} \right.$	$\left\{ \begin{array}{l} 25 \\ 25 \end{array} \right.$	$\left\{ \begin{array}{l} 2^8 \\ 2^8 \end{array} \right.$	$\left\{ \begin{array}{l} \text{feet} \\ \text{feet} \end{array} \right.$	$\left\{ \begin{array}{l} \text{Differential LM-CSM altitude after CDH burn} \\ \text{Transfer orbit perifocus altitude in TPI} \end{array} \right.$
47	57	q_{LT}	25	2^8	feet	Perifocus altitude of present LM orbit
48	60	$\left\{ \begin{array}{l} V_d \\ \end{array} \right.$	15	2^{-2}	fps	Sensed velocity increments along LM body axds
49	61					
50	o2					

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

The identification code represents the sequential position of the telemetry word in the word list (01g to 62g). The data "equation symbol" are the same as those used in the AGS software program. The central computer word contains codes or numeric data. The numeric data are in binary 2's complement format. The most significant bit (bit No. 0) is a sign bit (0 for positive data; 1 for negative data). The remaining bits are numbered (1 through 17), from most significant to least significant.

DEDA Register. The DEDA register is a four-bit input-output register, which receives or sends four bits of serial data. A 1 in the serial data is represented by a pulse; a 0, by lack of a pulse. Data are shifted in or out by asynchronous, externally controlled, shift pulses at a rate of 64 kpps. These input shift pulses are gated out as data pulses when the DEDA register is in an output mode. Bits No. 1 through 4 of the DEDA Register are transferred to bits No. 1 through 4 of the accumulator during an input instruction and loaded with bits No. 14 through 17 of the accumulator during an output instruction. The DEDA register consists of a shift pulse buffer section and a four-bit shift register with control logic.

Clock Countdown Subassembly. The clock countdown subassembly develops 512-k, 128-k, and 64-kpps and 50-pps timing signals from the 1.024-mc system clock. The clock countdown subassembly consists of a toggle flip-flop, three counter sections, and associated gating.

Power Subassemblies. Two independent power supplies are associated with the central computer. One functions in the standby and operate mode; it supplies power to clock-countdown circuits and to three integrating registers. The other power supply functions only in the operate mode; it supplies power to the remainder of the central computer. The outputs of the latter power supply are available in sequence so as not to compromise the memory, monitors the input voltage and output voltages and provides a control signal (PS13). The control signal is true if the central computer is in the operate mode, the input voltage is at least 20 volts dc and all output voltages are at the proper levels.

Startup. To place the central computer in operation, the operator sets the AGS STATUS switch from STANDBY to OPERATE. The power subassemblies then apply power to the central computer in a predetermined sequence. When all operate voltages are at their proper levels, power supply signal PS13 remains 0 for at least 10 microseconds. During this 10-microsecond period, certain initial conditions are established in the central computer. These conditions are such that the central computer automatically accesses an instruction from a predetermined memory location. Central computer operation then continues under normal program control.

Shutdown. The central computer shuts down if the power supply input voltage drops below 20 volts dc; it restarts automatically when the proper input voltage is restored. If the central computer shuts down after reading from memory, but before restoring memory, information is lost. This is prevented by completing the current memory cycle while preventing initiation of additional memory cycles during shutdown. When the power supply input voltage is restored (20 volts dc) and the current memory cycle is completed, the central computer is ready to go through its startup sequence.

2.2.4.2.3 Data Entry and Display Assembly. (See figure 2.2-49B.)

DEDA operation involves the display; shift register; bit, digit and cycle counters; a data transfer sequencer; pushbutton entry, and data-AEA transfer.

Display. The three most significant digit positions (1, 2, and 3) of the shift register are decoded to form the address display. The next position (4) is decoded to form the sign display. The remainder of the digit positions (5, 6, 7, 8, and 9) are decoded to form the data display. The decoded digit for each numeric display provides an output for each segment of the display. The display segments for each digit are activated, via drivers, by outputs from the decoding logic associated with the respective four-bit digit positions of the shift register. The all 1's code in any digit position of the shift register deactivates the decode outputs, blanking the digit display.

Shift Register. The shift register accumulates and holds all information entered with the pushbuttons or transferred from the central computer for display. The shift register contains 36 bits comprising nine digits. Codes presented from the pushbuttons are entered directly into the four least significant bits of the register and positioned for display. Address and data bits transferred from the central computer are entered into the least significant bit position of the register. Address and data bits to be transferred to the central computer are taken from the most significant bit position of the register.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

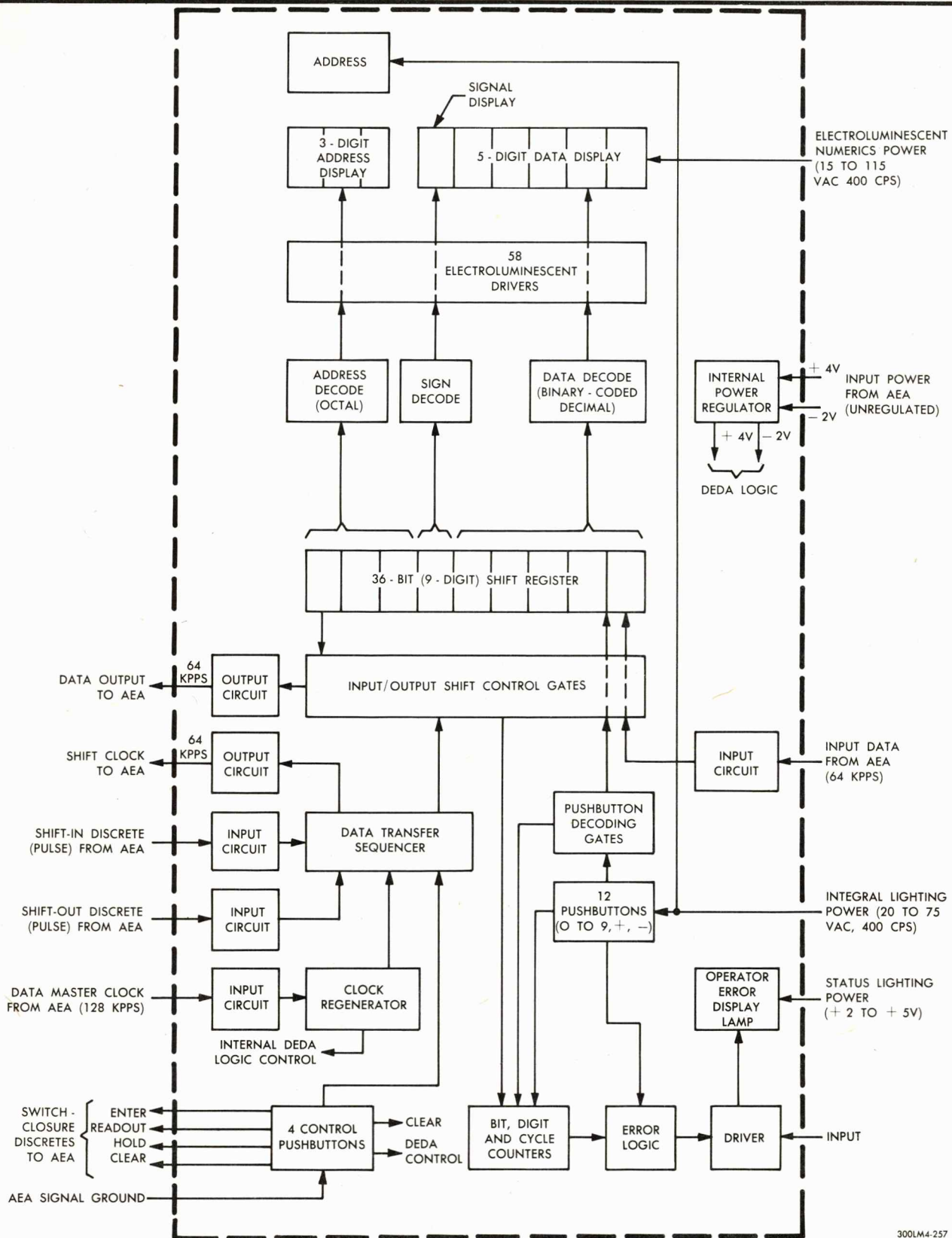


Figure 2.2-49B. Data Entry Display Assembly - Detailed Block Diagram

300LM4-257

Bit, Digit, and Cycle Counters. The bit counter is a six-bit ripple counter. The counter accumulates the number of clock pulses gated to the shift register and, in conjunction with the digit and cycle counter, controls positioning of digit codes in the shift register as they are entered with the pushbuttons. The bit counter also controls the transfer sequence as information is transferred to or from the central computer. The digit counter, a five-bit shift register counter that is incremented on the final count of the bit counter, detects operator errors. Errors arise when pushbuttons are pressed out of sequence, while entering data into the shift register. Such errors cause the OPR ERR light to go on. The cycle counter is a four-bit shift register whose main purpose is to control the cycling of inputs to the shift register.

Data Transfer Sequencer. The data transfer sequencer consists of a two-step counter and flip-flops that buffer and control data transferred from the central computer. The sequencer controls the timing of shift pulses and information bits transferred to or from the central computer and provides control for shifting the contents of the shift register during the transfer.

Pushbutton Entry. To enter data with the pushbuttons, the CLR pushbutton must be pressed. This action sets the entire shift register, which blanks the display and resets the digit, bit, and cycle control counters, and operator error logic circuits. A flip-flop that enables the keyboard (pushbutton) control logic is also set. This flip-flop is reset when the ENTR or READOUT pushbutton is pressed.

As the pushbuttons are pressed, binary coded decimal (BCD) signals are fed to gates, which, in parallel, set or reset the four least significant flip-flops of the shift register. These gates are controlled by the pushbutton strobe, a delayed pulse generated in the keyboard control logic. The pushbutton strobe also sets bit counter enable flip-flop and cycle control flip-flop. The bit counter enable flip-flop enables a gate, which feeds clock pulses to the shift register. When clock pulses add up to 31, as counted by the bit counter, another gate is enabled. This gate resets the bit counter enable flip-flop on the next clock (pulse 32), terminating the shift register clocks, and enables a clock pulse that is fed to the digit counter, incrementing the counter by one.

Digit codes entered into the shift register are positioned in their proper location by the cycle counter as the register is clocked. The first digit is positioned in the most significant end of the shift register, the second digit in the next most significant digit location, and so on until the ninth (last) digit is entered. The cycle counter positions the digit codes by enabling gates, which cause the information in the shift register to circulate out of the most significant end into the eighth digit position in the register while holding the current pushbutton input in the least significant digit position. At a time determined by the current state of the digit and bit counters, a gate that resets the cycle control flip-flop to allow the current input to shift through the register, behind the information previously entered, is enabled.

If pressing the ENTR or READOUT pushbutton results in an operator error or if either pushbutton is pressed during an operator error condition, the central computer recognizes the error by detecting 1's in positions that normally contains 0's. Pressing the CLR pushbutton resets the operator flip-flop.

Data AEA Transfer. Upon entering nine digits into the shift register with the pushbuttons, and visual verification that an operator error condition does not exist, the ENTER pushbutton may be pressed to start the data transfer sequencer. The central computer responds to the enter signal by sending a shift-in discrete to the DEDA. This discrete sets a flip-flop, which initiates data transfer to the central computer. Data are transferred in a two-step operation controlled by the data transfer sequencer. During the first step of the shift-in sequence, the most significant bit of the shift register is gated with a 2-microsecond-wide clock pulse and fed to an output circuit. During the second step of the shift-in sequence, a shift pulse is fed to the central computer, the contents of the shift register are shifted one bit position, and the bit counter is incremented by one.

Each shift-in discrete results in four cycles of the data transfer sequencer or transfer of one digit. The central computer sends shift-in pulses until the entire word is transferred. As the contents of the register are shifted from the most significant end, 1's are shifted into the least significant end. The display is blank after all digits have been transferred. Because of the data transfer sequencer four-cycle operation, the address digits, which are coded in three bits (octal), and the sign digit, which is coded in one bit, are allotted four bits in the shift register. The extra bits are used to provide redundant codes for blanking the respective displays or detection of an operator error by the central computer.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.2.4.3 Rendezvous Radar.

2.2.4.3.1 Antenna Assembly. (See figure 2.2-19.)

The RR antenna is an amplitude-comparison, simultaneous lobing, directional device that provides accurate angle and angle rate tracking information. The 24-inch-diameter parabolic primary reflector of the radar antenna has a conventional four-port Cassegrainian feedhorn. The radiation from the feedhorn illuminates a hyperbolic secondary reflector located to reflect the feedhorn-radiated RF energy back to the primary reflector. Use of the hyperbolic reflector uniformly illuminates the parabolic reflector with RF energy and results in the formation of a highly narrow directional beam. The antenna operates in the X-band region of the microwave spectrum and possesses an overall power gain of 32 db. The signal radiated from the antenna is circularly polarized to minimize signal losses due to variations in attitude that occur between the LM and CSM. The antenna beamwidth is nominally 3.5° .

The antenna pedestal and the base of the antenna assembly are mounted on the external structural members of the LM. The antenna pedestal supports the shaft axis members, including the shaft resolver and servo motor. The trunnion axis is perpendicular to, and intersects the shaft axis. The trunnion axis members, including the trunnion resolver and servo motor, are vertically supported in a cascade arrangement. The antenna reflectors and the microwave and RF electronics components are assembled at the top of the trunnion axis. This assembly is counterbalanced by the trunnion-axis gyroscope package, mounted below the shaft axis. The plane of the trunnion axis is determined by the angular position of the shaft axis. Both groups of components, mounted opposite each other on the trunnion axis, revolve about the shaft axis forming a balanced arrangement which enables balanced driving torques and reduces the servo requirements and weight. The microwave, radiating, and gimballing components, and other internally mounted components, have flexible cables that connect the outboard antenna components to the inboard electronics assembly. A flexible cable wrap-up system is used at the trunnion and shaft bearing joints.

The RR shaft and trunnion axes resolvers transmit angular position to the LGC via the RR CDU Channel 3. The LGC uses these angular position data in conjunction with range and range rate to update the state vector. The RR also provides data to LM cabin displays for use by the astronauts in evaluating the rendezvous phase of the mission.

2.2.4.3.2 Electronics Assembly. (See figure 2.2-18.)

The electronics assembly consists of the following subassemblies: receiver, frequency synthesizer, frequency tracker, self-test, antenna control amplifier, range tracker, signal data converter, and instrumentation converter.

Receiver Subassembly. The receiver subassembly is an amplitude-comparison, triple-conversion, three-IF-channel unit that provides shaft and trunnion error signals to an antenna servo loop when angle-tracking the target with the radar antenna. It also functions as an intermediate signal path for received signals, which are then processed before being picked off by the frequency tracker and range tracker. An AGC loop in the receiver subassembly maintains a constant signal output to the other subassemblies despite variation in signal input. This ensures that the angle error is a function of the angle error signals, not a function of signal amplitude. The receiver subassembly consists of three channels: sum channel (reference), trunnion-axis difference channel, and shaft-axis difference channel.

The receiver subassembly also contains phase-sensitive detectors for generating angle error signals, an IF distribution amplifier unit for supplying reference channel signals to the range and frequency trackers, and a local oscillator-mixer for generating the second local oscillator signal. The second local oscillator frequency is obtained by heterodyning the frequency tracker VCO output and a reference frequency. This produces a sum frequency that is 6.8 mc lower than the incoming 40.8-mc Doppler-shifted frequency. A feedback loop removes the Doppler frequency shift; all subsequent signal processing is accomplished at fixed carrier frequencies.

Frequency Synthesizer Subassembly. The frequency synthesizer subassembly supplies the other RR subassemblies and the antenna assembly with coherent reference frequencies, which provide extreme accuracy and a high degree of stability. Multiplying and dividing circuits, local oscillators, amplifiers, and mixers are used in the frequency synthesizer to provide five output frequencies whose function and destination are listed in table 2.2-21.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.2-21. Frequency Synthesizer - Output Frequencies

GN&CS

Frequency (mc)	Function	Destination
5.1	Third mixer excitation	Receiver
6.8	Reference signal	Frequency tracker, and range tracker analog processor
7.0125	Doppler bias mixer excitation	Frequency tracker
27.2	Second mixer reference	Receiver
102.425	Transmitter excitation	Antenna assembly, multiplier chain

The frequency synthesizer subassembly contains two oscillators. One oscillator operates at 102.425 mc; it provides an input to a chain multiplier in the antenna assembly. The chain multiplier multiplies the input by a factor of 96 to produce the radar transmitter frequency (9832.8 mc). The other oscillator produces a frequency of 1.7 mc. Multiplication and division of this frequency provides the other frequencies required for RR operation. (The 1.7-mc oscillator is locked in frequency to the 102.425-mc oscillator.)

Frequency Tracker Subassembly. The frequency tracker subassembly is a phase-lock loop that detects, acquires, and phase-tracks the received signal in the receiver subassembly sum channel. In addition, the frequency tracker provides Doppler frequency and sense outputs, and frequency lock-on indication.

The frequency tracker phase-locks a VCO with the return signal. A sweep circuit varies the VCO frequency across the expected Doppler frequency range (-100 to +100 kc). The presence of a carrier signal within the locking range, when sensed through a narrow 1 kc filter by a threshold circuit, stops the sweep and permits the VCO to phase-lock. The phase detector has a 6.8-mc reference signal, supplied by the frequency synthesizer subassembly. An error signal from the receiver subassembly drives the VCO to a frequency, which, when mixed with a 27.2-mc signal at the receiver second IF mixer produces a 1.7 mc signal which passes through another 1 kc filter. A feedback loop removes the Doppler frequency shift from all signals in the succeeding IF stages. When the sweep generator stops sweeping the VCO, the acquisition logic and the threshold detector produce a frequency tracker lock-on discrete. When present together this discrete and the auto-track enable discrete, the range tracker subassembly and the servo mechanisms are enabled.

Self-Test Subassembly. The self-test subassembly is used to perform a qualitative test to verify proper operation of the RR electronics. The test provides range, range rate, and angle error data for a simulated 200-nautical-mile signal. Basically, the self-test subassembly provides a means of inserting, into the antenna assembly, signals that simulate RF inputs to the RR. Range rate information is contained in fixed IF offset of 10 kc, corresponding to a velocity of 500 fps. Range information is placed on the crystal oscillator frequency by a varactor phase modulator driven by three-tone phase shifters. Angle error information is simulated by square-wave phase modulation of the difference channel signals, which switches the phase of the difference channels between 90° and 270° with respect to the sum channel signal.

The self-test circuits are enabled when the RADAR TEST switch is set to RNDZ. The RANGE and RANGE RATE indicators should read 200 nm in range and -500 fps in range rate and the antenna should be driven in alternate directions, as indicated by the FDAI error needles.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEM DATA

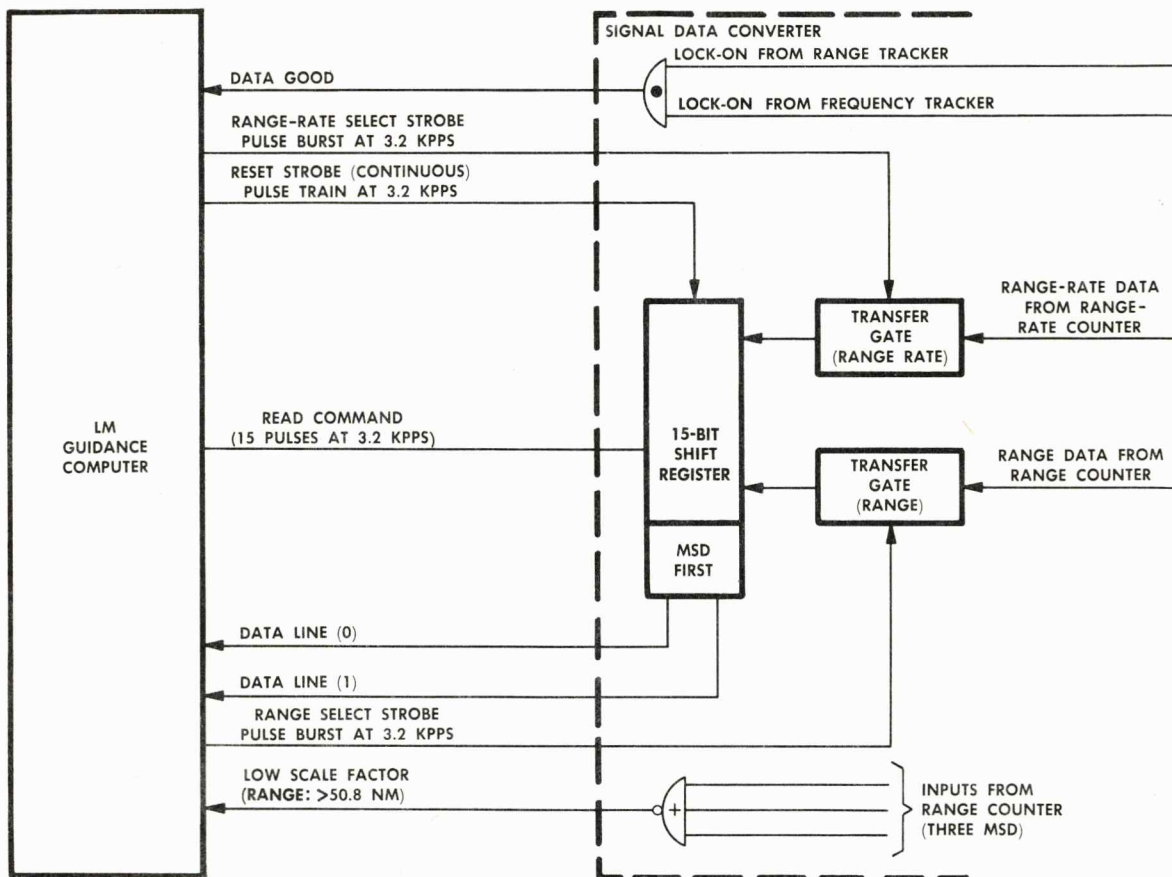
Antenna Control Amplifier Subassembly. The antenna control amplifier subassembly contains amplifiers for driving the antenna shaft-and trunnion-axis servomotors, amplifiers for driving the gyro torquer coils, and voting logic for switching between the redundant and primary gyros in the event of a malfunction of the pair in gyros in use. The amplifiers, in conjunction with the antenna components and receiver subassembly, form inner and outer closed loops for each axis. The inner (stabilization) loop maintains the antenna boresight axis fixed in inertial space in the presence of vehicle motion, as sensed by the gyros. The outer (tracking) loop further maintains the antenna boresight on the target, using the tracking error signals from the receiver. In the designate mode, the tracking loop is open and LGC-designated antenna position data, or manual slew data, are accepted by the subassembly. The LGC designates the antenna boresight to the target, and supplies an automatic tracking enable signal for the RR when within 0.5° of the computer target LOS. The possibility of side-lobe lock-on is made negligible by limiting the antenna boresight displacement from the computed target LOS to within 0.5° before closing the tracking loop. Allowing for a 1° error in the computer LOS with respect to the actual LOS to the CSM and a 1° boresight error for the RR-to-IMU alignment, the total initial misalignment is limited to 2.5° , which is adequate to prevent an antenna side-lobe lock-on. The first side-lobe is 6° off boresight. The antenna then continuously tracks the target by maintaining the monopulse receiver angle error signals at null. The antenna may also be slewed manually, at either of two selected LOS rates. In this case, the enable signal required to close the automatic tracking loop is supplied by setting the RENDEZVOUS RADAR selector switch (panel 3) to AUTO TRACK.

The antenna shaft and trunnion are positioned by 32-pole, brushless, permanent-magnet-rotor-type motors. The motors are driven by pulse-width-modulated drive signals applied to their sine and cosine windings. The direction of motor rotation is reversed by reversing the motor winding across the pulse-width-modulated drive voltage.

A gyro voting circuit consisting of performance comparison and logic-switching circuits, automatically detects a malfunctioning gyro and removes it from the control loop. There are two pairs of gyros; one pair at a time is used to stabilize the antenna. Either pair can be used. The voting system determines whether the active pair contains a failed gyro by comparing the output of each of the gyros, three at a time. If a failure or degradation occurs, the redundant pair is switched automatically into the antenna-stabilizing network.

Range Tracker Subassembly. The range tracker subassembly determines the range to the transponder by measuring the relative phase delay between the transmitted signals and the received signals modulating tones. The transponder reply signal, after undergoing dual frequency conversion in the RR antenna assembly, and receiver subassembly is applied as a 6.8-mc signal to the coherent detector of the range tracker subassembly. This signal is demodulated in the coherent detector, which uses a 6.8-mc quadrature reference. The individual sine-wave range tone signals are extracted from the receiver noise by use of bandpass filters. The range phase delay, proportional to range, is measured independently on each of the three tone signals in a closed tracking loop, using three locally generated reference square waves each of which has variable phase with respect to the transmitted signals. The phase of the reference square waves is adjusted until they have matching phase with respect to each of the received signals. The matching phase is when the reference square wave is in quadrature with the phase of the received signals. The reference square waves are produced digitally by comparison between a running high-speed counter and a low-speed range counter. The low-speed range counter is driven upward or downward until phase null is achieved in each of three phase detectors by incremental range pulses obtained from a voltage-to-PRF converter. The converter is controlled above 50.6 nm by weighted integration of the medium and high tone phase-detector error signals, and by the high tone error below 50.6 nm.

Signal Data Converter Subassembly. (See figure 2.2-50.) The signal data converter subassembly consists of a computer interface board and an input-output amplifier board; it provides the electrical interface between the RR and the LGC for range and range-rate data. It also supplies an 18-bit range word for the displays and routes the RR control signal from the control panel. The computer interface board contains a 15-bit binary counter, a 15-bit shift register, two strobe gate generators and associated data transfer logic, and six discrete interface circuits. The input-output amplifier board contains relay drivers, pulse shapers, a signal shaper, and antenna servo control logic. Depending on the control signals from the LGC, the computer interface board provides either range or range rate data to the LGC.



300LM4-171

Figure 2.2-50. Rendezvous Radar Digital Data Transfer - Flow Diagram

The LGC requests range data by sending strobe pulses on three lines at a rate of 3,200 pps. One strobe line supplies a continuous train of reset pulses for timing. A burst of 256 pulses is sent on a second strobe line to read the range. After a strobe interval of 80 milliseconds (read time), the LGC sends a burst of readout pulses on a third line to obtain the data. In this operation, the range data are transmitted to the LGC in true and complement forms on two separate lines, one line for one bits and the other line for zero bits.

The LGC requests range rate data by sending pulses on the reset strobe line and a burst of pulses on the range rate gate strobe line at a rate of 3,200 pps. Both sets of pulses are routed through a gate generator to provide three outputs, as follows:

- Reset pulses to clear the 15-bit binary counter
- Range rate pulses to control the duration of the counting period
- Transfer pulses to activate the 15-bit transfer logic.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEM DATA

The range data routed to the LGC can have either of two different scale factors. Range low scale factor switching is implemented automatically by the RR at ranges of 50.58 nautical miles or less. The RR has an 18-bit accumulator (counter) for transferring range data to the 15-bit shift register. Only 15 of 18 bits are gated to the shift register, to be sent to the LGC. At ranges greater than 50.58 nautical miles, the 15 most significant bits are transferred. At ranges less than 50.58 nautical miles the least significant bits are transferred. The range scale factor determines whether the 15 least significant bits or 15 most significant bits from the counter are gated to the shift register. The RR sends a discrete signal (RR range low scale factor) to inform the LGC that low-scale-factor radar operation is in effect. The low scale factor of the transferred range data is 9.38 feet/bit; the high scale factor, 75.04 feet/bit. RR scale factor switching does not occur during range data transfer.

Instrumentation Converter Subassembly. The instrumentation converter subassembly provides the following signals:

- Transmitter power output to cabin display (SIGNAL STRENGTH indicator on panel 3)
- Data-no-good signal to telemetry
- No-track signal to cabin display (NO TRACK light on panel 3).

The instrumentation converter subassembly consists of a power monitor converter and a data no good and no track converter. The power monitor converter output indicates the RR transmitter power output. A dc analog voltage from a directional coupler in the antenna assembly is chopped by a 20 kc square wave. This signal is amplified, demodulated, and rectified. The rectified dc analog signal is filtered and then routed to the RADAR: SIGNAL STRENGTH indicator (panel 3) where it indicates RR transmitter power output.

The data no good and no track converter monitors the signal data converter subassembly for a data no good condition. A data-no-good condition exists when range, range-rate or both are not locked up. This converter generates data no good and no track signals upon detection of a data no good condition. Routed to the IS, the data no good signal enables the lighting of the RNDZ RDR caution light (panel) and the telemetering of the no track condition to MSFN. The no track signal enables the lighting of the RNDZ RADAR: NO TRACK light (panel 3).

2.2.4.4 Rendezvous Radar Transponder.

2.2.4.4.1 Antenna Assembly (See figure 2.2-51.)

The transponder antenna assembly is mounted tilted upward (+X direction) at a 15° angle from the CSM Service module surface. The antenna assembly, located close to the transponder electronic assembly, is approximately 159 inches forward of the Service module aft section with CSM axis coordinates of 40° from +Z-axis, 50° from +Y-axis, and 75° off +X-axis. The antenna, a wide angle of coverage horn, has an on-axis gain of approximately +7 db. Out to the limits of antenna coverage ($\pm 60^\circ$), the antenna gain is never less than 0 db.

The antenna part of the assembly consists of a Y-type, two porthole, waveguide horn with a circular ground plane. The antenna is protected by, and isolated from, an aluminum-phosphate radome connected to a cork-insulated fiberglass housing, mounted on the Service module fairing. The radome and housing protect the antenna from dust and thermal and abrasive-blastoff effects. The radome composition permits optimum microwave reception and transmission with minimum deterioration of signal strength and characteristics.

The antenna is coupled, by waveguide section, to the transponder electronics assembly. A waveguide flange is used to connect the antenna base to an antenna waveguide flexible adapter. A coupling flange connects the flexible adapter to a waveguide transmission line.

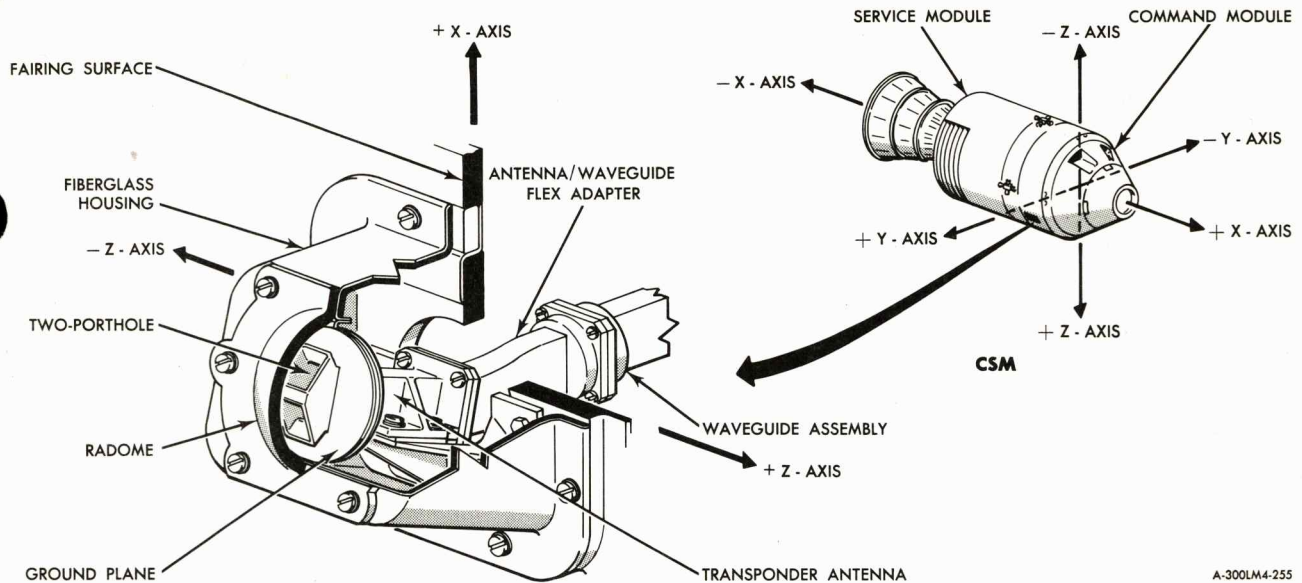


Figure 2.2-51. Transponder Antenna Assembly

2.2.4.4.2 Electronics Assembly. (See figure 2.2-23.)

The electronics assembly consists of the following subassemblies: microwave, receiver carrier lock loop, tone modulation detector, and a power supply.

Microwave Subassembly. The microwave subassembly couples the antenna to the receiving and transmitting circuits of the transponder. A circulator in the microwave subassembly permits dual use of the antenna for both transmitting and receiving without significant leakage of the transmitted signal into the receiver circuits. The signals transmitted from the RR are routed through the circulator to the mixer. The circulator permits signal transfer in the direction of the mixer only. The mixer combines the received signal with a 9792-mc signal developed in the X96 frequency multiplier. The mixer output, a 40.8-mc IF signal, is amplified by the preamplifier and routed to the receiver subassembly. The X96 frequency multiplier develops the 9792-mc signal by multiplying the 102-mc output of the carrier lock loop subassembly by 96. The 9792-mc signal is used as the carrier frequency for the transponder transmitter signal. The 9792-mc signal is applied to the tone phase modulator where the range tones from the originally received signal are phase modulated onto the newly generated transponder carrier signal (9792-mc plus Doppler); this signal is transmitted for RR acquisition.

The modulated output of the tone phase modulator is coupled to the antenna waveguide through the circulator. The modulated transmitted energy enters one port of the circulator whose magnetic and RF fields cause it to be rotated clockwise to the output port and out to the antenna.

When enabled, the self-test circuit applies 12 volts dc to an oscillator. The oscillator provides a 40.8-mc test signal, which is applied to the preamplifier of the first mixer. This self-test signal takes the place of the 40.8-mc output of the mixer and is used to self-test the receiver subassembly.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Receiver Subassembly. The 40.8-mc IF signal from the microwave subassembly is amplified by the 1st IF amplifier and applied to the 2nd mixer. The amplifier output is mixed with a 34-mc signal from the carrier lock loop subassembly to produce a 6.8-mc output.

The 6.8-mc signal is routed through an 8-kc band-pass filter to produce a narrow-band signal and through a 200-kc band-pass filter to produce a wide-band signal. The narrow-and-wide-band signals are both fed to the threshold detector of the carrier lock loop subassembly to permit comparison of the noise levels of the two signals. The 6.8-mc IF signal is also routed to the tone modulation detector subassembly through an output buffer and gating circuit.

The receiver subassembly also provides an AGC loop for the transponder electronics assembly. The AGC loop controls the preamplifier of the 1st mixer in the microwave subassembly and the 1st IF amplifier in the receiver subassembly. The narrow-band output of the 8-kc band-pass filter is passed, through a 1-kc filter, to the AGC amplifier and detector. By filtering out the noise, the narrow-band filter assures that AGC response is mostly determined by the received signal level. The AGC loop maintains the 6.8 mc receiver output signal constant despite the variation of the signal input from the RR.

Carrier Lock Loop Subassembly. By use of a difference amplifier, the lock-up and threshold detector compares the respective signal powers of a 6.8-mc wide-band (100 kc) signal with a 1-kc limited-band signal (+10 db), extracted from the wide-band signal. When the signal power of the ± 500 cps signal (1-kc limited-band) exceeds the 6.8-mc wide-band signal, a stop sweep signal is sent to the acquisition logic and sweep generator. The sweep generator stops its sawtooth sweep of the VCO. The VCO phase positioning is now controlled by an error signal generated by the modulator, which derived this signal by comparing the VCO output with the 6.8-mc narrow-band signal. The difference amplifier switches in a notch filter to remove the 200-cps phase modulation from the error signal. The error signal enables a relay to produce the VCO tune signal. Through the acquisition logic and sweep generator, the VCO tune signal drives the VCO phase positioning to make and keep the VCO output phase-coherent with the 6.8-mc narrow-band signal.

The lock-up detector also compares the 6.8 mc narrow-band signal with the output of the VCO. When the two signals are phase-locked, the detector produces a frequency lock-on discrete, which is used to gate the 6.8-mc signal out of the receiver subassembly. The acquisition logic and sweep generator circuit produces a sawtooth output, which sweeps the VCO linearly with time during search. The 6.8-mc output of the VCO is used as a reference signal in the tone modulation detector subassembly. This 6.8-mc signal is also multiplied by five in a frequency multiplier to produce a 34-mc signal, which is routed to the 2nd mixer in the receiver subassembly and to the X3 frequency multiplier. The 102 mc output of the X3 frequency multiplier is applied to the microwave subassembly.

Tone Modulation Detector Subassembly. Routed from the output buffer and gate (receiver subassembly), the 6.8-mc signal, with RR transmitted tones and Doppler is compared with the 6.8-mc reference signal at the coherent detector (tone modulation detector subassembly). The coherent detector uses the reference signal to ensure that the transponder transmitter signal will be phase coherent with the received signal. The output of the coherent detector is routed to three parallel band-pass filters and amplifier and phase adjust circuits, one each for 200-cps, 6.4-kc and 204.8-kc. The 6.4-kc and 204.8-kc tone filters are in an oven whose temperature is controlled to $160 \pm 1^\circ/\text{F}$ to stabilize tone phase shift. Each strip of circuits filters, amplifies and phase adjusts its respective tone before it is routed to the phase modulator in the microwave subassembly.

Power Supply Subassembly. The power supply subassembly provides all the secondary power (+25, +12, and -12 volts dc) required by the transponder electronics assembly.

2.2.4.5 Landing Radar.

2.2.4.5.1 Antenna Assembly. (See figure 2.2-52.)

The LR antenna assembly is mounted on an antenna pedestal which is bolted to the underside of the descent stage. In antenna position No. 2 (hover), the antenna transmitting and receiving arrays face directly downward; the antenna is parallel to the LM-axis and perpendicular to the LM X-axis. Two interlaced transmitting phased arrays propagate cross-polarized (90° out-of-phase) velocity (10.51 gc) and altimeter (9.58 gc) microwave energy. Four receiving broadside arrays (DVS beams No. 1, 2, and 3, and the altimeter beam) are each structurally positioned so that their maximum-gain field of view coincides with the center of the corresponding received microwave beam. To effect this cross-polarization, the velocity sensor transmitting and receiving arrays have E-wall slots in their multiwaveguides, and the altimeter transmitting and receiving arrays have H-wall slots in their multiwaveguides. A non-absorptive finish, thermal blanket and external thermal shield partially protect the antenna assembly against the effects of blastoff and the thermal conditions of space.

Inside the antenna assembly, four microwave mixers, four dual audio frequency preamplifiers, a velocity sensor transmitter, an altimeter transmitter and a frequency modulator are mounted on a platform, underneath the interlaced transmitting arrays. Specially cut waveguide sections, phase shifters, isolators, and waveguide couplers provide optimum matching, minimize VSWR, and obtain proper phase relationships between the transmitters and transmitting phased arrays and between the four microwave mixers and the four receiving arrays. At the back of each receiving array and at right angles to its parallel waveguides, a coupling waveguide is positioned to collect received microwave energy equally from each of the multiple waveguides. The microwave energy is routed through specially cut waveguide sections to a microwave, quadrature-pair balanced mixer. Phase shifters and isolators within the waveguide sections maintain the $0/90$ phase relationships and thereby retain the Doppler velocity sign sense of the received microwave energy. At a dual preamplifier (figure 2.2-26), this Doppler sign sense is further retained by quadrature-pair lines ($<0^\circ$ and $\leq +90^\circ$). When the received microwave energy conveys a closing Doppler velocity, the $\leq +90^\circ$ line leads the $<0^\circ$ line by $+90^\circ$. A microwave leveler, between the altimeter transmitting array and the altimeter transmitter multiplier, minimizes amplitude modulation of the frequency-modulated transmitter signal by smoothing the amplitude fluctuations.

Velocity Sensor Transmitters. The velocity sensor transmitter is a continuous-wave radar velocity transmitter. It operates at 10.51 gc (X-band) in conjunction with an electronically tilted, transmitting, four-beam, broadside-array antenna. The reflected X-band energy in each beam is frequency shifted in proportion to the relative velocities and beam angles of the LM and the reflecting surfaces. The velocity sensor transmitter has a self-contained, crystal-controlled oscillator.

Altimeter Transmitter. The altimeter transmitter is an FM/CW transmitter, which does not contain an oscillator. The transmitter receives its input, which is a sawtooth frequency modulated at a repetition rate of 130 cps, from the frequency modulator. The altimeter X-band (9.58 gc) microwave energy is applied to the antenna. The transmitted energy is continuously sampled, and mixed with any returned energy. The resultant difference of the mixing process (range frequency) is proportional to the sawtooth frequency deviation rate and twice the distance to the reflecting surface. In addition to range frequency, there is a frequency component (Doppler) due to relative velocity along the altimeter beam. This Doppler component (D_4) is proportional to velocity along the altimeter beam and adds to, or subtracts from, the range frequency, depending on the velocity. A closing velocity results in a positive Doppler signal.

Frequency Modulator. The frequency modulator output is an FM signal that is sawtooth modulated at a repetition rate of 130 cps. The average center frequency of the modulator is 99.8 mc; average output is 14 millivolts. One of two FM deviation magnitudes are used, depending on the altitude at which the LR is used.

Antenna Heaters and Temperature Control. (See figure 2.2-27.) The antenna assembly, designed for a temperature range of 0° to 185°F , is mounted off a surface whose temperature can vary from -260° to $+260^\circ\text{F}$ during translunar and lunar-orbital periods. An external thermal shield, thermal blankets, and a non-absorptive finish partially protect the antenna from spatial temperature extremes. When the LR is operating, sufficient heat is dissipated to maintain antenna temperature within $+55^\circ$ and $+150^\circ\text{F}$.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

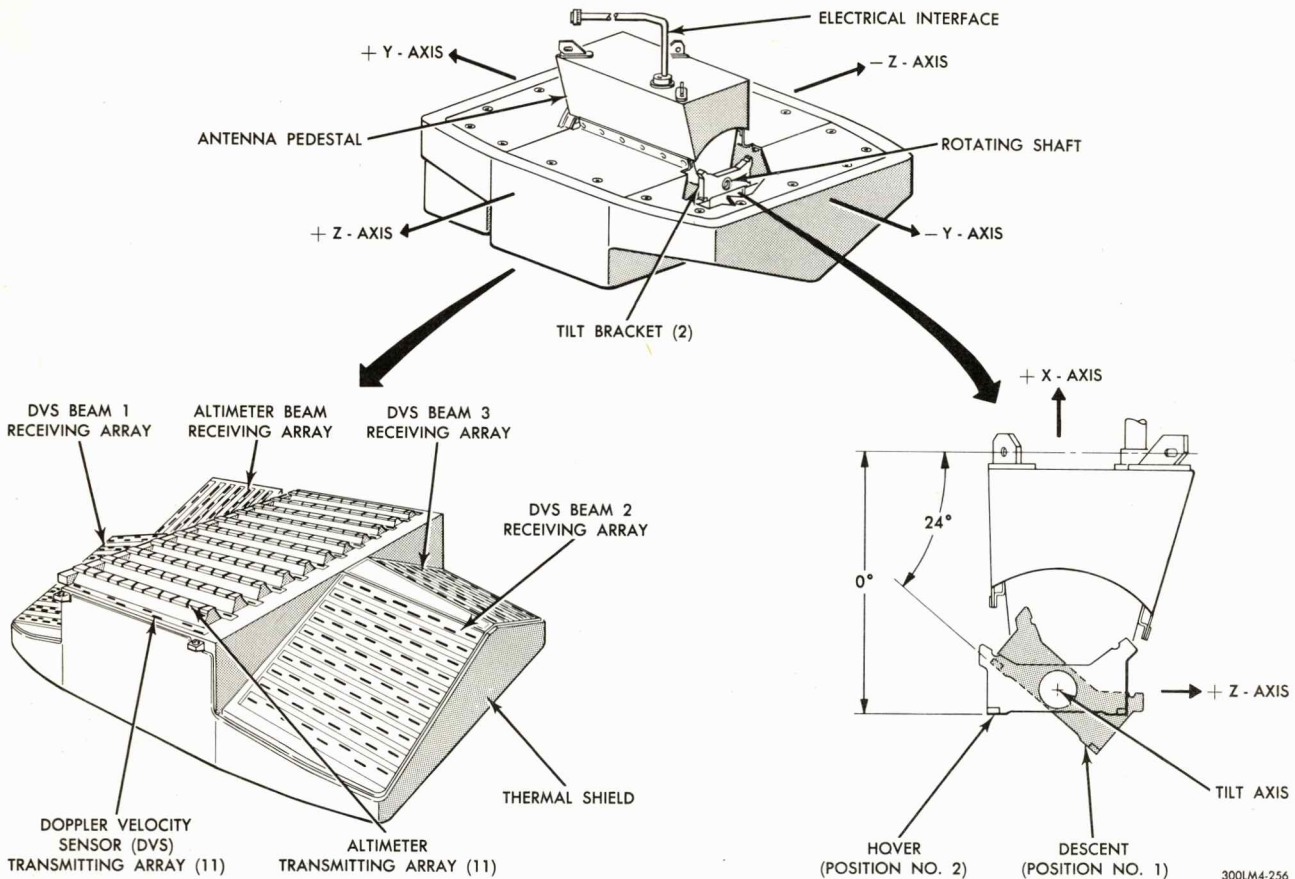


Figure 2.2-52. Landing Radar Antenna Assembly

When the LR is not operating, an antenna temperature control and antenna heaters are used to maintain antenna temperature within $+55^{\circ}\text{F}$ and $+70^{\circ}\text{F}$. The antenna heaters are cal-rod heating elements that are imbedded in the antenna transmitting array support structure. With the HEATERS: LDG RDR circuit breaker closed and the PGNS: LDG RDR circuit breaker open, $+28$ volts dc is applied to the antenna heater control, which then furnishes heater power, through a power amplifier, to the antenna heaters. Using thermistors to monitor the antenna temperature and controlling switch circuits, the antenna heater control switches in the power amplifier to apply power to the heaters when the antenna temperature drops below $+55^{\circ}\text{F}$; it switches out the power amplifier when antenna temperature exceeds $+70^{\circ}\text{F}$.

To assure accurate LR frequency tracking, the monitored antenna temperature when the LR is not operating, must not be less than $+50^{\circ}\text{F}$ and should not exceed $+70^{\circ}\text{F}$. With the LR turned on, the antenna temperature must be within $+50^{\circ}$ to $+150^{\circ}\text{F}$ to assure velocity and range data accuracy. When the PGNS: LDG RDR circuit breaker is closed, the LR is activated and antenna heater power is turned off.

An antenna temperature sensor conveys antenna temperature signals to the TEMP indicator (panel 3) and to the CWEA (IS) through the SCEA (IS). When antenna temperature is not within -15.6° and $+148.8^{\circ}\text{F}$, the CWEA causes the HEATER caution light to go on. Setting the TEMP MONITOR switch to LDG causes the HEATER caution light to go off.

Antenna Pedestal Tilt Mechanism. The antenna pedestal tilt mechanism serves as the electrical and mechanical interface between the antenna assembly and the LM. The mechanism places the antenna in either of two operating positions. The antenna is positioned by a tilt actuator, which is controlled manually by setting the LDG ANT switch to the desired position, or automatically by an LGC command if the LDG ANT switch is set to AUTO.

The tilt actuator is a reversible d-c motor with internal limit switches that remove drive power once the extreme of either position is reached. Travel time of the actuator is approximately 10 seconds. Antenna position No. 1 or 2 is indicated by closing of a set of antenna pedestal switch contacts, which generate antenna position No. 1 or 2 discrete signal. This signal is routed to the LGC and to the IS for telemetering to MSFN.

2.2.4.5.2 Electronics Assembly. (See figure 2.2-26.)

The electronics assembly contains frequency trackers (one for each velocity beam), a range frequency tracker, a velocity converter and computer, a range computer, a signal data converter, and a data-good/no-good logic circuit. The electronics assembly for lunar-landing LR provides the LGC with digital outputs that correspond to the range along the altitude beam and velocity in relation to LM coordinates.

Velocity Frequency Trackers. The velocity trackers convert the broadband outputs of the dual AF preamplifiers (antenna assembly) to discrete frequency components. By use of a variable oscillator and a narrow-band filter, each frequency tracker acquires and tracks the center of each doppler spectrum of the returned radar beam. The resultant AF Doppler is mixed with a 614.4 kc subcarrier for further processing in a velocity data converter. Doppler sense representative of opening or closing velocity is interpreted and preserved through the frequency trackers, and a tracker lock-on indication is given to the LGC when a signal of proper characteristics and amplitude is acquired. (For LM-4, this lock-on signal is hardware to be present at all times.) The velocity frequency trackers comprise three identical subassemblies; each one tracks one of the three Doppler velocity sensor beams.

Range Frequency Tracker. The range frequency tracker operates in a manner similar to that of the velocity frequency trackers. The preamplifier output is the sum of the range signal and the Doppler signal. This output is processed by the range frequency tracker to produce two outputs: a tracker lock signal and a range signal that is routed to the range computer.

Velocity Converter and Computer. The velocity converter contains three identical Doppler-processing circuits (D1, D2 and D3 converters). These circuits process the Doppler signals, indicate Doppler sense, and, based on Doppler sense, provide an output to the velocity computer. In addition to the D1, D2, and D3 converters, the velocity converter contains a velocity sensor bandwidth control circuit. This circuit, basically a frequency comparator, switches each velocity frequency tracker to its narrow band-pass mode when tracker output frequency is less than 1.47 kc and switches all these trackers to the narrow band-pass mode when the altitude is less than 2,500 feet.

The velocity computer normally converts the beam Doppler products into antenna velocity coordinate outputs (V_{xa} , V_{ya} , and V_{za}) and applies them to coordinate computers. There are three coordinate computers, one associated with each coordinate. The outputs of these computers are pulse trains whose repetition rates are directly proportional to the measured velocity. During the LM-4 mission, the velocity computer converts the stray beam returns into pulse trains proportional to the stray returns.

Range Computer. The range computer calculates the range from the LR to the lunar surface and removes the Doppler effect along the altimeter beam. The Doppler effect is removed by an adder-subtractor circuit in the computer. The velocity-compensated range signal is then routed to a frequency comparator and to a pulse shaper and gating circuit. The output of the range computer is routed to the signal data converter and to the RANGE indicator.

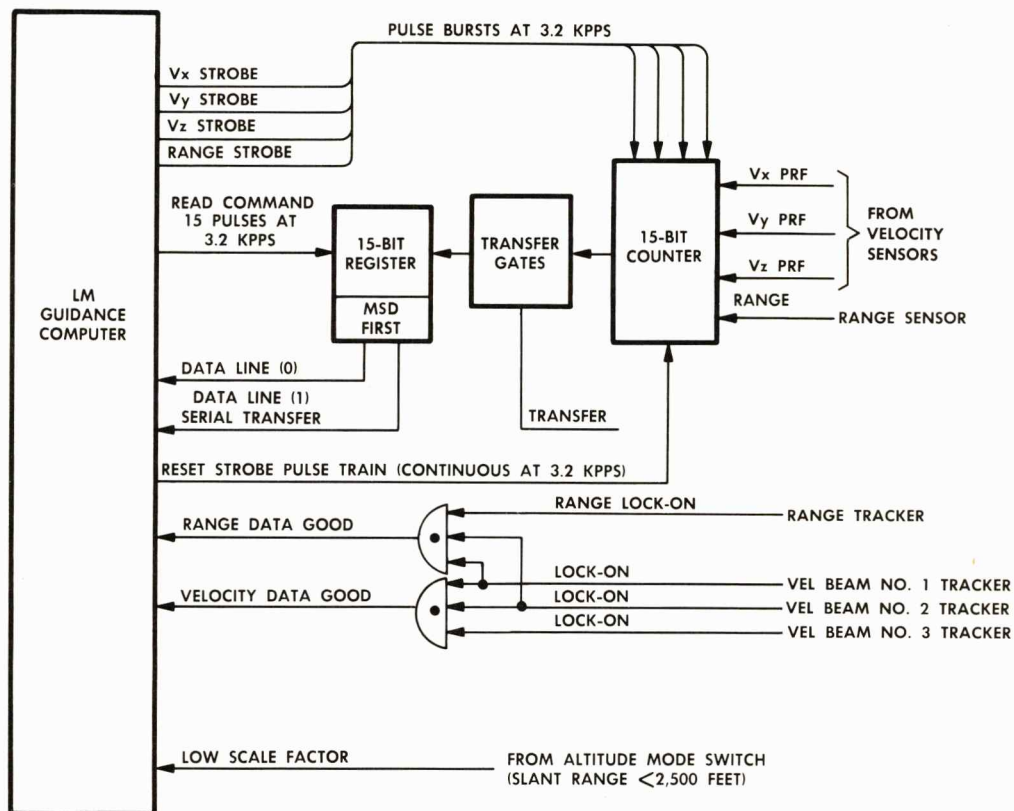
Signal Data Converter. The basic function of the signal data converter is to convert the LR computer outputs to the 15-bit format required by the LGC. To accomplish this, the signal data converter uses a 15-bit counter, transfer gates, and a 15-bit shift register. (See figure 2.2-54.) The LGC

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Figure 2.2-53. Deleted (See figure 2.2-26.)

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Page 2.2-174 Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968



A-300LM4-167

Figure 2.2-54. Landing Radar Digital Data Transfer - Functional Diagram

provides the timing signals that sequentially select the PRF input to be counted, determine the counting time, and provide the data readout command required to shift out the contents of the shift register. Before the LGC issues transfer commands, range (altitude) data-good and velocity data-good discretes must be received from the radar. During the LM-4 mission, these two discretes are present as soon as the LR is turned on. Upon receipt of these discretes, the LGC issues strobe pulses, alternately, to the 15-bit accumulator or counter in the digital data converter. This permits stray return data to accumulate in the counter, depending upon which strobe line is used. The data are then transferred to the 15-bit shift register by an internal timing pulse. The LGC initiates a readout command, consisting of a train of 15 pulses, to the shift register. Each bit in the shift register is then transferred to the LGC input registers over a data line (0 or 1). The 15-bit data word is shifted out, most significant digit (MSD) first. The two data lines permit a parity check of the transfer (serial) functions to be made by the LGC, because both lines (1 and 0) are not simultaneously high or low.

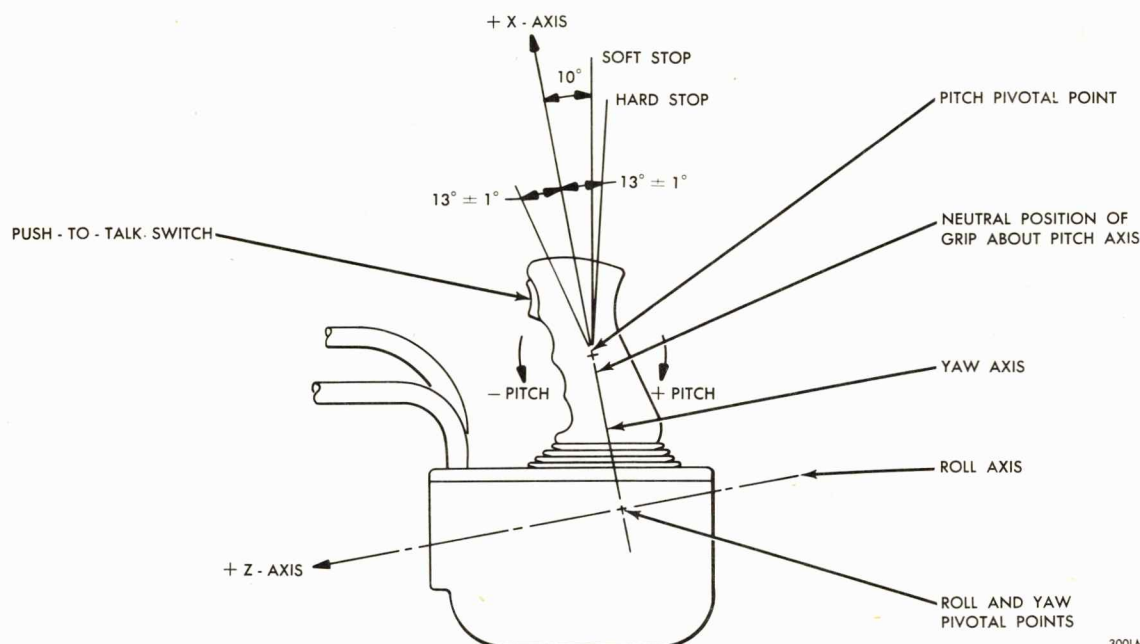
2.2.4.6 Control Electronics Section.

2.2.4.6.1 Attitude Controller Assemblies. (See figure 2.2-55.)

The ACA is used for closed-loop proportional rate commands in the attitude hold mode and for open-loop acceleration commands in the pulse, direct, and hardover modes (see figure 2.2-56).

For ACA proportional control, a 28-volt, 800-cps signal is applied to the primary coil of three proportional transducers. The secondary voltage of each transducer is proportional to ACA displacement about its input axis (one each for the roll, pitch, and yaw axes). When ACA displacement exceeds 0.5° , two sets of switch contacts close. One set of contacts sends an out-of-detent discrete to the LGC and a followup discrete to the AEA if the PGNS MODE CONTROL switch is set to ATT HOLD.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



300LM4-204

Figure 2.2-55. Attitude Controller Assembly

The other set of contacts sends a proportional secondary voltage to the LGC or the ATCA. The output voltages are 0.28 volt degree of displacement (1° to 11°). Positive rotation commands generate in-phase voltage; negative commands, 180° out-of-phase voltage.

During attitude control (ROLL, PITCH, or YAW ATTITUDE CONTROL switch set to MODE CONT) when the minimum pulse mode is selected via the DSKY and the PGNS is in control, the LGC common applied to the plus and minus switches of the ACA is returned to the LGC when the ACA is displaced more than 2.5° . When the AGS is in control and the pulse mode is selected for a given axis (ROLL, PITCH, or YAW ATTITUDE CONTROL switch set to PULSE), positive or negative d-c voltages are applied to the plus and minus switches of the ACA. When the ACA is displaced 2.5° or more from neutral, a positive or negative d-c voltage is applied to the ATCA. A positive voltage commands a positive acceleration (pulsed) about the selected axis; a negative voltage commands negative acceleration. With the ROLL, PITCH, or YAW ATTITUDE CONTROL switch set to DIR, 28 volts dc is applied to two RCS secondary solenoids when the ACA is displaced more than 2.5° .

In any attitude control operation, if the ACA is moved 12° or more in any axis (hardover mode), the ACA activates four TCA's via the secondary coils. With either the PGNS or AGS MODE CONTROL switch set to ATT HOLD and the ROLL, PITCH, and YAW ATTITUDE CONTROL switches (one or more) set to MODE CONT, the ACA can command LM attitude changes about the selected axis or about as many as three axes simultaneously.

2.2.4.6.2 Thrust/Translation Controller Assemblies. (See figures 2.2-57 and 2.2-58.)

A pair of detent switches and a pair of limit switches are located about each TTCA axis; a nonlinear potentiometer is located along the longitudinal axis. The detent switches provide a discrete to the LGC when the astronaut operates the TTCA 0.25 inch out of its detent (neutral) position. The discrete commands the LGC to issue translation change commands in accordance with the TTCA motion. When the AGS is controlling the vehicle, the limit switches issue on and off translation commands directly to the RCS jet logic and driver circuitry. The nonlinear potentiometer provides an electrical output, which is proportional to TTCA displacement, when the THROTTLE-JETS select lever is set to THROTTLE. These signals are routed to the DECA to command proportional control of thrust (magnitude). The nonlinear characteristic provides increased servo characteristics.

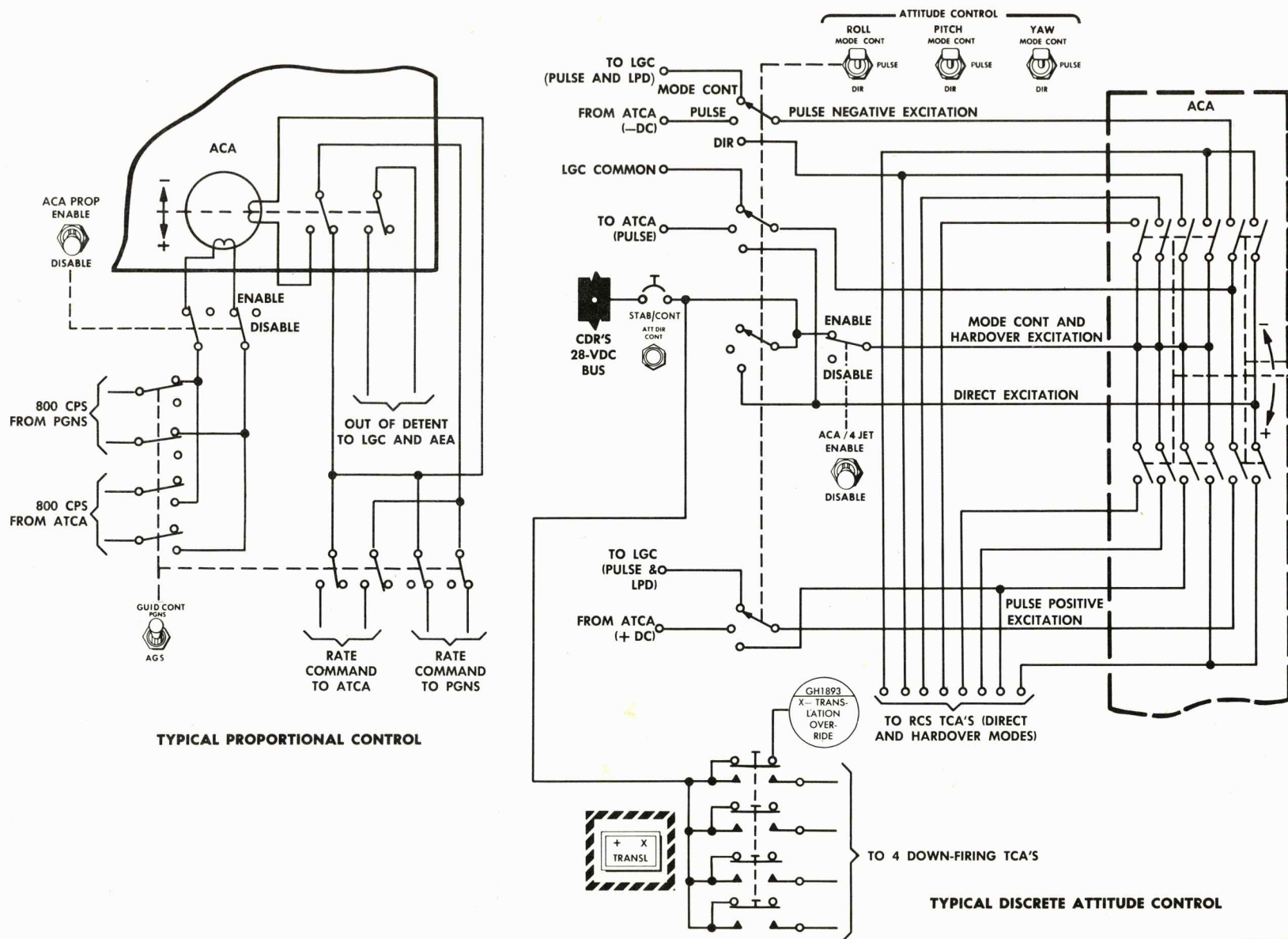
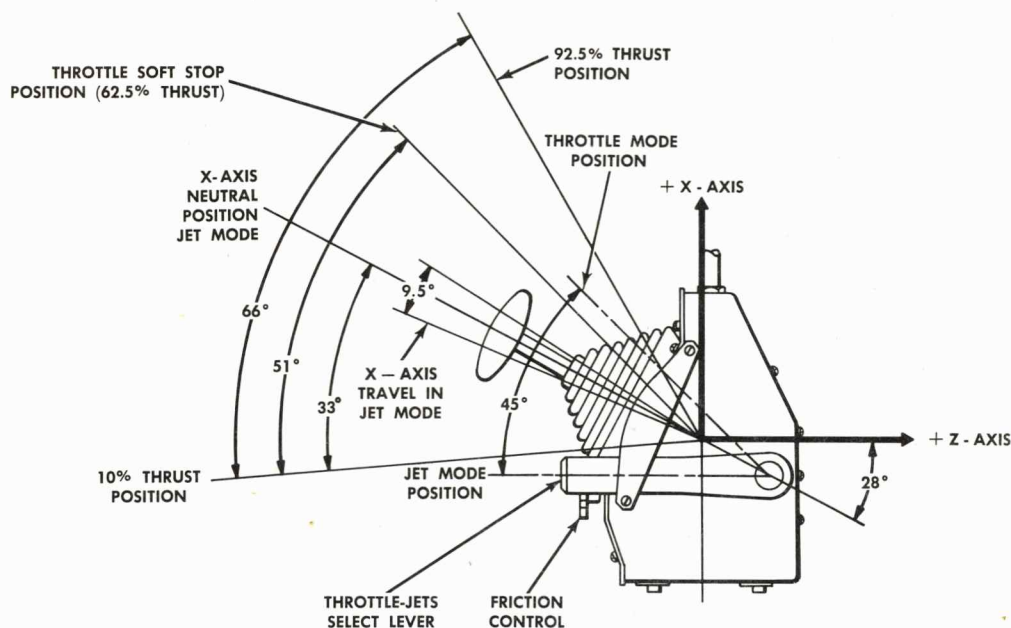


Figure 2.2-56. Attitude Controller Assembly - Functional Diagram



300LM4-51

Figure 2.2-57. Thrust/Translation Controller Assembly

2.2.4.6.3 Attitude and Translation Control Assembly. (See figure 2.2-58A.)

The ATCA consists of an attitude error signal conditioner-limiter, rate signal summation amplifier, demodulator-filter network, deadband circuits, jet select logic, pulse ratio modulators, and solenoid driver preamplifiers and jet solenoid drivers.

Attitude Error Signal Conditioner-Limiter. To provide the proper scale factor, impedance match, and limiting, the 800-cps attitude error signal is passed through a signal conditioner-limiter before summation with the rate gyro signal. Limiting the attitude error prevents command of excessive vehicle rates during narrow deadband operation only. The limiting action of the limiter is eliminated during wide deadband operation.

Rate Signal Summation Amplifier. The attitude error and rate and ACA signals are a-c summed and amplified before demodulation by the amplifier, because they are all synchronized to the 800-cps power source. Amplification and summing at this point helps to minimize error caused by drift.

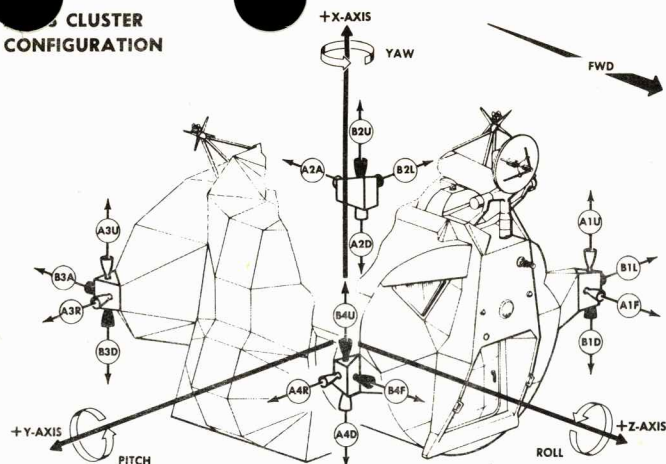
Demodulator-Filter Network. A keyed demodulator is used to convert summed and amplified error signals to dc. The synchronized 800-cps power source of the ATCA is used for the demodulator reference. An active filter is used to obtain 12-db/octave attenuation for frequencies above 18 cps. The filtered roll and pitch signals are routed to the DECA for descent engine trim.

Deadband Circuits. Either of two deadband circuits can be selected manually with the DEAD BAND switch. The wide deadband (DEAD BAND switch set to MAX) is used for coasting-period limit cycling to conserve fuel. The narrow deadband (DEAD BAND switch set to MIN) is used during periods when the ascent or descent engine is thrusting. In addition to the manual capability, an automatic minimum deadband override is provided for use during ascent or descent engine thrusting.

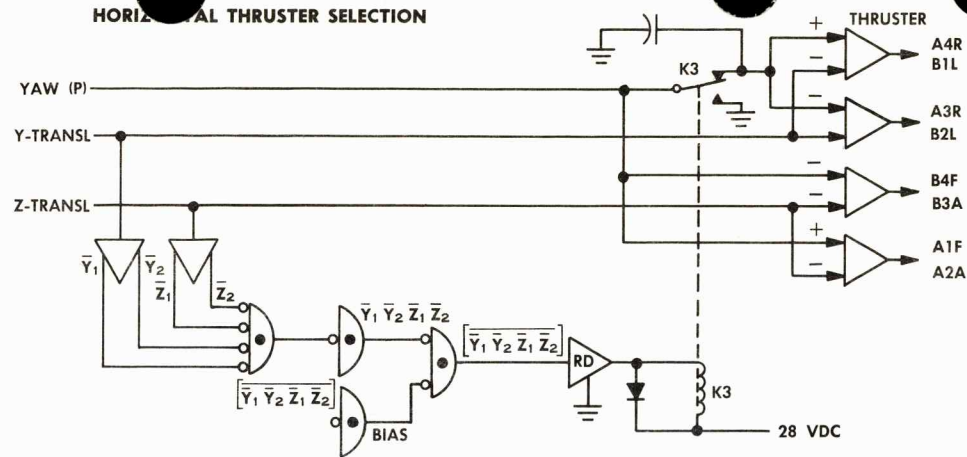


GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM			
Mission LM 4	Basic Date	15 June 1968	Change Date 15 September 1968 Page 2.2-180A/2.2-180B

CLUSTER CONFIGURATION



HORIZONTAL THRUSTER SELECTION



VERTICAL THRUSTER SELECTION

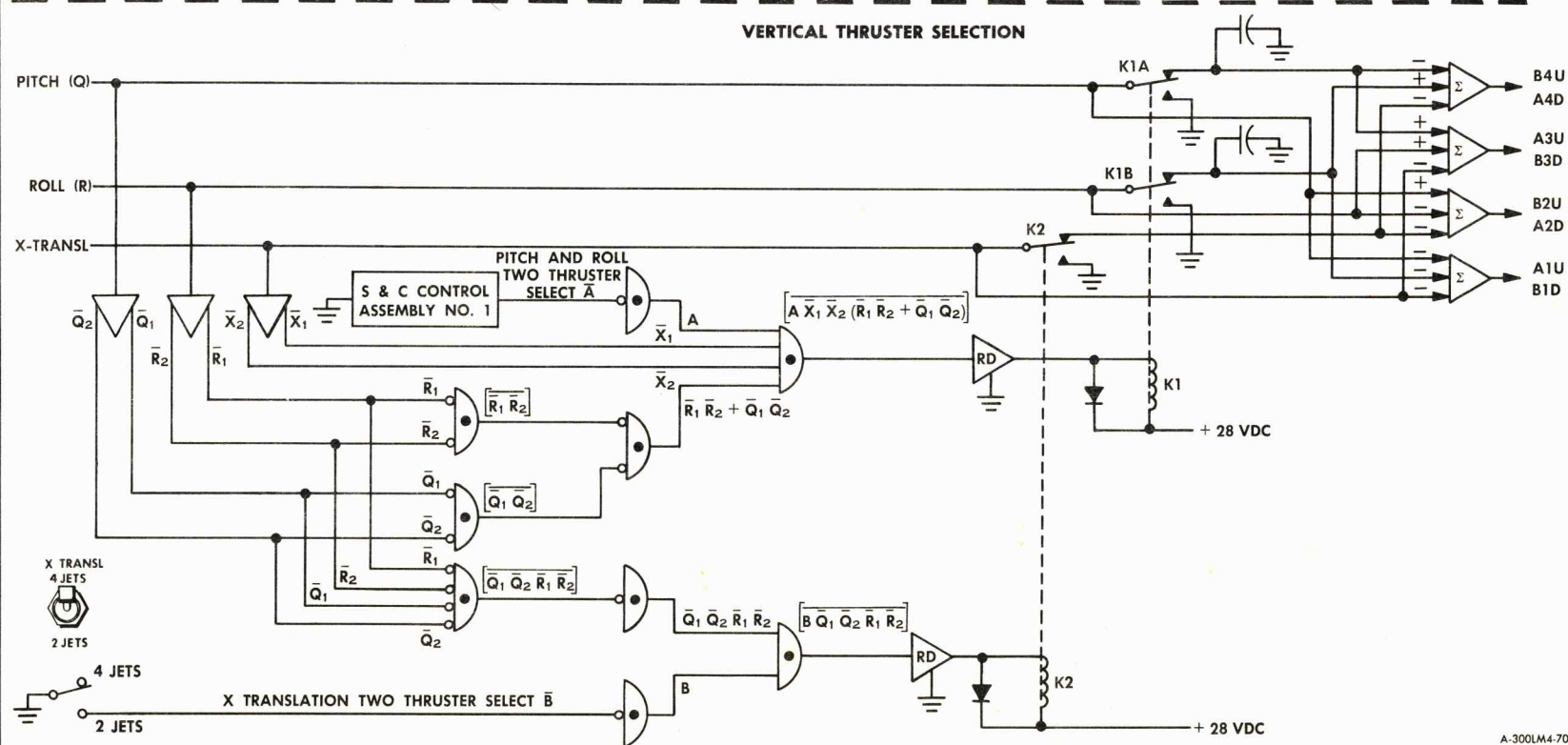
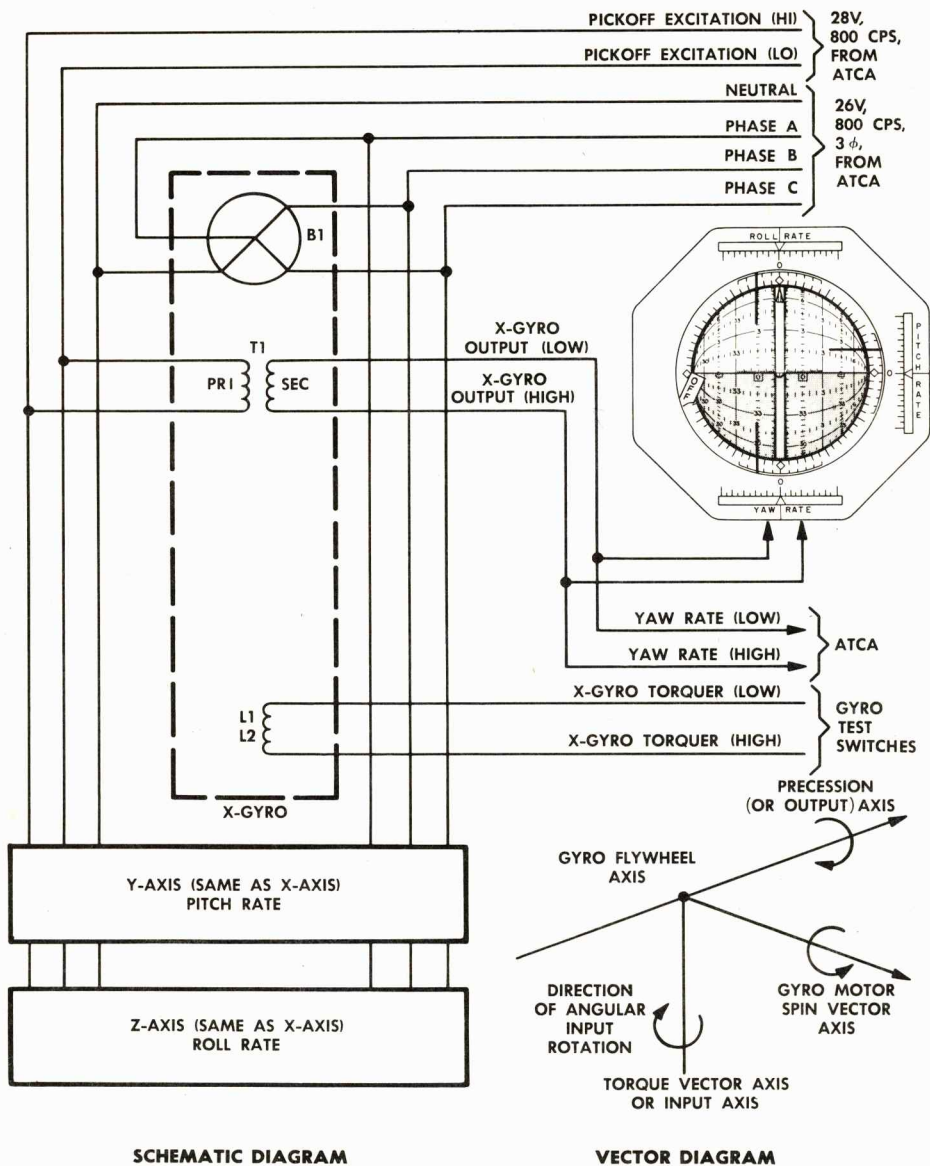


Figure 2.2-59. Attitude and Translation Control Assembly - Jet Select Logic

A-300LM4-70

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

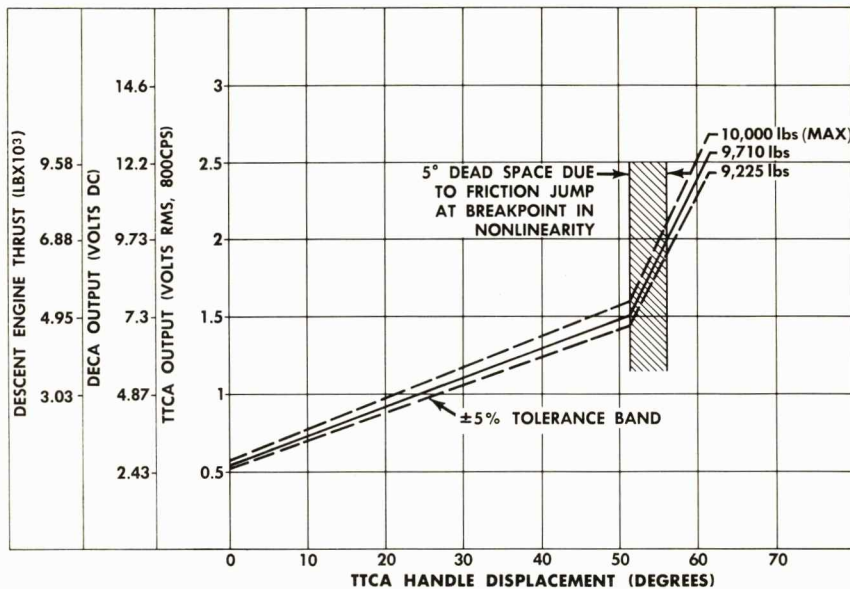


300LM4-92

Figure 2.2-60. Rate Gyro Assembly - Schematic and Vector Diagrams

The active TTCA always provides at least a 10% command. These commands drive a nonlinear circuit that provides the desired thrust level. With nonlinear thrust levels, more resolution is obtained at the lower levels. (See figure 2.2-66.) Once the nonlinear region of the TTCA is reached, the TTCA should be displaced to its hard stop to prevent erratic descent engine operation in the thrust range between 60% and 90%.

To apply automatic throttle commands to the descent engine, the THR CONT switch must be set to AUTO. The automatic commands appear on two separate lines (throttle increase and throttle decrease) as 3,200-cps pulse inputs to an integrating d-c counter (up-down counter). Each pulse corresponds to a 2.7-pound thrust increment. Throttle increases can drive the counter up to 12 volts, which, when combined with the TTCA fixed 10% output in the descent engine throttle servoamplifier, corresponds to 92.5% thrust.



A-300LM4-131

Figure 2.2-65. Throttle Control vs TTCA Displacement

On and off signals are supplied to the GDA's from the power gate circuits in the DECA. The DECA provides 115-volt, 400-cps power to the GDA motor, which controls the stroke of the GDA at a rate of 0.0662 inch per second (7.5 volts per inch) $\pm 10\%$, in a positive or negative direction along the Y-axis and Z-axis. Gimbal position signals are supplied by the feedback potentiometer to a failure-detection circuit in the DECA.

2.2.4.6.7 800-cps Synchronization Loop. (See figure 2.2-67.)

Because the CES uses 800-cps analog signals as a reference, the various assemblies must be synchronized. In the primary guidance mode, the ACA's and TTCA's receive 28-volt, 800-cps signals from the power and servo assembly of the PGNS. The proportional attitude commands to the LGC are either in phase or 180° out of phase with this 800-cps signal. In the abort guidance mode, the ACA's and TTCA's receive 28-volt, 800-cps signals from the ATCA. The proportional attitude commands fed back to the ATCA are now synchronized with the ATCA power supply. The AGS and RGA receive 28-volt, 800-cps signals from the ATCA; their outputs returned to the ATCA are synchronized. The FDAI's also receive the 800-cps synchronization voltage to properly display the RGA signals.

2.2.4.6.8 S&C Control Assemblies.

The S&C control assemblies contain three similar assemblies. The S&C control assemblies process, switch, and/or distribute the various signals associated with the GN&CS. One of the assemblies also processes automatic and manual ascent or descent engine on and off commands. The circuits used for the ascent engine commands are called the ascent engine latching device (AELD). The ascent engine commands are processed, and engine ignition and shutdown commands are sent to the ascent engine and staging sequence commands are sent to explosive devices. The first time that the ascent engine is ignited, the engine-on command starts the staging sequence, using the Explosive Devices Subsystem. The S&C control assemblies contain a memory circuit that maintains the ascent engine on if the automatic on signal is lost, until an automatic off signal or a manual stop command is received.

Descent engine commands are processed and sent to the DECA. These commands (descent engine arm, and automatic and manual engine on and engine off) are processed again by the DECA for descent engine ignition and shutdown.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

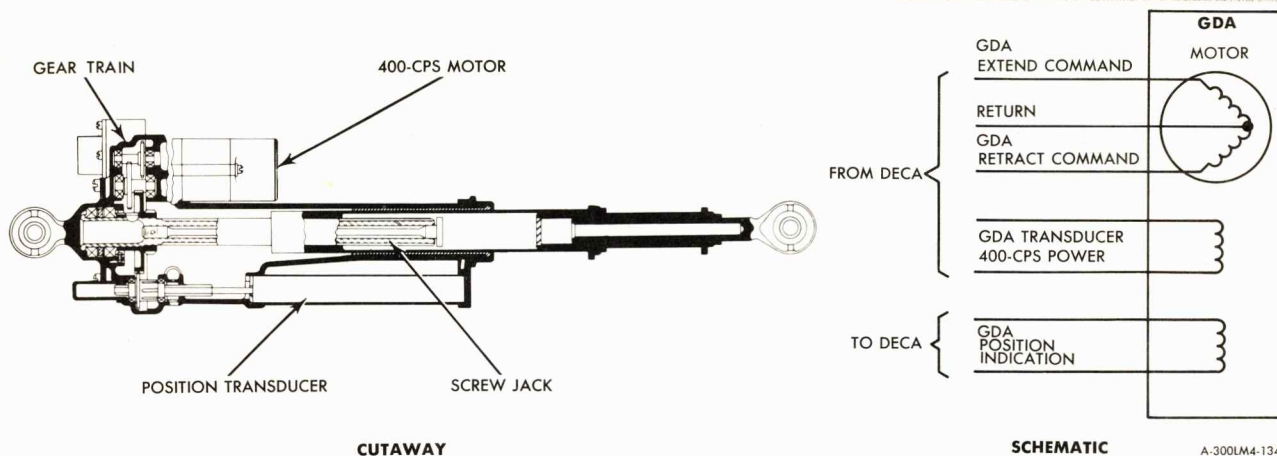


Figure 2.2-66. Gimbal Drive Actuator - Cutaway

2.2.4.6.9 Gimbal Angle Sequencing Transformation Assembly. (See figure 2.2-68.)

The GASTA operates in conjunction with the FDAI. The GASTA modifies the analog gimbal angle signals from the IMU before they are applied to the FDAI. Modification conditions the signals to drive the FDAI sphere. The GASTA converts 26-volt, 800-cps signals from the IMU to 15-volt, 400-cps signals. The signals from the IMU must be routed through the GASTA for conditioning. Because the AGS has its own GASTA (or conditioning) function, its gimbal angle signals are routed directly to the FDAI.

2.2.4.7 Orbital Rate Display - Earth and Lunar. (See figure 2.2-69.)

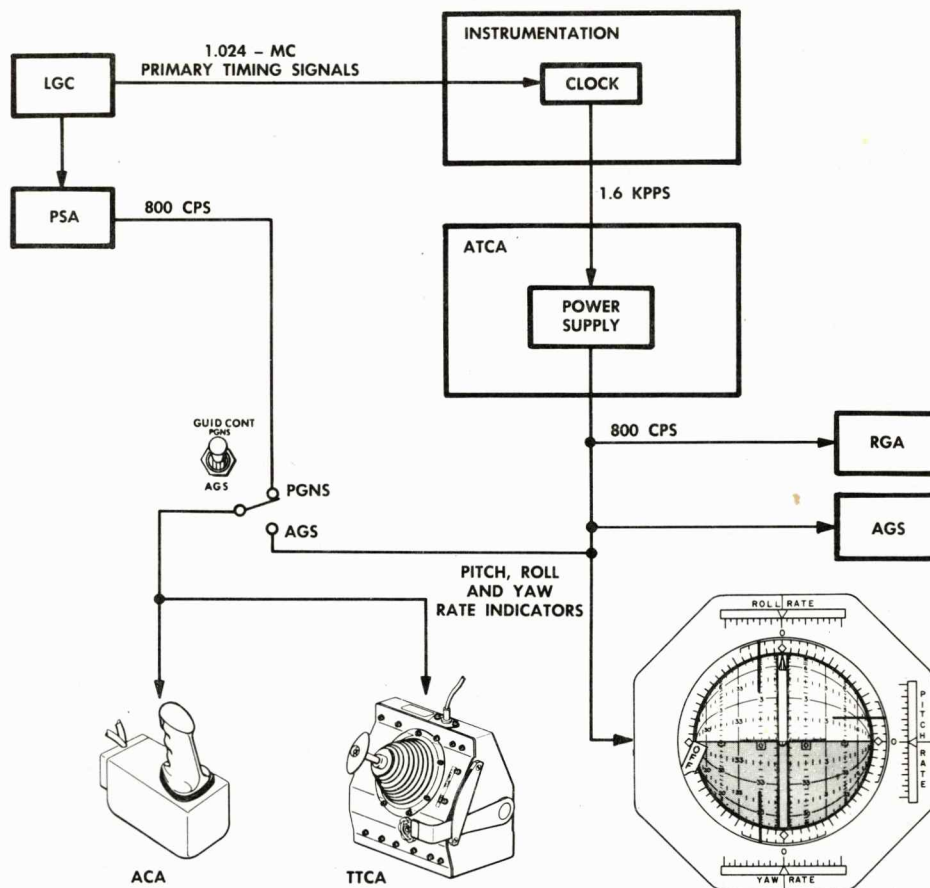
The ORDEAL is an electromechanical device that contains a pair of resolvers, one for each FDAI. The resolvers are driven by a stepper motor, at the orbital rate selected with the ALT SET control on the ORDEAL panel. They may be switched directly into the pitch total attitude channels of the FDAI's.

The resolvers provide a display of spacecraft attitude with respect to a local horizontal reference frame. They transform the pitch angle in accordance with their shaft position, which is driven by a digital rate servo. The desired shaft rate is obtained by controlling the pulse rate which is applied to the stepper motor. This pulse rate is derived from the LM 400-cps frequency supply, as required by digital logic. Because the transformation is made in the pitch channel only, it is imperative that the inertial reference be aligned with the +Y-axis of the reference along the cross product of the velocity vector (V) and the position vector from the center of the earth (\bar{R}).

The ORDEAL is used after obtaining the LM attitude with respect to the local horizontal, from the LGC. The selected FDAI (commander's or LM Pilot's) is slewed to this angle. The ORDEAL then torques its resolvers at the orbital rate, specified by the ALT SET control, of a circular orbit.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

GN&CS



300LM4-130

Figure 2.2-67. Control Electronics Section - 800 cps Synchronization Loop.

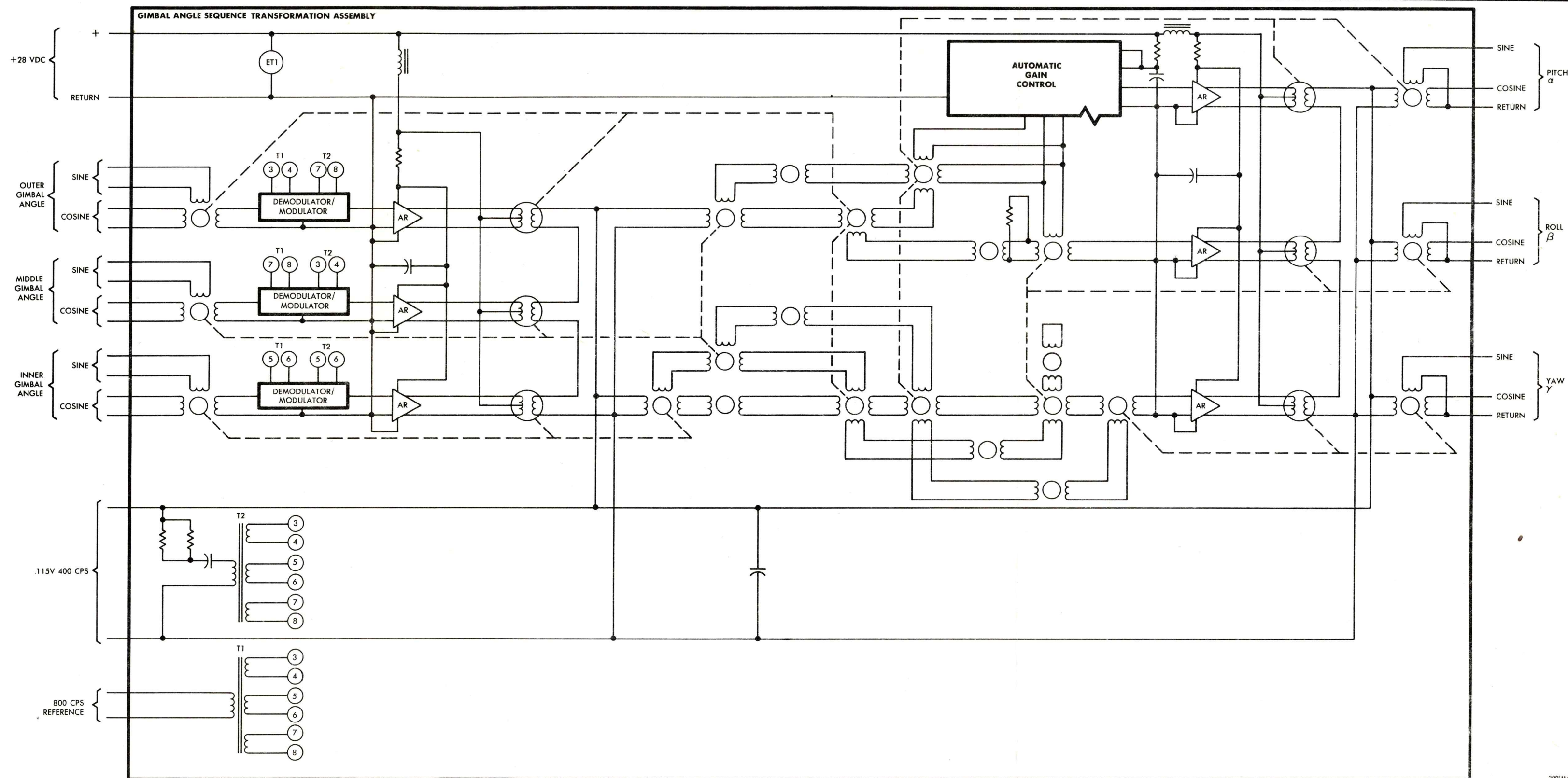


Figure 2.2-68. Gimbal Angle Sequence Transformation Assembly - Schematic Diagram

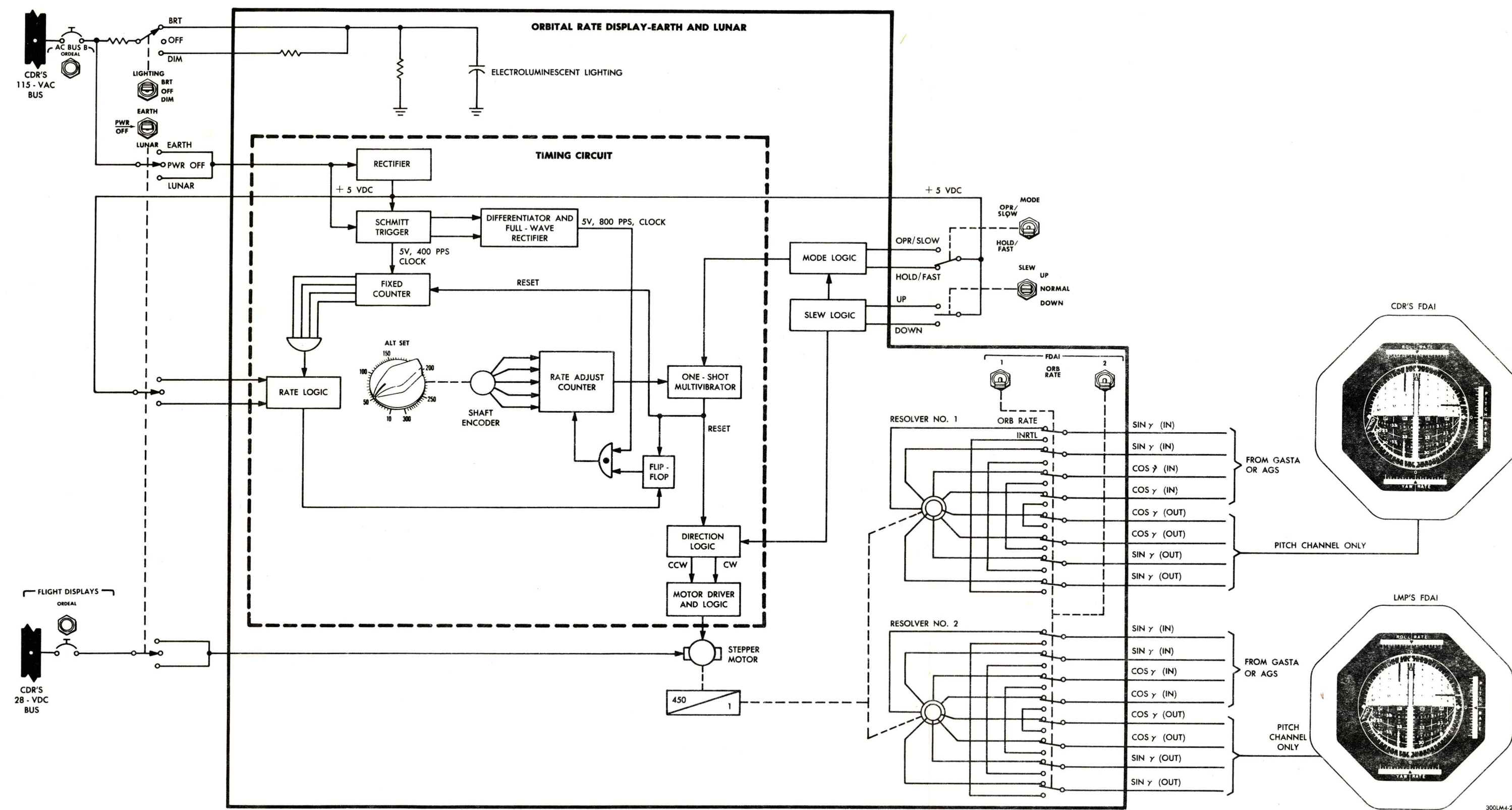


Figure 2.2-69. Orbital Rate Display - Earth and Lunar - Schematic Diagram

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

2.3 MAIN PROPULSION SUBSYSTEM.

2.3.1 INTRODUCTION.

The Main Propulsion Subsystem (MPS) consists of the descent propulsion section (DPS) and the ascent propulsion section (APS). Each section is complete and independent of the other and consists of a liquid-propellant rocket engine with its own propellant storage, pressurization, and feed components. Although the LM 4 vehicle is used for an earth-orbital mission, the DPS can provide the thrust to control descent to the lunar surface. The APS can provide the thrust for ascent from the lunar surface. In case of mission abort, the APS and/or DPS propulsion section can place the LM into a rendezvous trajectory with the CSM from any point in the descent trajectory. (For a lunar-landing mission, there is a deadman zone immediately above the lunar surface, where abort cannot be accomplished.) The choice of engine to be used depends on the cause for abort, on how long the descent engine has been operating, and on the quantity of propellant remaining in the descent stage.

Both propulsion sections use identical hypergolic propellants: a 50-50 mixture, by weight, of hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine (UDMH) as the fuel; nitrogen tetroxide (N_2O_4), as the oxidizer. The injection ratio of oxidizer to fuel is 1.6 to 1, by weight.

Basic operation of the two propulsion sections is similar. In each section, gaseous helium forces the propellants from their tanks, through control valves; to the engine injectors. The DPS uses supercritical helium for propellant pressurization; the APS uses ambient gaseous helium. The primary reason for using supercritical helium is the weight saving obtained, due to its greater density. Both the descent and ascent engine assemblies consist of a combustion chamber, where the propellants are mixed and burned; an injector that determines the spray pattern of the propellants injected into the combustion chamber; and propellant control valves and orifices that meter, start, and stop propellant flow to the engine upon command. The descent engine is larger and produces more thrust than the ascent engine. It is throttleable for thrust control, and is gimballed for thrust vector control. The ascent engine is neither throttleable nor gimballed. Redundancy of vital components in both propulsion sections provides a high reliability factor.

Before starting the descent or ascent engine, proper propellant settling must be established. This is accomplished by moving the LM in the +X-direction to cause the propellants to settle at the bottom of the tanks. As the propellants are consumed, tank ullage increases and more propellant settling time is required for each subsequent engine start. The +X-translation is accomplished by operating the Reaction Control Subsystem (RCS) downward-firing thrust chamber assemblies (TCA's). Two or all four downward-firing TCA's can be selected; depending upon whether RCS propellant conservation (two TCA's) or a shorter RCS firing time (four TCA's) is the major consideration.

2.3.2 DESCENT PROPULSION SECTION INTERFACES. (See figure 2.3-1.)

The DPS receives 28-volt d-c and 115-volt a-c primary power through the Commander's and LM Pilot's buses of the Electrical Power Subsystem (EPS). The outputs of the DPS pressure and temperature transducers, liquid-level sensors, and valve position indicator switches are processed in the Instrumentation Subsystem (IS) and are transmitted via the Communications Subsystem (CS) to MSFN. The IS also processes the DPS caution and warning and display signals. The Explosive Devices Subsystem (EDS) opens explosive valves in the DPS to enable propellant tank pressurization and venting.

The Guidance, Navigation and Control Subsystem (GN&CS) issues automatic on and off commands, gimbal drive actuator commands, and thrust level commands to the descent engine. The automatic on and off commands and thrust level commands can be overridden manually. Descent engine arming and ignition are controlled by automatic guidance equipment, or by the astronauts through the stabilization and control (S&C) control assembly and the descent engine control assembly (DECA). A descent engine arm signal is sent to the S&C control assembly when an astronaut sets the ENG ARM switch (panel 1) to DES or when he presses the ABORT pushbutton (panel 1) preparatory to starting a mission abort program, using the descent engine. Engine-on signals from the LM guidance computer (LGC) or abort guidance section (AGS) are sent to the DECA through the S&C control assembly. The DECA turns the descent engine on upon receiving the arm and the engine-on signals. If DECA power fails, the DES ENG CMD OVRD switch (panel 3), in the ON position, will supply an alternate voltage source to keep the engine firing. The engine remains on until the engine-off command is received from the automatic guidance equipment. The astronauts can also generate engine on and off commands

manually; these commands override the automatic commands. A manual start is accomplished (after propellant tank pressurization with ambient helium) by arming the descent engine and pressing the START pushbutton (panel 5). Either astronaut can shut off the descent engine by pressing his STOP pushbutton (panels 5 and 6) or by pressing the ABORT STAGE pushbutton (panel 1). An abort-stage command results in immediate descent engine shutdown, automatically followed by ascent propellant tank pressurization, stage separation, and ascent engine firing.

Descent engine throttling is controlled by the LGC or the astronauts. The throttling-range limitations are from minimum thrust (approximately 10% of 10,500 pounds) to approximately 60%, and full throttle (approximately 92.5%). The range between 60% and 92.5% is a transient region that should not be used for extended periods because of excessive engine erosion and mixture ratio shifts. Under normal conditions, the engine is not operated in the transient region because above 60% thrust LGC throttle commands are generated to produce a full throttle output. Only in case of malfunction can inadvertent throttling occur in the transient region, in which case manual correction must be made. Automatic throttle increase and decrease signals from the LGC are sent to an integrating counter in the DECA. The analog output of the DECA controls descent engine thrust. In the automatic mode, the thrust-translation controller assemblies (TTCA's) can be used by the astronauts to increase descent engine throttle (over-riding the automatic throttle command); the TTCA's cannot be used, however, to decrease the throttle command. (Refer to paragraph 2.3.3.3.1.) In the manual throttle mode, the astronauts have complete control over descent engine thrust.

The primary guidance and navigation section (PGNS) or the AGS automatically controls descent engine gimbal trim, to compensate for center-of-gravity offsets during descent engine firing. In PGNS operation, the LGC sends trim on and off signals in two directions, for each gimbal axis, to the DECA. These signals operate power control circuitry, which drives the two gimbal drive actuators (GDA's). In AGS operation, Y- and Z-axis error signals from the attitude and translation control assembly (ATCA) are sent to the DECA to drive the GDA's. The GDA's tilt the descent engine along the Y-axis and Z-axis a maximum of 6° from the center position. The ENG GMBL switch (panel 3) permits removing GDA power to interrupt the tilt capability if the ENG GMBL caution light (panel 2) goes on, indicating a malfunction.

The DPS modes of operation are discussed in detail in paragraph 2.2.3.7. The control circuitry is shown in figure 2.2-33.

2.3.3 DESCENT PROPULSION SECTION FUNCTIONAL DESCRIPTION.

The DPS consists of an ambient helium bottle and a cryogenic helium storage vessel with associated helium pressurization components; two fuel and two oxidizer tanks with associated feed components; and a pressure-fed, ablative, throttleable rocket engine. The engine can be shut down and restarted, within operational limitations and restrictions (paragraph 2.3.7), as required by the mission. At the fixed full-throttle position, the engine develops a nominal thrust of 9,710 pounds; it can also be operated within a nominal range of 1,050 to 6,300 pounds of thrust.

The engine is mounted in the center compartment of the descent stage cruciform (figure 2.3-2); it is suspended, at the throat of the combustion chamber, on a gimbal ring that is part of the engine assembly. The gimbal ring is pivoted in the descent stage structure, along an axis normal to that of the engine pivots. The engine can be tilted up to +6° or -6°, by means of the GDA's, to ensure that the thrust vector passes through the LM center of gravity.

Functionally, the DPS can be subdivided into a pressurization section, a propellant feed section, and an engine assembly.

2.3.3.1 Pressurization Section. (See figure 2.3-3.)

Before earth launch, the propellant tanks are only partly pressurized, so that the tanks will be maintained within a safe pressure level under the temperature changes experienced during launch and earth orbit. Before initial engine start, the ullage space in each propellant tank requires additional pressurization. This initial pressurization (prepressurization) is accomplished with ambient helium. (Supercritical helium cannot be used because the helium circulating through the fuel/helium heat exchanger may freeze the fuel before fuel flow is established.) A pressure transducer at the outlet port of the ambient helium bottle supplies a signal through the HELIUM MON selector switch (panel 1), when

MAIN PROPULSION SUBSYSTEM

and descent stages separate.) Under regulator failure conditions, the astronauts must close the solenoid valve in the malfunctioning leg and open the solenoid valve in the redundant leg, to restore normal propellant tank pressurization. The normally open (primary) solenoid valve is closed by momentarily setting the DESCENT He REG 1 switch (panel 1) to CLOSE; the DESCENT He REG 1 talkback above the switch then provides a barber-pole display. The normally closed (secondary) solenoid valve is opened by momentarily setting the DESCENT He REG 2 switch to OPEN; the DESCENT He REG 2 talkback above the switch then provides a gray display. (Both solenoid valves may be closed during the coast periods of descent, to prevent inadvertent tank overpressurization due to possible helium leakage through the pressure regulators.)

The primary and secondary helium flow paths merge downstream of the regulators to form a common helium pressurization manifold. Transducers monitor the manifold pressure; they provide continuous telemetry signals to MSFN, and signals that cause the DES REG warning light to go on when the sensed pressure exceeds 260 psia or drops below 222 psia. The manifold routes the helium into two flow paths: one path leads to the oxidizer tanks; the other, to the fuel tanks. Each path has a quadruple check valve assembly in a series-parallel arrangement. The quadruple check valves isolate the upstream components from the corrosive propellant vapors and prevent hypergolic action, as a result of backflow from the propellant tanks, in the helium pressurization manifold. After passing through the compatibility explosive valves, the helium flows into the top of the fuel and oxidizer tanks. Diffusers at the top of the tanks uniformly distribute the helium throughout the ullage space. Helium crossover lines maintain a balanced ullage pressure in the tanks containing the same propellants.

Immediately upstream and in parallel with the propellant tanks, each helium flow path contains a relief valve assembly to protect the propellant tanks against overpressurization. The assembly (a burst disk in series with a relief valve) vents pressure in excess of approximately 275 psia and re-seals the flow path after overpressurization is relieved (254 psia). A thrust neutralizer eliminates unidirectional thrust generated by the escaping gas. To prevent leakage through a single point relief valve during normal operation, the burst disk is located upstream of the relief valve. The burst disk ruptures at a pressure between 260 and 275 psi; the relief valve opens fully at 275 psi to pass the entire helium flow from a failed-open regulator preventing damage to the propellant tanks.

Two vent lines, in parallel with the relief valve assemblies, include an explosive valve in series with a normally closed solenoid valve for each propellant tank. The vent lines are intended for planned depressurization of the tanks after lunar landing, when temperature rise of the supercritical helium and heat soak-back from the engine (after shutoff) causes pressure buildup in the tanks. (The propellants need not be settled during the venting procedure.) The planned venting arrangement protects the astronauts against untimely venting of the tanks through the relief valve assemblies. The fumes are vented overboard, one propellant at a time, through the relief valve thrust neutralizers at the fuel and oxidizer pressurization line vents. (See figure 2.3-2.) Because the helium pressurization line is open, the supercritical helium in the cryogenic storage vessel will be vented together with the propellant tanks. The supercritical helium will vent rapidly until pressure drops to approximately 350 psia, then the pressure remaining in the cryogenic storage vessel will slowly bleed off. To open the vent lines, the MASTER ARM switch is set to ON and the DES VENT switch (panel 8) is set momentarily to FIRE, opening both explosive vent valves simultaneously. The MASTER ARM switch is then set to OFF. The oxidizer tank and pressurization line is vented by setting the OXID VENT switch (panel 8) to OPEN; the OXID VENT talkback will change to gray. Venting of the line is monitored by setting the PRPLNT TEMP/PRESS MON switch (panel 1) to DES 1 and the HELIUM MON selector switch (panel 1) to SUPCRIT PRESS. When the OXID PRESS indicator and the HELIUM indicator indicate less than 20 psia, the HELIUM MON selector switch is set to OFF, the OXID VENT switch is set to CLOSE, and the OXID VENT talkback will change to a barber-pole display. To vent the fuel tank and pressurization line, the FUEL VENT switch is set to OPEN, changing the FUEL VENT talkback to gray. When the FUEL PRESS indicator indicates less than 20 psia, the FUEL VENT switch is set to CLOSE, causing the FUEL VENT talkback to provide a barber-pole display.

2.3.3.2 Propellant Feed Section. (See figure 2.3-3.)

Each pair of propellant tanks (containing like propellants) is manifolded into a common delivery line. Each pair is also interconnected by a double crossfeed piping arrangement to maintain a positive pressure balance across the upper (helium) and lower (propellant) portions.

Helium pressure in the propellant tanks is monitored on the PRESS indicator (panel 1), propellant temperature in the tanks is monitored on the TEMP indicator. The PRPLNT TEMP/PRESS MON switch selects the set of fuel and oxidizer tanks (No. 1 or No. 2) for monitoring. Each propellant tank has its own temperature transducer to supply temperature signals to the indicator. One pressure transducer in the fuel pressurization line and one in the oxidizer pressurization line supply pressure signals to the indicator. Therefore, the pressure reading remains constant regardless of whether tanks No. 1 or 2 are monitored. Propellant quantity remaining in the tanks is monitored on the OXIDIZER and FUEL QUANTITY indicators (panel 1). The PRPLNT QTY MON switch selects the set of fuel and oxidizer tanks (No. 1 or No. 2) for monitoring.

Pressurized helium, acting on the surface of the propellant, forces the fuel and oxidizer into the delivery lines through a propellant retention device that maintains the propellant in the delivery lines during negative-g acceleration (up to acceleration in excess of -2g). The oxidizer is piped directly to the engine assembly; the fuel circulates through the fuel/helium heat exchanger before it is routed to the engine assembly. Each delivery line contains a trim orifice and a filter. The trim orifices provide engine interface pressure of approximately 222 psia at full throttle position for proper propellant use. The filters prevent debris, originating at the explosive valves or in the propellant tanks, from contaminating downstream components.

2.3.3.2.1 Propellant Quantity Gaging System. (See figure 2.3-4.)

The propellant quantity gaging system (PQGS) enables the astronauts to continuously monitor the quantity (from 95% full) of propellants remaining in the four tanks. The PQGS consists of four quantity-sensing probes with low-level sensors, a control unit, two QUANTITY indicators, the PRPLNT QTY MON switch, and the DES QTY warning light.

During the lunar-landing mission, the PQGS will operate continuously for 15 minutes; the system will be in operation during the final powered descent phase, from start of the braking maneuver (10 seconds after engine turn on) until Lunar touchdown. The low-level sensors provide a discrete signal to cause the DES QTY warning light to go on when the propellant level in any tank is down to 9.4 inches (equivalent to 5.6% propellant remaining). When the DES QTY warning light goes on, the quantity of propellant remaining is sufficient for only 2 minutes of engine burn at 24% thrust.

The quantity-sensing probes are concentric tubes; one probe is installed to run vertically through the center of each propellant tank. Each probe is powered by 20 volts d-c. The output signals vary from 5 volts d-c to zero in direct proportion to propellant quantity. The output signals are sent directly to the control unit. The measurement of oxidizer (a dielectric) is based on change in capacitance as a function of propellant quantity. The measurement of fuel (a conductor) is based on a change in conductance as a function of propellant quantity.

The control unit processes the probe output signals to provide the following:

- Digital signals to the QUANTITY indicators corresponding to the propellant quantity remaining in the tanks.
- Analog signals to the telemetry register corresponding to propellant quantities in each of the four tanks.
- A warning signal when the propellant in any of the four tanks is depleted to 9.6 inches (5.6% propellant remaining).

While the engine is shut off, the QUANTITY indicators display the propellant quantity that was in the tanks when the last engine burn was terminated.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEM DATA

2.3.3.3 Engine Assembly. (See figures 2.3-5 and 2.3-6.)

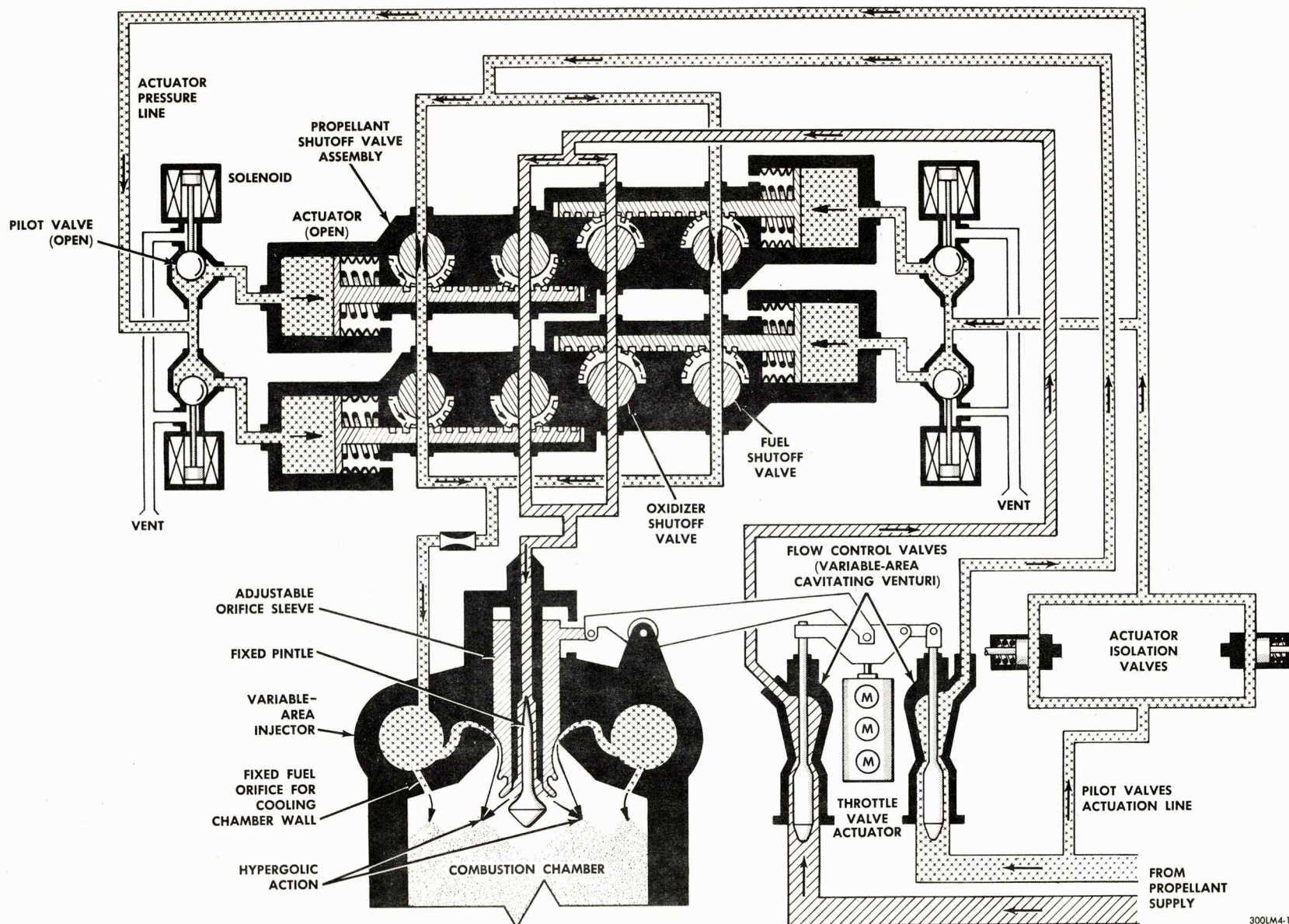
Fuel and oxidizer enter the engine assembly through interface flanges on opposite sides of the engine. The fuel line has a tap-off branch to pre valve assembly that leads through two actuator isolation valves (arranged in parallel for redundancy) to the four solenoid-operated pilot valves. The fuel in this line is routed, through the pilot valves, to the actuators, where it is used as actuation fluid to open the propellant shutoff valves. The main fuel and oxidizer flow is routed through respective flow control valves, then each flow path splits into two parallel paths that route the propellants through the redundant propellant shutoff valves. The propellant shutoff valve assemblies are in a series-parallel arrangement. The series redundancy prevents open failure; the parallel redundancy prevents closed failure. The valves open simultaneously to permit propellant flow to the engine while it is operating; they close simultaneously to terminate propellant flow at engine shutdown. At the two upstream fuel shutoff valves, venturis restrict the fuel flow so that the oxidizer reaches the injector between 40 and 50 milliseconds before the fuel. This precludes the possibility of a fuel lead, which would result in rough engine starts. Downstream of the propellant shutoff valves, the parallel paths merge to form one fuel and one oxidizer path. The fuel passes through a final trim orifice and enters the variable-area injector manifold, where a concentric annulus of fuel flow is formed. The oxidizer is routed, through the center element of the injector, to the combustion chamber, where it mixes with the fuel for combustion.

Before initial engine operation and during engine shutdown, the solenoid-operated actuator isolation valves (pre-valves) are closed to prevent possible fuel loss in the pilot valve actuation line due to leakage at the pilot valves. The actuator isolation valves are opened by setting the ENG ARM switch to DES. This enables the actuation fuel to flow to the pilot valves just before the pilot valves are opened. When the START pushbutton is pressed (or an engine-on command initiated), the four solenoid-operated pilot valves open simultaneously, permitting the actuation fuel to open the propellant shutoff valves, thus routing fuel and oxidizer to the combustion chamber. During the start, the solenoids in the pilot valves unseat the caged balls from the inlet ports and allow them to seat against the overboard vent ports; fuel enters the actuator cavities. The actuator pistons are connected to rack-and-pinion linkages that rotate the balls of the shutoff valves 90° to the open position to permit propellant flow to the injector. The series-parallel redundancy in the valve arrangement provides for positive start and shutdown. During shutdown, the solenoids in the pilot valves are deenergized returning the ball poppets to the inlet port seats, sealing the actuation fluid, and opening the actuator to the open vent ports. The spring-loaded actuators close the shutoff valves. Residual actuation fuel is vented overboard through four separate lines that lead to vent ports at the bottom of the descent stage. (See figure 2.3-2.)

The propellant in the main fuel and oxidizer lines flows through cavitating-venturi-type flow control valves that control propellant flow to the engine below the 60% throttle setting. Transition from cavitation to noncavitation occurs between 70% and 80%. A full throttle, and during momentary transition through the full throttle to 60% range, engine throttling takes place primarily in the pintle assembly of the injector and in the flow control valves. At approximately 70% of maximum thrust, cavitation commences in the throats of the flow control valves, causing the valves to function as cavitating venturis down to minimum thrust. Once cavitation begins, the propellant-metering function is entirely removed from the injector; flow is controlled entirely by the flow control valves.

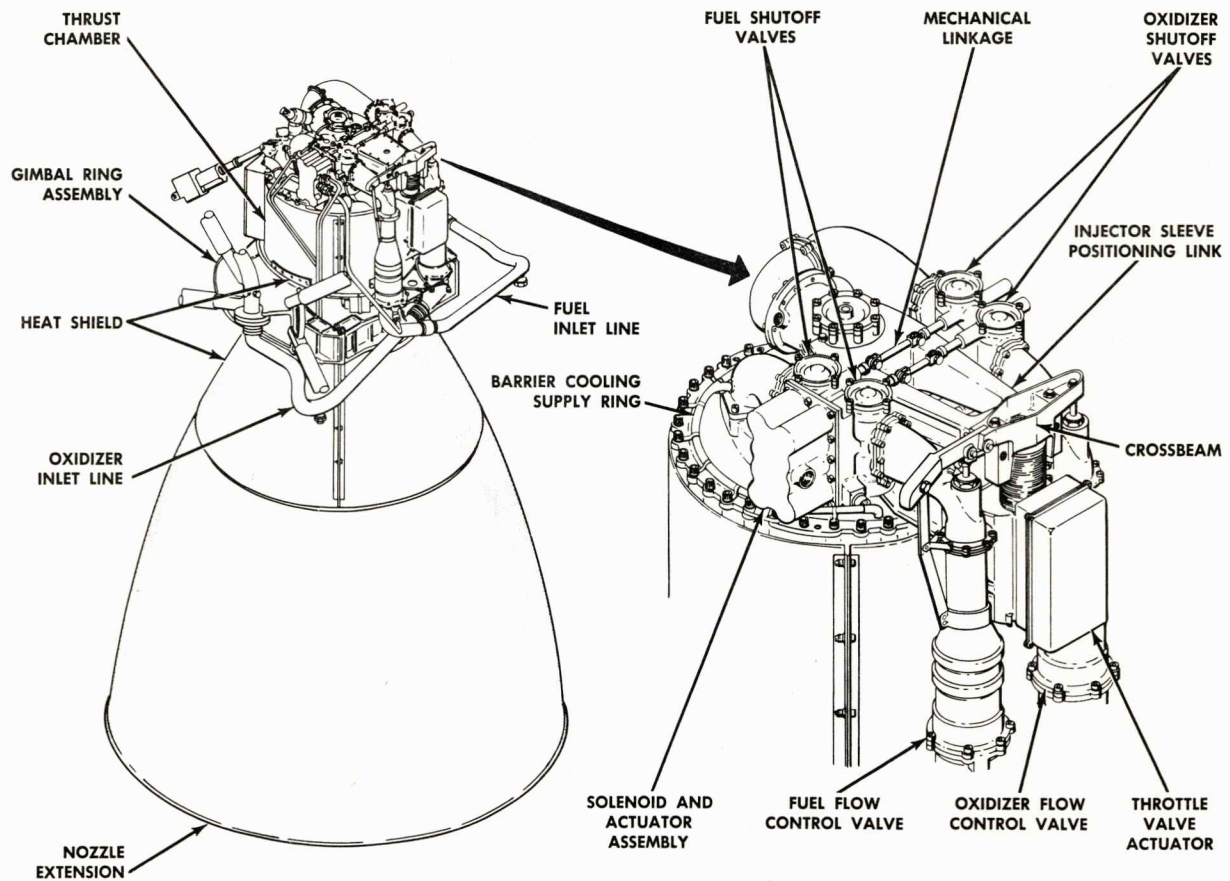
The throat area of the flow control valves is regulated by a close-tolerance, contoured, metering pintle that is linked directly to the injector sleeve. The linkage is operated by a single actuator so that movement of the actuator simultaneously adjusts the pintles in the flow control valves and the movable sleeve in the injector. The fuel and oxidizer are injected into the chamber at velocities and angles compatible with variations in weight flow. At full throttle, engine operation is conventional. As the engine is throttled down, the pintles in the flow control valves are moved to decrease the flow control area in the venturis so that the pressure drop across the valves balances out the differential between engine and injector inlet pressures. At the same time, the injector orifice areas are decreased so that the injection velocities and impingement angles of fuel and oxidizer are maintained at near-optimum condition for combustion efficiency.

The injector consists of a faceplate and fuel manifold assembly with a coaxial oxidizer feed tube and an adjustable (metering) orifice sleeve. Oxidizer flows through the center tube and out between a fixed pintle and the bottom of the sleeve; the fuel orifice is an annular opening between the sleeve contour and the injector faceplate. The fuel flows behind the face of the injector to cool the faceplate. Some fuel is tapped off near the chamber wall to be used for barrier cooling of the combustion chamber wall. The fuel is emitted from the injector in the form of a thin cylindrical sheet; the oxidizer, in a series of individual sprays. The oxidizer sprays break up the fuel stream and establish the injector pattern at all thrust settings.



300LM4-10

Figure 2.3-5. Descent Engine Assembly - Flow Diagram



MPS

300LM4-174

Figure 2.3-6. Descent Engine and Head End Assembly

The mechanical linkage that connects the pintle of the flow control valves and the injector sleeve is pivoted about a fulcrum attached to the injector body. Throttling is controlled by the throttle valve actuator which positions the linkage in response to electrical input signals. At maximum thrust, the actuator strokes the flow control valve pintles and the injector sleeve in the closing direction to reduce the metering areas thereby reducing the propellant flow.

2.3.3.3.1 Descent Engine Control.

After the engine is manually armed by setting the ENG ARM switch to DES, it can be fired automatically or manually. Under manual control, the engine can be started and stopped by the Commander by pressing the START and STOP pushbuttons on panel 5; it can be stopped by the LM Pilot by pressing the STOP pushbutton on panel 6. The mode of thrust control is determined by the THR CONT switch (panel 1). When this switch is set to AUTO, engine thrust is controlled by the LGC. When the switch is set to MAN, the Commander's or LM Pilot's TTCA (depending on the setting of the MAN THROT switch, panel 1) controls the engine thrust. In the automatic mode, the TTCA is still operational. It is normally set against its hard (low) stop where it supplies a 10% thrust command that is summed with the LGC command, resulting in a combined thrust command to the descent engine. For example, if the required thrust is 50%, the LGC commands 40%, which is augmented by the 10% obtained from the TTCA. If the TTCA is moved from the hard stop, it supplies a greater portion of the combined command and the LGC command decreases accordingly. Thus, for the required 50% thrust, the TTCA may now command

20%; the LGC, 30%. If the TTCA is moved to a setting such that it commands more than the required thrust, it overrides the automatic command (the LGC portion becomes zero) and descent engine thrust is determined entirely by the TTCA setting.

The dual-scale THRUST indicator (panel 1) displays commanded manual or automatic thrust on the CMD scale and actual engine thrust on the ENG scale. The ENG scale input is derived from a pressure transducer in the combustion chamber, because thrust is proportional to chamber pressure. At full throttle position (FTP), the ENG scale reads 92.5% (actual FTP thrust) while the CMD scale reads between 92.5% and 100%. At all other throttle settings (10% to 60% throttling range) the ENG and CMD scales normally display identical readings of the actual engine thrust. Display of dissimilar readings indicates that the engine is not following the thrust commands or that transfer from automatic to manual throttle control is in process. As shown in the example given previously, if the TTCA is displaced from the hard stop in the automatic mode, for 50% required thrust, the TTCA may command 20% while the LGC contributes 30%. The ENG scale of the THRUST indicator will read 50%; however, the CMD scale (where LGC command is summed with a 10% bias) will read 40%. As the TTCA is moved to increasing throttle settings, the CMD scale readings decrease. When the CMD scale reading has dropped to 10%, the LGC no longer supplies any portion of the thrust command and the TTCA is in control. At this point, a smooth transfer from the automatic mode to the manual mode is accomplished by setting the THR CONT switch to MAN. The CMD and ENG scales will now indicate identical readings. (For the above example, both pointers will align at 50%.) Very slight deviations between CMD and ENG scale readings may occur as engine operating time increases. The deviations are due to combustion chamber erosion, which causes chamber pressure to decay slightly.

The T/W (thrust/weight) indicator (panel 1) is used primarily to monitor X-axis acceleration during lunar landing and liftoff. The T/W indicator is a self-contained accelerometer that displays instantaneous X-axis acceleration in lunar-g units (1 lunar g = 5.23 ft/sec²). Inasmuch as a given throttle setting provides a specific acceleration when the vehicle has a given mass, the T/W indicator can be used as backup for the THRUST indicators to monitor engine performance. Information regarding use of the T/W indicator during the earth-orbital mission and the relation of the percentage of thrust indicated on the ENG scale of the THRUST indicator relates to the lunar-g reading on the T/W indicator at specific time intervals during the mission will be supplied when available.

2.3.4 DESCENT PROPULSION SECTION MAJOR COMPONENT/FUNCTIONAL DESCRIPTION.

2.3.4.1 Explosive Valves. (See figure 2.3-7.)

An ambient helium isolation valve, a supercritical helium isolation valve, a fuel compatibility valve, an oxidizer compatibility valve, a fuel vent valve, and an oxidizer vent valve are the explosive valves used in the DPS. These valves normally are closed; they are controlled by the EDS control and fire circuits and, when fired, fully open and remain open. To prevent valve failure in the closed position, each explosive valve has two cartridges that are fired from redundant systems in the EDS. A cartridge is fired by applying power to the initiator bridgewire for a few milliseconds. The resultant heat fires the initiator, generating gases in the valve explosion chamber at an extremely high rate. The gases drive the valve piston into the valve housing to open the valve by shearing a closure disk and aligning the piston port permanently with the pressure line plumbing.

2.3.4.2 Cryogenic Helium Storage Vessel.

The double-walled cryogenic helium storage vessel consists of an inner spherical tank and an outer jacket. The void between the tank and the jacket is filled with aluminized mylar insulation and evacuated to minimize ambient heat transfer into the tank. The vessel has fill and vent ports, a burst disk assembly, and an internal helium/helium heat exchanger. The inner tank is initially vented and loaded with liquid helium; the fill sequence is completed by closing the vent and introducing a high-pressure head of gaseous helium at a higher temperature than that of the liquid helium. As this occurs, the heat transfer causes the liquid helium to evaporate into a high-density gas.

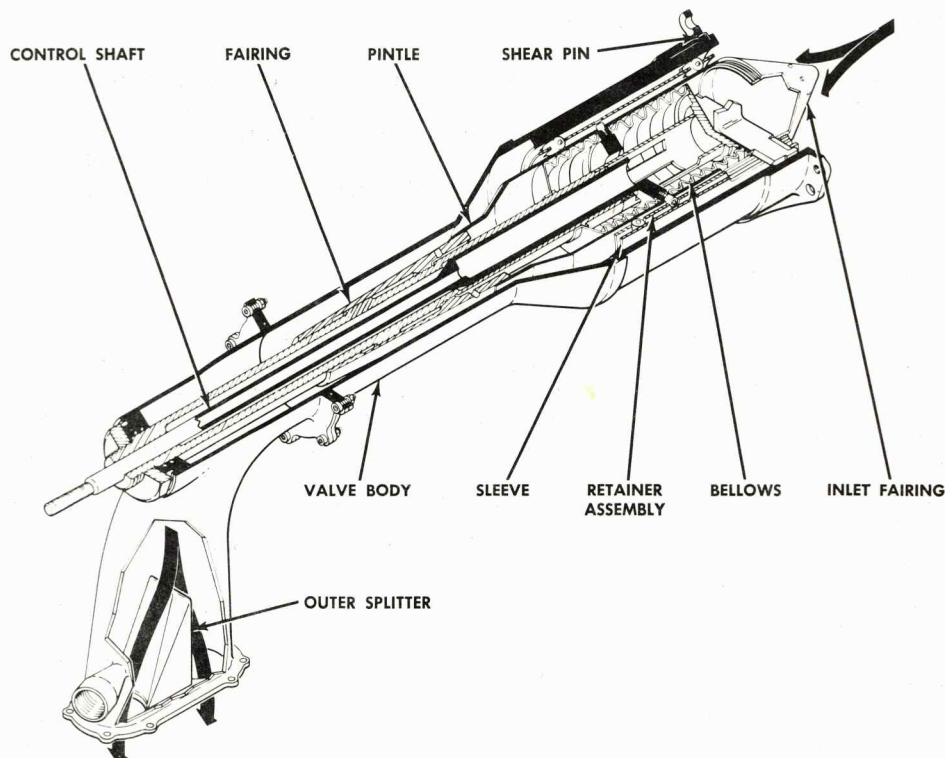
The burst disk assembly prevents hazardous overpressurization within the vessel. It consists of two burst disks in series, with a normally open, low-pressure vent valve between the disks. The burst disks are identical; they burst at a pressure between 1,881 and 1,967 psid to vent the entire supercritical helium supply overboard. A thrust neutralizer at the outlet of the downstream burst disk

throttle position by the hydraulic loads imposed by the propellant pressures on the flow control valves and the injector. (If there are no hydraulic loads, the throttle valve actuator moves to the position of balanced mechanical loads.) Loss of the -10-volt reference voltage causes the potentiometer feedback voltage to drop to zero. As a result, only the positive command voltage is in control, and the throttle valve actuator moves to the full throttle position. If the +10-volt reference voltage fails, the failure-detection circuit that monitors this voltage provides a signal that causes the throttle valve actuator to move to the full throttle position. The power source for the command voltage is the +15 volt power supply that supplies the input to the plus side of the throttle valve actuator reference. Therefore, loss of the command voltage also results in loss of the +10-volt d-c reference voltage, which causes the throttle valve actuator to move to the full throttle position. (Loss of the command voltage alone would cause the throttle valve actuator to move to the 10% thrust position.)

2.3.4.9 Flow Control Valves. (See figure 2.3-12.)

The oxidizer and fuel flow control valves are on the side of the engine, immediately downstream of the propellant inlet lines. They are secured to the throttle valve actuator mounting bracket, the shutoff valves, and the injector faceplate. The flow control valve pintle assemblies are mechanically linked to the throttle valve actuator by a crossbeam.

The flow control valves are nonredundant cavitating venturis with movable pintle sleeves. Engine throttling is initiated by an electrical signal to the throttle valve actuator, commanding an increase or decrease in engine thrust. Operation of the throttle valve actuator drives the crossbeam to a new position, thus changing the position of the pintles in the flow control valves. This axial movement of the pintles decreases or increases the pintle flow areas to control propellant flow rate and thrust. Below an approximate 70% thrust setting, flow through the valves cavitates and hydraulically uncouples the propellant transfer system (and thereby, the flow rate) from variations in combustion chamber pressure. The movable pintle, shaft-driven through the valve case, is sealed by two flexible bellows.



300LM4-177

Figure 2.3-12. Flow Control Valve

Shear pins in the bearing sleeves permit the valve bearings to operate in redundant modes (rolling or sliding) to improve reliability. Normally, the bearings roll; however, under an excessive load, the shear pins give to permit the sleeve bearings to slide. Both pintle sleeve shafts are attached to a cross beam that is driven by the throttle valve actuator.

2.3.4.10 Propellant Shutoff Valve Assemblies.

Each of the four propellant shutoff valve assemblies consists of a fuel shutoff valve, an oxidizer shutoff valve, a pilot valve, and a shutoff valve actuator. The shutoff valve actuator and the fuel shutoff valve are in a common housing. The oxidizer shutoff valve is in a separate housing to isolate fuel from oxidizer. It is actuated by a mechanical linkage driven from the fuel shutoff valve. The fuel and oxidizer shutoff valves, downstream of the flow control valves, are exposed to the full propellant pressure before engine firing. Actuation fuel is vented overboard.

2.3.4.10.1 Pilot Valves and Shutoff Valve Actuators.

The four solenoid-operated pilot valves control the actuation fluid (fuel) for the fuel and oxidizer shutoff valves. Normally, the pilot valves are closed, the solenoids are deenergized, and the actuation fluid is shut off by action of the spring-loaded solenoid plungers on the ball poppet seated in the lower pilot valve seats. The actuator chamber is vented to preclude a fuel buildup caused by leakage of the pilot valves, or overpressurization caused by an excessive fuel temperature increase. When the pilot valves are open, all solenoids are energized, and the actuation fluid (at approximately 235 psia) acts against the spring-loaded actuator piston to open the oxidizer and fuel shutoff valves. At this point, the pilot valve poppets are forced against the upper valve seats, sealing off the vent port. When the electrical signal is removed, the valve poppets return to their lower seat to seal off the actuation fluid. The propellant shutoff valves are closed by the return action of the actuator piston springs, which expels the fuel entrapped in the cylinders and valve passages through the pilot vent port.

2.3.4.10.2 Fuel and Oxidizer Shutoff Valves.

The fuel and oxidizer shutoff valves are mechanically linked, fuel-actuated ball valves that are arranged in a series-parallel configuration. The valve housings are made of aluminum alloy. The ball element operates against a spring-loaded teflon seat to assure positive sealing when the valve is closed. All ball valves are supported by ball bearings. The individual valves are rotated by a rack-and-pinion-gear arrangement, which translates the linear displacement of the pistons in the shutoff valve actuators.

2.3.4.11 Variable-Area Injector. (See figure 2.3-13.)

The variable-area injector consists of a pintle assembly, drive assembly, and fuel manifold assembly. The pintle assembly introduces the propellant uniformly into the combustion chamber. The drive assembly has a twofold function: first, it serves as a passage for conducting the oxidizer into the pintle assembly; second, it contains the bearing and sealing components that permit accurate positioning of the injector sleeve. The injector sleeve varies the injection area so that near-optimum injector pressure drops and propellant velocities are maintained at each thrust level. The fuel manifold assembly consists of a toroidal barrier-cooling manifold and a main manifold. Fuel that enters the barrier-cooling manifold at two inlets is injected onto the chamber wall through 36 orifices. Fuel that enters the main manifold at its single inlet passes through a series of distribution plates near the outer periphery of the assembly. From the plates, the fuel is admitted into a narrow passage formed by two nearly parallel conical surfaces (the manifold body and the injector faceplate). The passage smoothes fuel discontinuities and cools the faceplate. Near the center of the manifold, the fuel passes through a hole ring that structurally connects the faceplate to the manifold. The fuel then passes onto the outer surface of the sleeve, past a fuel-metering lip. The fuel is injected as a hollow cylinder so that it reaches the impingement zone with a uniform circumferential velocity profile and without atomizing, at all flow rates. The oxidizer, injected through a multislot sleeve, forms a large number of radial filaments. Each filament partially penetrates the fuel cylinder and is enfolded by fuel in such a way that, following fast liquid-phase reactions and gas evolution, little preferential separation of oxidizer and fuel can occur. This provides high combustion efficiency.

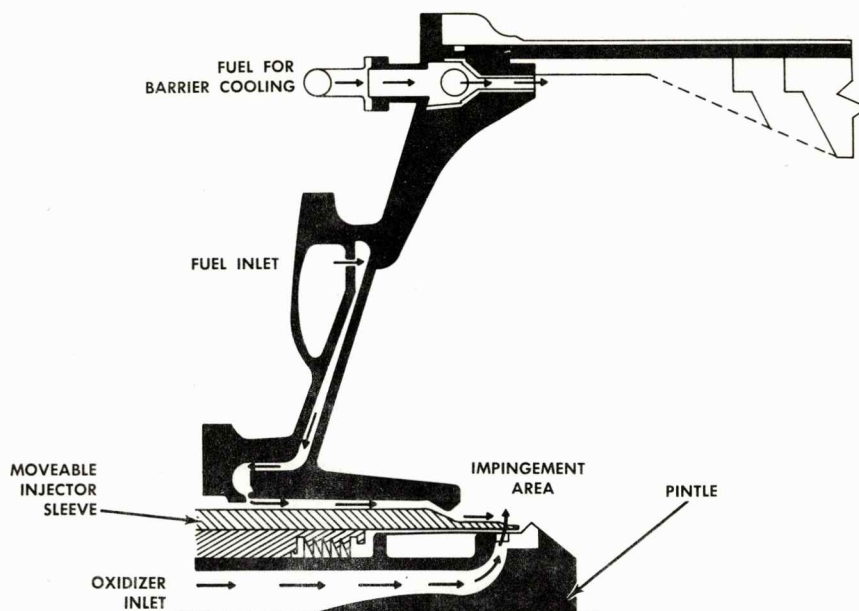


Figure 2.3-13. Variable-Area Injector - Quarter Section

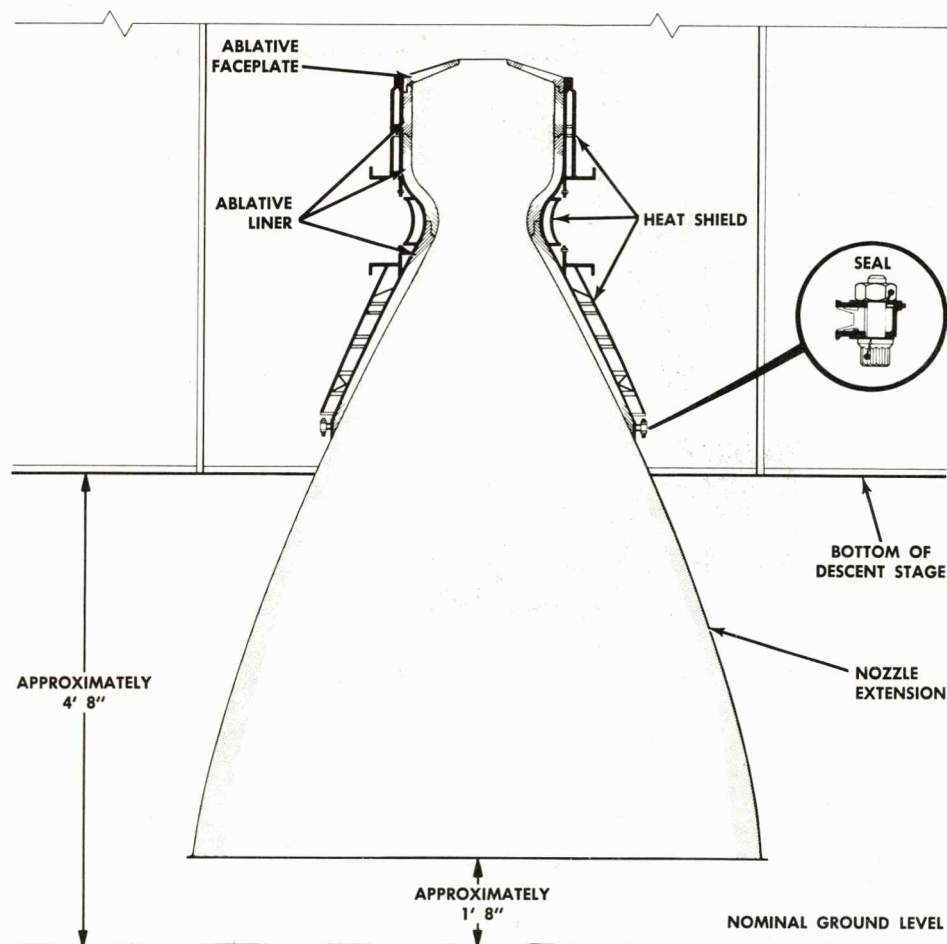
2.3.4.12 Combustion Chamber and Nozzle Extension. (See figure 2.3-14.)

The combustion chamber consists of an ablative-cooled chamber section, nozzle throat, and nozzle divergent section. The ablative sections are enclosed in a continuous titanium shell and jacketed in a thermal blanket composed of stainless-steel foil and glass wool. A seal prevents leakage between the combustion chamber and nozzle extension flanges.

The nozzle extension is a radiation-cooled, crushable skirt; it can collapse a distance of 28 inches on lunar impact so as not to affect the stability of the LM. The nozzle extension is made of columbium coated with aluminide. It is attached to the combustion chamber case at a nozzle area ratio of 16 to 1 and extends to an exit area ratio of 47.4 to 1.

2.3.4.13 Gimbal Ring and Gimbal Drive Actuators.

The gimbal ring is located at the plane of the combustion chamber throat. It consists of a rectangular beam frame and four trunnion subassemblies. The assembly permits up to +6° or -6° roll and pitch trim control of the engine. A description of the two gimbal drive actuators is given in paragraph 2.2.4.6.



300LM4-179

Figure 2.3-14. Descent Engine Combustion Chamber and Nozzle Extension

2.3.5 DESCENT PROPULSION SECTION FAILURE MODES.

The DPS failure modes and the effect on mission capability are given in table 2.3-1.

Table 2.3-1. Descent Propulsion Section - Failure Modes

Failure	Indication	Vehicle Capability
Primary helium isolation solenoid valve failed closed.	DESCENT He REG 1 talkback provides barber-pole display.	Secondary helium isolation solenoid valve must be opened.
Secondary helium isolation solenoid valve failed open.	DESCENT He REG 2 talkback provides gray display.	Failure does not affect vehicle capability.
Primary pressure regulator failed open.	DES REG warning light goes on.	Vehicle capability is not affected, except for loss of pressure regulator redundancy. Primary helium isolation solenoid valve must be closed immediately (to minimize helium loss through relief valve assemblies) and secondary helium isolation valve must be opened.

MAIN PROPULSION SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.3-2. Descent Propulsion Section - Performance and Design Data (cont)

MPS

Pressurization section (cont)

Quadruple check valve assemblies

Opening pressure	2 to 2.88 psid
Maximum pressure drop	4.0 psi at full throttle

Relief valve assembly

Burst-disk rupture pressure	260 to 275 psi
Relief valve cracking pressure	260 psi
Full-flow pressure	275 psi
Fully open flow rate	10 pounds of helium per minute
Reseat pressure (minimum)	254 psi

Propellant feed section

Propellant tanks

Capacity (each tank)	62.8 cubic feet
Total fuel	6,982 pounds
Total oxidizer	11,067 pounds
Ullage volume (minimum)	1,728 cubic inches
Usable fuel	6,759 pounds
Usable oxidizer	10,730 pounds
Unusable fuel	222.8 pounds
Unusable oxidizer	336.4 pounds
Nominal tank ullage pressure (at full throttle position)	235 psia
Proof pressure	360 psia
Burst pressure	405 psia
Nominal propellant temperature	+70° F
Propellant temperature range	+50° to +90° F

Engine interface inlet pressure (trim orifice output at full throttle position)	222±2 psia
---	------------

Propellant filters

Nominal filtration	40 microns
Absolute filtration	60 microns
Fuel filter rated flow	13.3 pounds per second
Fuel filter maximum pressure drop	2 psi
Oxidizer filter rated flow	21.2 pounds per second
Oxidizer filter maximum pressure drop	3 psi

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.3-2. Descent Propulsion Section - Performance and Design Data (cont)

Engine Assembly	
Nominal engine thrust (full throttle)	9,870 pounds (94%)
Minimum engine thrust	1,050 pounds (10%)
Nominal combustion chamber pressure	103.4 psia
Engine-gimbaling capability	+6° or -6° for trim control along Y-axis and Z-axis
Propellant injection ratio (oxidizer to fuel)	1.6 to 1
Approximate weight	360 pounds
Overall length	85 inches
Nozzle exit diameter	59 inches

2.3.7 DESCENT PROPULSION SECTION OPERATIONAL LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions for the DPS are as follows:

- The landing gear must be deployed before descent engine firing. If not deployed, the landing gear would be in the path of the descent engine plume and would be damaged.
- Before descent engine starts, the RCS +X-axis thrusters must be fired to settle the propellants under weightless conditions. Unsettled or insufficiently settled propellants will result in rough or erratic starts that could lead to engine failure.
- Normal start profile for all descent engine starts must be at 10% throttle setting for 26 seconds. Because the thrust vector at engine start may not be directed through the LM center of gravity, a low-thrust start (10%) will permit corrective gimbaling. If the engine is started at high thrust, RCS propellants must be used to stabilize the vehicle. In the worst case, the momentary unbalance may exceed the RCS compensation capability.
- Propellant bulk temperatures before descent engine start must be between +50° and +90° F. If temperature limits are exceeded, the resultant rough combustion may adversely affect engine-component performance, and ullage pressure limitations may be exceeded.
- Propellant tank pressure during earth launch must be less than 230 psia. If this limit is exceeded, possible structural failure in the propellant tanks will cause loss of the descent engine.
- The first descent engine firing must last at least 30 seconds. This minimum initial operating time is required to ensure full pressurization of the ullage space in the propellant tanks. If this requirement is not met, fuel may freeze in the fuel/helium heat exchanger.
- The descent engine must not be operated in the throttling range of 60% to 92.5%. In this range, operation of the cavitating venturis of the flow control valves may cause excessive variations in fuel-oxidizer mixture ratio resulting in improper propellant utilization.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

MPS

- The descent engine must be shut down before propellants are depleted to within 10 seconds of burn time remaining, until capability to burn safely to depletion is established. If this requirement is not met, fuel or oxidizer may become depleted, causing possible ingestion of helium into the combustion chamber.
- The PQGS must be used only during the descent engine firing phase. It must be turned on approximately 10 seconds before the start of engine firing and turned off 10 seconds after engine shutdown. Failure to meet this requirement may lead to loss of the PQGS, due to overheating of the PQGS electronics after 15 minutes of continuous operation.
- The DPS must not remain pressurized longer than 3.5 days before the anticipated termination of use. If this limit is exceeded, the pressurization section components may fail to operate because of the corrosive nature of the propellants.
- The descent engine combustion chamber must not be subjected to more than 910 seconds of engine operation or 17,510 pounds of propellant burn (tanks depleted), whichever comes first. Exceeding this limitation will cause the engine to operate with a severely charred combustion chamber, possibly resulting in burn-through.
- The descent engine must not be started more than 20 times.

2.3.8 DESCENT PROPULSION SECTION TELEMETRY MEASUREMENTS.

The DPS telemetry data are given in table 2.3-3. "(R)" in the "Telemetry Range" column signifies a real-time telemetry measurement. Real-time telemetry points are shown in figure 2.3-3.

Table 2.3-3. Descent Propulsion Section - Telemetry Measurements

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GQ3015P	Ambient helium prepressurization bottle pressure	0 to 1,750 psia	0 to 1,750 psia	HELIUM indicator
GQ3018P	Helium regulator outlet manifold pressure	0 to 300 psia (235 psia) (R)	< 219.2 psia; > 259.1 psia	DES REG warning light
GQ3025P	Helium regulator outlet manifold pressure	0 to 300 psia (235 psia) (R)		None
GQ3309X	Helium primary solenoid valve closed	Contact closure (R)		DESCENT He REG 1 talkback
GQ3310X	Helium secondary solenoid valve open	Contact closure (R)		DESCENT He REG 2 talkback
GQ3435P	Supercritical helium supply tank pressure	0 to 2,000 psia (1,250 psia) (R)		None
GQ3436P	Supercritical helium supply tank pressure	0 to 1,750 psia (1,250 psia)	0 to 1,750 psia	HELIUM indicator
GQ3500X	Fuel vent solenoid valve open	Contact closure		FUEL VENT talkback
GQ3501P	Fuel tank No. 1 ullage pressure	0 to 300 psia	0 to 300 psia	FUEL PRESS indicator
GQ3603Q	Fuel tank No. 1 quantity	0% to 95% (R)		None
GQ3604Q	Fuel tank No. 2 quantity	0% to 95% (R)		None

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.3-3. Descent Propulsion Section - Telemetry Measurements (cont)

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GQ3605Q	Fuel tank No. 1 or 2 (as selected) quantity	0% to 99%	0% to 99%	FUEL QUANTITY indicator
GQ3611P	Engine interface fuel pressure	0 to 300 psia (222 psia) (R)		None
GQ3718T	Fuel tank No. 1 fuel bulk temperature	+20° to +120° F (+70° F) (R)	+40° to +100° F	FUEL TEMP indicator
GQ3719T	Fuel tank No. 2 fuel bulk temperature	+20° to +120° F (+70° F) (R)	+40° to +100° F	FUEL TEMP indicator
GQ4000X	Oxidizer vent solenoid valve open	Contact closure		OXID VENT talkback
GQ4001P	Oxidizer tank No. 1 ullage pressure	0 to 300 psia (235 psia)	0 to 300 psia	OXID PRESS indicator
GQ4103Q	Oxidizer tank No. 1 quantity	0% to 95% (R)		None
GQ4104Q	Oxidizer tank No. 2 quantity	0% to 95% (R)		None
GQ4105Q	Oxidizer tank No. 1 or 2 (as selected) quantity	0% to 99%	0% to 99%	OXIDIZER QUANTITY indicator
GQ4111P	Engine interface oxidizer pressure	0 to 300 psia (222 psia) (R)		None
GQ4218T	Oxidizer tank No. 1 oxidizer bulk temperature	+40° to +100° F (+70° F) (R)	+40° to +100° F	OXID TEMP indicator
GQ4219T	Oxidizer tank No. 2 oxidizer bulk temperature	+20° to +120° F (+70° F) (R)	+40° to +100° F	OXID TEMP indicator
GQ4455X	Propellant tanks liquid level low	Contact closure (R)	120 seconds burn time at 25% thrust	DES QTY warning light
GQ6297V	Actuator isolation valve A current shunt voltage	14 to 21 mv		None
GQ6299V	Actuator isolation valve B current shunt voltage	14 to 21 mv		None
GQ6331V	Shutoff valve A positive current shunt voltage	8 to 13 mv		None
GQ6332V	Shutoff valve A return current shunt voltage	8 to 13 mv		None
GQ6333V	Shutoff valve B positive current shunt voltage	8 to 13 mv		None
GQ6334V	Shutoff valve B return current shunt voltage	8 to 13 mv		None
GQ6335V	Shutoff valve C positive current shunt voltage	8 to 13 mv		None
GQ6336V	Shutoff valve C return current shunt voltage	8 to 13 mv		None

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.3-3. Descent Propulsion Section - Telemetry Measurements (cont)

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GQ6336V	Shutoff valve D positive current shunt voltage	8 to 13 mv		None
GQ6338V	Shutoff valve D return current shunt voltage	8 to 13 mv		None
GQ6441V	Channel No. 1 motor positive current shunt voltage	< 5 to 70 mv		None
GQ6442V	Channel No. 1 motor return current shunt voltage	< 5 to 70 mv		None
GQ6443V	Channel No. 2 motor positive current shunt voltage	< 5 to 70 mv		None
GQ6444V	Channel No. 2 motor return current shunt voltage	< 5 to 70 mv		None
GQ6445V	Channel No. 3 motor positive current shunt voltage	< 5 to 70 mv		None
GQ6446V	Channel No. 3 motor return current shunt voltage	< 5 to 70 mv		None
GQ6510P	Thrust chamber pressure	0 to 200 psia (104 psia) (R)	0% to 100%	ENG THRUST indicator
GQ6806H	Throttle valve actuator position	0% to 100% (R)		None
GQ7498U	Shutoff valves A and B position. (Composite of GQ6306X - Shutoff valves A not open, GQ6307X - Shutoff valves A not closed, GQ6308X - Shutoff valves B not open, GQ6309X - Shutoff valves B not closed.)	Contact closure (R)		None
GQ7499U	Shutoff valves C and D position. (Composite of GQ6314X - Shutoff valves C not open, GQ6315X - Shutoff valves C not closed, GQ6316X - Shutoff valves D not open, GQ6317X - Shutoff valves D not closed.)	Contact closure (R)		None

MPS

2.3.9 ASCENT PROPULSION SECTION INTERFACES. (See figure 2.3-15.)

The APS receives 28-volt d-c and 115-volt a-c primary power through the Commander's and LM Pilot's buses of the EPS. The outputs of the APS pressure and temperature transducers, low-level sensors, and valve position indicator switches are processed in the IS and are transmitted via the CS to MSFN. The IS also processes the APS caution and warning signals. The EDS opens explosive valves in the APS to enable propellant tank pressurization. Interconnect plumbing between the APS propellant tanks and the RCS thrust chamber assembly feed lines permits the RCS to use APS propellants during certain mission phases, thereby conserving the RCS propellant supply.

The GN&CS issues automatic on and off commands to the ascent engine. Ascent engine arming, ignition, and shutdown can be initiated by automatic guidance equipment (PGNS or AGS) or by the astronauts. The automatic and manual commands are sent to the S&C control assemblies, which provide sequential control of LM staging and engine on and off commands. The ascent engine is armed manually by setting the ENG ARM switch to ASC or pressing the ABORT STAGE pushbutton. (Either action automatically shuts off the descent engine if it is firing.) After setting the ENG ARM switch to ASC (and after initial propellant tank pressurization), the ascent engine can be started manually by pressing the START pushbutton. Pressing either STOP pushbutton shuts down the engine. In the event of an abort-stage command while the descent engine is firing, the S&C control assemblies provide a time delay before commanding LM staging and ascent engine firing. This delay ensures that the descent engine has stopped thrusting before staging occurs. The manual commands override the commands issued by the automatic guidance equipment.

The APS modes of operation are discussed in detail in paragraph 2.2.3.7. The control circuitry is shown in figure 2.2-34.

2.3.10 ASCENT PROPULSION SECTION FUNCTIONAL DESCRIPTION.

The APS consists of a constant-thrust rocket engine that is not gimbaled, two propellant tanks, two helium tanks, and associated propellant feed and helium pressurization components. (See figure 2.3-16.) The engine is installed in the center section of the ascent stage; it is canted so that the center line is tilted 1.5° from the X-axis, in the +Z-direction. The engine develops 3,500 pounds of thrust in a vacuum, sufficient to launch the ascent stage from the lunar surface and place it into a pre-determined lunar orbit. The engine can be shut down and restarted, within the operational limitations and restrictions (paragraph 2.3.14), as required by the mission.

Functionally, the APS can be subdivided into a pressurization section, a propellant feed section, and an engine assembly.

2.3.10.1 Pressurization Section. (See figure 2.3-17.)

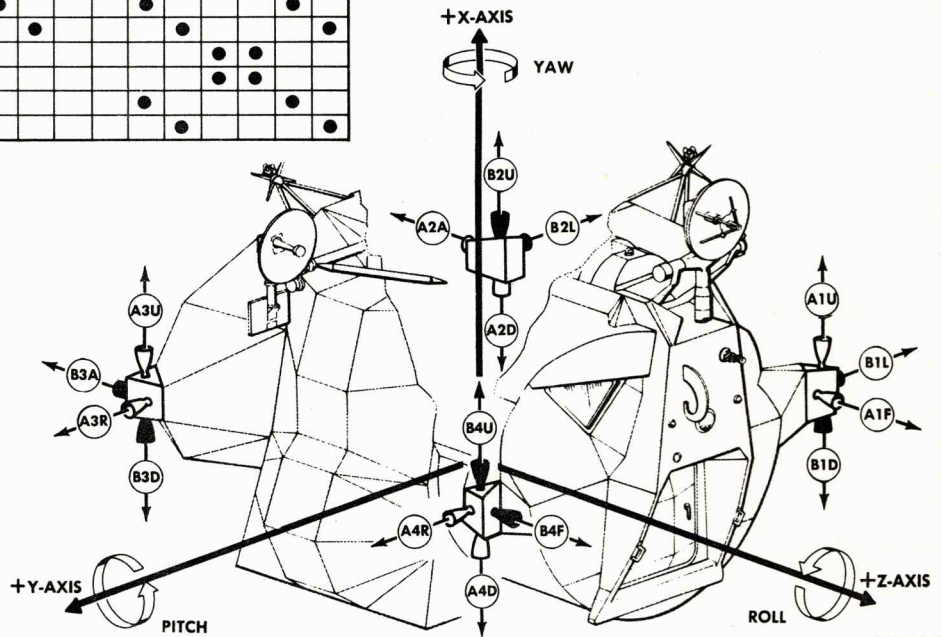
The propellants are pressurized by gaseous helium, supplied from two identical tanks and routed through redundant flow lines into the propellant tanks. The helium is stored at a nominal pressure of 3,050 psia at a nominal temperature of +70° F. A pressure transducer and a temperature transducer at each tank outlet port are connected to the HELIUM indicator via the HELIUM MON selector switch. The pressure transducers also supply a signal to the IS that will cause the ASC PRESS warning light to go on when the pressure in either helium tank is less than 2,773 psia. This alerts the astronauts to identify and isolate the faulty tank. The HELIUM MON selector switch selects the tank temperature or pressure to be displayed. When a caution or warning light goes on, a signal is routed from the caution and warning electronics assembly in the IS to light the MASTER ALARM pushbutton/lights (panels 1 and 2) and to provide a 3-kc tone in the astronaut headsets. Pressing either MASTER ALARM pushbutton turns off both lights and terminates the tone, but has no effect on the caution or warning light. The ASC PRESS warning light goes off upon separation of descent and ascent stage. After staging, the input to this warning light originates at the fuel and oxidizer engine inlet pressure transducers. (Refer to paragraph 2.3.10.3.)

Before initial ascent engine operation, the helium isolation explosive valves prevent the helium from leaving the tanks. These valves can be opened individually or simultaneously. The propellants are not pressurized until shortly before initial ascent engine start. To accomplish initial pressurization, the helium isolation explosive valves and the fuel and oxidizer compatibility explosive valves

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

RCS

THRUSTER FAILED ON	ON COMMAND SIGNALS															
	A1U	B1D	A1F	B1L	B2U	A2D	A2A	B2L	A3U	B3D	B3A	A3R	B4U	A4D	B4F	A4R
A1U	●	●							●							
B1D	●	●								●						
A1F			●				●								●	
B1L				●				●								●
B2U					●	●							●			
A2D					●	●								●		
A2A			●				●				●					
B2L				●				●				●				
A3U	●								●	●						
B3D		●							●	●						
B3A							●				●				●	
A3R								●				●				●
B4U					●								●	●		
A4D						●							●	●		
B4F			●								●				●	
A4R				●								●				●



300LM4-147

Figure 2.4-10. Reaction Control Subsystem - Thruster-On Failure Correction

Thruster-Off Failures. A thruster-off failure is defined as a nonthrusting condition with an on command from its jet driver. With an on command to TCA A4D (figure 2.4-7), and the absence of a response from the pressure switch in the TCA, an off-failure signal is initiated at the AND gate. The TCA pressure switch closes by the time a maximum pressure of 10.5 psia is sensed in the combustion chamber; it opens by the time pressure has decayed to a minimum of 3 psia. Under normal operating conditions, the TCA's may not respond to all individual on commands, if these commands (pulses) are of short duration (less than 80 milliseconds). Pulse skipping is not critical until a number of consecutive short pulses are skipped. When six short pulses have not produced a thruster-on response, the seventh consecutive pulse, regardless of duration or whether it produces TCA firing or not, will cause the counter to generate a failure output. The counter output sets the flip-flop via the OR gate, causing the RCS TCA warning light and the respective QUAD switch to provide failure indication.

2.4.3.4.2 Heaters.

Thirty-two heaters are used to heat the 16 TCA's; two of the heaters are attached to each TCA flange. The heaters are fed by a 28-volt d-c input; each heater consumes 17.5 watts at 24 volts. One heater of each TCA receives power from the SYS A/B-1 circuit breaker of its respective cluster; the other heater, from the SYS A/B-2 circuit breaker. In effect, two redundant, independently

REACTION CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

operating heating systems are used to heat each TCA cluster. The heaters normally operate in an automatic mode, in which redundant thermal switches (two connected in parallel for each heater) sense the TCA injector head temperature and turn the heaters on and off, as required, to maintain TCA temperature within the required range. The primary thermal switches are set to maintain TCA temperature at $+140^{\circ}\pm 8^{\circ}$ F; the redundant thermal switches are set for $+147^{\circ}\pm 7^{\circ}$ F. A 2° deadband permits the temperature to drop 2° below the turn-on temperature.

The heaters of system A/B-1 are connected directly to their circuit breakers and cycle on and off, as required, unless the appropriate circuit breaker (HEATERS RCS SYS A/B-1: QUAD circuit breakers on panel 11) is opened. The heaters of system A/B-2 are connected to their circuit breakers (HEATERS: RCS SYS A/B-2 QUAD circuit breakers on panel 16) through four three-position switches (one for each cluster). Normally, these switches (RCS SYS A/B-2 QUAD 1 through QUAD 4 on panel 3) are set to AUTO to cycle the heaters on and off to maintain the desired temperature. With the switch set to OFF, the circuit is interrupted and the heaters in the particular cluster are off. With the switch set to MAN, the thermal switches are bypassed and the heaters remain on regardless of the temperature.

If an out-of-limit temperature (less than $+119^{\circ}$ F or more than $+190^{\circ}$ F) occurs, the HEATER caution light (panel 2) goes on and the astronaut can determine the off-temperature cluster by using the TEMP MONITOR selector switch and TEMP indicator (panel 3). The inputs to the TEMP indicator originate at four temperature transducers, one for each quad. The heaters of the off-temperature cluster can be disconnected from the electrical power supply by opening the appropriate heater QUAD circuit breaker.

2.4.4 MAJOR COMPONENT/FUNCTIONAL DESCRIPTION.

2.4.4.1 Explosive Valves.

The explosive valves are single-cartridge-actuated, normally closed valves. Setting the MASTER ARM switch (panel 8) to ON and momentarily holding the RCS He PRESS switch to FIRE completes the circuit between the explosive-valve cartridge and the EDS buses. A cartridge is fired by applying power to the initiator bridgewire for a few milliseconds. The resultant heat fires the initiator, generating gases in the valve explosion chamber at an extremely high rate. The gases drive the valve piston into the valve housing to open the valve by shearing a closure disk and aligning the piston port permanently with the pressure line plumbing.

2.4.4.2 Helium Pressure Regulator Assembly. (See figure 2.4-11.)

The helium pressure regulator assembly consists of two individual pressure regulator units connected in series. Both regulator units function in the same manner, however, the downstream regulator is set to produce a higher outlet pressure so that it becomes a secondary unit that will only be in control if the upstream regulator (primary unit) fails open.

Each pressure regulator unit consists of a direct-sensing main stage and a pilot stage. The valve in the main stage is controlled by the valve in the pilot stage, which senses small changes in the outlet pressure (P_r) and converts these changes to proportionally larger changes in control pressure (P_c). The main stage valve poppet is positioned for varying flow demands by changes in control pressure acting on the main stage piston.

During normal RCS operation, the helium flow rate through the regulator assembly is relatively low. Consequently, the main stage valve poppet remains closed. The pilot stage valve poppet is held open by the pilot stage spring, permitting helium to flow into the regulator outlet line through a fixed orifice in the main stage piston. The outlet pressure, sensed by the pilot stage bellows, acts against the force of the pilot stage spring to maintain the regulator output at a constant preset pressure. The flow limiter at the regulator assembly outlet port restricts maximum flow through the regulator assembly so that the propellant tanks are protected if both regulator units fail open. The filter at the regulator assembly inlet prevents particles, which could cause excessive leakage at lockup, from reaching the valve seats.

2.5 ELECTRICAL POWER SUBSYSTEM.

2.5.1 INTRODUCTION

The Electrical Power Subsystem (EPS), the principal source of electrical power for the LM mission, has a d-c section and an a-c section. Both sections supply operating power to respective electrical buses, which supply all LM subsystems through circuit breakers. The descent stage batteries (No. 1 through 4) provide power to the LM until the docked phases of the mission, at which time the batteries are disconnected and the LM receives electrical power from the CSM. During descent propulsion section (DPS) engine burns, the ascent stage batteries (No. 5 and 6) are paralleled with the descent stage batteries to ensure at least the minimum voltage output (26.5 volts dc) at the subsystem's buses in the event of a descent battery failure. Ascent stage battery power is introduced before staging, at which time the descent battery power is terminated and descent feeder lines are severed. Ascent stage battery power is then used until final docking and astronaut transfer to the CSM are completed.

EPS

The EPS batteries are controlled and protected by electrical control assemblies, a relay junction box, and a deadface relay box, in conjunction with the control and display panel. Primary a-c power is provided by two inverters, which supply 115 ± 2.5 -volts, 400 ± 0.4 -cps (syncd condition), a-c power to LM subsystems. The operating frequency of the inverters is 400 ± 10 cps, in the nonsyncd, free-running condition.

Electrical power is distributed to LM subsystems via the LM Pilot's d-c bus, the Commander's d-c bus, and a-c buses A and B. A complete description of the d-c and a-c circuit breakers is in section 3. Secondary d-c power distribution is provided by two sensor power fuse assemblies, which supply d-c power to sensors of the Environmental Control Subsystem (ECS), Reaction Control Subsystem (RCS), and Main Propulsion Subsystem (MPS).

Batteries in the Explosive Device Subsystem (EDS) provide primary power to trigger LM explosive devices. (Refer to subsection 2.8.) These batteries are in addition to those in the EPS.

2.5.2 SUBSYSTEM INTERFACES.

2.5.2.1 External Power and Signal Interface. (See figure 2.5-1.)

The EPS provides the primary source of electrical power for the LM during the mission and is the distribution point for externally generated power during prelaunch and docked operations. The EPS interfaces with all functional subsystems of the LM. The Commander's and LM Pilot's buses supply 28 volts dc; a-c buses A and B supply 115-volt, 400-cps ac. Prelaunch d-c and a-c power is initially supplied to the EPS from external ground power supplies: d-c power to the Commander's and LM Pilot's buses and a-c power to a-c bus A. Approximately 7 hours before launch, ground-supplied power is discontinued and d-c power is connected from the launch umbilical tower (LUT) to the LM Pilot's bus via the relay junction box (RJB). The EPS distributes internally generated d-c power from launch until LM-CSM docking, at which time LM power is shut down and the CSM supplies 28 volts dc to the Commander's bus. Ground return is effected via the floating translunar bus. (Ground return in the LM is generally accomplished through the metal structure, whereas ground return in the CSM is isolated from the CSM structure.) Before LM-CSM separation, all LM internally supplied electrical power (ac and dc) is restored.

Circuits in electrical control assemblies (ECA's) of the EPS monitor and control LM electrical power. The circuits route discrete output signals if operating limits are exceeded. The output signals are fed to the Instrumentation Subsystem (IS) for conditioning before subsequent panel display and telemetering. The EPS monitoring signals are representative of malfunctioning batteries, battery terminal voltages, and feed line/battery status. The conditioned analogs of these signals are then routed from electronic replaceable assemblies (ERA's) No. 1 and No. 2 in the IS to EPS panel 14 for related meter, talkback, or component caution light display. In addition, primary-bus-monitoring signals are connected from the LM Pilot's and Commander's 28-volt buses and a-c bus A to the IS for conditioning and subsequent voltage display on panel 14. Synchronization pulses (6.4 kpps) are supplied from the pulse-code-modulation and timing electronics assembly (PCMTEA) of the IS to inverters to accurately control a-c output frequency. An inverter inhibit signal path is completed in the caution and warning electronics assembly (CWEA) of the IS when the INVERTER switch (panel 14) is set to OFF.

ELECTRICAL POWER SUBSYSTEM

LMA790-3-LM 4
 APOLLO OPERATIONS HANDBOOK
 SUBSYSTEMS DATA

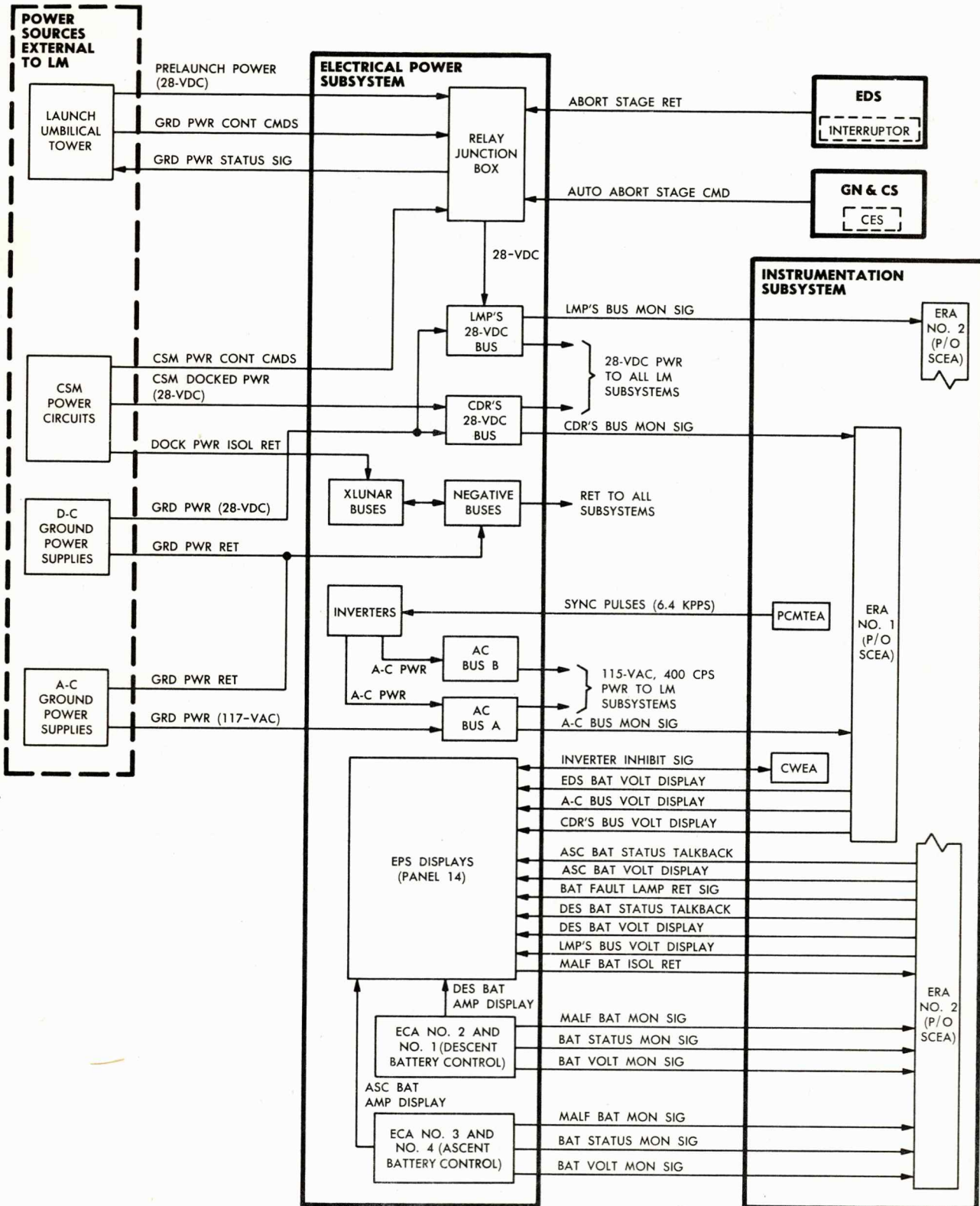


Figure 2.5-1. Electrical Power Subsystem - Signal Interface Block Diagram

300LM4-216

Pressing the ABORT STAGE pushbutton (panel 1) supplies an automatic abort stage command from the Guidance, Navigation, and Control Subsystem (GN&CS) to the RJB with a return path via the Explosive Devices Subsystem (EDS) circuit interrupters. Abort staging initiates a switchover from descent d-c power to ascent a-c power before stage separation.

2.5.2.2 Internal Power Interface. (See figure 2.5-2.)

Battery-generated d-c power is initially supplied to the Commander's and LM Pilot's buses for distribution through circuit breakers. Inverter-generated a-c power is initially supplied to a-c buses A and B for distribution to all subsystems, except the ECS and EDS. The a-c buses supply 115-volt, 400-cps, regulated power through circuit breakers. Figure 2.5-2 shows the circuit breakers associated with specific subsystems, circuit breaker amperage ratings, and reference designations.

2.5.3 FUNCTIONAL DESCRIPTION.

2.5.3.1 General. (See figures 2.5-3 through 2.5-5.)

Four descent stage batteries and two ascent stage batteries make up the EPS d-c section power source. The output of each battery is applied to an ECA. The descent stage ECA's provide an independent control circuit for each descent battery. The ascent stage ECA's provide four independent battery control circuits, two control circuits for each ascent battery. Overtemperature, reverse-current, and overcurrent conditions are monitored by the ECA's. Each ECA battery control circuit can detect a bus or feeder malfunction and operate a main feeder contactor (MFC) associated with the malfunctioning battery to remove the battery from the EPS distribution system (overcurrent only). Only the ascent power backup contactors do not have overcurrent protection.

Descent stage battery power (figure 2.5-4), controlled and monitored by the descent ECA's, is switched and applied to the RJB and the deadface relay box (DRB). The power feeders of descent batteries No. 1 and 2 are routed through the RJB deadface relay to the LM Pilot's circuit breaker panel (16). The power feeder of batteries No. 1 and 2 also route power to the circuit breaker and receptacle on the scientific equipment rack. The power feeders of batteries No. 3 and 4 are routed through the DRB deadface relay to the Commander's circuit breaker panel (11). This circuit arrangement provides for deadfacing all descent battery power in the event of normal staging or abort staging situations. Ascent stage battery power (figure 2.5-5), controlled and monitored by the ascent ECA's, is switched and applied, through power feeders, to terminal junction points at the RJB and DRB.

Ascent and descent battery main power feeders are routed through circuit breakers to the d-c buses. From the d-c buses, d-c power is distributed through circuit breakers to all LM systems.

Two inverters, which make up the EPS a-c section power source, supply all a-c power requirements of the LM. The INVERTER switch (panel 14) connects either inverter to the circuit breakers that supply the a-c buses. Closing the AC BUS A and AC BUS B circuit breakers on panel 11 selects the inverters to feed the a-c buses. Either inverter can supply the LM a-c load requirements.

2.5.3.2 D-C Section.

The EPS d-c section consists of the four descent stage batteries, two descent stage ECA's, two relay switching boxes, two ascent stage batteries, two ascent stage ECA's, a control panel, two circuit breaker panels, and two sensor power fuse assemblies.

The basic EPS power distribution is shown in figure 2.5-3. After ground facility power is disconnected, the descent stage batteries are connected at their low-voltage (LV) taps. These batteries are under light load (less than 200 watts) at this time; therefore, the LV taps (at the 17th cell of these 20-cell batteries) are used to facilitate discharge of the inherent initial high voltage of the battery.

During normal EPS operation, beginning with subsystem activation, the descent battery high-voltage (HV) taps are selected with the HI VOLTAGE switches (panel 14). Power is then fed from the descent stage batteries, through their associated ECA's (batteries No. 1 and 2, and No. 3 and 4, through ECA No. 1 and 2, respectively), to the ascent stage via the RJB and DRB, terminating at the LM Pilot's and Commander's d-c buses. The battery feeders are connected to the d-c buses through the 100-ampere EPS: BAT FEED TIE circuit breakers (panels 11 and 16).

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

D-C power is then distributed to the LM subsystems and to the EPS inverters through circuit breakers on panels 16 and 11. During noncritical phases of normal operation, the 30-ampere EPS: CROSS TIE BAL LOADS circuit breakers (panels 11 and 16) are closed to distribute unbalanced loads between buses so that the batteries discharge evenly. The ascent stage batteries supply d-c power to the LM Pilot's and Commander's buses in essentially the same manner as the descent stage batteries. The batteries are selected with the ASCENT POWER switches (panel 14). The corresponding descent and ascent stage batteries (by respective bus) are paralleled during DPS engine burns, and the EPS: CROSS TIE circuit breakers (panels 11 and 16) are opened, thereby isolating the Commander's and LM Pilot's d-c buses.

Each descent battery is connected to a separate power feeder (jumped to one other battery power feeder at the output of the ECA's and at the 100-ampere EPS: BAT FEED TIE circuit breakers). Failure of one descent battery therefore permits its associated descent battery to supply both power feeders. A power feeder short while operating on descent battery power is detected by overcurrent-sensing circuits in the respective ECA, which disconnects the two descent batteries associated with the shorted power feeder. The d-c buses are isolated from the shorted feeder by opening the respective EPS: BAT FEED TIE circuit breakers. During an emergency, either operating d-c bus can supply power to the other (inoperative) isolated bus if the EPS: CROSS TIE BUS circuit breakers (panels 11 and 16) are closed. A similar condition, whereby a shorted feeder is automatically isolated, exists when operating on both ascent batteries. However, the battery associated with the shorted feeder can be connected manually to the power feeder of the other ascent battery, using the BACK UP FEED switches (panel 14), thereby permitting continued use of the battery.

2.5.3.3 A-C Section. (See figure 2.5-3.)

A-C power is provided to LM subsystems by either of two identical, redundant inverters; it is controlled by the EPS: INV 1 and INV 2 circuit breakers (panel 11 and 16, respectively), the INVERTER switch, and the AC BUS: BUS TIE circuit breakers on panel 11. The EPS: INV 1 or INV 2 circuit breaker supplies 28 volts dc from the LM Pilot's or Commander's 28-volt d-c bus to the chosen normally synched inverter, where the d-c is changed to 115-volt, 400-cps, a-c power (350 volt-amperes steady-state). The INVERTER switch selects and routes the output of either inverter to the a-c buses via the respective AC BUS A: BUS TIE or AC BUS B: BUS TIE circuit breakers. Normally, inverter No. 2 is energized when the LM subsystems are first activated and connected to the a-c buses. Inverter No. 1 functions as backup during the mission, except that it is the operating inverter during DPS and APS engine burns. An a-c bus voltage and frequency readout signal is supplied from a-c bus A via the AC BUS A: AC BUS VOLT circuit breaker (panel 11) to the IS for telemetry and caution light display. An out-of-tolerance frequency (less than 398 cps or more than 402 cps) or low-voltage condition (less than 112 volts ac) causes the INVERTER caution light (panel 2) to go on. The astronaut determines the cause of the malfunction and performs corrective action. The INVERTER caution light goes out when the malfunction is remedied. (When a caution or warning light goes on, the MASTER ALARM pushbutton/light also goes on and a tone is generated. Pressing the MASTER ALARM pushbutton/light extinguishes the pushbutton/light and terminates the tone.) When set to AC BUS, the POWER/TEMP MON selector switch (panel 14) selects a-c bus A for voltage display on the VOLTS indicator (panel 14).

2.5.3.4 Power Monitoring.

The primary a-c and d-c voltage levels, d-c current consumption, and the status of all main power feeders must be monitored periodically to ensure availability of proper power for all LM subsystems, throughout the mission. This monitoring is normally the responsibility of the LM Pilot, who controls the ELECTRICAL POWER panel (14). This panel has talkbacks that indicate main power feeder status, indicators that display battery and bus voltages and battery current, and component caution lights that are used for detecting shorted buses or main power feeder lines and for isolating a malfunctioning battery. Availability of redundant a-c and d-c power permits the astronaut to disconnect, substitute, or reconnect batteries, feeder lines, buses, or inverters to assure a continuous electrical supply.

2.5.3.4.1 Battery Status-Monitoring Circuits. (See figure 2.5-6.)

Talkbacks on panel 14 indicate the status of the descent and ascent batteries (on or off the line). A DESCENT POWER talkback barber-pole display signifies that the related battery is disconnected from its main power feeder; a LO display on a gray background, that the battery low-voltage tap is connected to the

ELECTRICAL POWER SUBSYSTEM

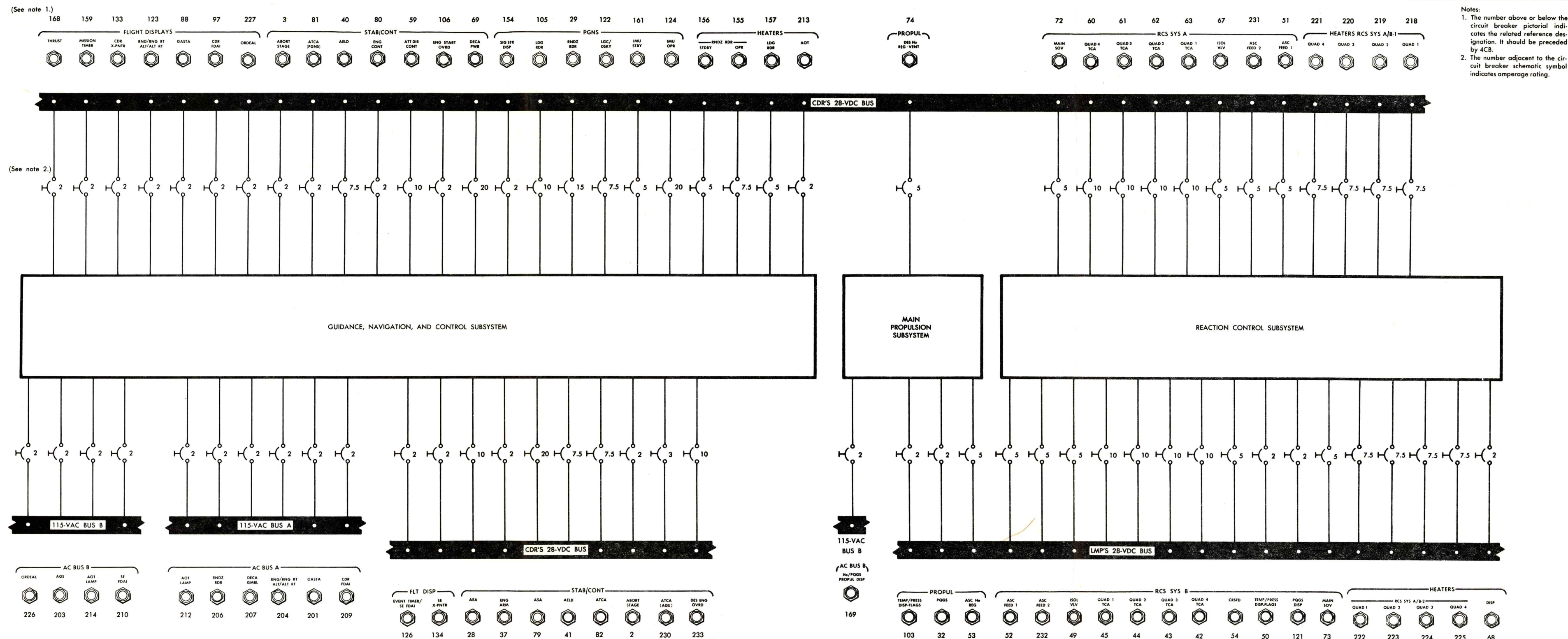


Figure 2.5-2. Electrical Power Subsystem - Power Interface Diagram (Sheet 1 of 2)

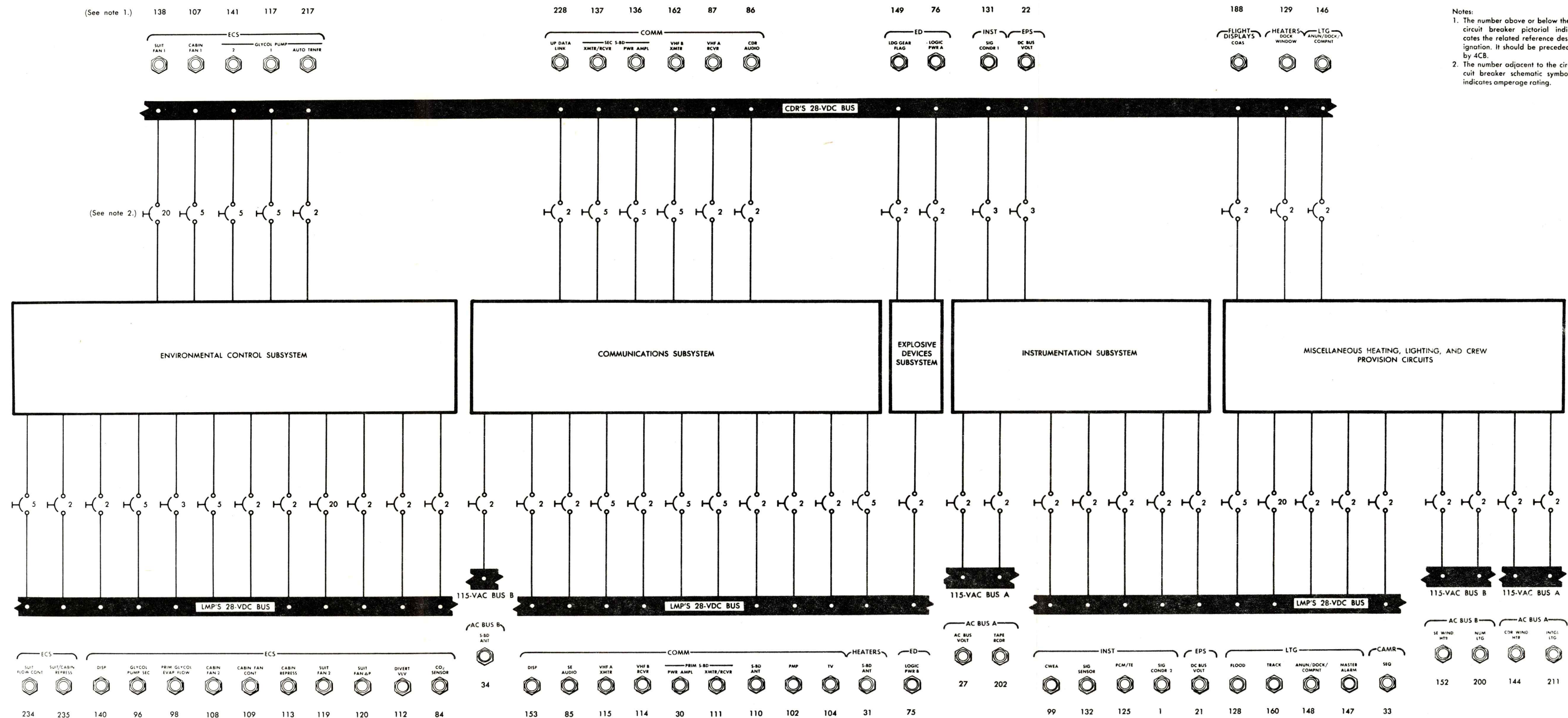
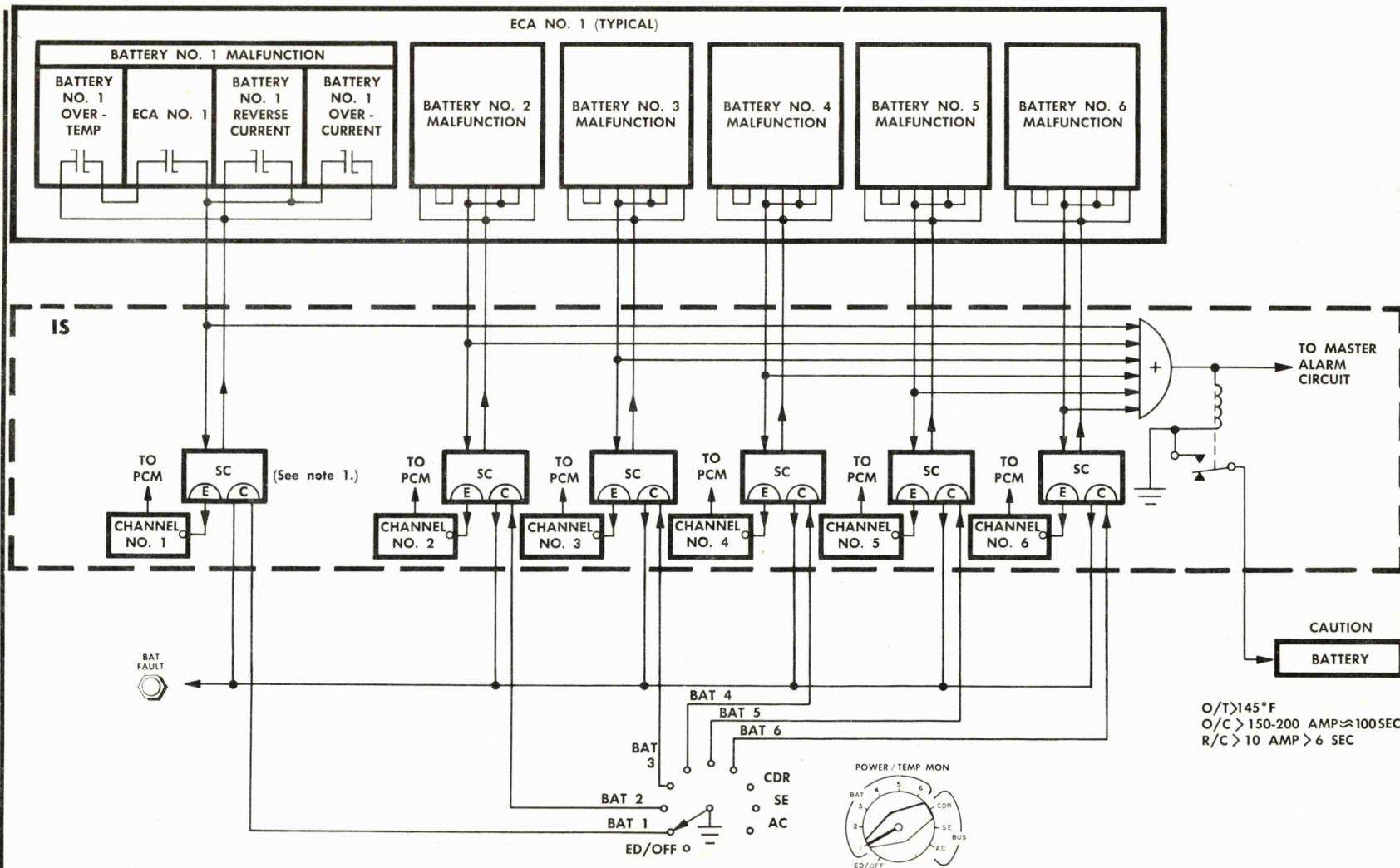


Figure 2.5-2. Electrical Power Subsystem - Power Interface Diagram (Sheet 2 of 2)



Notes:

1. "E" represents an event gate, which provides a channelled signal to the PCM. "C" represents a contact closure gate.
2. Refer to the overall functional block diagram of the IS, for related input and output signals.

Figure 2.5-9. Malfunctioning-Battery Isolation Circuitry - Simplified Schematic Diagram

300LM4-113

removed from its main power feeder line by momentarily setting the respective switch (panel 14) to OFF/RESET.

2.5.3.5 EPS Stages of Operation.

The EPS operates in seven basic stages, as follows:

- GSE-vehicle ground power supply (VGPS) to LM until T-7 hours (ac and dc)
- GSE-launch umbilical tower (LUT) power to LM from T-7 hours until T-30 minutes (dc only)
- Descent stage power from T-30 minutes through CSM rendezvous and docking (descent batteries on low-voltage taps)
- Earth orbit - CSM/LM power
- Descent stage power (descent batteries manually switched to high-voltage taps)
- Ascent stage power (on normal main feeders)
- Staging (normal or abort)

2.5.3.5.1 GSE-Supplied Prelaunch Power. (See figure 2.5-10.)

Before earth launch, a-c and d-c power are supplied to the LM by the VGPS. An umbilical cable or a portable power supply is carried onboard the LM and connected to two GSE connectors: one located under the lower circuit breaker tier (panel 11), the other under the lower circuit breaker tier (panel 14). At approximately T-7 hours, the VGPS power is replaced by LUT power. The SLA/LM assembly is now connected to the LUT by umbilical cable. LUT power is supplied to the LM Pilot's bus through the RJB. LUT remote control includes LUT deadface relay (LDR) set and reset commands, and LM high-voltage and low-voltage off and low-voltage on signals. The LDR supplies power through the LM Pilot's main power feeders to the LM Pilot's d-c bus. D-C power is supplied to the Commander's bus through the EPS: CROSS TIE circuit breakers.

2.5.3.5.2 Descent Stage Power (Low-Voltage Taps).

At T-30 minutes, the LUT resets the LDR and connects the EPS descent battery low-voltage taps to the respective feeder lines. The descent batteries supply limited power to certain critical equipments from T-30 minutes until completion of CSM transposition and docking.

2.5.3.5.3 Earth Orbit - CSM/LM Power. (See figure 2.5-11.)

On completion of CSM rendezvous and docking, an astronaut connects power and control umbilical cables from the LM to the CSM. The CSM deactivates LM power and then supplies power to the LM critical equipment, using the LM translunar negative bus. CSM/LM control logic prohibits CSM power to the LM power distribution system while the LM descent stage batteries are on the main power feeder lines. CSM/LM control logic is as follows:

- CSM deactivates LM power (high voltage and low voltage off)
- CSM initiates CSM power to LM
- CSM deactivates CSM power to LM
- Astronauts enter LM. Select low-voltage taps and set to ON, thereby automatically deactivating CSM power.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

After staging, power to the LM ascent stage can again be supplied by the CSM. This ensures that LM heaters and other essential equipment will operate continuously until the final ascent engine burn is initiated. The normally open pins of a circuit interrupter in the ascent stage close when staging occurs. This completes a circuit that simulates the presence of descent batteries in shut off condition, thus enabling the CSM to supply power to the LM when the two vehicles are docked. The LM circuit breakers must be returned to the prelaunch configuration before the astronaut in the CSM initiates the power transfer.

2.5.3.5.4 Descent Stage Power (High-Voltage Taps).

Each of the four descent batteries has a HI VOLTAGE and LOW VOLTAGE switch and a talk-back on panel 14. Descent stage battery low-voltage and off/reset control lines are routed through the

EPS

ELECTRICAL POWER SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.5-20A/2.5-20B

ELECTRICAL POWER SUBSYSTEM

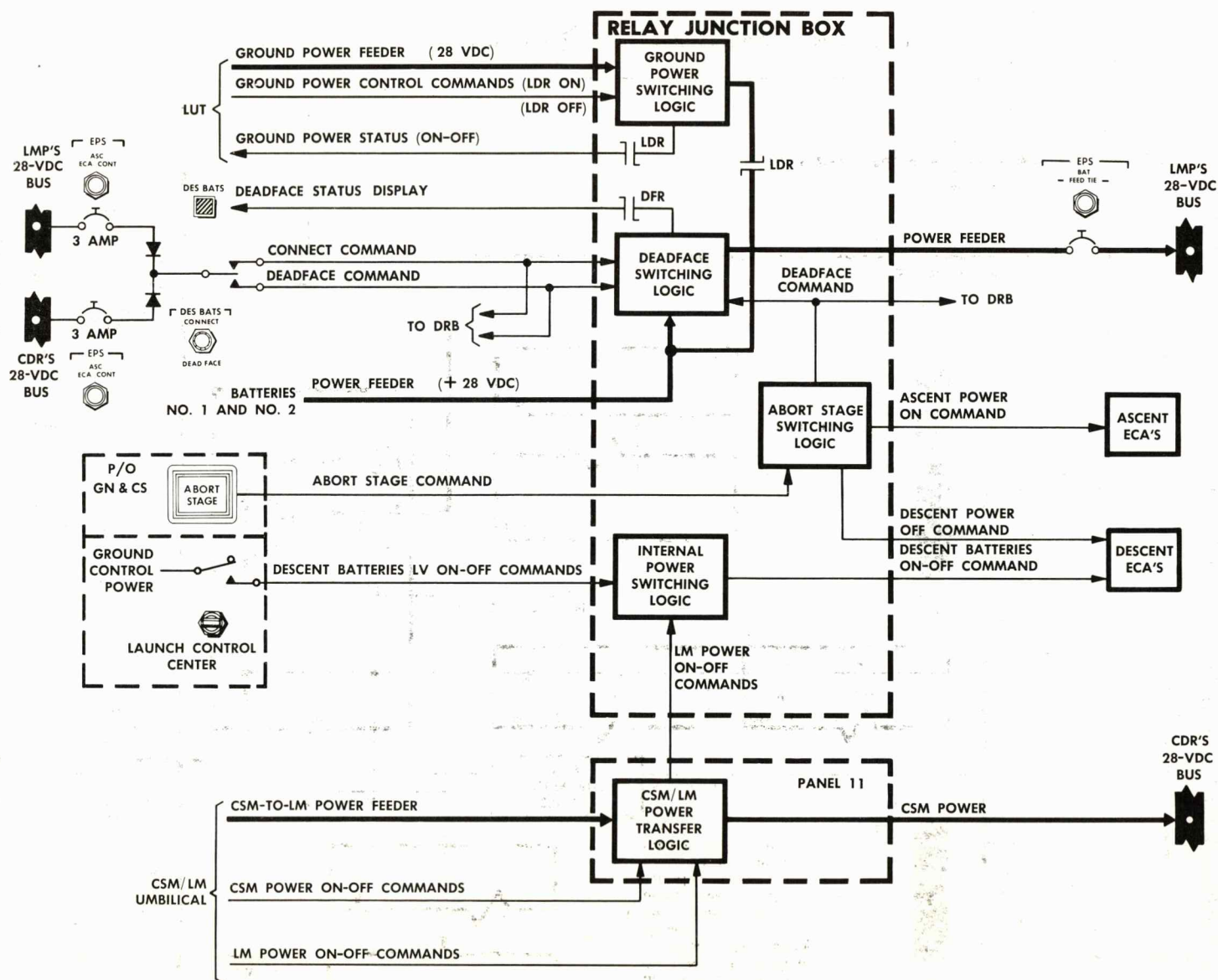


Figure 2.5-10. Relay Junction Box - Functional Block Diagram

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

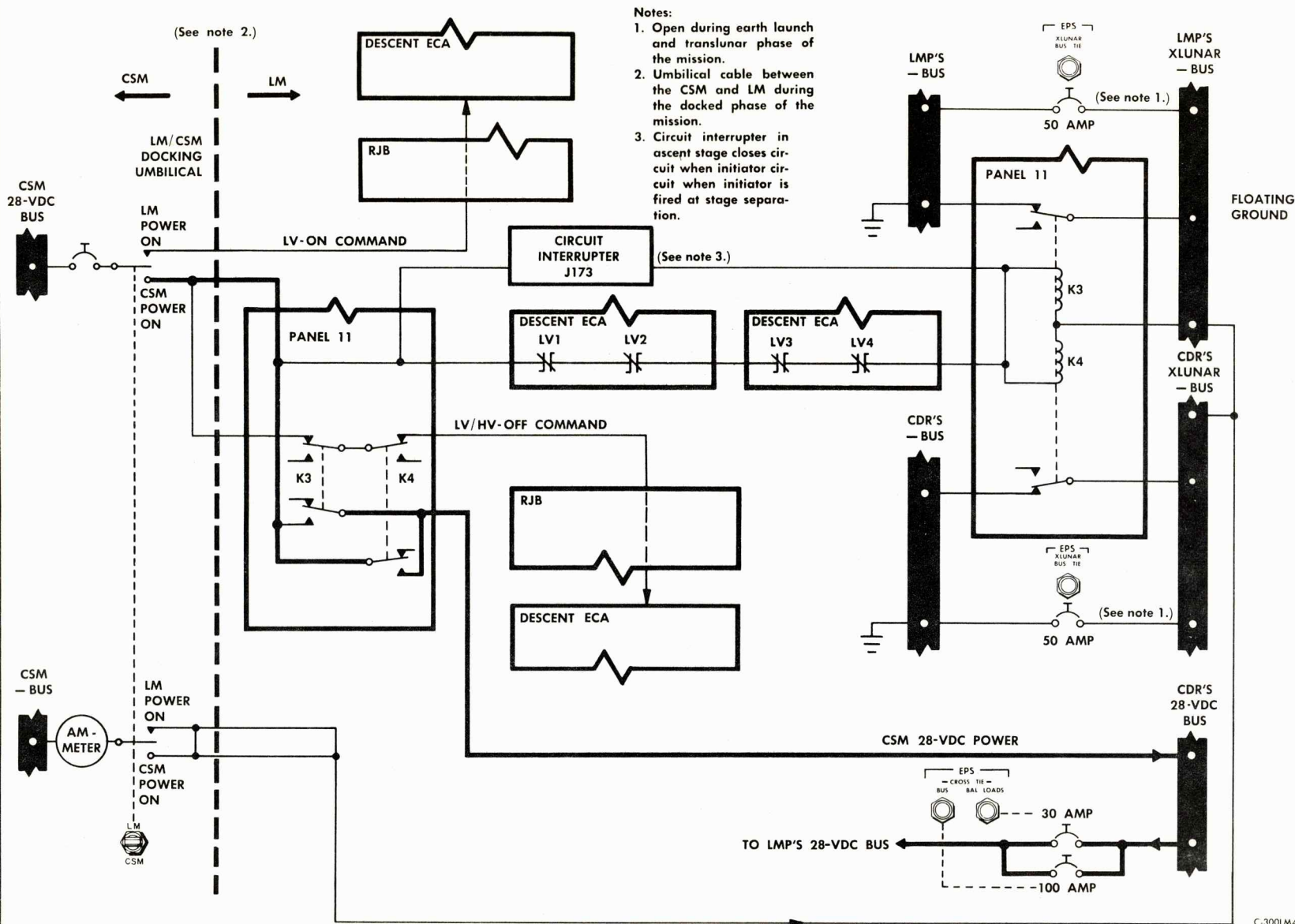


Figure 2.5-11. CSM/LM Primary Power Transfer Logic - Simplified Schematic Diagram

C-300LM4-105

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. Battery temperature in excess of +1450° ±5°F closes a thermal sensor, causing the ascent battery normal main feed contactor is open). The astronaut then takes corrective action to eliminate the faulty battery. Two ascent stage batteries ordinarily supply the d-c power requirements, 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 voltage. Normally, the ascent stage batteries are paralleled to ensure that both batteries discharge evenly. See figure 2.5-5 for a functional block diagram of ascent stage battery power distribution. 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.

2.5.4.3 Descent Stage Electrical Control Assemblies. (See figure 2.5-13.)

The two descent stage ECA's protect and control the four descent stage batteries. Each ECA has a set of control circuits for each battery accommodated. A failure in one set of battery control circuits does not affect the other set. The primary function of the ECA is to connect the two related batteries (normal) or one battery (contingency) to the associated main feeder line through the LV or HV MFC's. The protective circuits of the ECA automatically disconnect a descent stage battery if an over-current condition is detected, and cause the BATTERY caution light and MASTER ALARM pushbutton lights to go on if a battery overcurrent, reverse-current, or over-temperature condition is detected.

Analogous of battery voltage and current (monitored constantly in the ECA) are routed to the IS to provide indications on the VOLTS and AMPS indicators (panel 14). Battery talkbacks (low voltage, high voltage, and off) receive drive signals derived from ECA-supplied discrete levels processed through the IS. The major elements in the descent stage ECA's are LV and HV MFC's, current monitors, over-current relays, reverse-current relays, and power supplies. An auxiliary relay supplies system logic contact closures to other control assemblies in the LM power distribution system.

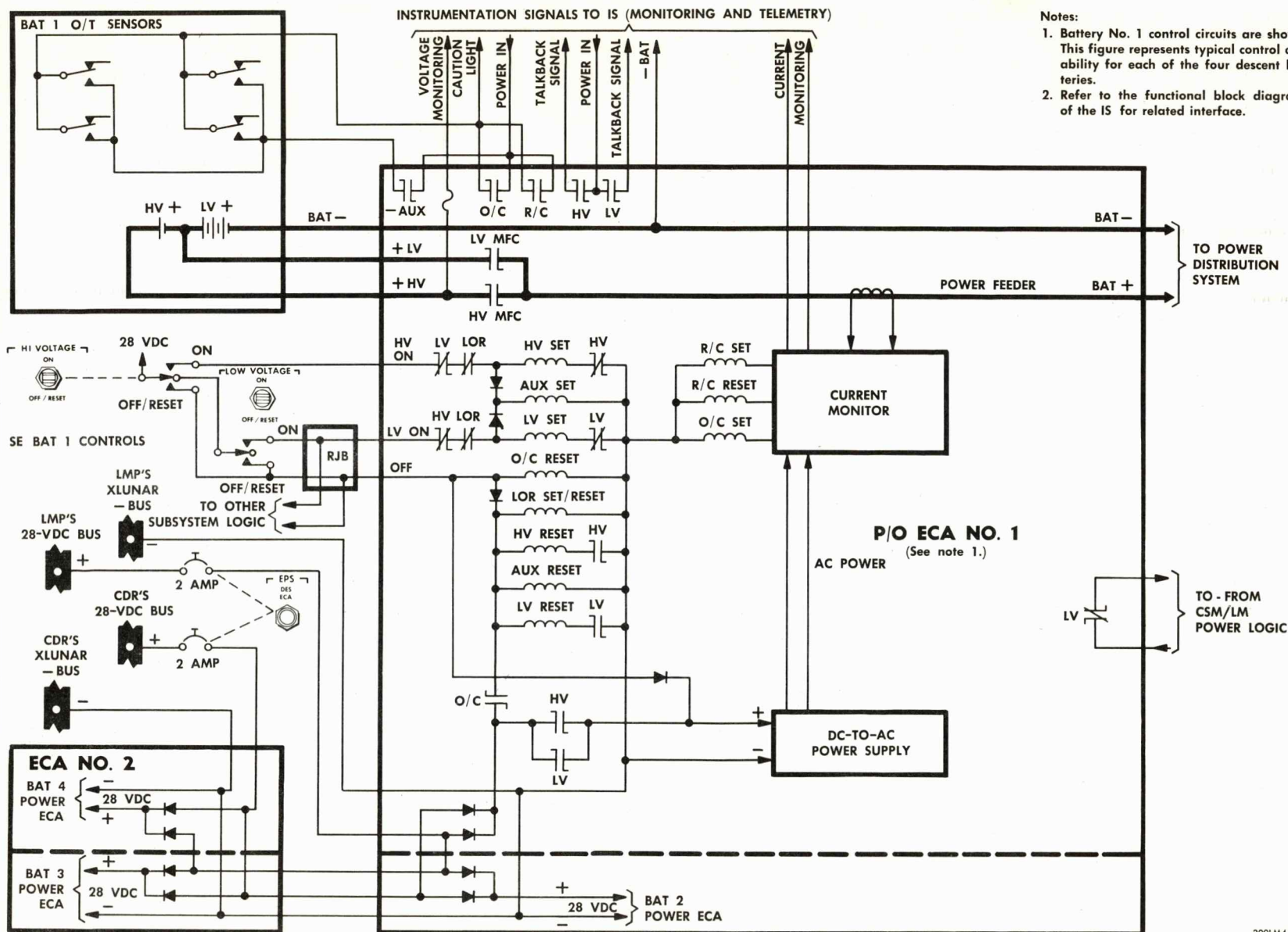
The MFC's are electrically interlocked to prevent closing an HV contactor when an LV contactor is already closed, and vice versa. This prevents short-circuiting the tapped sections of the respective descent stage battery. The MFC's are operated with the DESCENT POWER HI VOLTAGE and LOW VOLTAGE switches.

The ECA current monitor produces isolated outputs for the IS, overcurrent relay, and reverse-current relay from the battery current that flows through the HV or LV MFC's. The current monitor supplies a d-c level (proportional to battery current) to the overcurrent relay. Voltage in the relay circuit builds up until the close coil of the overcurrent relay is energized (reflecting 150 to 200 amperes for approximately 100 seconds minimum), at which time the HV or LV contactor opens to disconnect the battery from its power feeder. The magnitude of battery current is inversely proportional to the time it takes for the close coil of the relay to be energized; the greater the initial overcurrent, the less time required for tripping. The overcurrent relay is sensitive to magnitude; it therefore follows the overcurrent trip characteristics. The relay must be reset manually (OFF/RESET position of the DESCENT POWER HI VOLTAGE and LOW VOLTAGE switches) once it is tripped. A lockout relay (LOR) ensures that the battery cannot be turned on unless the overcurrent relay has been reset. For each over-current circuit, the ECA provides an alarm signal to the caution and warning circuits in the IS, causing the BATTERY caution light and MASTER ALARM pushbutton/lights (panels 1 and 2) to go on, and a tone to be generated.

The reverse-current relay causes the BATTERY caution light to go on when the current flow in the direction opposite to normal current flow exceeds 10 amperes for at least 6 seconds. Unlike the

ELECTRICAL POWER SUBSYSTEM

ELECTRICAL POWER SUBSYSTEM



LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

overcurrent relay, the reverse-current relay does not open the related MFC's and is self-resetting when the current monitor ceases to detect a reverse-current condition. During reverse-current conditions, the related MFC must be switched open manually by setting the DESCENT POWER switches to OFF/RESET. The descent stage ECA power supplies provide ac for current-monitor excitation, and regulated dc for the other ECA circuits.

2.5.4.4 Ascent Stage Electrical Control Assemblies. (See figure 2.5-14.)

The two ascent stage ECA's individually control and protect the two ascent stage batteries (No. 5 and 6) in nearly the same manner as the descent stage ECA's control and protect the descent stage batteries. Each ascent stage ECA contains electrical power feeder contactors, an overcurrent relay, a reverse-current relay, and a current monitor. Each ascent stage battery can be connected to its normal or backup main feeder line via the normal MFC or the backup main feed contactor (BU MFC) in its respective ECA. Both batteries are thereby connected to the primary d-c power buses. The normal feeder line has overcurrent protection; the backup feeder line does not. Otherwise, the ascent stage ECA's function the same as the descent stage ECA's. The ascent stage ECA's provide reserve-current monitoring, overcurrent protection, and contact closures for indicators. The MFC's and BU MFC's in the ascent stage ECA's are manually operated with the respective ASCENT POWER switches.

2.5.4.5 Relay Junction Box. (See figure 2.5-10.)

The RJB provides the following:

- Control logic and junction points for connecting external prelaunch power (via the LUT) to the LM Pilot's d-c bus.
- Control and power junction points for connecting descent stage and ascent stage ECA's to the LM Pilot's bus
- Deadfacing of half the power feeders between the descent and ascent stages.

The RJB provides control of all LV contactors (on and off) from the LUT and CSM, and control of all low- and high-voltage descent power contactors (off) on receipt of an abort stage command. On and off control of ground facility power through the LUT is provided by the LDR. The RJB includes abort logic relays, which, when energized by an abort stage command, close the ascent stage battery MFC's and open the deadface relays. The RJB deadface relays provide on and off control of the main power feeders of descent batteries No. 1 and 2. These relays are opened and closed with the DES BATS switch, or are opened automatically when the abort logic relays close. The deadface relays in the RJB deadface half of the main power feeders between the descent and ascent stages; the other half of the power feeders is deadfaced by the deadface relays in the DRB. The ascent stage then provides primary d-c power to the LM.

2.5.4.6 Deadface Relay Box. (See figure 2.5-12.)

The DRB contains a deadface relay, which controls the output of descent stage batteries No. 3 and 4 in the same manner as the RJB deadface relay controls descent stage batteries No. 1 and 2. Because the Commander's and LM Pilot's circuit breaker panels are located on opposite sides of the LM cabin, two individual deadfacing facilities (one for each circuit breaker panel) provide a physically expedient deadfacing process.

2.5.4.7 Inverters. (See figure 2.5-15.)

Two redundant 400-cps inverters individually supply the primary a-c power required in the LM. Inverter output is controlled by application of 28 volts dc from the Commander's or LM Pilot's 28-volt d-c bus through the EPS: INV 1 and INV 2 circuit breakers, the INVERTER switch, and the AC BUS A and AC BUS B: BUS TIE circuit breakers. The inverters are identical; therefore, only the inverter No. 1 circuitry is discussed. The 28 volts dc is applied through an electromagnetic interference (EMI) filter and an input filter to a dc-to-dc converter. The regulated output of the converter is changed to a 400-cps square wave in the inverter stage. The output of the inverter stage is controlled by 400-cps

ELECTRICAL POWER SUBSYSTEM

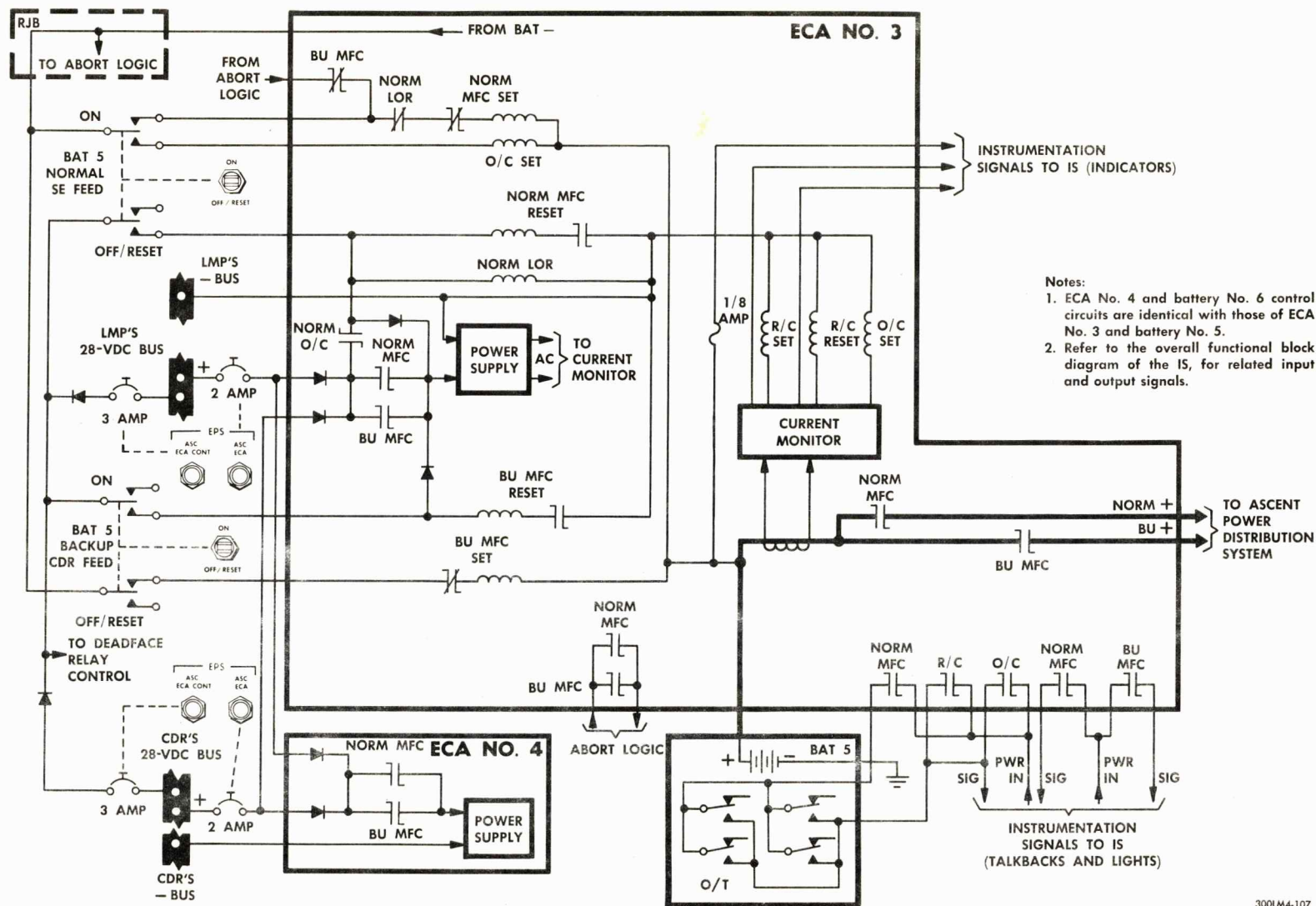


Figure 2.5-14. Ascent Stage Electrical Control Assembly - Simplified Schematic Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

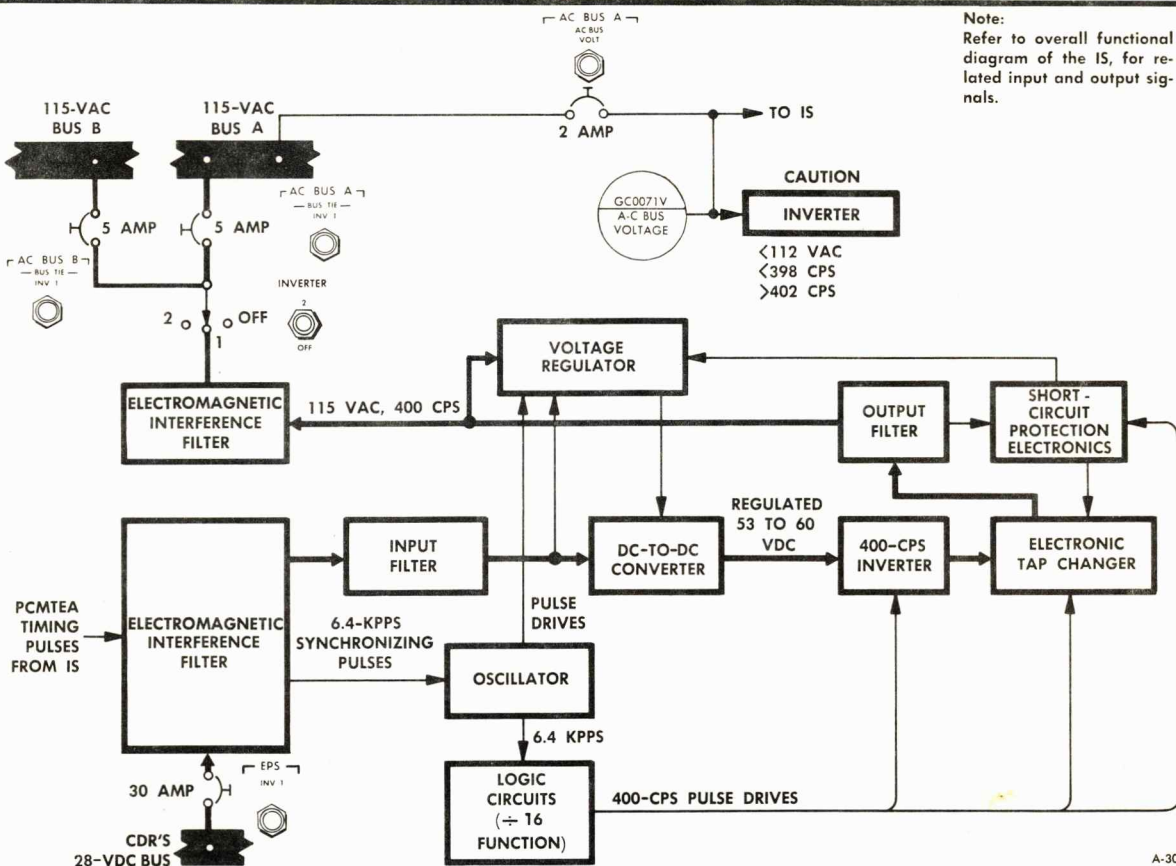


Figure 2.5-15. Inverter No. 1 - Block Diagram

pulse drives developed from a 6.4-kilopulse-per-second (kpps) oscillator, which is, in turn, synchronized by timing pulses from the IS. The output of the inverter stage is 400 ± 0.4 cps, synchronized; 400 ± 10 cps, with a free-running oscillator. The 6.4-kpps oscillator output is divided precisely by 16 in the divider logic circuit and routed to the inverter stage, the electronic tap changer, and short-circuit protection circuits. The electronic tap changer sequentially selects the output of the tapped transformer in the inverter stage, converting the 400-cps square wave to an approximate sine wave of the same frequency. The output filter minimizes harmonic distortion. The voltage regulator maintains the inverter output at 115 volts ac $\pm 1\%$ during normal load conditions by controlling the amplitude of the dc-to-dc converter output. The voltage regulator also compensates for variations in the d-c input and a-c output load.

Setting the INVERTER switch to 1 routes the inverter No. 1 output to the AC BUS A: BUS TIE INV 1 circuit breaker for a-c bus A application and to the AC BUS B: BUS TIE INV 1 circuit breaker for a-c bus B application. A-C power at bus A is monitored for proper voltage amplitude and frequency by the IS via the AC BUS A: AC BUS VOLT circuit breaker (panel 11). When the voltage at bus A is less than 112 volts ac, or the frequency is less than 398 cps or more than 402 cps, the INVERTER caution light goes on. In addition, the master alarm circuit is completed. The INVERTER caution light goes off when the malfunction is remedied.

2.5.4.8 Circuit Breaker and EPS Control Panels.

All primary a-c and d-c power feed circuits are protected by appropriately rated circuit breakers on the Commander's and LM Pilot's buses. The two d-c buses are electrically connected by the main power feeder network. Functionally redundant LM equipment is placed on both d-c buses (one on each bus), so that each bus can individually perform an abort mission during emergencies. EPS circuit controls and monitoring displays are on panel 14. Section 3 contains a description of all EPS controls, displays, and circuit breakers. Circuit breaker characteristics are given in table 2.5-1.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.5-1. Circuit Breaker Characteristics

D-C circuit breaker	
Operating voltage	28 volts
Operating voltage limit	32.5 volts
Transient voltage limits	50 volts above, 100 volts below operating voltage
Ripple voltage	Less than 1.5 volts, peak
A-C circuit breaker	
Steady-state operating voltage	115 \pm 2.5 volts rms
Transient voltage limits	90 to 188 volts, peak
Transient voltage recovery time	170 milliseconds

2.5.4.9 Sensor Power Fuse Assemblies. (See figure 2.5-16.)

Two sensor power fuse assemblies, in the aft equipment bay, provide a secondary d-c bus system that supplies excitation to ECS, RCS, and MPS transducers that develop display and telemetry data. During prelaunch procedures, the INST: SIG SENSOR circuit breaker (panel 16) is open, and primary power is supplied to the assemblies from the Commander's 28-volt d-c bus, via the EPS: DC BUS VOLT circuit breaker (panel 11) and the closed contacts of relay K6 in panel 11. The coil of K6 is energized by the ground-supplied GSE umbilical cable. Before launch, the GSE umbilical is disconnected and K6 is deenergized. Power is then supplied directly to the sensor power fuse assemblies from the LM Pilot's 28-volt d-c bus by closing the INST: SIG SENSOR circuit breaker. Each assembly comprises a positive d-c bus, negative return bus, and 40 fuses. All sensor return lines are routed to a common ground bus.

2.5.5 FAILURE MODES.

The EPS failure modes and the effect on mission capability are given in table 2.5-2.

Table 2.5-2. Electrical Power Subsystem - Failure Modes

Failure	Indication	Vehicle Capability	
		Before Staging	After Staging
Descent Stage Power			
One descent battery failed.	BATTERY caution light goes on. BAT FAULT component caution light goes on when POWER/TEMP MON switch is set to failed battery.	Abbreviated mission may still be effected, using remaining descent batteries.	N/A

ELECTRICAL POWER SUBSYSTEM

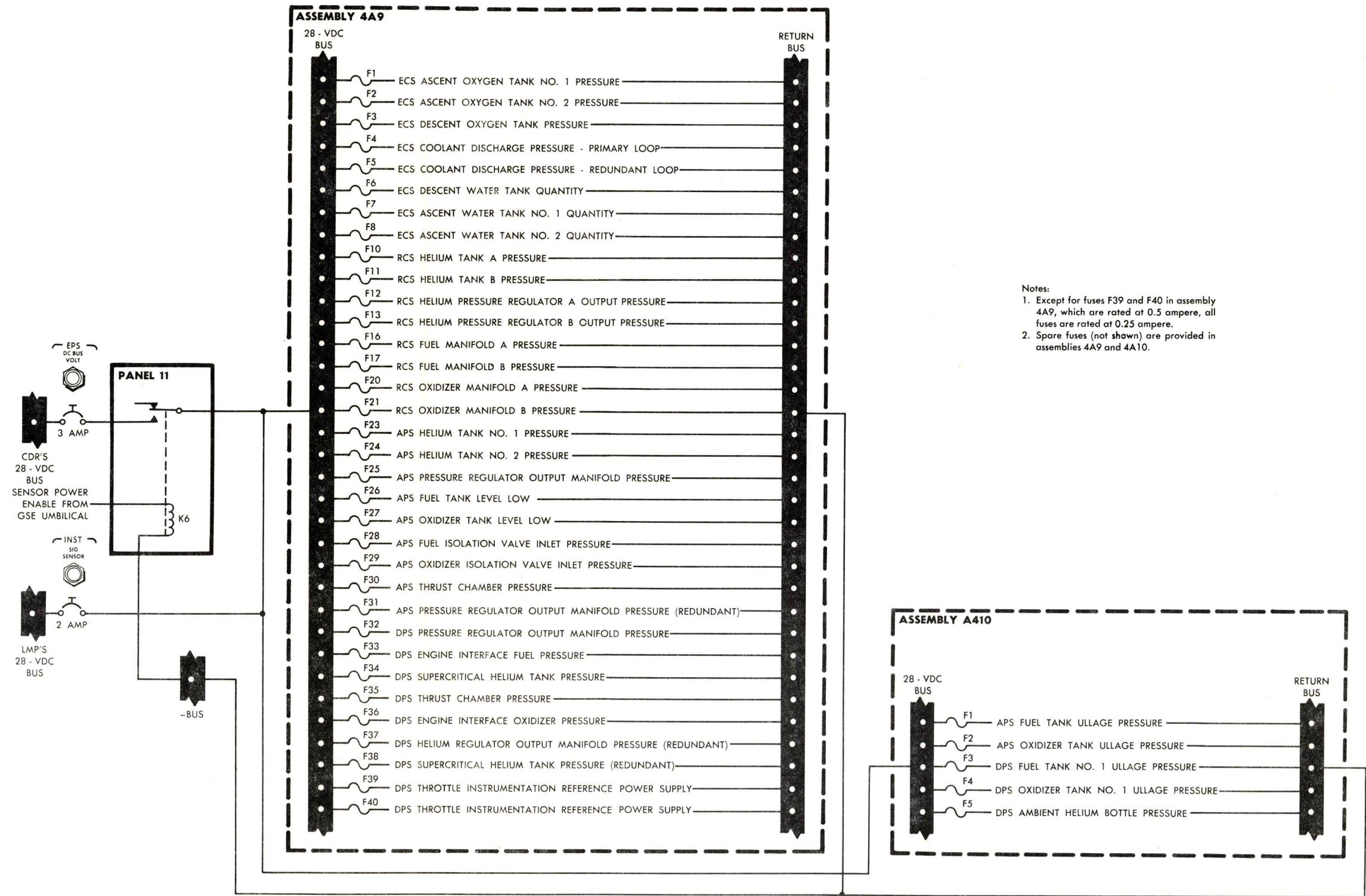


Figure 2.5-16. Sensor Power Fuse Assemblies - Simplified Power Distribution Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.5-2. Electrical Power Subsystem - Failure Modes (cont)

Failure	Indication	Vehicle Capability	
		Before Staging	After Staging
Descent Stage Power			
Two or more descent batteries failed.	BATTERY caution light goes on. BAT FAULT component caution light goes on when POWER/TEMP MON switch is set to failed battery.	Mission must be severely curtailed. Mission may be aborted, with unstaged vehicle rendezvousing with CSM (using ascent power and remaining descent power); vehicle may be staged, followed by normal resumption of ascent stage mission procedures (using ascent power) and rendezvous and docking with the CSM.	N/A
One bus or main feeder line failed (panel 11 EPS: CROSS TIE circuit breakers open). *	DC BUS warning light and DC BUS FAULT component caution light go on, and associated DESCENT STAGE talkbacks provide barber-pole display.	Mission must be aborted after faulty bus or main feeder line is isolated from power distribution network and ascent power is supplied to operational feeder.	See ascent stage power section of table.
One bus or main feeder line failed (panel 16 EPS: CROSS TIE circuit breakers open). *	C/W PWR caution light, and DC BUS FAULT component caution light go on, and all DESCENT POWER talkbacks provide barber-pole display.	Mission must be aborted after faulty bus or main feeder line is isolated from power distribution network and ascent power is supplied to operational feeder.	See ascent stage power section of table.
Ascent Stage Power			
One ascent battery failed.	BATTERY caution light goes on. BAT FAULT component caution light goes on when POWER/TEMP MON switch is set to failed battery.	Mission must be aborted. Rendezvous and docking by unstaged vehicle may be achieved, using one ascent battery and remaining descent power.	Mission must be aborted. Remaining ascent battery is sufficient to ensure safe rendezvous and docking of LM ascent stage and CSM.
One bus or main feeder line failed (panel 11 or 16 EPS: CROSS TIE circuit breakers open). *	DC BUS warning light, C/W PWR caution light, and DC BUS FAULT component caution light go on and associated ASCENT POWER talkbacks provide barber-pole display.	N/A	Mission must be aborted. Faulty bus or main feeder line must be isolated from power distribution network, and associated battery must be removed from faulty feeder line (and respective bus) and connected to remaining bus.
* A d-c bus or main feeder line short with the EPS: CROSS TIE circuit breakers closed results in automatic opening of these circuit breakers. (Batteries feeding into the short will automatically be removed). The effect of the short on vehicle capability is then as stated.			

EPS

ELECTRICAL POWER SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.5-2. Electrical Power Subsystem - Failure Modes (cont)

Failure	Indication	Vehicle Capability	
		Before Staging	After Staging
Ascent Stage Power (cont)			
Inverter No. 1 failed.	INVERTER caution light goes on with INVERTER switch set to 1.	Mission must be aborted after inverter No. 2 is selected to replace inverter No. 1.	Same as mission effect before staging.
Inverter No. 2 failed.	INVERTER caution light goes on with INVERTER switch set to 2.	Redundant inverter No. 1 output is sufficient to continue nominal mission.	Same as mission effect before staging.
A-C bus failed.	INVERTER caution light goes on with INVERTER switch set to 1 or 2.	Faulty bus must be isolated from a-c power distribution network, with remaining a-c bus continuing to operate normally.	Same as mission effect before staging.
D-C Electrical Power - General			
Control of any ascent battery or any two descent batteries is lost.	Talkbacks (panel 14) do not display proper command feedback.	Mission must be aborted, using remaining available power.	Same as mission effect before staging.

2.5.6 PERFORMANCE AND DESIGN DATA.

The performance and design data for the EPS are given in table 2.5-3.

Table 2.5-3. Electrical Power Subsystem - Performance and Design Data

A-C Section	
Inverter input voltage	24 to 32 volts dc
Inverter output (with PCMTEA sync)	115±1.2 volts rms, 400±0.4 cps, single-phase
Inverter output (with internal sync)	115±1.2 volts rms, 400±10 cps, single-phase
Normal load range	0 to 350 volt-amperes (at power factors 0.65 lagging to 0.80 leading)
Maximum overload at constant voltage output	525 volt-amperes for 10 minutes
D-C Section	
Steady-state bus voltage limits	26.5 to 32.5 volts dc
Nominal supply bus voltage	28 volts dc
Transient voltages	50 volts above or below nominal supply voltage

ELECTRICAL POWER SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEM DATA

Table 2.5-3. Electrical Power Subsystem - Performance and Design Data (cont)

Power requirements	
Ascent ECA's	84 watts, momentarily during battery switching; 10 watts, steady-state
Descent ECA's	84 watts, momentarily during battery switching; 20 watts, steady-state
EPS displays	2 watts

2.5.7 OPERATIONAL LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions for the EPS are as follows:

- When EPS batteries are in use, the primary glycol cooling loop (ECS) must be operating. If this restriction is not observed, EPS components will be damaged. The only deviation to this is at low power levels (200 watts or less); then batteries can operate for 8 hours without cooling.
- When supplying power, temperature must be maintained between +29° and +100°F. If this restriction is exceeded, battery could be damaged or voltage output could go out of specification (less than 26.5 volts dc).
- Maximum continuous inverter power output must not exceed 350 volt-amperes. Demand can go to 525 volt-amperes for up to 10 minutes. If this limitation is exceeded, inverter components could be damaged.
- Maximum CSM power to LM during docked coast phases is approximately 296 watts peak (average is approximately 108 watts). If this restriction is exceeded, circuit breakers in the CSM will trip, shutting off power to the LM.
- Available ascent electrical energy is 17.8 kilowatt-hours at maximum steady-state current drain of 50 amperes per battery, at 28 volts dc. Available descent electrical energy is 46.9 kilowatt-hours maximum at steady-state current drain of 25 amperes per battery, at 28 volts dc.
- During descent engine burns, the corresponding descent and ascent stage batteries (by supplied bus) must be paralleled to preclude a low bus voltage due to descent battery failure.
- All EPS: CROSS TIE circuit breakers must be opened during descent engine burns. If this restriction is not observed, a short on one bus will affect the other bus.
- Maximum current drain on one ascent battery during ascent engine burns (abort condition) is 104 amperes. At higher current drains, the d-c bus voltage may drop below the low operating limit.
- Inverter output voltage must never drop below 110 volts ac; otherwise the DECA may become inoperative.

2.5.8 TELEMETRY MEASUREMENTS.

The EPS telemetry data are given in table 2.5-4. "(R)" in the "Telemetry Range" column signifies a real-time telemetry measurement. Real-time telemetry points are shown in figures 2.5-4 and 2.5-5 (d-c circuits) and 2.5-15 (a-c circuits).

ELECTRICAL POWER SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.5-4. Electrical Power Subsystem - Telemetry Measurements

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GC0020X	LM umbilical release	0 to 28 vdc		None
GC0071V	A-C bus voltage	0 to 125 vac (115 vac) (R)	62.5 to 125 vac (20 to 40 vdc)	VOLTS indicator
GC0155F	A-C bus frequency	380 to 420 cps (400 cps) (R)		INVERTER caution light
GC0201V	Battery No. 1 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0202V	Battery No. 2 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0203V	Battery No. 3 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0204V	Battery No. 4 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0205V	Battery No. 5 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0206V	Battery No. 6 voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0301V	Commander's bus voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC0302V	LM Pilot's bus voltage	0 to 40 vdc (28 vdc) (R)	20 to 40 vdc	VOLTS indicator
GC1201C	Battery No. 1 current	0 to 60 amperes (R)	0 to 120 amperes ÷ 2	AMPS indicator
GC1202C	Battery No. 2 current	0 to 60 amperes (R)	0 to 120 amperes ÷ 2	AMPS indicator
GC1203C	Battery No. 3 current	0 to 60 amperes (R)	0 to 120 amperes ÷ 2	AMPS indicator
GC1204C	Battery No. 4 current	0 to 60 amperes (R)	0 to 120 amperes ÷ 2	AMPS indicator
GC1205C	Battery No. 5 current	0 to 120 amperes (R)	0 to 120 amperes	AMPS indicator
GC1206C	Battery No. 6 current	0 to 120 amperes (R)	0 to 120 amperes	AMPS indicator
GC4361X	Battery No. 1 high-voltage tap	Contact closure (R)		SE BAT 1 talk- back (gray)
GC4362X	Battery No. 1 low-voltage tap	Contact closure (R)		SE BAT 1 talk- back (white)
GC4363X	Battery No. 2 high-voltage tap	Contact closure (R)		SE BAT 2 talk- back (gray)
GC4364X	Battery No. 2 low-voltage tap	Contact closure (R)		SE BAT 2 talk- back (white)
GC4365X	Battery No. 3 high-voltage tap	Contact closure (R)		CDR BAT 3 talk- back (gray)
GC4366X	Battery No. 3 low-voltage tap	Contact closure (R)		CDR BAT 3 talk- back (white)

ELECTRICAL POWER SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.5-4. Electrical Power Subsystem - Telemetry Measurements (cont)

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GC4367X	Battery No. 4 high-voltage tap	Contact closure (R)		CDR BAT 4 talk-back (gray)
GC4368X	Battery No. 4 low-voltage tap	Contact closure (R)		CDR BAT 4 talk-back (white)
GC4369X	Battery No. 5 on Commander's bus	Contact closure (R)		BAT 5 BACK UP CDR FEED talkback
GC4370X	Battery No. 6 on Commander's bus	Contact closure (R)		BAT 6 NORMAL CDR FEED talkback
GC4371X	Battery No. 5 on LM Pilot's bus	Contact closure (R)		BAT 5 NORMAL SE FEED talkback
GC4372X	Battery No. 6 on LM Pilot's bus	Contact closure (R)		BAT 6 BACK UP SE FEED talkback
GC4381X	Auxiliary relay No. 1 condition	Contact closure		None
GC9961U	Battery No. 1 malfunction (Composite of GC3511X - O/T, GC4301X - O/C, GC4321X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light
GC9962U	Battery No. 2 malfunction (Composite of GC3512X - O/T, GC4302X - O/C, GC4322X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light
GC9963U	Battery No. 3 malfunction (Composite of GC3513X - O/T, GC4303X - O/C, GC4323X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light
GC9964U	Battery No. 4 malfunction (Composite of GC3514X - O/T, GC4304X - O/C, GC4324X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light
GC9965U	Battery No. 5 malfunction (Composite of GC3515X - O/T, GC4305X - O/C, GC4325X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light
GC9966U	Battery No. 6 malfunction (Composite of GC3516X - O/T, GC4306X - O/C, GC4326X - R/C)	Contact closure (R)		BATTERY caution light and BAT FAULT component caution light

ELECTRICAL POWER SUBSYSTEM

2.6 ENVIRONMENTAL CONTROL SUBSYSTEM.

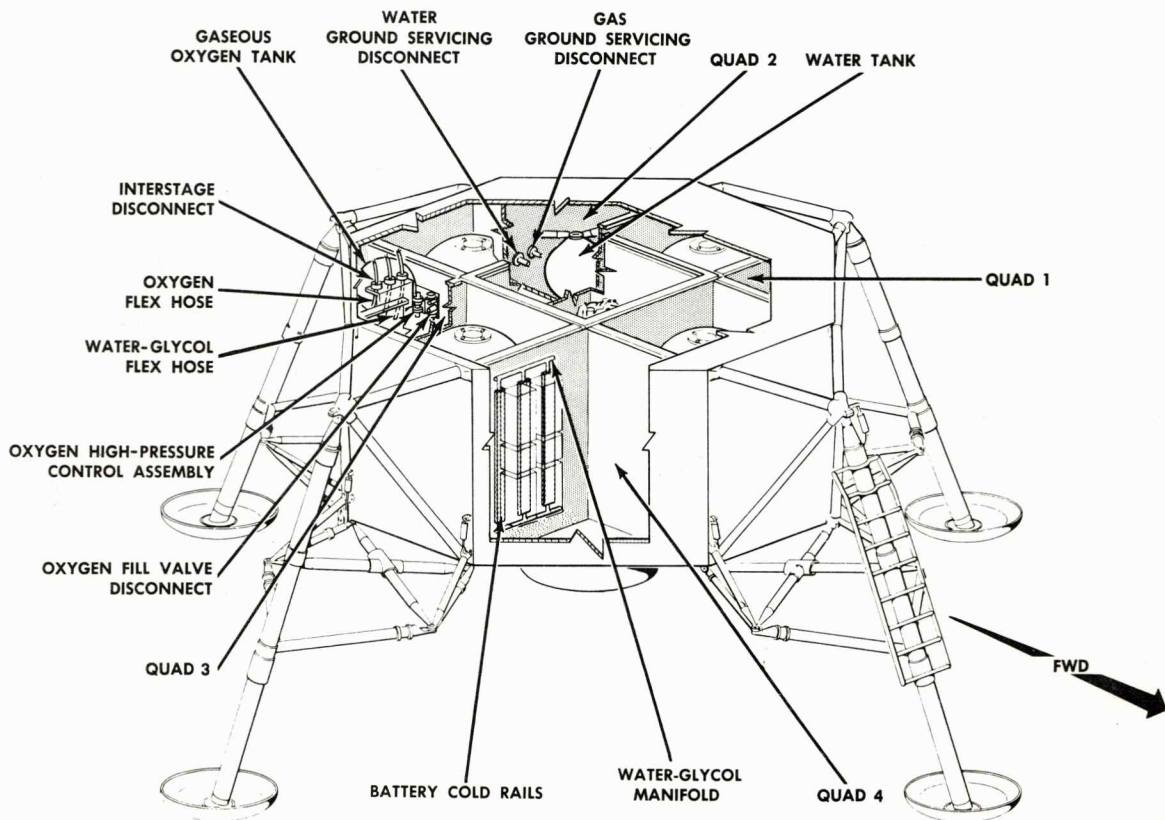
2.6.1 INTRODUCTION.

The Environmental Control Subsystem (ECS) provides a habitable environment for two astronauts for a maximum of 48 hours while the LM is separated from the CSM, and permits the astronauts to decompress and repressurize the cabin up to four times. It also controls the temperature of electronic equipment and provides water for drinking, cooling, fire extinguishing, and food preparation.

The major portion of the ECS is in the cabin. Part of the equipment-cooling loop and two oxygen tanks are in the aft equipment bay. Two water tanks are in the upper midsection. A third (larger) oxygen tank and a third (larger) water tank are in the descent stage. (See figures 2.6-1 and 2.6-2.) Power for the ECS is supplied from the Electrical Power Subsystem. (See figure 2.6-3.) The ECS (figure 2.6-4) comprises an atmosphere revitalization section (ARS), an oxygen supply and cabin pressure control section (OSPCPS), a water management section (WMS), a heat transport section (HTS), and provisions for supplying oxygen and water to the Portable Life Support System (PLSS).

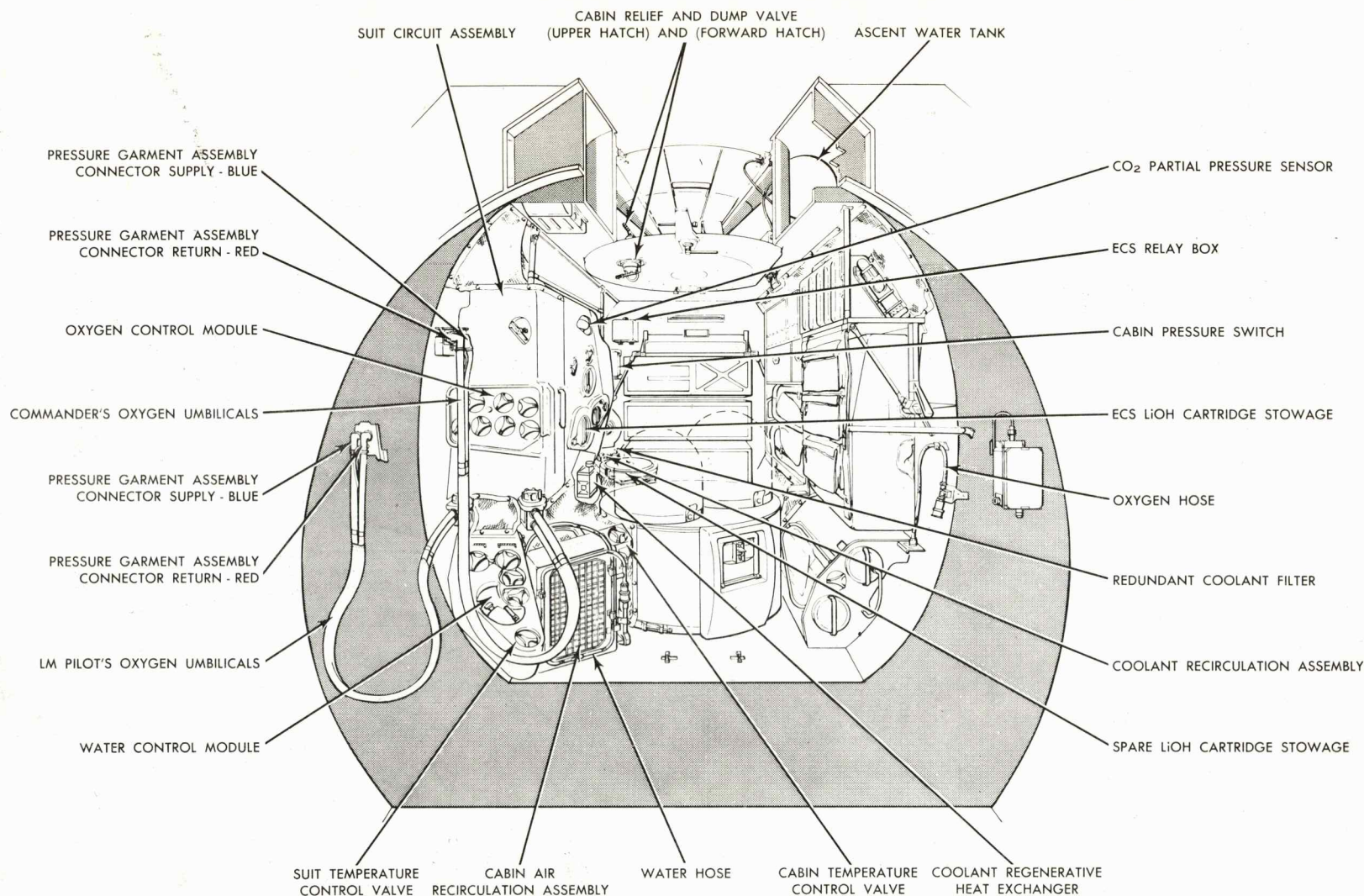
2.6.1.1 Atmosphere Revitalization Section.

The ARS consists of the suit circuit assembly, the cabin recirculation assembly, and steam flex vent. The suit circuit assembly is a closed-loop recirculation system that cools and ventilates the pressure garment assemblies (PGA's); maintains a desirable level of carbon dioxide in the atmosphere; removes odors, noxious gases, and excessive moisture; enables control of the temperature of oxygen flow to the PGA's and if required, automatically isolates the PGA's from the system. The cabin recirculation assembly circulates and controls the temperature of the cabin atmosphere. The steam flex duct carries steam discharged from the suit circuit water sublimator to ambient vacuum.



A-300LM4-67

Figure 2.6-1. Environmental Control Subsystem - Component Location (Sheet 1 of 3)



A-300LM4-66-1

Figure 2.6-1. Environmental Control Subsystem - Component Location (Sheet 2 of 3)

ENVIRONMENTAL CONTROL SUBSYSTEM

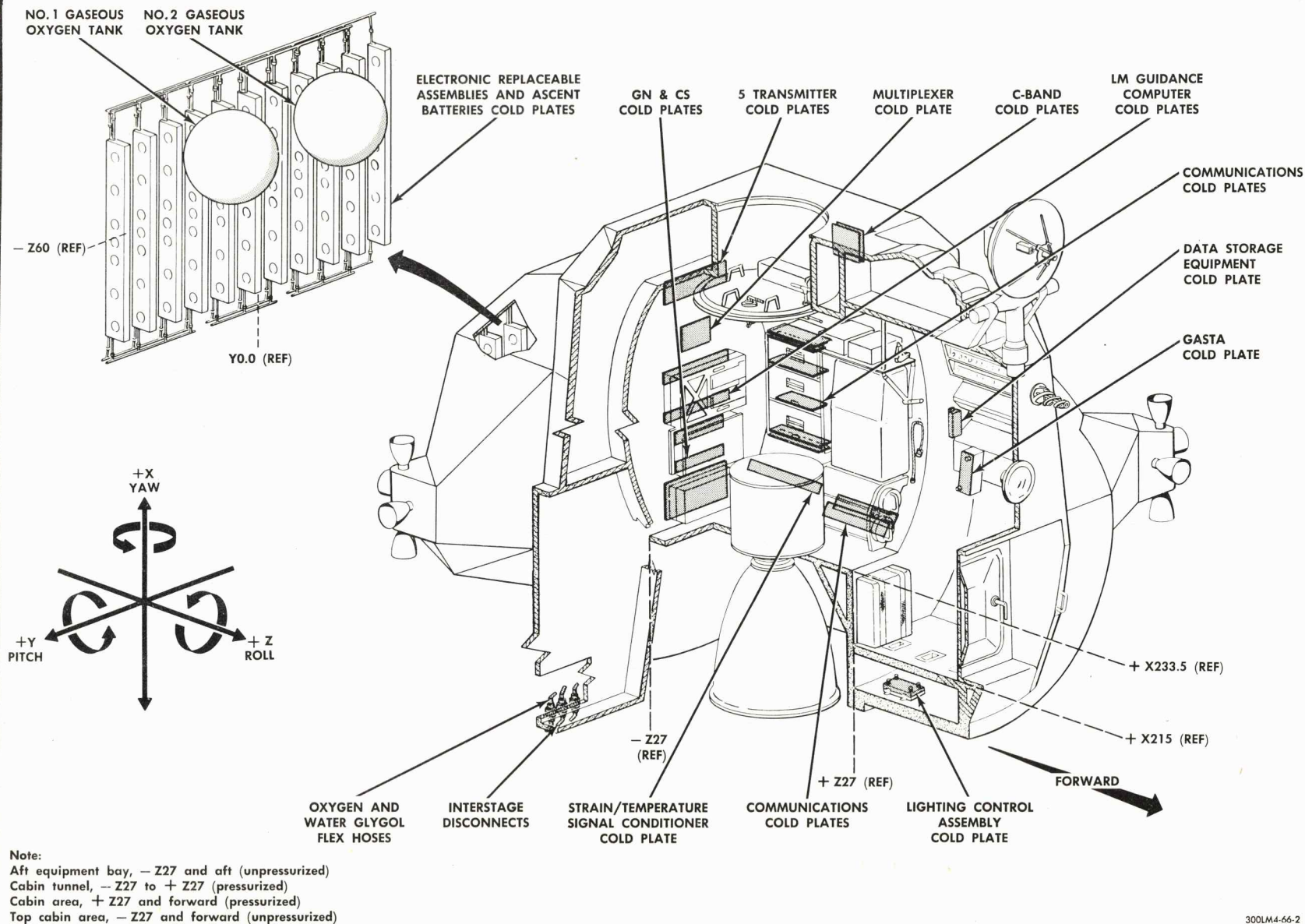
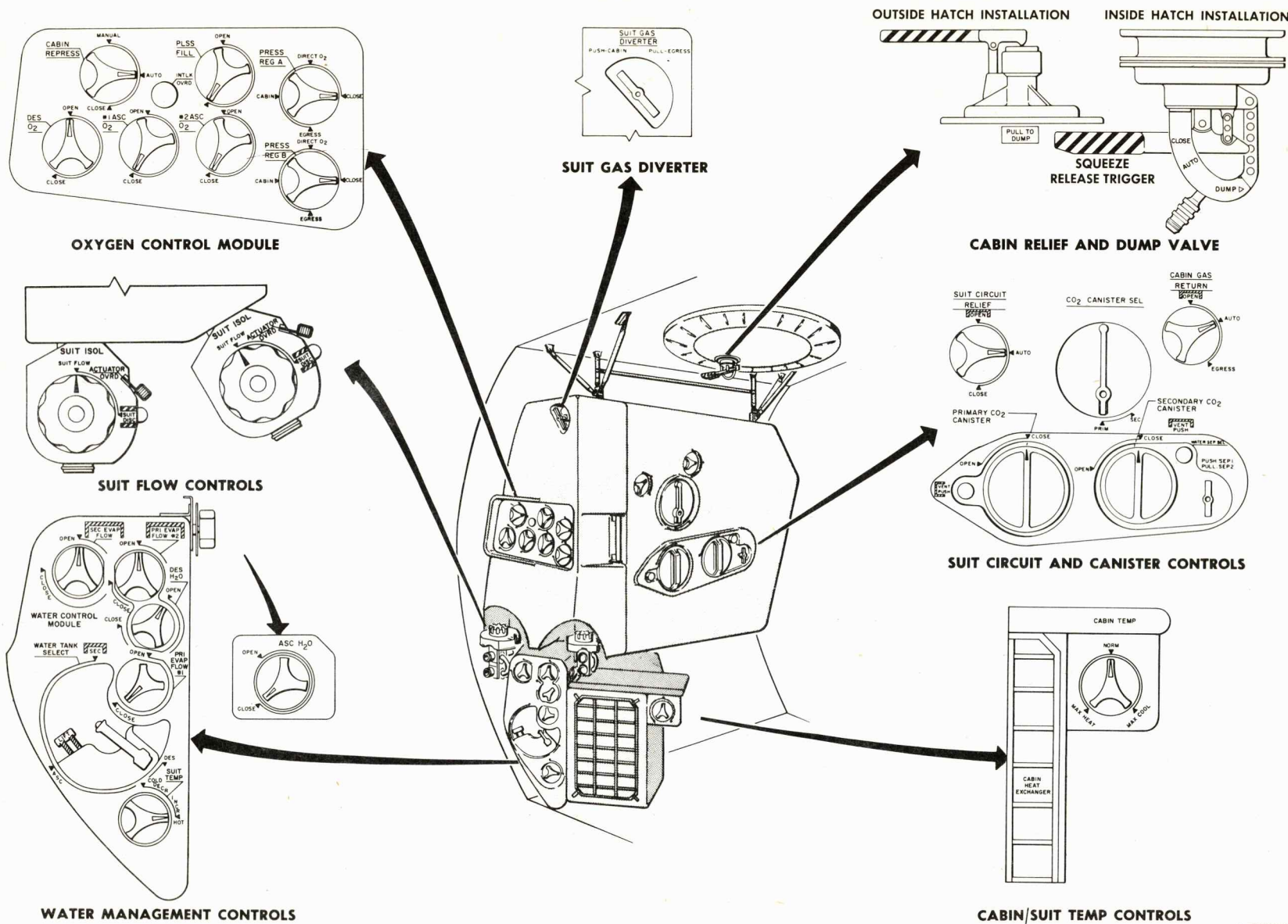


Figure 2.6-1. Environmental Control Subsystem - Component Location (Sheet 3 of 3)



A-300LM4-8

Figure 2.6-2. Environmental Control Subsystem - Aft Bulkhead Controls

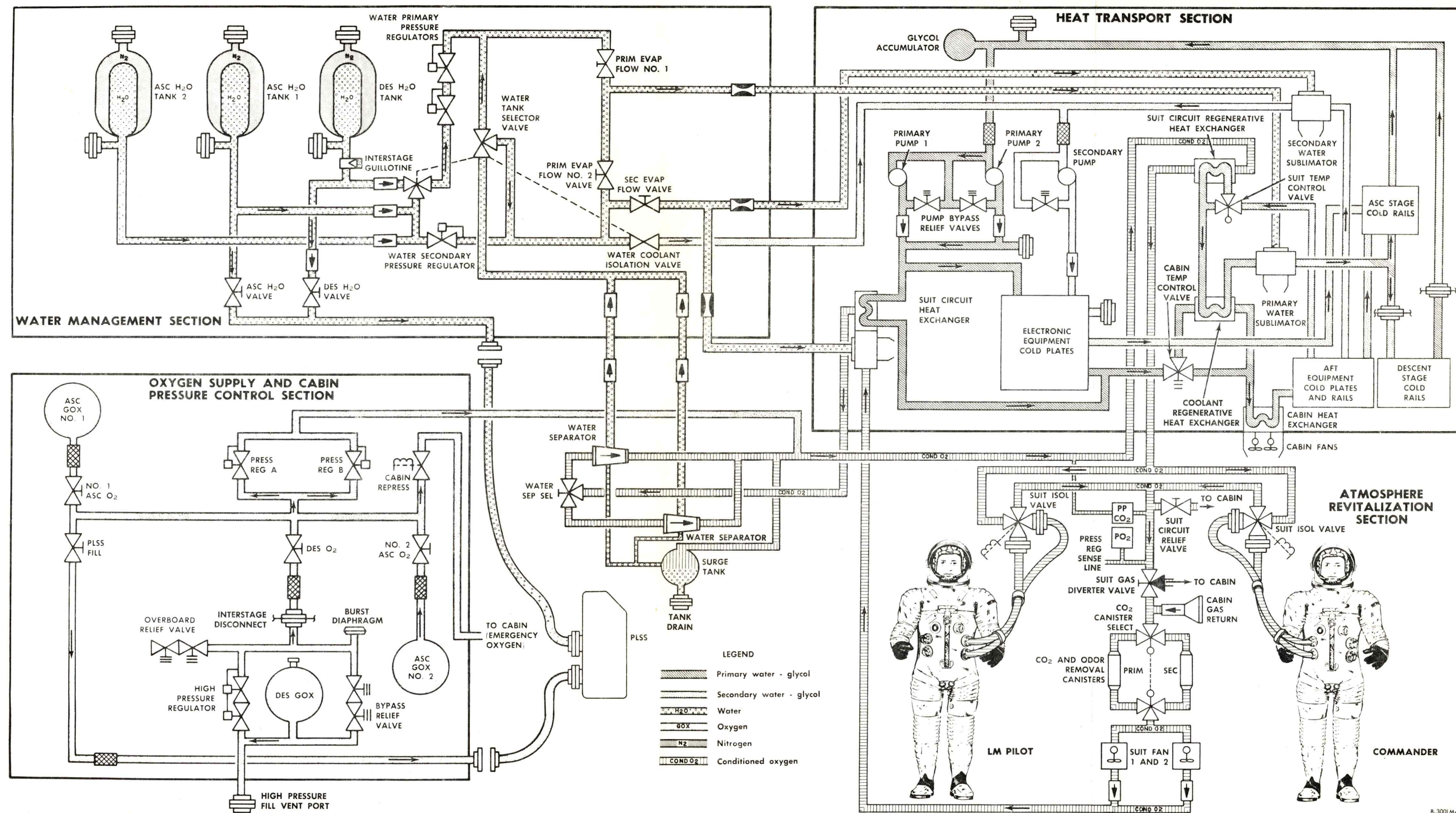


Figure 2.6-4. Environmental Control Subsystems - Functional Flow Diagram

Oxygen circulation is maintained by centrifugal fans. Heating and cooling is accomplished by passing the oxygen through heat exchangers, where heat is either surrendered to, or picked up from, the coolant of the HTS. Water separators are used to remove excess moisture after the cooling phase.

2.6.1.2 Oxygen Supply and Cabin Pressure Control Section.

The OSCPCS stores gaseous oxygen and maintains cabin and suit pressure by supplying oxygen to the ARS. This replenishes losses due to crew metabolic consumption and cabin or suit leakage. There are two oxygen supplies; one in the descent stage provides oxygen during the descent and lunar-stay phases of the mission, and another one, in the ascent stage, is used during the ascent and rendezvous phases of the mission.

2.6.1.3 Water Management Section.

The WMS supplies water for drinking, cooling, fire extinguishing, and food preparation; for refilling the PLSS cooling water tank; and for pressurization of the secondary coolant loop of the HTS. It also provides for delivery of water from ARS water separators to HTS sublimators.

ECS

The water tanks are pressurized before launch to maintain the required pumping pressure in the tanks. The descent stage tank supplies most of the water required until staging occurs. After staging, water is supplied by the ascent storage tanks. Water from ARS water separators is used in the sublimators. A self-sealing PLSS DRINK valve delivers water for drinking and food preparation.

2.6.1.4 Heat Transport Section.

The HTS consists of a primary coolant loop and a secondary coolant loop. The secondary loop serves as a backup loop and functions in the event the primary loop fails. A water-glycol solution circulates through each loop. The primary loop provides temperature control for batteries, electronic equipment that requires active thermal control, and for the oxygen that circulates through the cabin and PGA's. The batteries and electronic equipment are mounted on cold plates and rails through which coolant is routed to remove waste heat.

The cold plates used for equipment that is required for mission abort contain two separate coolant passages; one for the primary loop and one for the secondary loop. The secondary coolant loop, which is used only if the primary loop is inoperative, serves only these cold plates.

In-flight waste heat rejection from both coolant loops is achieved by the primary and secondary sublimators which are vented overboard. A coolant pump recirculation assembly contains all the HTS coolant pumps and associated check and relief valves. Coolant flow from the assembly is directed through parallel circuits to the cold plates for the electronic equipment and the oxygen-to-glycol heat exchanger in the ARS.

2.6.2 SUBSYSTEM INTERFACES.

An absolute pressure transducer in the descent oxygen tank feed line generates an output proportional to tank pressure. The output is routed to the IS, where it is conditioned to provide a telemetry signal, a caution indication, and, through the O₂/H₂O QTY MON selector switch (panel 2), a display on the O₂ QUANTITY indicator. When tank pressure drops below 135 psia (approximately 5% of capacity), a signal is routed to the O₂ QTY caution light. Momentarily setting the O₂/H₂O QTY MON selector switch to C/W RESET extinguishes the light.

A pressure transducer in the fill line of each ascent oxygen tank generates an output that is also conditioned in the IS to provide a telemetry signal, a caution indication, and a display on the O₂ QUANTITY indicator. The quantity of oxygen remaining in the tanks is read on the indicator by setting the selector switch to ASC 1 or ASC 2, as applicable. The O₂ QTY caution light goes on if, before staging, the pressure in either ascent oxygen tank is less than 684 psia (less-than-full condition). After staging, the signal that causes this indication is inhibited; instead, the light goes on when pressure in ascent tank No. 1 is less than 100 psia. The caution light is extinguished by setting the selector switch to C/W RESET.

ENVIRONMENTAL CONTROL SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.6-9

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.6.3 FUNCTIONAL DESCRIPTION.

The functional description of each of the four major ECS sections is supported by a functional flow diagram which, to reduce complexity, does not contain electrical circuitry. Figure 2.6-5 shows all ECS electrical circuits, and all ECS interfaces with the Electrical Power Subsystem (EPS) and Instrumentation Subsystem (IS).

2.6.3.1 Atmosphere Revitalization Section. (See figure 2.6-6.)

The ARS is a recirculation system that conditions oxygen from the OSCPS by cooling or heating, and dehumidifying and deodorizing it before use within the PGA's and cabin.

In normal operation the SUIT ISOL valve is set to SUIT FLOW and conditioned oxygen flows to the PGA's and to discharge through the return umbilical to the suit circuit. Suit circuit pressure, sensed at a point downstream of the suits, is referenced to the oxygen regulators that control pressure by supplying makeup oxygen to the suit circuit. The suit circuit relief valve protects the suit circuit against overpressurization, by venting to cabin ambient.

The CABIN position of the suit gas diverter valve is used during pressurized-cabin operation, to divert sufficient conditioned oxygen to the cabin to control carbon dioxide and humidity levels. Pulling the valve handle selects the EGRESS position, isolating the suit circuit from the cabin. The EGRESS position is used for all unpressurized cabin operations as well as closed-suit mode with pressurized cabin. An electrical solenoid override automatically repositions the valve from CABIN to EGRESS when cabin pressure drops below 4.15 ± 0.030 psia, or when EGRESS is selected on PRESS REG A or PRESS REG B.

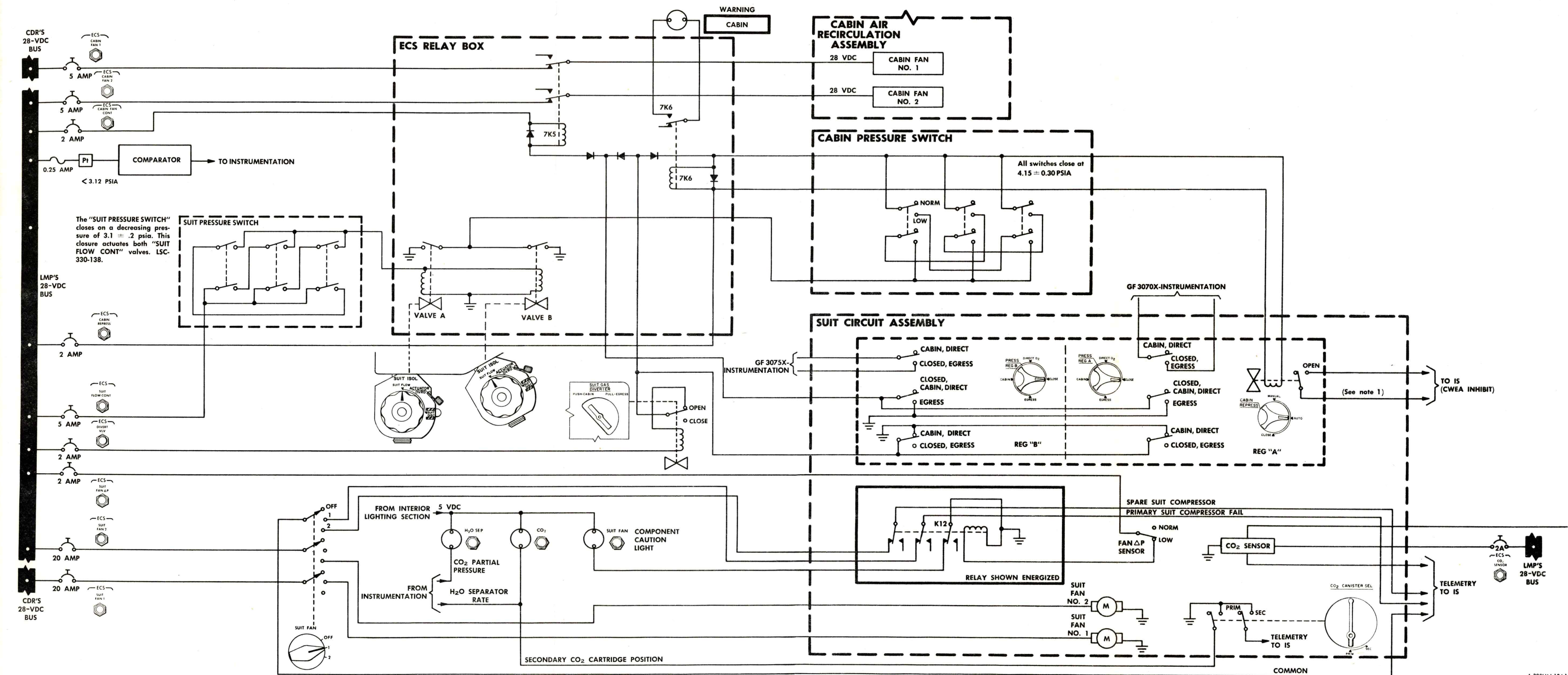
With the suit gas diverter valve set to CABIN, cabin discharge oxygen is returned to the recirculation system through the cabin gas return valve. Setting the CABIN GAS RETURN valve to AUTO enables cabin pressure to open the valve. When the cabin is depressurized, differential pressure closes the valve, preventing suit pressure loss.

A small amount of oxygen is tapped from the suit circuit upstream of the PGA inlets and fed to the carbon dioxide partial pressure sensor, which provides a voltage to the PART PRESS CO₂ indicator (panel 2).

The primary and secondary carbon dioxide (CO₂) and odor removal canisters are connected to form a parallel loop. The primary canister contains a LM-size cartridge; the secondary canister, a PLSS-size cartridge. A debris trap in the primary canister cover prevents particulate matter from entering the cartridge. A relief valve in the primary canister permits flow to bypass the debris trap if it becomes clogged. Oxygen is routed to the CO₂ and odor removal canisters through the canister selector valve. The carbon dioxide level is controlled by passing the flow across a bed of LiOH; odors are removed by absorption in activated charcoal. When carbon dioxide partial pressure reaches or exceeds 7.6 mm Hg nominal, as indicated on the PART PRESS CO₂ indicator, the CO₂ component caution light and ECS caution light go on. The CO₂ CANISTER SEL valve is then set to SEC, placing the secondary canister onstream. This unlocks the cover on the PRIM CO₂ CANISTER. The PRIM CO₂ CANISTER VENT push-button is pushed to release pressure from the canister before the canister cover is removed. The primary cartridge is replaced, the canister cover is installed, and the CO₂ CANISTER SEL valve is set to PRIM, placing the primary canister back onstream. The ECS caution lights is extinguished by the lowering of the carbon dioxide partial pressure below 7.6 mm Hg. The CO₂ component light is on whenever the CO₂ level is above the 7.6 mm Hg nominal, or the secondary canister is selected.

From the canisters, conditioned oxygen flows to the suit fan assembly, which maintains circulation in the suit circuit. Only one fan operates at a time. The CB ECS: SUIT FAN 1 is closed and the SUIT FAN selector switch is set to 1 to initiate suit fan operation. At startup, a fan differential pressure sensor is in the low position (low ΔP), which, through the fan condition signal control, energizes the ECS caution light and SUIT FAN component caution light. The lights remain on until the differential pressure across the operating fan increases sufficiently to cause the differential pressure sensor to move to the normal position. If the differential pressure drops to 6.0 inches of water or less,

ENVIRONMENTAL CONTROL SUBSYSTEM



A-300LM4-104-1



GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.6-12

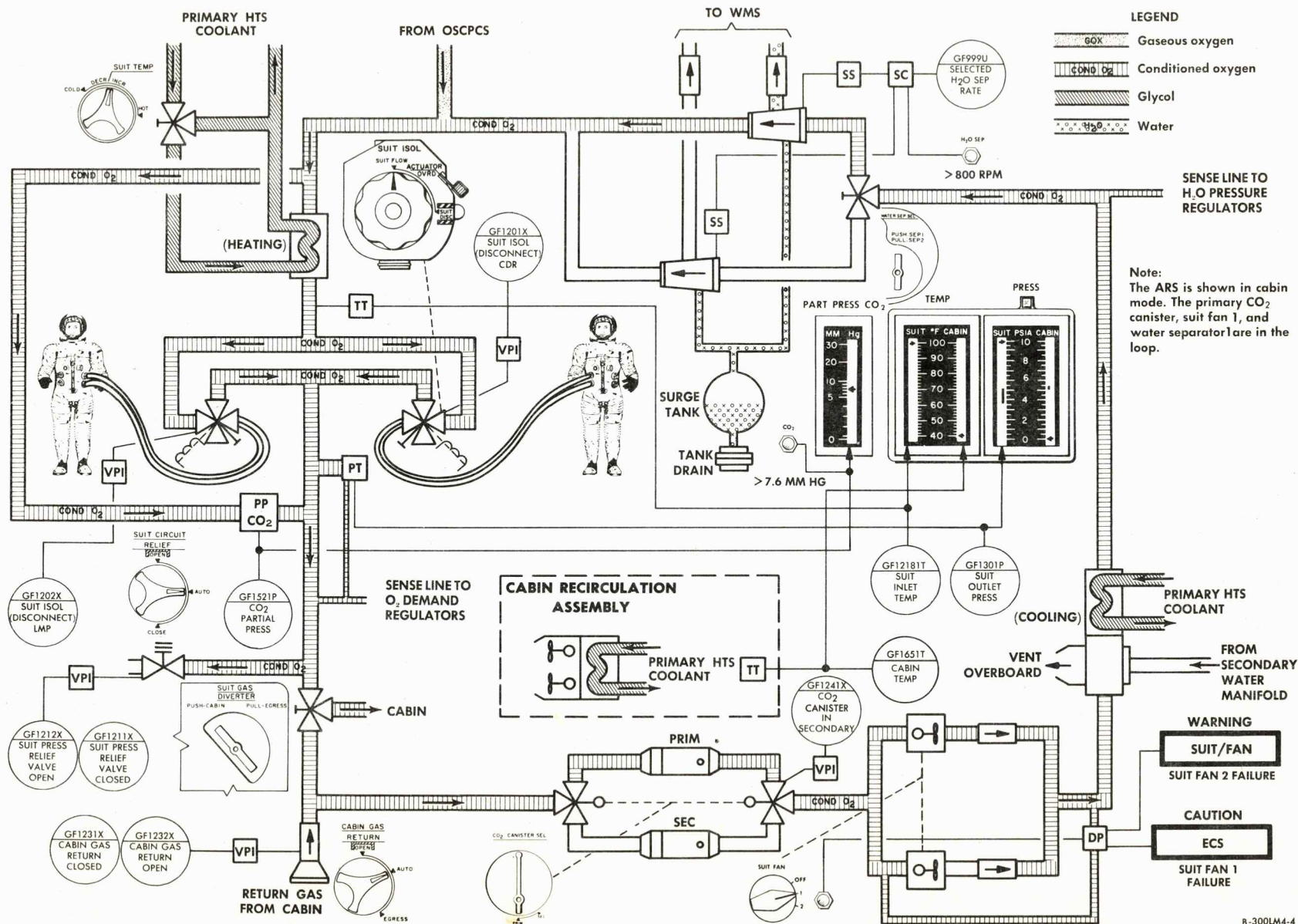


Figure 2.6-6. Atmosphere Revitalization Section - Simplified Functional Flow Diagram

the lights go on and switchover to fan No. 2 is required. The ECS caution light goes off upon selection of suit fan No. 2 and the suit fan component light goes off when fan No. 2 comes up to speed and builds up normal differential pressure. The SUIT/FAN warning light, ECS caution light, and SUIT FAN component caution light go on if fan No. 2 differential pressure does not reach 8.0 inches of water. The fan check valve permits air to pass from the operating fan without backflow through the inoperative fan.

From the check valve, the conditioned oxygen passes through a sublimator to the cooling heat exchanger. The sublimator cools the oxygen under emergency conditions. Normally, the sublimator is inoperative; it is placed in operation when the secondary evaporator flow (SEC EVAP FLOW) valve of the water control module is opened because a failure in the primary heat transport loop renders the cooling heat exchanger inoperative. Heat transfer to the coolant in the heat exchanger reduces gas temperature and causes some condensation of water vapor.

The conditioned oxygen is next routed to two, parallel-redundant water separators through the water separator selector (WATER SEP SEL) valve. One separator, selected by pushing (SEP 1) or pulling (SEP 2) the WATER SEP SEL valve handle, is operated at a time. The separator is driven by the gas flowing through it. Moisture removed from the oxygen is discharged under a dynamic head of pressure sufficient to ensure positive flow from the separator to the WMS. A water drain carries some water from the separators to a surge (collection) tank outside the recirculation system.

The conditioned oxygen from the water separator is mixed with makeup oxygen from the OSCPCS to maintain system pressure. The mixture flows through the regenerative (heating) heat exchanger, where the temperature is adjusted, to the suit isolation valves. The suit temperature control (SUIT TEMP) valve on the water control module controls the flow of coolant through the regenerative heat exchanger. Setting the valve to INCR HOT increases oxygen temperature; setting it to DECR COLD reduces the temperature.

If a PGA tears while the SUIT ISOL valves are set to SUIT FLOW, the valves are automatically repositioned to SUIT DISC when suit circuit pressure drops to $3.12 \pm .18$ psia while in a depressurized cabin. This action isolates the PGA's from the ARS. At the same time, the CABIN REPRESS valve automatically opens, repressurizing the cabin. The cabin repress valve is electrically deenergized when the cabin pressure switch opens at $4.65 \pm .25$ psia.

2.6.3.1.1 Cabin Air Recirculation Assembly.

During pressurized-cabin operation, the cabin air recirculation assembly cools or heats cabin oxygen by circulating the oxygen through the cabin heat exchanger. The assembly maintains cabin temperature at approximately $+75^{\circ}$ F.

Initially, the assembly fans are turned on by closing the ECS: CABIN FAN 1, CABIN FAN 2, and CABIN FAN CONT circuit breakers and setting the oxygen demand regulators (PRESS REG A and PRESS REG B valves) to CABIN. The fans are turned off by setting the oxygen demand regulators to EGRESS or by opening the circuit breakers. The fans operate independently, but simultaneously; each supplies one-half of the maximum required circulation. The fans pull the cabin oxygen directly into the cabin heat exchanger. Cabin temperature is controlled by the HTS cabin temperature control valve.

2.6.3.2 Oxygen Supply and Cabin Pressure Control Section. (See figure 2.6-7.)

The ECS descent stage oxygen supply hardware consists of the following: descent oxygen tank, high pressure fill coupling, high pressure oxygen control assembly, interstage flex line, descent stage disconnect. The descent tank pressure transducer is part of the IS and generates an output proportionate to tank pressure.

The ascent stage oxygen supply hardware consists of the following: ascent stage disconnect, interstage flex line, oxygen module, two ascent oxygen tanks, PLSS hose, PLSS oxygen disconnect (GFE), and the cabin pressure switch. Two automatic pressure relief and dump valves, one in each hatch, are provided to protect the cabin from overpressurization. Two ascent stage tank pressure transducers and a selected oxygen supply transducer are part of the IS and operate in conjunction with the OSCPS.

The OSCPCS stores gaseous oxygen, replenishes the ARS oxygen, and provides refills for the PLSS oxygen storage tank. Before staging, oxygen is supplied from the descent stage oxygen tank.

ENVIRONMENTAL CONTROL SUBSYSTEM

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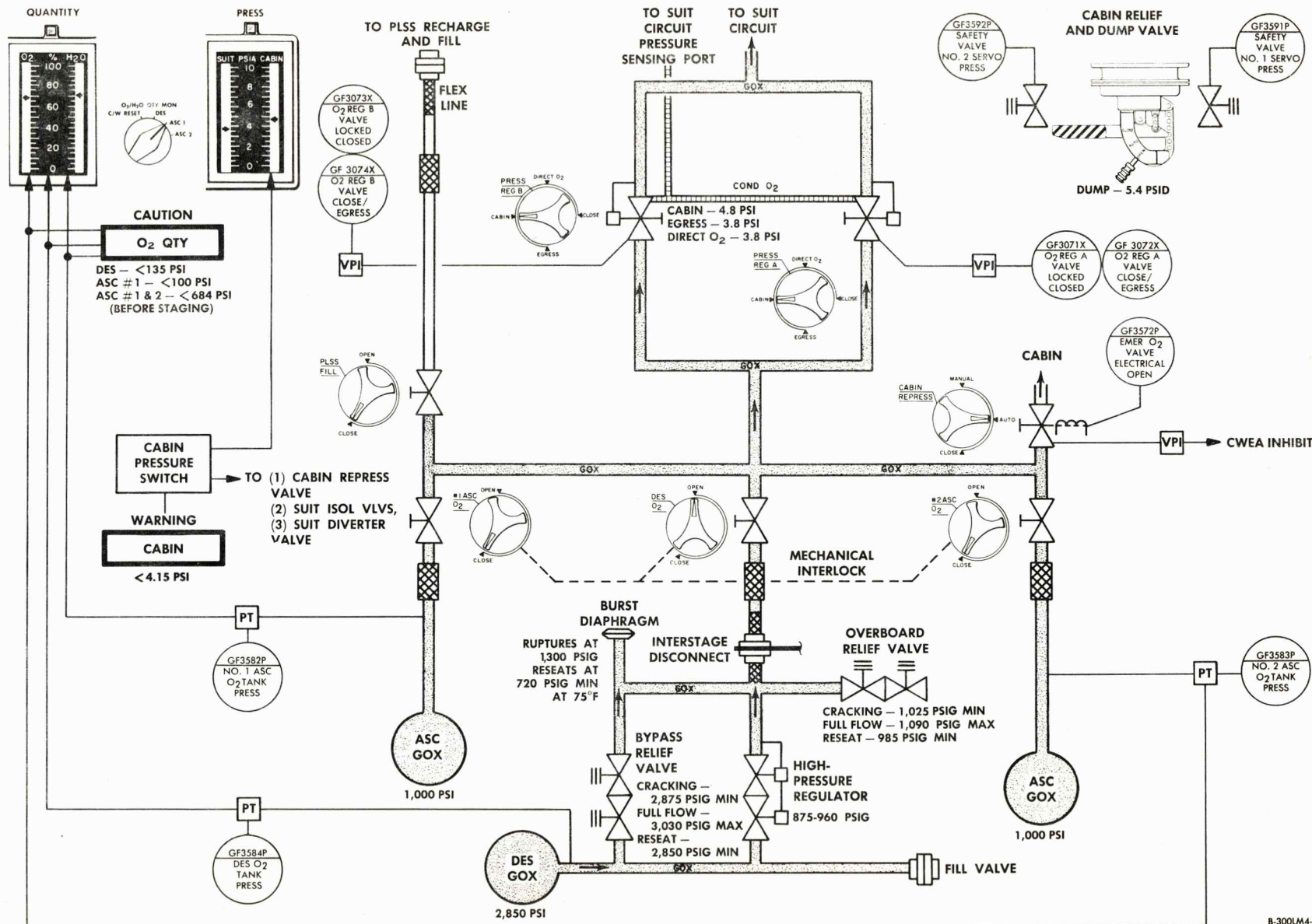


Figure 2.6-7. Oxygen Supply and Cabin Pressure Control Section - Simplified Functional Flow-Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

After staging, or if the descent tank is depleted, the ascent stage oxygen tanks supply oxygen to the oxygen control module. The high-pressure assembly in the descent stage, and the oxygen control module in the ascent stage, contain the valves and regulators necessary to control oxygen in the OSCPCS. Two cabin relief and dump valves vent excess cabin pressure.

The descent tank pressure, approximately 2,850 psig, is reduced to a level that is compatible with the components of the oxygen control module, approximately 900 psig, by a high-pressure regulator. A series-redundant bypass relief valve protects the descent oxygen tank against over-pressurization. In addition, a series-redundant overboard relief valve protects the oxygen control module against excessive pressure caused by a defective regulator or by flow through the bypass relief valve. If the pressure on the outlet side of the regulator rises to a dangerous level, the burst diaphragm assembly vents the high pressure assembly to ambient. A poppet in the burst diaphragm assembly reseats when tank pressure is reduced to approximately 1,000 psig. Descent oxygen flows through the interstage disconnect to the oxygen module and is controlled with the descent oxygen shutoff (DES O₂) valve. The interstage disconnect acts as a redundant seal to prevent loss of oxygen overboard after staging.

When ascent stage oxygen is required, the ascent oxygen shutoff (#1 and #2 ASC O₂) valves are used to select their respective tank. A mechanical interlock prevents the valves from being opened unless the descent oxygen shutoff valve is closed. The mechanical interlock may be overridden (if the descent oxygen shutoff valve cannot be closed and the ascent oxygen shutoff valves must be opened) by pressing the INTLK OVRD pushbutton on the oxygen control module.

From the oxygen shutoff valves, oxygen is routed to oxygen demand regulators (PRESS REG A and PRESS REG B), a PLSS FILL valve, and a cabin repressurization and emergency oxygen (CABIN REPRESS) valve. The PLSS FILL valve connects to the PLSS oxygen tank through a flexible service hose. A check valve in the PLSS disconnect is opened when the hose is connected. The valve automatically closes when the hose is disconnected. The oxygen demand regulators maintain the pressure to the suit circuit at a level consistent with normal requirements. Both regulators are manually controlled with four-position handles; both are ordinarily set to the same position. The CABIN position is selected during normal pressurized-cabin operations, to provide oxygen at 4.8 ± 0.2 psia. Setting the regulators to EGRESS maintains suit circuit pressure at 3.8 ± 0.2 psia. The DIRECT O₂ position provides an unregulated flow of oxygen into the suit circuit. The CLOSE position shuts off all flow through the regulator. In the CABIN and EGRESS positions, the regulator is internally modulated by a reference pressure from the suit circuit. The demand regulators are redundant; either one can fulfill the ARS oxygen requirements.

If both demand regulators are set to CABIN or DIRECT O₂ and cabin pressure drops to 4.15 ± 0.30 psia, the cabin pressure switch energizes the cabin repressurization valve, and oxygen flows through the valve into the cabin. If cabin pressure builds up to 4.65 ± 0.25 psia, the cabin pressure switch deenergizes the valve solenoid, shutting off the oxygen flow. The valve can maintain cabin pressure at 3.5 psia for at least 2 minutes following a 0.5-inch-diameter puncture of the cabin. It responds to signals from the cabin pressure switch during pressurized-cabin operation and to a suit pressure sensor during unpressurized operation. Manual override capabilities are provided.

The two cabin relief and dump valves are manually and pneumatically operated. They prevent excessive cabin pressure and permit deliberate cabin decompression. The valves automatically relieve cabin pressure when the cabin-to-ambient differential reaches -5.8 psia. When set to AUTO, the valves can be opened manually with their external handle. The valve in the overhead hatch can dump cabin pressure from 5.0 to 0.08 psia in 180 seconds without cabin inflow. The valve in the forward hatch requires 310 seconds to dump the same amount of cabin pressure because of the flow restriction caused by the bacteria filter. In addition to relieving positive pressure, the valves relieve a negative cabin pressure condition.

To egress from the LM, the oxygen demand regulators are set to EGRESS, turning off the cabin fans and closing the SUIT GAS DIVERter valve; the CABIN GAS RETURN valve is set to EGRESS; and cabin pressure is dumped by setting the cabin relief and dump valve to OPEN. A bacteria filter on the cabin side of the cabin relief and dump valve in the forward hatch removes 95% of all bacteria larger than 0.5 micron from the dumped oxygen. When repressurizing the cabin, the cabin relief and dump valve is set to AUTO, the CABIN REPRESS valve is set to AUTO, the PRESS REG A and PRESS REG B

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

valves are set to CABIN, and the CABIN GAS RETURN valve is set to AUTO. The CABIN warning light goes on when the regulators are set to cabin, and goes off when the cabin reaches the actuation pressure of the cabin pressure switch.

2.6.3.3 Water Management Section. (See figure 2-6-8.)

The WMS stores water for metabolic consumption, evaporative cooling, and PLSS water tank refill. It controls the distribution of this stored water and the water reclaimed from the ARS by the water separators. Reclaimed water is used only for evaporative cooling, in the ECS sublimators. Water is stored in a tank in the descent stage and two identical smaller tanks in the upper midsection of the ascent stage. All three tanks are bladder-type vessels, which are pressurized with nitrogen before launch. The controls for the WMS are grouped together on the water control module located to the right rear of the LM Pilot's station.

Water from the descent stage water tank is fed through check valves to the water control module and, through the descent water (DES H₂O) valve, to the PLSS H₂O DISCONNECT. Water quantity is determined by a water quantity measuring device, which senses the temperature/pressure ratio of the tank-pressurizing nitrogen, computes the volume of the water, and generates an output analogous to that volume. The output is displayed on the H₂O QUANTITY indicator (panel 2) after the O₂/H₂O QTY MON selector switch is set to DES. If the quantity of water in the tank drops to $16\% \pm 3\%$, the WATER QTY caution light goes on. When the DES H₂O valve is set to OPEN, high-pressure water is available for drinking, food preparation, PLSS fill, and fire extinguishing.

ECS

When the vehicle is staged, the descent interstage water feed line is severed by the inter-stage umbilical guillotine, and water is supplied from the ascent stage water tanks. Water quantity in each ascent stage tank is measured and displayed in the same manner as descent water quantity, the O₂/H₂O QTY MON switch is set to ASC 1 or ASC 2, as required. The WATER QTY caution light goes on when a less-than-full condition (less than 95%) exists in either tank or when the tank water levels are unequal (greater than 15%). Water from ascent stage water tank No. 1 is fed to the PLSS H₂O DISCONNECT through the ascent water (ASC H₂O) valve for drinking, food preparation, PLSS fill, and fire extinguishing.

Water from the three water tanks enters the water tank selector (WATER TANK SELECT) valve which consists of two water diverting spools. Setting the valve to DES or ASC determines which tank is on-line.

When using the descent tank, water is supplied to the primary manifold (which consists of the primary pressure regulators and the primary evaporator flow No. 2 valve) by setting the WATER TANK SELECT valve to DES. The water flows through the series primary pressure regulators, which control water discharge pressure in response to suit circuit gas reference pressure, at 0.5 to 1.0 psi above this gas pressure. With the PRIM EVAP FLOW valve set to OPEN, the water is routed to the primary sublimator. Discharge water from the water separator is routed through the secondary spool of the selector valve and joins the water from the primary pressure regulators. Setting the selector valve to ASC routes water from the ascent tanks through the primary pressure regulators and, with PRIM EVAP FLOW No. 1 valve set to OPEN, to the primary sublimator. Water flow from the water separators is not changed by selection of the ASC position. If the primary pressure regulators fail, an alternative path to the primary sublimator is provided by setting the primary evaporator flow No. 2 (PRIM EVAP FLOW #2) valve to OPEN. Water then flows directly from the ascent water tanks through the secondary pressure regulator, and the primary evaporator flow No. 2 valve to the primary sublimator.

Under emergency conditions (failure of the primary HTS loop), water from the ascent tanks is directed through the secondary manifold (secondary pressure regulator) to the secondary sublimator and the suit circuit sublimator by setting the SEC EVAP FLOW valve to OPEN. Discharge water from the water separators is also directed to the sublimator. Water is diverted to the secondary HTS loop to provide a head of pressure for its pump. To divert the water, the finger latch on the WATER TANK SELECT valve is lifted and the valve is set to SEC. This action punctures the coolant isolation valve and starts the water flow to the secondary HTS loop.

ENVIRONMENTAL CONTROL SUBSYSTEM

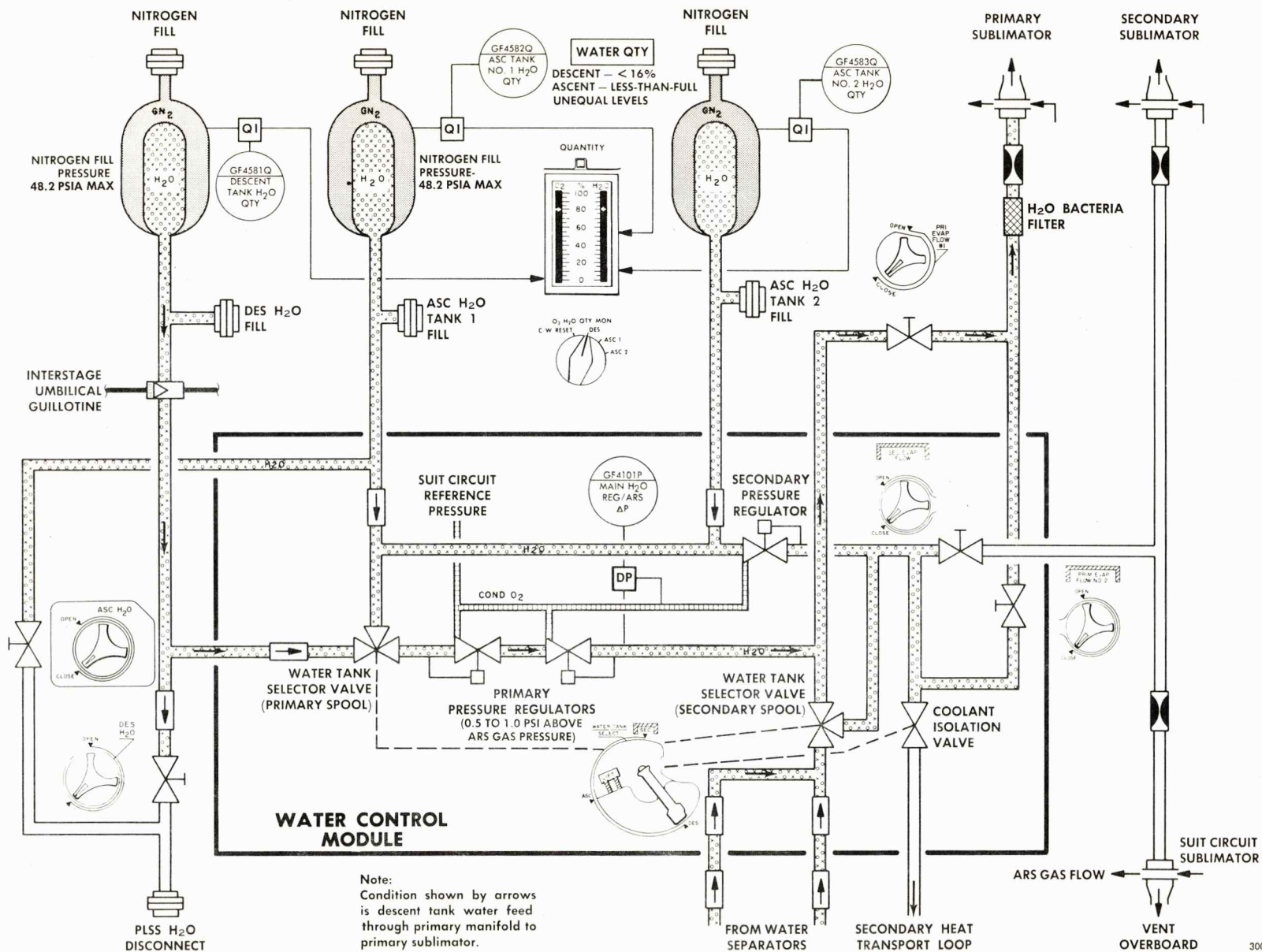


Figure 2.6-8. Water Management Section - Simplified Functional Flow Diagram

2.6.3.4 Heat Transport Section. (See figure 2.6-9.)

The HTS consists of two closed loops (primary and secondary) through which a water-glycol solution is circulated to cool the suit circuits, cabin atmosphere, and electronic equipment. Coolant is continuously circulated through cold plates and cold rails to remove heat from electronic equipment and batteries. For the purpose of clarity, the primary and secondary coolant loops, and the primary and secondary coolant loop cold plates and rails are discussed separately in the following paragraphs. When necessary, the primary loop is also a heat source for the cabin atmosphere. Heat is removed by absorption and is radiated to space by sublimation.

2.6.3.4.1 Primary Coolant Loop.

The primary coolant loop is charged with coolant at GSE fill points and is then sealed. The glycol pumps force the coolant through the loop. The glycol accumulator maintains a constant head of pressure (5.5 to 8 psi, depending on coolant level) at the inlets of the primary loop glycol pumps to prevent cavitation. Coolant temperature at the inlets is approximately +40° F. The accumulator also replaces coolant lost through subsystem leakage. A switch in a low-level sensor trips when only 10±5% of coolant volume remains in the accumulator. When tripped, the switch provides a telemetry signal and causes the GLYCOL caution light (panel 2) to go on.

The coolant is routed to the pumps through a filter which has a bypass to maintain flow if the element becomes clogged. Each pump increases coolant pressure by 30 psia. They are started by closing the ECS: GLYCOL PUMP 1 and 2 circuit breakers and setting the GLYCOL selector switch to PUMP 1 or PUMP 2. If the operating pump does not develop a minimum differential pressure (ΔP) of 3.0 psi, the ΔP switch generates a signal to energize the ECS caution light and the GLYCOL PUMP component caution light. Selecting the other pump deenergizes the lights when the onstream pump develops a ΔP of 7±2 psid. If both pumps fail, the secondary loop is activated by setting the WATER TANK SELECT valve to SEC, setting the GLYCOL pump switch to INST (SEC), and closing the ECS: GLYCOL PUMP SEC circuit breaker. Automatic transfer of primary pump operation is initiated by closing the ECS: GLYCOL AUTO TRNFR circuit breaker and setting the selector switch to PUMP 1. When transfer is necessary, the caution lights go on, the transfer is accomplished, and the ECS caution light goes off, but the GLYCOL PUMP component light remains on.

If primary loop ΔP exceeds 33 psi, the pump bypass relief valve opens and routes the coolant back to the pump inlet, relieving the pressure. The valves start to open at 33 psi, are fully open at a maximum of 39 psia, and reseal at a minimum of 32 psia. Check valves prevent coolant from feeding back through an inoperative pump.

Part of the coolant leaving the recirculation assembly flows to the suit circuit heat exchanger to cool some of the suit circuit gas of the ARS. The remainder of the coolant flows to electronic equipment within the cabin. The flow paths then converge and the coolant is directed to the cabin temperature control valve.

The cabin temperature control valve varies the flow rate to the coolant regenerative heat exchanger to regulate and maintain the temperature of the coolant entering the cabin heat exchanger. Setting the valve handle to NORM controls the percentage of coolant diverted to the coolant regenerative heat exchanger and to the cabin heat exchanger. Greater flow through the coolant regenerative heat exchanger causes a higher temperature in the cabin heat exchanger. This is caused by that portion of the coolant (flowing through the coolant regenerative heat exchanger) that is heated by the warmer coolant returning from the aft equipment bay cold plates. The valve is adjusted by an element which senses the coolant temperature downstream of the cabin heat exchanger. The operating characteristics (sensed temperature-flow percentage relation) are adjusted by selecting a position within the NORM range. Setting the valve to MAX COOL routes all the coolant directly to the cabin heat exchanger, lowering its temperature. Setting the valve to MAX HEAT routes all the coolant to the coolant regenerative heat exchanger, where the coolant picks up heat from the warmer coolant returning from the aft equipment bay cold plates. The warmer coolant flow can be diverted to the suit circuit regenerative heat exchanger through the suit temperature (SUIT TEMP) control valve to increase ARS gas temperature. The valve varies the coolant flow to the heat exchanger from 0 to 2 pounds per minute. All or part of the warm coolant may be diverted to bypass the heat exchanger. The flow from the heat exchanger, combined with the diverted coolant, passes through the warm side of the coolant regenerative heat exchanger.

The coolant from the warm side of the coolant regenerative heat exchanger is routed to the primary sublimator. The sublimator rejects the heat to space through a steam duct. Deflector plates, attached to the duct, prevent escaping steam from applying thrust to the vehicle. Water is fed to the sublimator at a pressure that exceeds 4.0 psia but is less than 8.0 psia. The water pressure must be less than the suit circuit static pressure plus the head pressure from the water separators to the sublimator. The water regulators, referenced to suit circuit pressure, are in the water feed line to the sublimator. Regulated water pressure varies from 0.5 to 1.0 psid above suit circuit pressure. The sublimator inlet temperature is sensed by a temperature transducer, which provides a telemetry signal. Coolant from the sublimator flows through the ascent and descent battery cold rails, then returns to the recirculation assembly.

Two, self-sealing disconnects upstream and downstream of the glycol pumps permit servicing and operation of the HTS by GSE. Interstage disconnects are installed in coolant lines that connect to the descent stage. Before staging, coolant flows through the ascent and descent stage battery cold rails. After staging, the interstage disconnects separate, the lines are sealed by spring-loaded check valves, and the full coolant flow enters the ascent stage battery cold rails.

2.6.3.4.2 Secondary Coolant Loop.

The secondary redundant coolant loop provides thermal control for those electronic assemblies and batteries whose performance is necessary to effect a safe return to the CSM. Cooling is provided by the secondary sublimator.

When the secondary loop is activated, water is routed through the water isolation valve and a filter to the secondary glycol pump, enabling pressurization of the secondary loop. A pump by-pass relief valve relieves excessive pressure by routing coolant back to the pump inlet. A check valve at the discharge side of the glycol pump prevents coolant flow from bypassing the HTS during GSE operation. The coolant from the pump passes through the check valve, to the secondary passages of the cold plates and cold rails of the electronics and batteries cold plate section. Waste heat is absorbed by the coolant. The warm coolant then flows to the secondary sublimator.

The secondary sublimator contains porous plates which reject heat to space. It operates in the same manner as the primary sublimator in the primary coolant loop. Water for the sublimator is provided when the SEC EVAP FLOW valve is set to OPEN. The coolant returns to the pump for recirculation.

2.6.3.4.3 Cold Plates and Rails. (See figure 2.6-9.)

Equipment essential for mission abort is mounted on cold plates and rails that have two independent coolant passages, one for the primary loop and one for the secondary loop.

Primary Coolant Loop Cold Plates and Rails.

The cold plates and rails in the primary coolant loop are arranged in three groups: upstream electronics, aft equipment bay, and batteries.

Coolant from the recirculation assembly flows into parallel paths that serve the upstream electronics cold plate group. This group, the data storage equipment (DSEA) is cooled by cold rails; the remainder of the group, by cold plates. The cold plates are located in the pressurized and unpressurized areas of the LM. The flow rates through the parallel paths are controlled by flow restrictors, installed downstream of the power and servo assembly (PSA) cold plate group. The first upstream electronics flow path cools the suit circuit heat exchanger. The second flow path cools five cold plates mounted on the pressurized side of the equipment tunnel back wall. The third path serves the integrally cooled IMU and the rate gyro assembly (RGA) cold plate, both located in the unpressurized area (on the navigation base). The fourth path cools the abort sensor assembly (ASA) and pulse torque assembly (PTA) cold plates. All the plates for the fourth path are in the unpressurized area above the cabin; the ASA is on the navigation base (AOT). The fifth path serves the tracking light electronics (TLE), gimbal angle sequencing transformation assembly (GASTA), lighting control assembly (LCA), and DSEA plates: one in the unpressurized area in front of the cabin, a second one in the control and display panel area, a third one below the cabin floor, and another one on the left wall of the cabin.

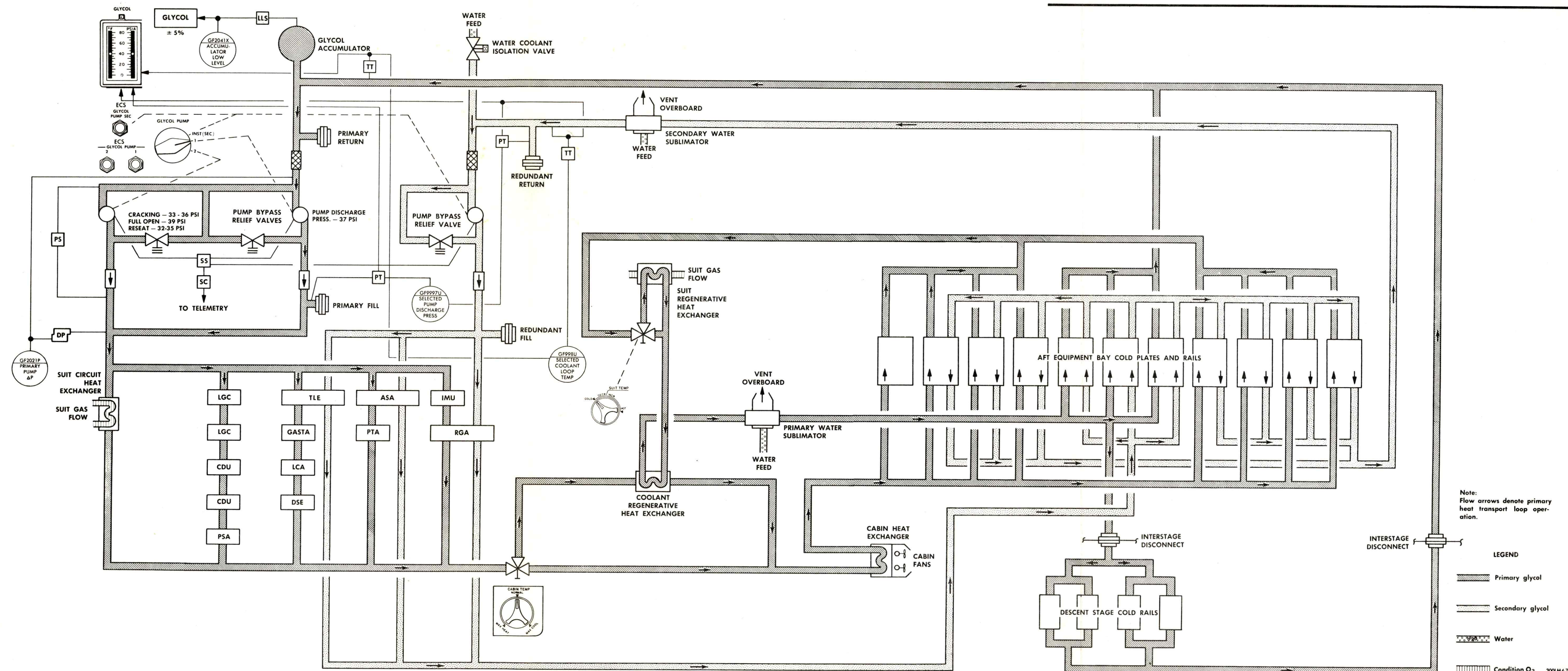


Figure 2.6-9. Heat Transport Section - Simplified Functional Flow Diagram

The aft equipment bay is cooled by eight cold rails; the flow is in parallel. The batteries are cooled by cold rails. The ascent batteries are in the center section of the aft equipment bay; the descent batteries are in the forward right area of the descent stage. During the descent phase, the coolant flow is split between the descent batteries and the ascent batteries; the ascent batteries are not used during this time. When the stages are separated, quick-disconnects break the coolant lines and seal the ends; all coolant then flows through the ascent battery cold rails.

Secondary Coolant Loop Cold Plates and Rails.

The secondary coolant loop is for emergency use. Only cold plates that have two independent passages (one for the primary loop and one for the secondary loop) are served by this loop.

In the upstream electronics area, the secondary coolant flow is split between three cold plates (RGA, ASA, and TLE) in parallel. The flow rate is controlled by a flow restrictor downstream of the TLE. After these three plates, the secondary loop cools the aft equipment bay cold rails in a series-parallel arrangement. The coolant first flows through three ascent battery cold rails in parallel, then through seven aft equipment bay cold rails in parallel.

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2.6.4 MAJOR COMPONENT/FUNCTIONAL DESCRIPTION.

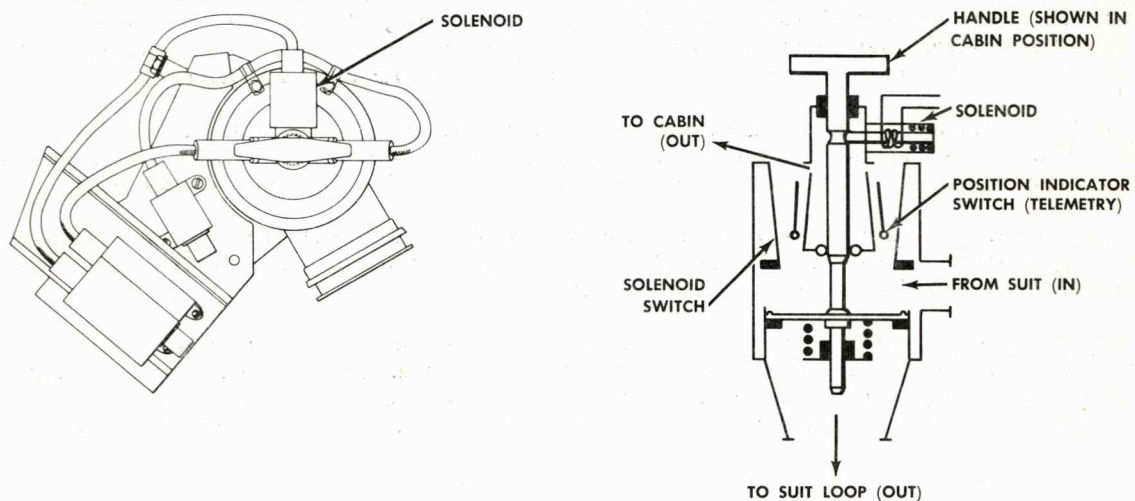
2.6.4.1 Atmosphere Revitalization Section.

2.6.4.1.1 Suit Circuit Assembly.

Suit Circuit Relief Valve. The SUIT CIRCUIT RELIEF valve is an aneroid-operated, poppet-type valve that protects the suit circuit against overpressurization. The valve has AUTO, OPEN, and CLOSE positions. Two externally mounted microswitches provide telemetry signals when the OPEN or CLOSE position is selected.

In the AUTO position, the valve responds to pressure sensed by the aneroid, and cracks open at approximately 4.3 psia to prevent overpressurization of the suit loop by allowing oxygen flow to the cabin. At 4.7 psia, the valve is fully open and flows approximately 7.8 pph at +90° F. The valve reseats at less than 4.3 psia. In the OPEN position, the valve handle displaces the poppet from the seat to open the valve, regardless of pressure. In the CLOSE position, selected if the valve fails to reset, the auto poppet is left open, but an auxiliary poppet is closed, maintaining pressure.

Suit Gas Diverter Valve. (See figure 2.6-10.) The SUIT GAS DIVERTER valve is a manually operated, two-way valve (one inlet and two outlets) with a solenoid override in one direction. The valve is on the ECS package above the oxygen control module. When the valve handle is pushed into CABIN position, oxygen is directed into the cabin. The valve is maintained in this position, against main valve spring



300LM4-93

Figure 2.6-10. Suit Gas Diverter Valve - Functional Schematic

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

pressure, by a deenergized solenoid. Pulling the valve handle to EGRESS position overrides the solenoid and permits the valve to reposition to its upper seat, blocking flow to the cabin.

- An automatic closed-to-cabin feature is provided. If cabin pressure falls below normal while the valve is set to CABIN, the solenoid is energized by the cabin pressure switch, and the main spring returns the valve to the EGRESS position. Electrical power is also supplied to the diverter valve when the PRESS REG A and PRESS REG B valves are set to EGRESS. The diverter valve has a valve-position-indicating switch that provides a telemetry signal when the valve is in the EGRESS position.

Cabin Gas Return Check Valve. The CABIN GAS RETURN check valve is a spring-loaded, flapper-type valve. The valve has AUTO, OPEN, and EGRESS (closed) positions. In the AUTO position, the valve automatically permits cabin gas to return to the suit circuit. When the cabin is depressurized, suit circuit pressure closes the valve, preventing backflow into the cabin. The OPEN and EGRESS positions provide manual override of the AUTO position. Valve-position-indicating switches provide telemetry signals when the valve is set to OPEN or EGRESS.

CO₂ Canister Selector Valve and CO₂ and Odor Removal Canisters. The CO₂ canister selector (CO₂ CANISTER SEL) valve is a dual-flapper-type valve that routes flow through the CO₂ and odor removal canisters. The valve has PRIM and SEC positions. The flapper assembly is connected to a control handle attached to a cable with an interlock, to preclude opening the canisters during use. One flapper is at the inlet to the canisters; the other, at the outlet.

Each canister contains a cartridge filled with lithium hydroxide (LiOH) and activated charcoal. The primary canister cover has a debris trap, which may be replaced before, but not during, flight. A relief valve in the cartridge permits flow to bypass the debris trap if it becomes clogged. The canister selector valve is sufficiently leakproof to permit replacement of cartridges, with the cabin unpressurized.

- The PRIM position permits suit circuit gas to flow through the primary canister. When replacing a saturated primary cartridge, the CO₂ CANISTER SEL handle is set to SEC. This unlocks the PRIMARY CO₂ CANISTER cover. A relief valve (VENT PUSH) equalizes the pressure of the cabin and the interior of the canister before canister cover removal. A finger latch, part of the canister cover, is released before the cover is rotated to OPEN.

Suit Circuit Fans. Each suit circuit fan is operated by a 28-volt d-c brushless motor. A full radial-type rotor with a vaneless diffuser is used; it moves approximately 24 cfm at 25,000 ± 500 rpm. A speed sensor, integral with the fan motor, senses motor rotation and develops an input signal to a frequency-to-voltage signal conditioner. Fan operation is controlled by the SUIT FAN selector switch. The fans permit operation at sea level for checkout purposes. An input voltage of 15 vdc is provided for this purpose.

Suit Circuit Sublimator. The suit circuit sublimator has a water inlet and a steam outlet that is vented overboard. Water and suit circuit gas both make a single pass through the unit, which is comprised of a stack of modules of several layers of porous plates, water, steam and suit circuit gas passages. The sublimator rejects suit circuit heat to space if the suit circuit heat exchanger is inoperative.

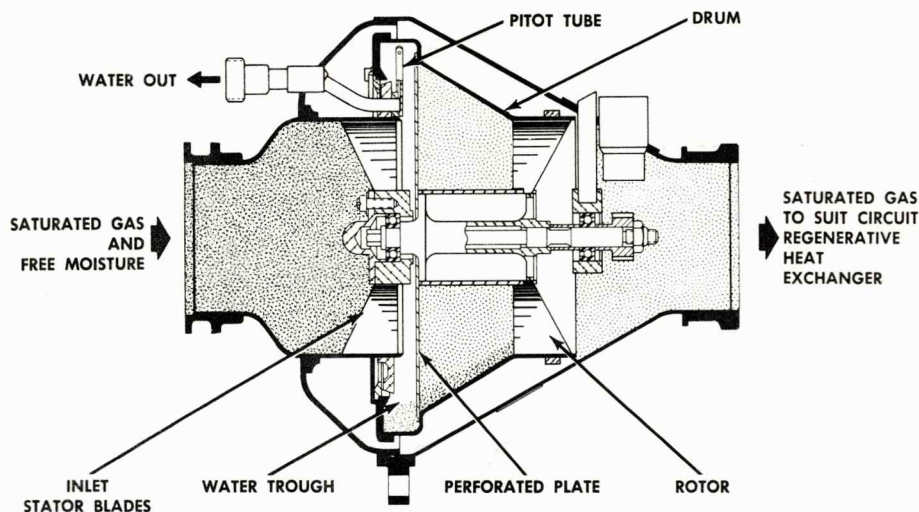
Suit Circuit Heat Exchanger. The suit circuit heat exchanger is a duct-shaped unit of aluminum plate-and-fin construction. It has a single pass for both the coolant and the suit circuit gas and is used to remove excess heat in the ARS. Heat is transferred to the HTS coolant supplied to the heat exchanger.

- Water Separator Selector Valve. The water separator selector (WATER SEP SEL) valve is a manually operated, flapper-type valve that enables selection of either of two water separators. The valve handle is pushed in for separator No. 1 and pulled for separator No. 2.

Water Separators. (See figure 2.6-11.) Two water separators are connected in parallel, but only one is used at a time. Saturated gas and free moisture fed into the separator come in contact with the inlet stator blades, which direct the flow onto a rotor at the proper angle. Most of the entrained moisture collects on the rotating perforated plate, which centrifuges the water into a rotating water trough. A stationary pitot

ENVIRONMENTAL CONTROL SUBSYSTEM

Page 2.6-24 Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968



300LM4-94

ECS

Figure 2.6-11. Water Separator - Functional Schematic

tube, placed tangentially to the rotating trough, picks up the removed water and discharges it to the WMS. The pitot pumping action creates a dynamic head of pressure sufficient to ensure positive flow from the water separator to the WMS. Water passing through the perforated plate collects on a rotating conical drum and is fed into the water trough. The oxygen flow drives the rotating parts of the separator.

Each water separator mounts an angular velocity transducer, which is a self-generating magnetic pickup that senses the rotation of the separator shaft. The pickup develops a signal whose frequency is converted to a voltage level with respect to separator rpm, in a signal conditioner unit. A water drain boss on each separator drains the cavity between the rotating drum and the outer shell of the unit. Plumbing attached to each drain boss carries water away from this area, and dumps it into a surge (collection) tank outside the suit circuit.

Suit Circuit Regenerative Heat Exchanger. The suit circuit regenerative heat exchanger is of the aluminum plate-and-fin type. Heat from the circulating warm coolant is transferred to the oxygen which makes a single pass through the unit while the coolant makes two passes.

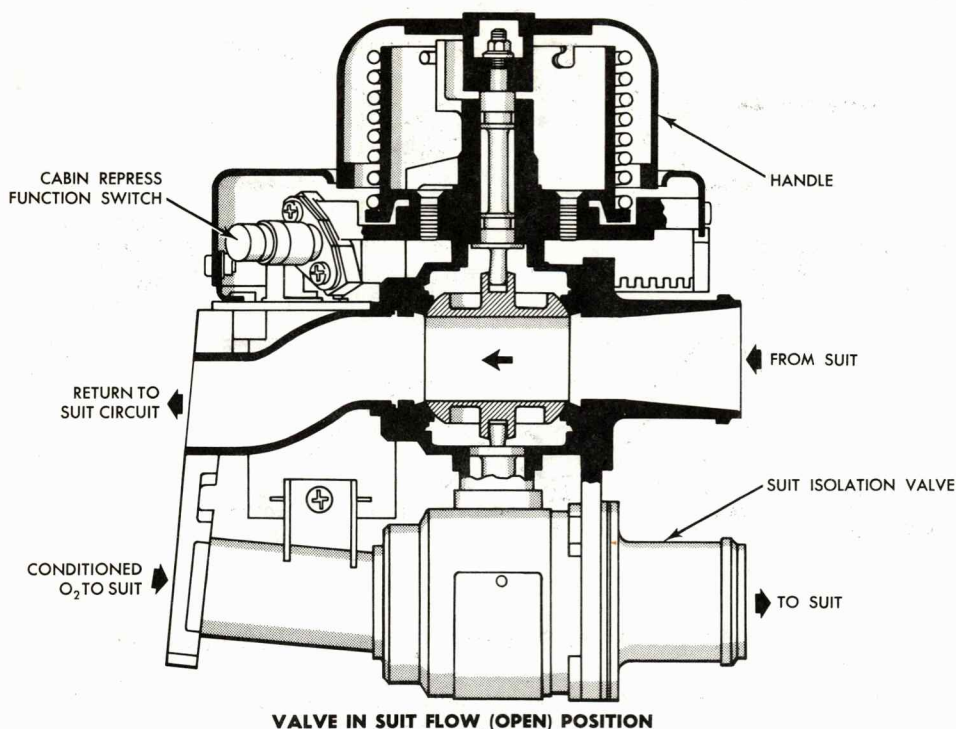
Suit Temperature Control Valve. The suit temperature control (SUIT TEMP) valve is a manually operated diverter valve that controls coolant flow through the suit circuit regenerative heat exchanger. Setting the valve to INCR HOT increases the temperature; setting it to DECR COLD reduces the temperature.

Suit Isolation Valves. (See figure 2.6-12.) The suit isolation (SUIT ISOL) valves are manually operated, two-position dual-ball valves. In the SUIT FLOW position, suit-circuit gas is directed through the valve into the PGA, and from the PGA back into the suit circuit. In the SUIT DISC position, the valves keep the gas in the suit circuit bypassing the PGA's and preventing flow in either direction between the suit circuit and PGA's.

Setting the valve handle to SUIT FLOW loads a solenoid operated spring return mechanism. A signal from the suit circuit low pressure sensor energizes the solenoid, releasing the return mechanism, which turns the valve to SUIT DISC. The ACTUATOR OVRD lever enables manual release of the return mechanism to the SUIT DISC position. A valve position indicator switch provides a telemetry signal for SUIT DISC position.

Carbon Dioxide Partial Pressure Sensor. (See figure 2.6-13.) The carbon dioxide partial pressure sensor, in the suit circuit assembly, is a single-beam, dual-wavelength, filter photometer, with ratio readout. The sensor operates on the infrared-absorption principle. It measures the amount of infrared energy absorbed by the carbon dioxide in a gas sample that passes through the sensor, by comparing transmitted energy of two different wavelengths in the infrared spectrum. (One wavelength is absorbed

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



A-300LM4-247

Figure 2.6-12. Suit Isolation Valve - Functional Schematic

by carbon dioxide; the other is a reference.) This establishes an amplified ratio signal that is indicated as a d-c voltage proportional to the partial pressure of carbon dioxide in the gas sample.

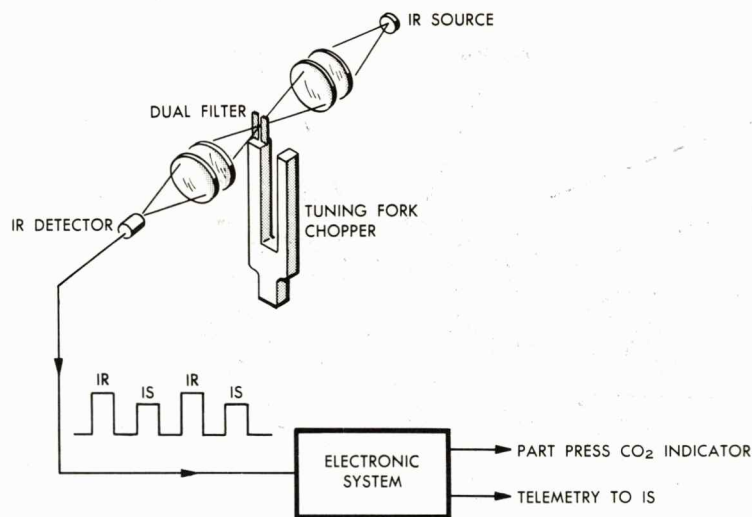
The sensor has two sections: optics and electronics. The optics section has the infrared energy source (a small tungsten lamp), a collimating lens, a lens that reimages the source on the dual filter, an aperture to fix the same image in the dual filter, and a lens that reimages the chopped and filtered source image onto the detector target. The electronics section detects and decodes the signal, computes the ratio and, then, reads out a continuous d-c voltage proportional to the partial pressure of carbon dioxide in the gas sample. The sensor provides an electrical signal to the PART PRESS CO₂ indicator and a telemetry signal to indicate the carbon dioxide level in the gas supplied to the astronauts.

2.6.4.1.2 Cabin Air Recirculation Assembly.

Cabin Heat Exchanger. The cabin heat exchanger is a cross-counter flow, single-pass oxygen, multipass coolant, aluminum plate-and-fin type of heat exchanger, in a welded, sheet-metal housing.

As flow passes through the heat exchanger, heat is transferred to, or from, the circulating coolant. The coolant flow is regulated by the cabin temp valve. A temperature transducer at the cabin heat exchanger outlet senses the coolant temperature and provides a signal to telemetry.

Cabin Fans. Two cabin fans with an interconnecting duct are mounted within a sheet-metal housing attached to the cabin heat exchanger. The fan motors are of the brushless, d-c type; they operate on 28 volts dc with an input power of 62 watts average, 210 watts peak. The fans circulate cabin gas at $+75^{\circ} \pm 5^{\circ}$ F. Each fan can move approximately 5 pounds of air per minute at 13,000 rpm. Fan inlet screens



300LM4-237

ECS

Figure 2.6-13. Carbon Dioxide Partial Pressure Sensor, Functional Schematic

filter out particles that could impair the performance of the cabin recirculation assembly. These screens are replaceable before, but not during, flight. The fans permit operation at sea level for checkout purposes. An input voltage of 15 volts dc is provided for this purpose.

2.6.4.2 Oxygen Supply and Cabin Pressure Control Section.

2.6.4.2.1 Descent Oxygen Tank.

The descent oxygen tank is in quadrant 3 of the descent stage. The tank holds 48.01 pounds of oxygen, stored at a pressure of 2,850 psia. The tank provides six PLSS refills at 0.92 pounds each and four cabin repressurizations at 6.6 pounds each. The tank is filled through a high-pressure fill port, which is capped and lockwired before launch.

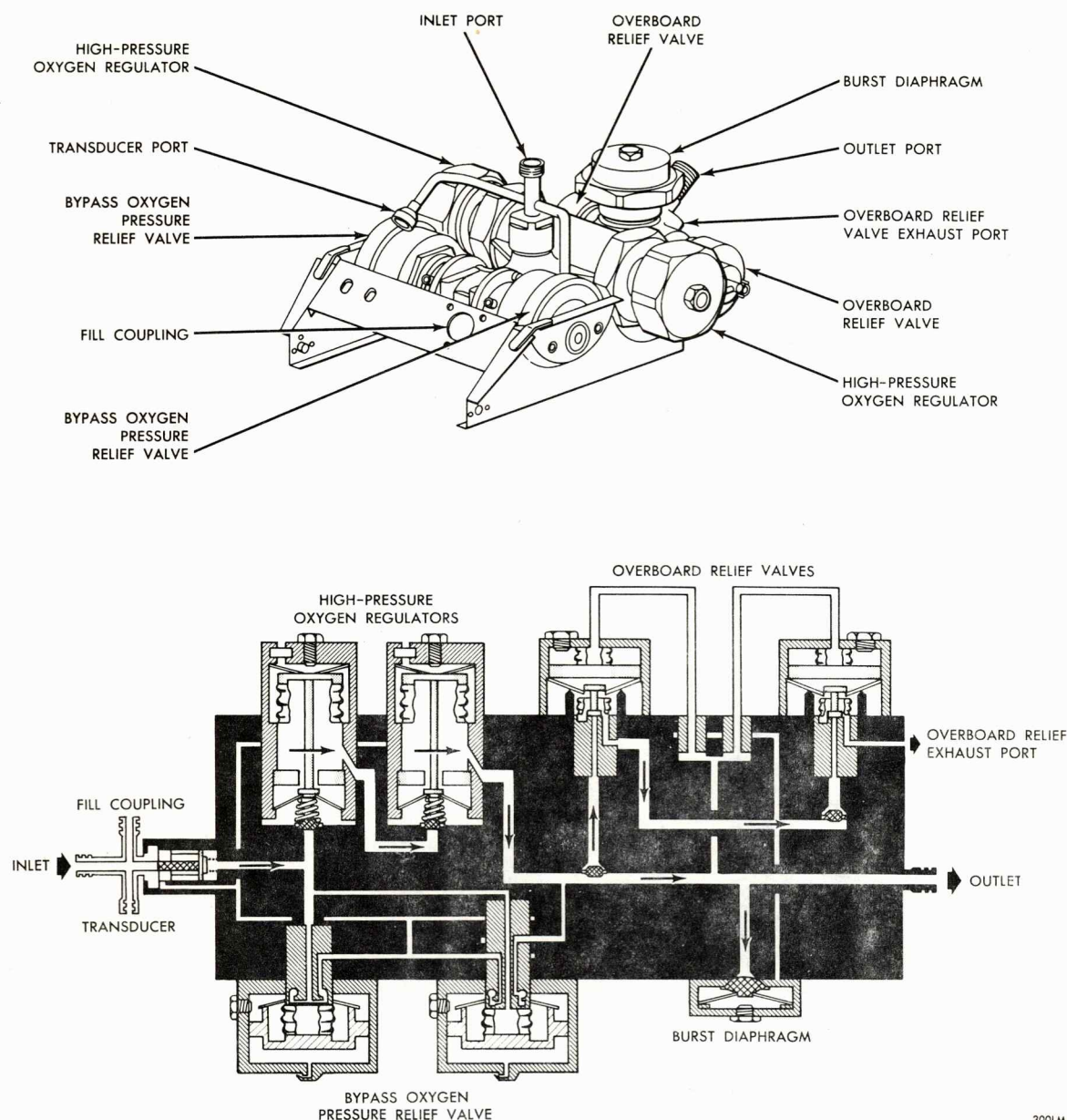
2.6.4.2.2 High-Pressure Oxygen Control Assembly. (See figure 2.6-14.)

The major components of the high-pressure oxygen control assembly are a high-pressure oxygen regulator, a bypass oxygen pressure relief valve, an overboard relief valve, and a burst diaphragm assembly.

High-Pressure Oxygen Regulator. (See figure 2.6-15.) The internally series redundant high-pressure oxygen regulator receives high-pressure oxygen from the descent oxygen tank and regulates the pressure to 875-960 psia. If the upstream (primary) regulator sensor fails, the valve fails open, permitting the downstream (secondary) regulator to control outlet pressure. Descent oxygen that enters the regulator is sensed by the primary sensor. As the pressure builds up inside the sensor, it expands and allows the primary valve poppet to move towards its seat, regulating the outlet pressure to the secondary regulator. The secondary regulator operates the same as the primary regulator.

Bypass Oxygen Pressure Relief Valve. (See figure 2.6-16.) The bypass oxygen pressure relief valve protects the descent oxygen tank against overpressurization by bypassing the high-pressure oxygen regulator. The valve is designed to fail in the open condition if it malfunctions. Pressure acting on the plunger-and-bellows assembly displaces a bellville spring causing its inside diameter to increase. This removes the force exerted by the bellville spring against the thin-walled tube around the plunger and allows inlet oxygen to vent to the outlet port. If relief valve malfunctions (bellows ruptures), oxygen is diverted to the

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



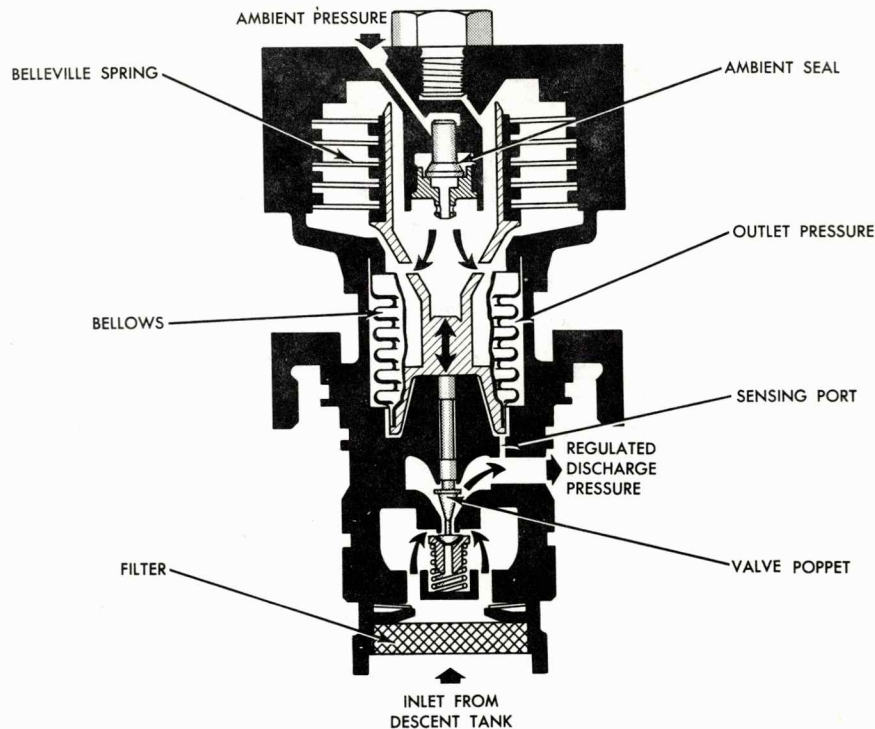
300LM4-236

Figure 2.6-14. High-Pressure Oxygen Control Assembly - Functional Schematic

guide chamber, causing the guide to close a seal. This places the valve in the failed-open condition and bypasses the oxygen to a secondary (identical) relief valve.

Overboard Relief Valve. (See figure 2.6-17.) The series-redundant overboard relief valve vents oxygen to ambient when the pressure downstream of the high-pressure oxygen regulator reaches 1,025 psig. The valve is fully open at 1,090 psig and reseats at 985 psig at +75° F. System pressure at the first stage relief valve is not relieved overboard, due to the force of the poppet load spring. System pressure is also fed to both capsule sensing elements. When system pressure reaches 1,025 psig, the capsule sensing

ENVIRONMENTAL CONTROL SUBSYSTEM



A-300LM4-233

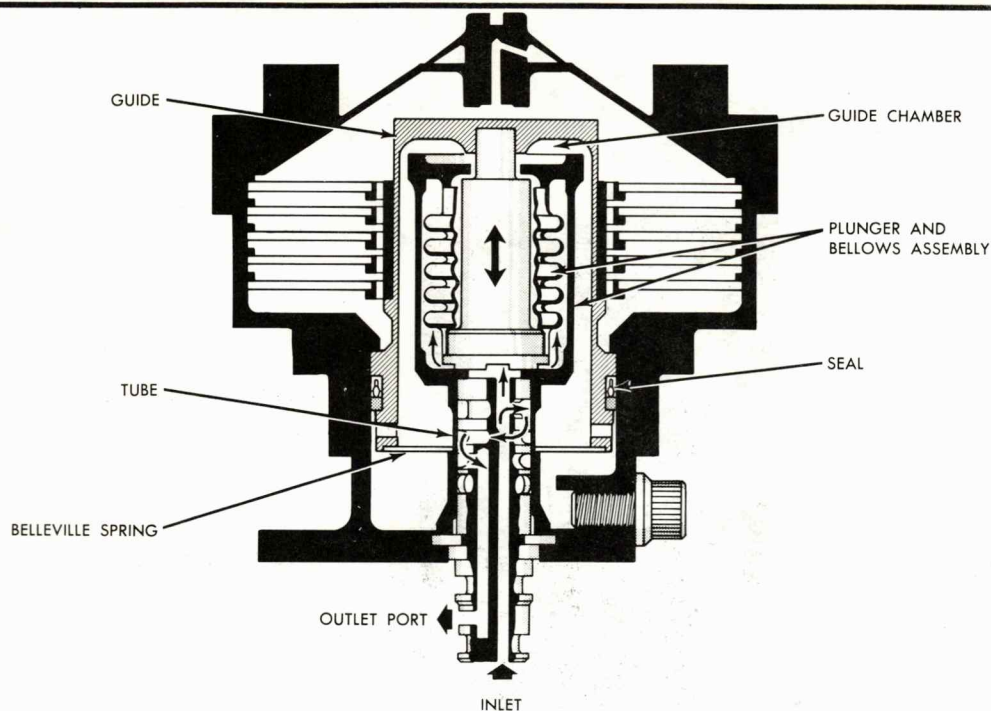
Figure 2.6-15. High-Pressure Oxygen Regulator - Functional Schematic

elements expand, displacing the fail-safe pistons and exerting a force on the belleville lever springs. Belleville-lever-spring motion about the fulcrums opens the poppet valves.

The relief valve has a fail-open feature. If a capsule sensing element leaks, the oxygen is dumped into the leakage chamber. This leakage bleeds overboard; however, its flow rate is restricted by a bleed orifice in the piston. This allows leakage chamber pressure to build up to force the poppet valve open. As the piston is displaced to its full travel, proof-pressure stops and seals make contact, preventing further loss of oxygen overboard. The valve can flow from 0 to 50 pounds per hour at an inlet pressure of 1,100 psig.

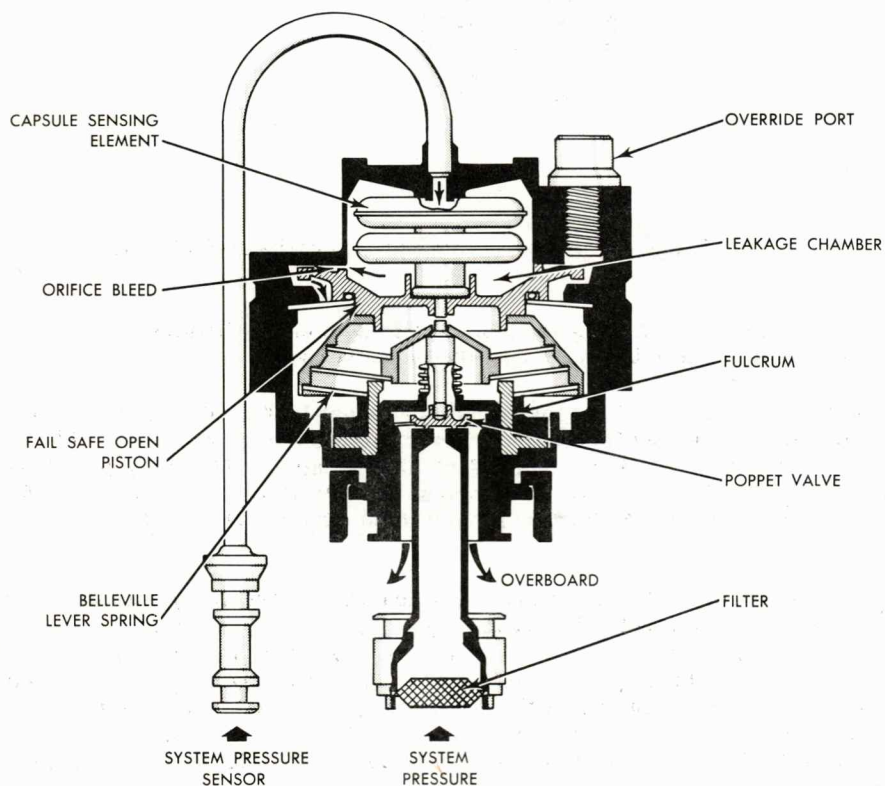
Burst Diaphragm Assembly. (See figure 2.6-18.) The burst diaphragm assembly opens when the flow from the descent oxygen tank exceeds the relieving capability of the overboard relief valve. An aluminum disk in the inlet port of the burst diaphragm assembly ruptures at a system pressure between 1,300 and 1,400 psig. System pressure causes the diaphragm support to move away from the disk, causing it to rupture. The disk support poppet opens, and vents the descent oxygen tank. The diaphragm assembly permits a minimum flow of approximately 10 pounds per minute. When descent tank pressure is reduced to 1,000 psia, the disk support poppet reseats to maintain sufficient oxygen for one cabin repressurization up to 1 hour after disk rupture.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



300LM4-240

Figure 2.6-16. Bypass Oxygen Pressure Relief Valve - Functional Schematic



300LM4-234

Figure 2.6-17. Overboard Relief Valve - Functional Schematic

ENVIRONMENTAL CONTROL SUBSYSTEM

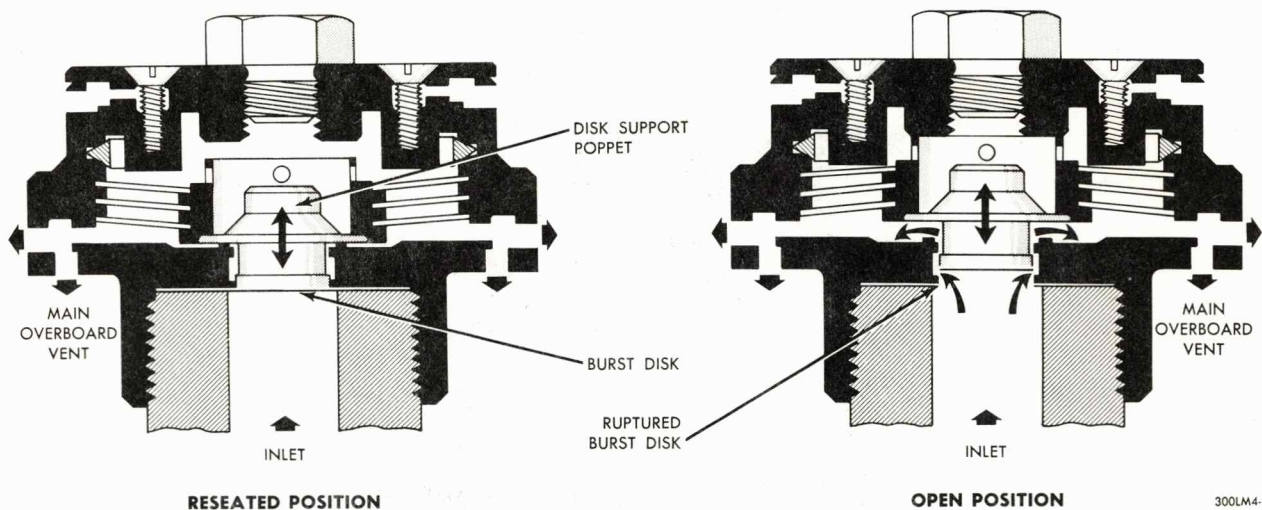


Figure 2.6-18. Burst Diaphragm Assembly - Functional Schematic

2.6.4.2.3 Ascent Oxygen Tanks.

Two identical tanks supply all the oxygen required for metabolic consumption and to compensate for cabin and/or suit circuit leakage, oxygen component leakage, and loss due to cabin puncture, subsequent to switchover to ascent consumables. Both tanks are in the aft equipment bay, which is unpressurized. Oxygen flow from either tank is controlled by individual oxygen shutoff valves on the oxygen control module.

2.6.4.2.4 Oxygen Filters.

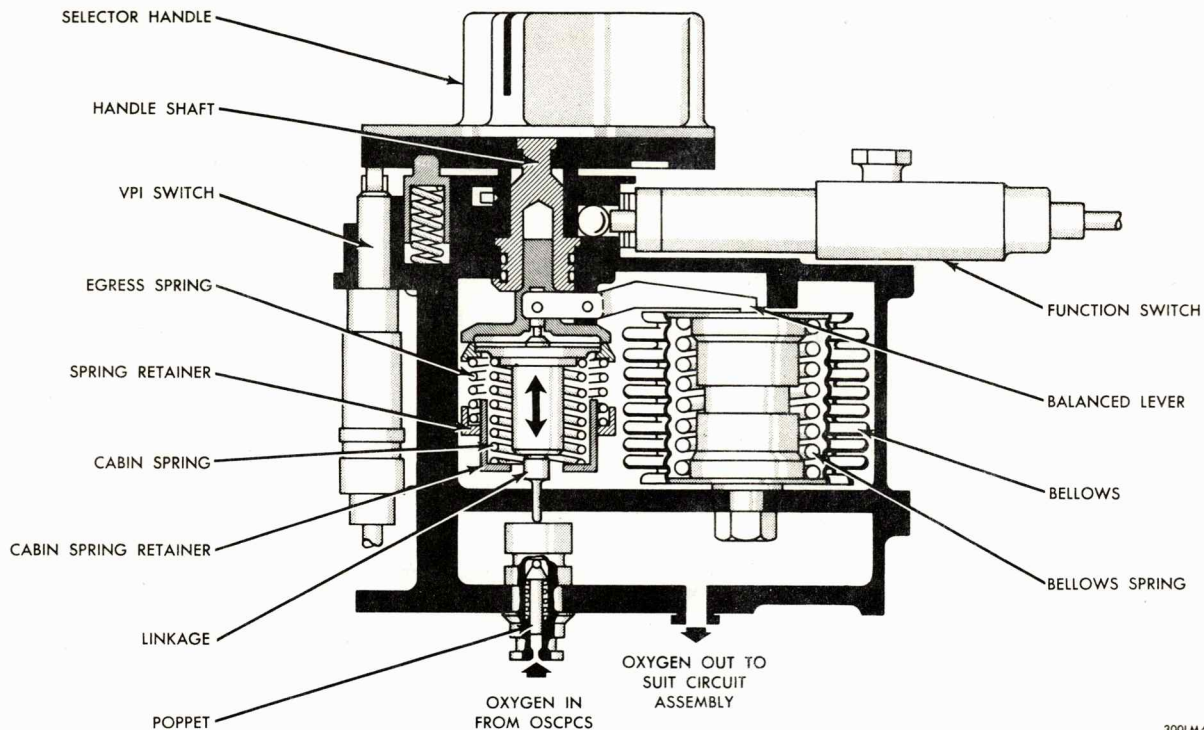
Filters at the inlets to the oxygen control module and the PLSS disconnect remove particulate matter from the oxygen. Filtering capability is 18 microns nominal and 40 microns absolute.

2.6.4.2.5 Oxygen Control Module. (See figure 2.6-2.)

The oxygen control module is mounted on the suit circuit package located to the right rear of the LM Pilot's station. It consists of a cast housing that contains filters, a PLSS oxygen shutoff valve (PLSS FILL), descent and ascent oxygen shutoff valves (DESCENT O₂, ASC O₂ No. 1 ASCENT O₂ No. 2), oxygen demand regulators (PRESS REG A, PRESS REG B), and a cabin repressurization and emergency oxygen valve (CABIN REPRESS). An interlock prevents opening the ascent oxygen shutoff valves until the descent oxygen shutoff valve is closed.

PLSS and Oxygen Tank Shutoff Valves. Four manually operated shutoff valves control oxygen flow into the oxygen control module. Rotating the valve handle to OPEN displaces a spring-loaded seal. The valves have detents in the OPEN and CLOSE positions; mechanical stops on the handle prevent overtravel.

Oxygen Demand Regulators. (See figure 2.6-19.) Two, parallel oxygen demand regulators (PRESS REG A and PRESS REG B) valve oxygen from the tanks to the suit circuit. Both regulators are manually controlled by individual four-position selector handles. ARS pressure is fed back to an aneroid bellows that actuates a poppet balanced lever system. The levers move a poppet that is biased by spring force. Rotating the valve handle to EGRESS changes the spring force acting on the poppet and establishes two outlet pressure levels.



300LM4-246

Figure 2.6-19. Oxygen Demand Regulator - Functional Schematic

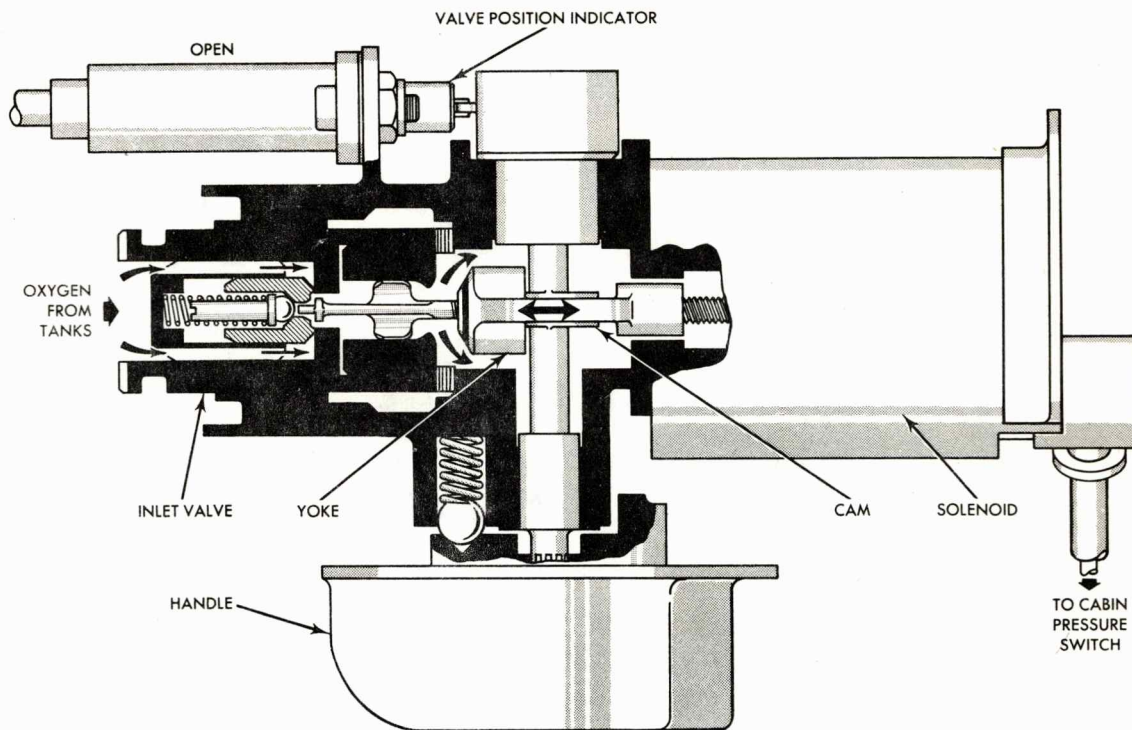
Selecting the DIRECT O₂ position fully opens the valve. With an upstream pressure of 900 psia and a temperature of +70°F, one regulator can flow approximately 7.0 pounds per hour into the suit circuit. In the CLOSE position, an auxiliary poppet stops all flow through the regulator. Each regulator has valve position indicator (VPI) switches, and two functional switches that control electrical circuits in the ECS.

Cabin Repressurization and Emergency Oxygen Valve. (See figure 2.6-20.) The cabin repressurization and emergency oxygen (CABIN REPRESS) valve is a solenoid-operated valve with manual override (MANUAL and CLOSE positions). It is used to repressurize the cabin after a deliberate decompression and provides an emergency flow of oxygen if the cabin is punctured. If the cabin is punctured and the diameter of the hole does not exceed 0.5 inch, the valve can maintain cabin pressure at 3.5 psia for at least 2 minutes, allowing the astronauts to return to a closed-suit environment. The cabin repressurization and emergency oxygen valve is controlled by a three-position handle on the oxygen control module.

When the AUTO position is selected, valve operation is controlled by the solenoid, which is actuated by the cabin pressure switch. If either oxygen demand regulator is set to CABIN or DIRECT O₂ and cabin pressure drops below normal pressure, the cabin pressure switch energizes the solenoid. This moves the valve yoke to open the inlet valve and permits oxygen from the oxygen manifold to flow through the valve into the cabin. If cabin pressure increases to normal, the cabin pressure switch deenergizes the solenoid and spring action closes the valve. Setting both oxygen demand regulators to CLOSE or EGRESS disables the circuitry to the solenoid through the cabin pressure switch, provided suit pressure does not drop below 3.12 psia. Flow is 4 pounds per minute regardless of pressure; in the ascent mode of operation flow is 8 pounds per minute at 700 psia. This protects the astronauts against catastrophic failure of the suit circuit while the cabin is unpressurized.

When the MANUAL position is selected, the handle shaft actuates a cam that moves the yoke to open the inlet valve and establish flow to the cabin.

In the CLOSE position an auxiliary poppet is forced onto its seat shutting off the flow.



A-300LM4-97

Figure 2.6-20. Cabin Repressurization and Emergency Oxygen Valve - Functional Schematic

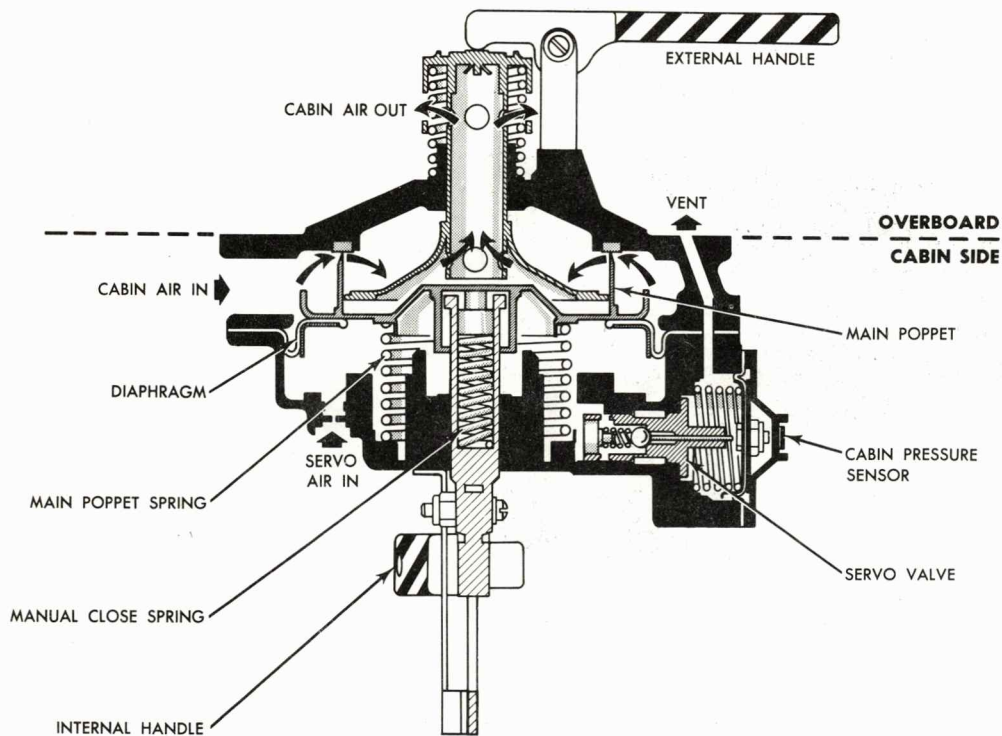
2.6.4.2.6 Cabin Relief and Dump Valves. (See figure 2.6-21.)

A cabin relief and dump valve is installed in each hatch door. The valve has a differential-pressure, servo-actuated poppet valve that prevents cabin overpressurization and permits deliberate (manual) dumping of cabin pressure. Each valve is controlled with either of two handles: one, inside the cabine; the other, outside. The inside handles can be used to select three positions (DUMP, AUTO, or CLOSE). The outside handles have only a DUMP position. A bacteria filter mounts on the forward cabin relief and dump valve.

Normally, the cabin relief and dump valves are set to AUTO. In this position, the valves are operated by the servo valve. One side of the servo valve diaphragm is exposed to cabin pressure; the other side is vented overboard. When the cabin-to-ambient pressure differential reaches approximately 5.4 psi, the servo valve diaphragm overcomes the servo spring pressure and unseats the servo valve. This vents the spring side of the main poppet to ambient and causes a lower pressure. Cabin pressure on the main poppet overcomes the force of the main poppet spring and opens the dump valve. Oxygen is dumped overboard and cabin pressure is reduced to an acceptable value. When the pressure differential is 5.8 psi, one fully open dump valve can dump 11.1 pounds of oxygen per minute overboard. When the valve handle inside the cabin is set to AUTO, the valve can be opened manually from outside.

Setting the internal handle to DUMP unseats the poppet. The valve in the overhead hatch can dump the cabin pressure from 5.0 to 0.08 psia in 180 seconds without cabin oxygen inflow; the forward hatch valve (with filter) in 310 seconds. Setting the handle to CLOSE prevents the valve from opening at normal pressures, if the servo valve fails.

Pressure transducers associated with each valve provides a telemetry signal to indicate when the valves are relieving cabin pressure.



300LM4-98

Figure 2.6-21. Cabin Relief and Dump Valve - Functional Schematic

Oxygen Bacteria Filter. A filter on the cabin side of the forward cabin relief and dump valve removes bacteria from the dumped cabin oxygen. The filter can be installed and removed by an astronaut in the cabin. It removes 95% of all bacteria larger than 0.5 micron.

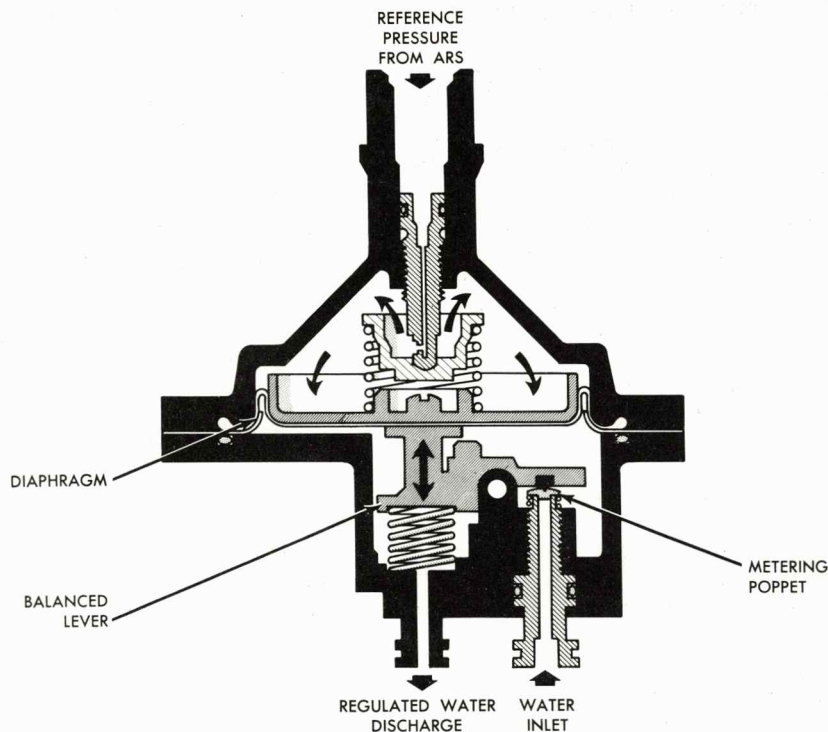
2.6.4.2.7 Cabin Pressure Switch.

The cabin pressure switch monitors cabin pressure and enables electrical signals to control related ECS functions. The switch is an absolute-pressure device that consists of three separate hermetically sealed microswitch capsules. The capsules are set to provide switch closure when cabin pressure decays to approximately 4.15 ± 3 psia during pressurized-cabin operation. If this occurs with either oxygen demand regulator in cabin mode, the suit gas diverter valve closes, the cabin repressurization and emergency oxygen valve opens, and the CABIN warning light goes on. Increasing cabin pressure to 4.65 ± 0.25 psia opens the cabin pressure switch circuits, closes the cabin repressurization and emergency oxygen valve, and deenergizes the CABIN warning light.

2.6.4.3 Water Management Section.

2.6.4.3.1 Descent Stage Water Tank.

The descent stage water tank is an aluminum tank with an internally mounted standpipe and bladder. The bladder contains the water; the space between the tank and bladder is charged with nitrogen according to a schedule dependent on the load (48.2 psia maximum at +80° F). The nitrogen squeezes the water out of the bladder through the standpipe and into the system. The tank outlet is connected to the water control module. Water from the descent water tank is routed through the WATER TANK SELECT valve by setting it to DES.



300LM4-232

Figure 2.6-23. Water Pressure Regulator - Functional Schematic

2.6.4.4 Heat Transport Section.

2.6.4.4.1 Coolant Accumulator.

The coolant accumulator consists of a two-piece aluminum cylinder that contains a movable spring-loaded piston bonded to a diaphragm. The fluid side contains approximately 46 cubic inches of fluid under pressure. The pressure varies, depending on fluid level, from 5.5 to 8 psi. The accumulator maintains a head of pressure on the glycol pump inlets to prevent pump cavitation and replaces coolant lost through subsystem leakage. The piston moves in response to volumetric changes caused by temperature variations in the primary loop or by leakage. The accumulator spring side is vented to space.

The accumulator has a low-level sensor with a switch that trips when only $10 \pm 5\%$ of the coolant volume remains. The switch is mounted on top of the accumulator. A tube extending from the top of the accumulator, houses a rod attached to the piston. The switch provides a telemetry signal.

2.6.4.4.2 Primary Coolant Filter.

The primary coolant filter has a filtering capability of 35 microns. It has an integral pressure relief bypass feature that operates at 0.4 psid to maintain coolant flow to the pumps if the filter becomes clogged.

2.6.4.4.3 Coolant Pumps.

The three coolant pumps are identical. Two pumps are connected in the primary loop; the third pump in the secondary loop. The pumps are of the sliding-vane, positive-displacement type; they are driven by 28-volt d-c brushless motors. The motors are of wet or submerged design and are cooled by the recirculating coolant. The pumps receive the coolant at a recirculation rate of approximately 3.7

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

pounds per minute at +40° F and 7.0-psia pump inlet pressure. The pumps increase coolant pressure by 30 psi (nominal).

2.6.4.4.4 Cabin Temperature Control Valve. (See figure 2.6-24.)

The cabin temperature control valve is a diverter valve of sliding-spool construction, with three positions; NORM (automatic), MAX COOL, and MAX HEAT. The NORM position is manually selected and is activated by a wax-filled sensor; all coolant discharged from the cabin heat exchanger is fed across the sensor. The sensor expands and contracts in response to coolant temperature changes and, by means of the plunger and spool return spring, repositions the valve spool to obtain the selected coolant and subsequent oxygen temperature. This splits the coolant flow; part goes to the cabin heat exchanger and part to the coolant regenerative heat exchanger. In the MAX COOL (override) position, the spool is positioned to the left, permitting all of the cold coolant to enter the valve and be fed directly to the cabin heat exchanger. In the MAX HEAT (override) position, the spool is repositioned to close the cabin heat exchanger feed port and opens the feed to the coolant regenerative heat exchanger for the hot fluid being returned to the coolant pump inlet from the cold plates. The hot fluid is bypassed to the cabin heat exchanger to perform its heating function.

2.6.4.4.5 Coolant Regenerative Heat Exchanger.

The coolant regenerative heat exchanger is of plate-and-fin construction. It has an inlet and outlet for cold coolant circulation and an inlet and outlet for warm coolant circulation past the coolant core. The cold side is single pass; the warm coolant makes four passes in a cross-counter-flow arrangement.

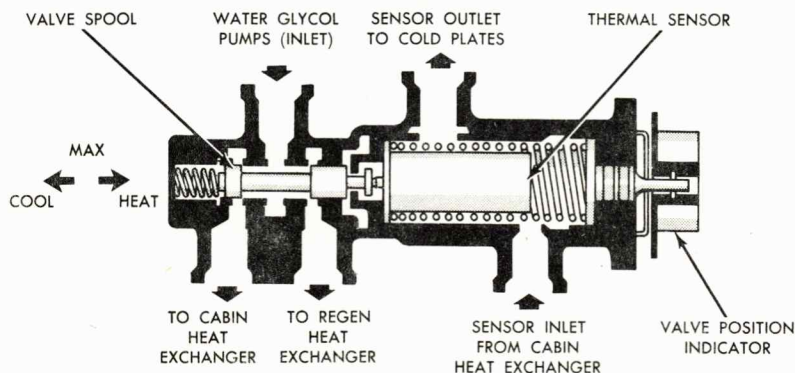
2.6.4.4.6 Suit Temperature Control Valve.

The suit temperature control valve is a manually operated diverter valve. The amount of warm coolant flowing through the suit circuit regenerative heat exchanger is regulated by the handle, which can be turned from HOT to COLD.

2.6.4.4.7 Sublimators

The porous-plate-type sublimators (one in the primary loop and another in the secondary loop) are identical, except that the primary sublimator has a larger capacity. Each sublimator has a coolant inlet and outlet, a water inlet, and a steam outlet. Water makes one pass through the unit; coolant makes six passes. For proper sublimator operation, water pressure must exceed 4.0 psia but be less than 8.0 psia. The water pressure must also be less than the suit circuit static pressure plus the head pressure from the water separators to the sublimator.

The unit rejects heat to space by sublimation of ice. Water from the WMS flows through the water passages, into the porous plates, and is exposed to space environment. The vacuum pressure is below the triple point of water; this causes an ice layer to develop within the pores and on the outer surface of the plates. As the hot coolant flows through the sublimator passages, heat transfers from the



300LM4-231

Figure 2.6-24. Cabin Temperature Control Valve - Functional Schematic

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

coolant to the water and to the ice layer. The ice sublimates from the solid state to steam without passing through a liquid state, rejecting its heat load overboard through a duct. The thickness of the ice layer varies with the heat load imposed on the sublimator, resulting in a regulated output temperature over a range of input temperatures.

If the heat load entering the sublimator drops below its low limit, the ice layer expands and may completely fill the water passages. The expanding ice layer may rupture the sublimator. If the heat load is too high, the ice layer melts. If the water pressure is too high, a breakthrough of the ice layer may occur, forcing water through the porous plates. In these cases, heat rejection by the sublimator is greatly reduced because ice is not sublimated. Water consumption is excessive because there is no restraining ice layer.

2.6.4.4.8 Cold Plates and Rails.

Electronic equipment that requires active temperature control is cooled by cold plates and cold rails. The cold plates are installed on the LM structure in a manner that minimizes heat transfer from the structure to the coolant, to avoid a reduction of the coolant cooling capacity. The surrounding structure and equipment may have a temperature range of 0° to +160° F. Cold rails are used in the aft equipment bay for batteries, and the DSEA. The IMU has an integral cooling circuit. All other cold plates are of the flat type. Cold plates for equipment essential for mission abort have two independent coolant passages, one for the primary loop and one for the secondary loop.

The flat cold plates are brazed assemblies with inlet and outlet fittings. The coolant flows between two parallel sheets, which are connected by fins for increased heat transfer and structural strength. The internal fin arrangement of these cold plates ensures sufficient flow distribution.

The cold rails are channel-and-tube type of extrusions; the tubular part forms the inside center of the channel. The tube has fins and, at the ends, coolant inlet and outlet fittings. The cold rails are installed in a parallel arrangement, with equal space between the rails to accommodate equipment designed for mounting on cold rails. Each cold rail (except the first and the last one) cools two adjacent rows of equipment.

To increase heat transfer, an interface material is used between the electronic equipment (except the DSEA) and the cold plates. The interface material (silicone rubber with an aluminum oxide additive) is applied in liquid form to the equipment surface before installation and is vulcanized at room temperature after the equipment is mounted. Electronic replaceable assemblies can be installed on, and removed from, their cold plates without disturbing the coolant circuit. Repeated removal and installation requires removal of old, and application of new, interface material. The IMU has flexible line connections and quick-disconnects to facilitate IMU replacement, if required.

2.6.5 FAILURE MODES.

The ECS failure modes and the effect on mission capability are given in table 2-41.

Table 2.6-1. Environmental Control Subsystem - Failure Modes

Failure	Indication	Vehicle Capability
Heat exchanger(s)		
1. Internal leak	Loss of suit circuit cooling	Remaining test objectives are lost.
2. External leak	Same as above	Remaining test objectives are lost. Primary coolant loop is lost and cabin becomes contaminated.
Suit circuit assembly cracked.	SUIT/FAN warning light goes on and SUIT PRESS indicator reads low.	Suit circuit gas goes to cabin, curtailing mission. Astronauts can live in cabin environment. Abort mission.

ECS

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-1. Environmental Control Subsystem - Failure Modes (cont)

Failure	Indication	Vehicle Capability
Descent oxygen tank to interface with the oxygen control module (OCM) has ruptured or has external leak.	Loss of descent tank pressure.	Abort mission.
Interface with OCM has external leak.	Cabin pressure may rise and O ₂ QUANTITY indicator reads low.	Descent oxygen may be lost, curtailing some mission objectives. Ascent oxygen used for ascent and rendezvous.
OCM cracked or ruptured.	O ₂ QTY caution light goes on and O ₂ QUANTITY indicator reads low.	Descent oxygen is lost, aborting mission.
Burst diaphragm ruptured.	O ₂ QTY caution light goes on and QUANTITY indicator reads low.	One more cabin pressurization is possible. Mission abort.
Oxygen interstage disconnect did not disconnect at staging.	No indication	Failure to separate stages with loss of crew.
Oxygen interstage disconnect leak before staging.	MSFN will notice pressure drop.	Loss of descent oxygen causes loss of descent primary mission objectives.
Interface between OCM and flex hose has external leak.	O ₂ QTY caution light goes on and O ₂ QUANTITY indicator reads low.	Mission objectives are curtailed.
Ascent oxygen tank ruptured or has external leak.	O ₂ QTY caution light goes on and O ₂ QUANTITY indicator reads low.	Loss of redundancy for ascent O ₂ .
Descent oxygen filter clogged.	Normal descent O ₂ pressure but low indication on PLSS line transducer.	Loss of descent oxygen curtails test objectives.
DES O ₂ valve jammed on opening or has external leak through stem.	Jammed valve is found during crew operations. O ₂ QUANTITY indicator reads low (for leak).	Loss of descent oxygen curtails mission.
#1 ASC O ₂ or #2 ASC O ₂ valve has external leak through stem.	Cabin pressure may rise and O ₂ QUANTITY indicator reads low.	Ascent oxygen may be lost, curtailing some mission objectives.
CABIN REPRESS valve failed open.	Cabin repress valve flow is very loud.	Close valve, some O ₂ loss.
PRESS REG A or PRESS REG B valve has external leak through stem.	Cabin pressure may rise and O ₂ QUANTITY indicator reads low.	Descent Oxygen is lost, aborting mission.
Plumbing between OCM and suit circuit assembly ruptured or has external leak.	Suit circuit leaks excessively during suit leak checkout.	Abort mission.
Descent water tank has external leak or is ruptured.	H ₂ O QUANTITY indicator reads low.	Remaining descent capability is lost.

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-2. Environmental Control Subsystem - Performance and Design Data (cont)

Atmosphere revitalization section (cont)

Suit circuit pressure

Cabin mode	4.8±0.2 psia (may exceed 5.0 psia during powered flight)
Egress mode	3.8±0.2 psia (4.7 psia maximum during powered flight)
Minimum	3.5 psia
Maximum	9.6 psia and 5.0 psia (with one oxygen demand regulator failed open)
During EMU checkout (measured downstream of PGA's)	8.7 psia (3.7 psig) for 5 minutes

LiOH dust concentration in atmosphere delivered to astronauts

0.02 mg/m³ (maximum)

Leakage

ARS external leakage (with CO₂ and odor removal canisters closed, exclusive of PGA's)

0.01 pound per hour (maximum) at 3.7 psia suit pressure with cabin unpressurized

Suit (one man)

0.035 pound per hour (maximum) with cabin unpressurized

Suit inlet temperature range

With SUIT TEMP control valve in full COLD position, temperature ranges from +38° to +65°F; in full HOT position from +42° to +100° F. Under certain maximum operating conditions suit inlet temperature may exceed +100°F.

Cabin temperature

+55° to +90° F

Relative humidity

40% to 80%

Suit circuit fan flow

At 4.8 psia - 36.0 pounds per hour
At 3.8 psia - 28.4 pounds per hour

Oxygen supply and cabin pressure control section

Suit pressure increase

First 1-psi increase may occur in less than 1 second. Each succeeding 1-psi increase occurs in not less than 8 seconds.

Cabin repressurization and emergency oxygen valve delivery rate

Descent mode

4 pounds per minute, maximum

Ascent mode

8 pounds per minute at 700 psia

PLSS refill

0.91 pound/refill at 850 psia or above. Can only be partially filled at lower pressures.

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-2. Environmental Control Subsystem - Performance and Design Data (cont)

Oxygen supply and cabin pressure control section (cont)

Suit circuit minimum inflow O ₂ regulator flow capability	O ₂ regulators are capable of sufficient flow to maintain minimum suit circuit pressure of 3.5 psia with suit circuit relief valve failed open and with 550 psia and +130° F at oxygen control module inlet
Cabin volume	235 cubic feet
Oxygen filter efficiency	40 microns absolute
Dump valve bacteria filter efficiency	Removes 95% of all bacteria larger than 0.5 micron
Cabin-to-ambient pressure differential	5.8 psi (maximum) steady state
Manual cabin pressure dump rate (without oxygen inflow)	Dump valve in forward hatch, with bacteria filter, from 5.0 psia down to 0.08 psia in 310 seconds; overhead hatch, without filter, in 180 seconds. Both valves without filter 90 seconds
Number of cabin repressurizations	4 at 6.6 pounds each
Cabin repressurization time	2 minutes
Cabin pressure switch settings	When cabin pressure drops to 4.15±0.3 psia, contacts close When cabin pressure increases to 4.65 ±0.25 psia, contacts open
Descent oxygen tank	
Capacity	48.01 pounds at 2730 psia and +70° F (residual oxygen: 1.0 pound)
Burst pressure	4,500 psi
Ascent oxygen tanks	
Capacity	2.41 pounds (each tank) at 840 psia and 70° F (residual oxygen: 0.24 pounds each tank)
Burst pressure	1,500 psia
Gaseous oxygen pressures and temperatures	
Normal pressure (regulated)	900 +60 -25 psia
Normal pressure (unregulated)	50 to 960 psia
Maximum pressure	1,100 psia
Normal temperature	0° to +160° F

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-2. Environmental Control Subsystem - Performance and Design Data (cont)

Oxygen supply and cabin pressure control section (cont)

Leakage

Oxygen supply components	Total external leakage of all oxygen supply components, under maximum operating pressures, does not exceed 0.010 pound per hour of oxygen.
Cabin relief and dump valve	0.005 pound per hour (maximum) for each valve, up to 5.25 psid.
Cabin	0.2 pound per hour (maximum) at 5.0 psia and +75° F

Bypass relief valve

Full flow pressure	3,030 psig (minimum) at +75° F
Cracking pressure	2,875 psig (maximum) at +75° F
Reseat pressure	2,850 psig (minimum) at +75° F

Overboard relief valve

Full flow pressure	1,090 psig (maximum) at +75° F
Cracking pressure	1,025 psig (minimum) at +75° F
Reseat pressure	985 psig (minimum) at +75° F

High-pressure regulator

Outlet pressure with primary and secondary regulators operating normally	At inlet pressure of 1,100 to 3,000 psig and flow of 0.1 to 4.0 pounds per hour, and inlet pressure of 975 to 1,100 psig and flow of 0.1 pound per hour will regulate to:
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Normal downstream regulated pressure (psig)

Temp (° F)	Minimum Pressure	Maximum Pressure
75	875	960

Downstream regulated pressure with a failed-open primary regulator (psig)

Temp (° F)	Minimum Pressure	Maximum Pressure
75	875	960

Downstream regulated pressure with a failed open secondary regulator (psig)

Temp (° F)	Minimum Pressure	Maximum Pressure
75	875	960

ECS

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-2. Environmental Control Subsystem - Performance and Design Data (cont)

Water management section

Descent water tank

Capacity	367 pounds
Initial fill pressure	48.2 psia (maximum) at +80° F
Expulsion efficiency	With initial fill pressure, 98% (minimum) of quantity loaded, down to 100 pounds (minimum)
Pressure upon expulsion of all expellable water	11.0 psia (minimum) at +35° F

Ascent water tanks

Capacity	47.5 pounds each
Initial fill pressure	48.2 psia (maximum) at +80° F
Expulsion efficiency	With initial fill pressure, 98% (minimum) of quantity loaded, down to 20 pounds (minimum) in each tank
Pressure upon expulsion of all expellable water	11.0 psia (minimum) at +35° F

External leakage

Normal mode (exclusive of sublimator in HTS primary loop)	0.700 cc/hr of water (maximum) at 50 psid and +70° F
Redundant mode (exclusive of sublimators in HTS secondary loop)	0.350 cc/hr of water (maximum) at 50 psid and +70° F
Sublimators in HTS primary loop	0.090 cc/hr of water (maximum combined) at 6.5 psid and +70° F
Sublimators in HTS secondary loop	0.090 cc/hr of water at 6.5 psid and +70° F
External-to-internal leakage ground service disconnects	1×10^{-2} scc/hr of helium (maximum each) at internal pressure of 500 microns and in helium ambient of 14.7 psia and +70° F
Pressure regulator discharge pressure	0.5 to 1.0 psi above ARS gas pressure
PLSS refill (each)	9.15 pounds of water

Heat transport section

Coolant	Solution of ethylene glycol and water (35% and 65%, respectively, by weight) with inhibitors (triethanolamino phosphate, to prevent decomposition; sodium mercaptobenzothiazol, to prevent electrolytic corrosion) Approximately 35 pounds of coolant is used in HTS.
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ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-2. Environmental Control Subsystem - Performance and Design Data (cont)

Heat transport section (cont)

Coolant slush point	-3° F
External leakage	
Primary coolant loop	1.12×10^{-2} cc/sec of helium (maximum) at 45 psid and +70° F
Secondary coolant loop	1.0×10^{-4} cc/sec of helium (maximum) at 45 psid and +70° F
Coolant pump pressure rise	30 psi minimum at 40° F and flow rate of 222 pounds per hour
Coolant pump bypass relief valve	
Cracking pressure	33 psi to 36 psi
Full open pressure	39 psi
Reseat pressure	1 psi less than cracking
Coolant temperatures	
Flow into primary sublimator	110° F (maximum)
Flow out of sublimator	29° to 36° F
Flow into secondary sublimator	60.1 (minimum)
Flow out of sublimator	47° F (maximum)
Vacuum fill requirements	HTS withstands internal pressure of 500 microns in sea-level ambient pressure environment.
Primary and secondary coolant filters	
Efficiency	35 microns absolute
Bypass valve operating pressure	0.4 psid
Coolant flow	
Primary coolant loop	222 pounds per hour (minimum)
Secondary coolant loop	222 pounds per hour (minimum)
Cold plates	
Coolant inlet operating temperature	+32° to +100° F
Coolant inlet operating pressure	5 to 45 psia
Coolant inlet operating flow	12 to 85 pounds per hour, depending on cold plates size

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.6.7 OPERATIONAL LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions for the ECS are as follows:

- The suit fans and cabin fans must be operated at a reduced voltage of 12 to 15 volts dc in earth ambient pressure, to prevent damage to the motors.
- The glycol pumps must be operating before water from the WMS or water separators is permitted to flow to the sublimators. Sublimator requires minimum heat load to prevent freeze up.
- The HTS must be stabilized at operating temperatures before IMU or LM guidance computer operation, to prevent erroneous and changing operation during temperature cycling resulting in invalid outputs.
- One suit fan must be on when the astronauts are on LM ECS otherwise atmosphere will not be reconditioned leading to possible loss of consciousness.
- The LiOH cartridge of the primary canister must be replaced with a new cartridge when carbon dioxide partial pressure, as indicated on the PART PRESS CO₂ indicator, reaches 7.6 mm Hg. Carbon dioxide level must be maintained below maximum acceptable partial pressure.
- The cabin fans must be turned off when cabin pressure drops below 3.8 psia or fans may overspeed and burn up.
- A minimum total heat load of 1,200 Btu/hr must be maintained when the primary water feed is on; otherwise sublimator may freeze and rupture, rendering the primary coolant loop inoperative. The heat loads result from crew activity, LiOH reaction, structural load, and equipment and battery heat loads.
- The temperature of the coolant flow to temperature-critical electronics must not exceed +50° F. Overheating may result and lead to degraded performance of critical equipment.
- A minimum total heat load of 500 Btu/hr must be maintained when the secondary water feed is on; otherwise, the sublimator may freeze and rupture, rendering the HTS inoperative.

2.6.8 TELEMETRY MEASUREMENTS.

The ECS telemetry data are given in table 2.6-3. "(R)" in the "Telemetry Range" column signifies a real-time telemetry measurement. Real-time telemetry points are shown in figures 2.6-6, 2.6-7, 2.6-8 and 2.6-9. For all ECS signals that require conditioning, refer to tables 2.9-1 and 2.9-2.

Table 2.6-3. Environmental Control Subsystem - Telemetry Measurements

Code	Description	Telemetry Range	Display Range	Crew Display
GF1083X	Prime suit compressor failure	Contact closure		None
GF1083X	Prime suit compressor failure			ECS caution light
GF1084X	Spare suit compressor failure	Contact closure		None

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-3. Environmental Control Subsystem - Telemetry Measurements (cont)

Code	Description	Telemetry Range	Display Range	Crew Display
GF1084X	Spare suit compressor failure			SUIT/FAN warning light
GF1087X	Selected suit compressor failure			SUIT/FAN component caution light
GF1111R	H ₂ O separator No. 1 rate	500 to 3,600 rpm		None
GF1112R	H ₂ O separator No. 2 rate	500 to 3,600 rpm		None
GF1201X	CMDR suit isolation valve VPI disconnect	Contact closure (R)		None
GF1202X	LMP suit isolation valve VPI disconnect	Contact closure (R)		None
GF1211X	Suit pressure relief VPI closed	Contact closure (R)		None
GF1212X	Suit pressure relief VPI open	Contact closure (R)		None
GF1221X	Suit diverter VPI closed	Contact closure (R)		None
GF1231X	Cabin gas return VPI closed	Contact closure (R)		None
GF1232X	Cabin gas return VPI open	Contact closure (R)		None
GF1241X	CO ₂ cartridge in SEC position	Contact closure (R)		None
GF1242X	CO ₂ cartridge in SEC position	Contact closure		CO ₂ component caution light
GF1281T	Suit inlet temperature	+20° to +120° F (R)		
GF1281T	Suit inlet temperature	+40° to +100° F	+40° to +100° F	SUIT TEMP indicator
GF1301P	Suit outlet pressure	0 to 10 psia (R)	0 to 10 psia	SUIT PRESS indicator
GF1521P	CO ₂ partial pressure	0 to 30 mm Hg (R)		None
GF1521P	CO ₂ partial pressure			CO ₂ component caution light
GF1521P	CO ₂ partial pressure	7.6 mm Hg		ECS caution light
GF1522P	CO ₂ partial pressure	0 to 30 mm Hg	0 to 30 mm Hg	PART PRESS CO ₂ indicator
GF1651T	Cabin temperature	+20° to +120° F (R)	+40° to 100° F	
GF1651T	Cabin temperature	+40° to +100° F	+40° to +100° F	CABIN TEMP indicator

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-3. Environmental Control Subsystem - Telemetry Measurements (cont)

Code	Description	Telemetry Range	Display Range	Crew Display
GF2021P	Primary coolant pump ΔP	0 to 50 psid (R)		None
GF2041X	Coolant accumulator fluid low level	Contact closure (R)		None
GF2041X	Coolant accumulator fluid low level	Contact closure		GLYCOL caution light
GF2521T	Main sublimator coolant inlet temperature	0° to +200° F		None
GF2571T	Redundant (GF9998U) sublimator coolant outlet temperature	+20° to +120° F		GLYCOL temperature indicator
GF2581T	Main sublimator coolant outlet temperature	+20° to +120° F		None
GF2681T	Temperature coolant upstream of critical electronics (GF9998U)	+20° to +120° F		GLYCOL temperature indicator
GF2741P	Primary pump discharge pressure (GF9997U)	0 to 60 psia	0 to 80 psia	GLYCOL pressure indicator
GF2921P	Redundant pump discharge pressure (GF9997U)	0 to 60 psia		GLYCOL pressure indicator
GF2935X	Selected coolant pump failure	Contact closure		ECS caution light
GF2936X	Selected coolant pump failure	Contact closure		GLYCOL PUMP component caution light
GF3071X	PRESS REG A valve locked closed	Contact closure (R)		None
GF3070X	PRESS REG A valve EGRESS/CLOSED	Contact closure (R)		None
GF3073X	PRESS REG B valve locked closed	Contact closure (R)		None
GF3075X	PRESS REG B valve EGRESS/CLOSED	Contact closure (R)		None
GF3571P	Cabin pressure	0 to 10 psia (R)	0 to 10 psia	None
GF3571P	Cabin pressure	0 to 10 psia	0 to 10 psia	CABIN PRESS indicator
GF3572P	Cabin repressurization and emergency oxygen valve electrically open	Contact closure (R)		None
GF3572P	Cabin repressurization and emergency oxygen valve electrically open	Contact closure		CABIN WARNING light

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-3. Environmental Control Subsystem - Telemetry Measurements (cont)

Code	Description	Telemetry Range	Display Range	Crew Display
GF3582P	No. 1 Ascent oxygen tank pressure	0 to 1000 psi (R)		None
GF3582P	Ascent oxygen tank No. 1 pressure	684/100 psia		O ₂ QTY caution light
GF3582P	Ascent oxygen tank No. 1 pressure	0% to 100%	0% to 100%	O ₂ QUANTITY indicator
GF3583P	Ascent oxygen tank No. 2 pressure	0 to 1,000 psia (R)		None
GF3583P	Ascent oxygen tank No. 2 pressure	684 psia		O ₂ QTY caution light
GF3583P	Ascent oxygen tank No. 2 pressure	0 to 854 psia	0% to 100%	O ₂ QUANTITY indicator
GF3584P	Descent oxygen tank pressure	0 to 3,000 psia (R)		None
GF3584P	Descent oxygen tank pressure	135 psia		O ₂ QTY caution light
GF3584P	Descent oxygen tank pressure	0% to 100%	0% to 100%	O ₂ QUANTITY indicator
GF3587T	Descent oxygen tank surface temperature	0° to +200° F (R)		None
GF3589P	Selected oxygen supply pressure	0 to 1,400 psia (R)		None
GF3591P	Safety valve No. 1 servo pressure	0 to 25 psia (R)		None
GF3592P	Safety valve No. 2 servo pressure	0 to 25 psia (R)		None
GF4101P	Main water regulator ARS Δ P	0 to 2 psid (R)		None
GF4511T	Main sublimator inlet water temperature	+20° to +160° F (R)		None
GF4581Q	Descent tank water quantity	0% to 100% (R)		
GF4581Q	Descent tank water quantity	15.96%	0% to 100%	WATER QTY caution light
GF4581Q	Descent tank water quantity	0% to 100%	0% to 100%	H ₂ O QUANTITY indicator
GF4582Q	Ascent tank No. 1 water quantity	0% to 100%	0% to 100%	H ₂ O QUANTITY indicator

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.6-3. Environmental Control Subsystem - Telemetry Measurements (cont)

Code	Description	Telemetry Range	Display Range	Crew Display
GF4582Q	Ascent tank No. 1 water quantity	94.78%		WATER QTY caution light
GF4582Q	Ascent tank No. 1 water quantity			WATER QTY caution light
GF4582Q	Ascent tank No. 1 water quantity	0% to 100% (R)	0% to 100%	
GF4583Q	Ascent tank No. 2 water quantity	94.78%		WATER QTY caution light
GF4583Q	Ascent tank No. 2 water quantity			WATER QTY caution light
GF4583Q	Ascent tank No. 2 water quantity	0% to 100% (R)	0% to 100%	H ₂ O QUANTITY indicator
GF8507V	Cabin pressure logic out voltage	0 to 28 vdc		None
GF8509V	PRESS REG A and B valve switch test voltage	0 to 28 vdc		None
GF9993U	LiOH select/high CO ₂ advisory	Contact closure		CO ₂ component caution light
GF9997U	Selected pump discharge pressure	0 to 60 psia (R)	0 to 80 psia	None
GF9997U	Selected pump discharge pressure	0 to 80 psia		GLYCOL PRESS indicator
GF9998U	Selected coolant loop temperature	+20 ⁰ to +120 ⁰ F (R)	+20 ⁰ to +120 ⁰ F	None
GF9998U	Selected coolant loop temperature	+50 ⁰ F	+49.98 ⁰ F	GLYCOL caution light
GF9998U	Selected coolant loop temperature	0 ⁰ to +80 ⁰ F	0 ⁰ to +80 ⁰ F	GLYCOL TEMP indicator
GF9999U	Selected water separator rate	600 to 3,500 rpm (R)	500 to 3,600 rpm	None
GF9999U	Selected water separator rate	800 rpm		ECS caution light
GF9999U	Selected water separator rate			H ₂ O SEP component caution light
JF8521V	Cabin fan No. 1 voltage	0 to 15 vdc		None
JF8522V	Cabin fan No. 1 voltage	0 to 15 vdc		None
JF8523V	Cabin fan No. 2 voltage	0 to 15 vdc		None
JF8524V	Cabin fan No. 2 voltage	0 to 15 vdc		None
JF8535V	Suit fan No. 2 voltage	0 to 15 vdc		None
JF8536V	Suit fan No. 1 voltage	0 to 15 vdc		None

ENVIRONMENTAL CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.7 COMMUNICATIONS SUBSYSTEM.

2.7.1 INTRODUCTION.

The Communications Subsystem (CS) provides the communications links between the LM and the Manned Space Flight Network (MSFN) and between the LM and CSM. Communications with the Extra Vehicular Astronaut (EVA) can also be accomplished via the CS; however EVA activity other than CSM/LM crew transfer is not planned for the LM 4 mission.

The following types of information can be handled by the communications links:

- Tracking and ranging
- Voice (LM/EVA, LM/CSM or LM/MSFN)
- PCM telemetry (LM status)
- Astronaut biomedical data
- LGC update
- Emergency key (Morse code)
- Television (not planned for the LM 4 mission)
- EVA/LM EMU data
- LM/CSM telemetry data (for retransmission to MSFN)

2.7.2 SUBSYSTEM INTERFACES. (See figure 2.7-1.)

The CS interfaces with the Electrical Power Subsystem (EPS) for primary a-c and d-c power. D-C operating power (28 volts) is supplied to the CS from the Commander's and LM Pilot's buses; a-c power (117 volts, 400 cps), from a-c bus B. Generally, redundant CS operating equipment is connected to separate d-c buses, thereby assuring an operational status if either primary d-c bus becomes inoperative.

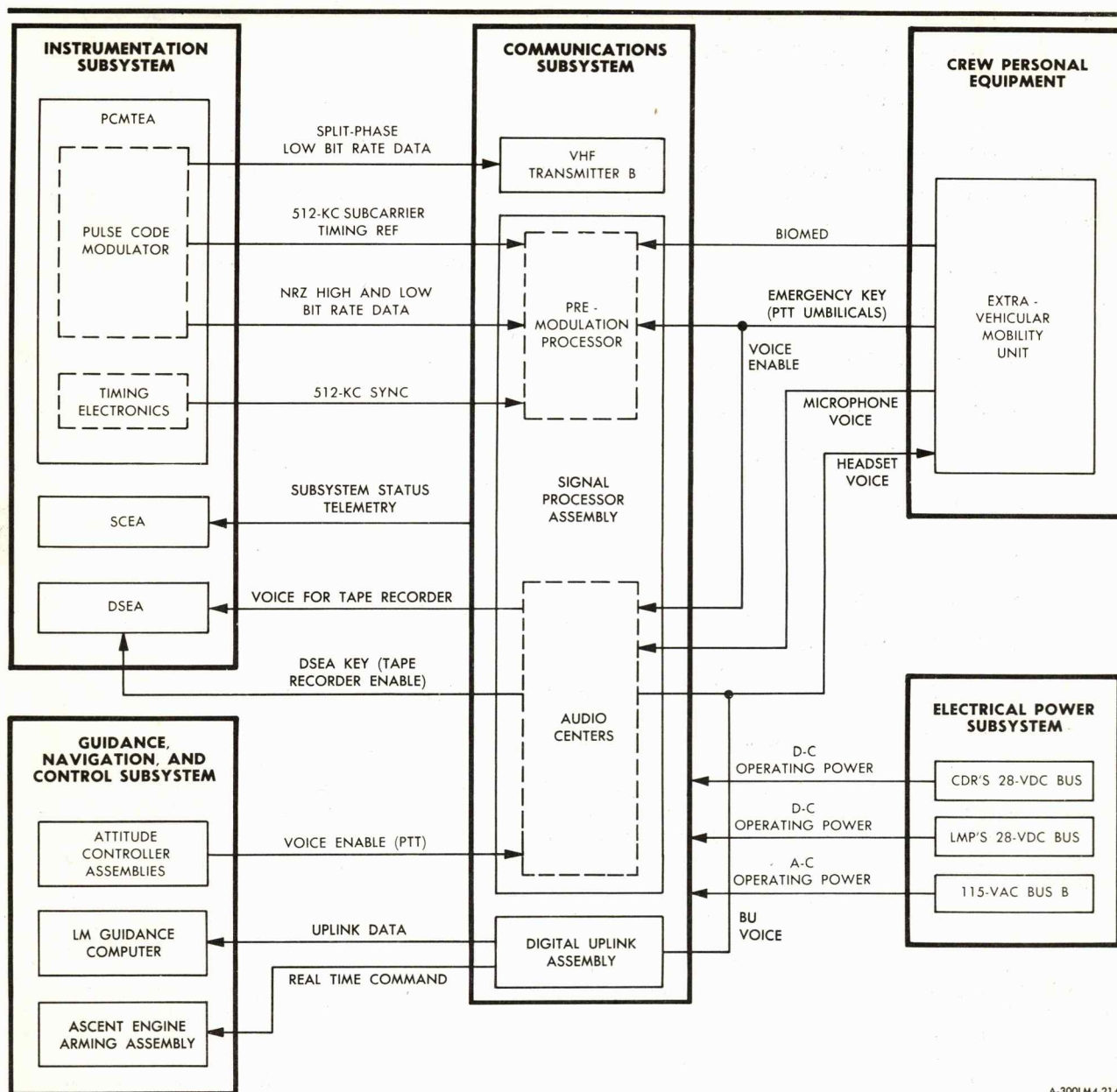
Telemetry data are routed to the CS from the Instrumentation Subsystem (IS) in nonreturn-to-zero (NRZ) form at a high or low bit rate. These data are fed through the S-band network for transmission to MSFN. When line of sight with MSFN is lost, these data are fed through VHF transmitter B to the CSM, but VHF B uses only low bit split phase data format. The IS supplies two 512-kc signals to the premodulation processor (PMP), a part of the signal processing assembly (SPA). One signal is the 512-kc synch signal from the timing electronics equipment (TE), a part of the pulse-code-modulation and timing electronics assembly (PCMTEA). This signal is used for subsequent generation of the 1.25-mc subcarrier. The other signal is the 512-kc subcarrier timing reference signal from the PCMTEA which is used to generate the 1.024-mc subcarrier. The data storage electronics assembly (DSEA) of the IS provides a voice recording capability. The DSEA receives its input from either audio centers' internal communications system (ICS) bus.

The Commander's and LM Pilot's attitude controller assemblies (ACA's) of the Guidance, Navigation, and Control Subsystem (GN&CS) each have a push-to-talk (PTT) switch for enabling the SPA audio centers permitting communications to commence. A PTT pushbutton is also located on each astronaut's electrical umbilical for enabling the SPA audio centers permitting voice communications. These PTT pushbuttons also provide an enabling signal for the PMP in the SPA for emergency keying if the emergency key mode is selected.

Voice signals are generated and received in the astronaut's communications carrier (headset) and are wired to the PMP for appropriate routing. Biomedical sensors in the astronaut's pressure garment assembly (PGA) continuously generate electrical analogs pertinent to heart beat. This signal is not routed through the IS, it is sent directly to the PMP for processing and subsequent transmission to the MSFN.

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



A-300LM4-214

Figure 2.7-1. Communications Subsystem - Interface Diagram

2.7.3 FUNCTIONAL DESCRIPTION (See figure 2.7-2.)

2.7.3.1 General.

The CS includes all S-band, VHF, and signal-processing equipment necessary to transmit and receive voice, and tracking and ranging data, and to transmit telemetry and emergency keying. Voice communications between the LM and MSFN are provided by the S-band transmitter-receiver assemblies, and between the LM and CSM by VHF transmitter-receiver assemblies. Telemetry data are transmitted to MSFN via S-band, which also provides the ranging and tracking functions. The communications links provided by the CS, and their functions, are listed in table 2.7-1.

COMMUNICATIONS SUBSYSTEM

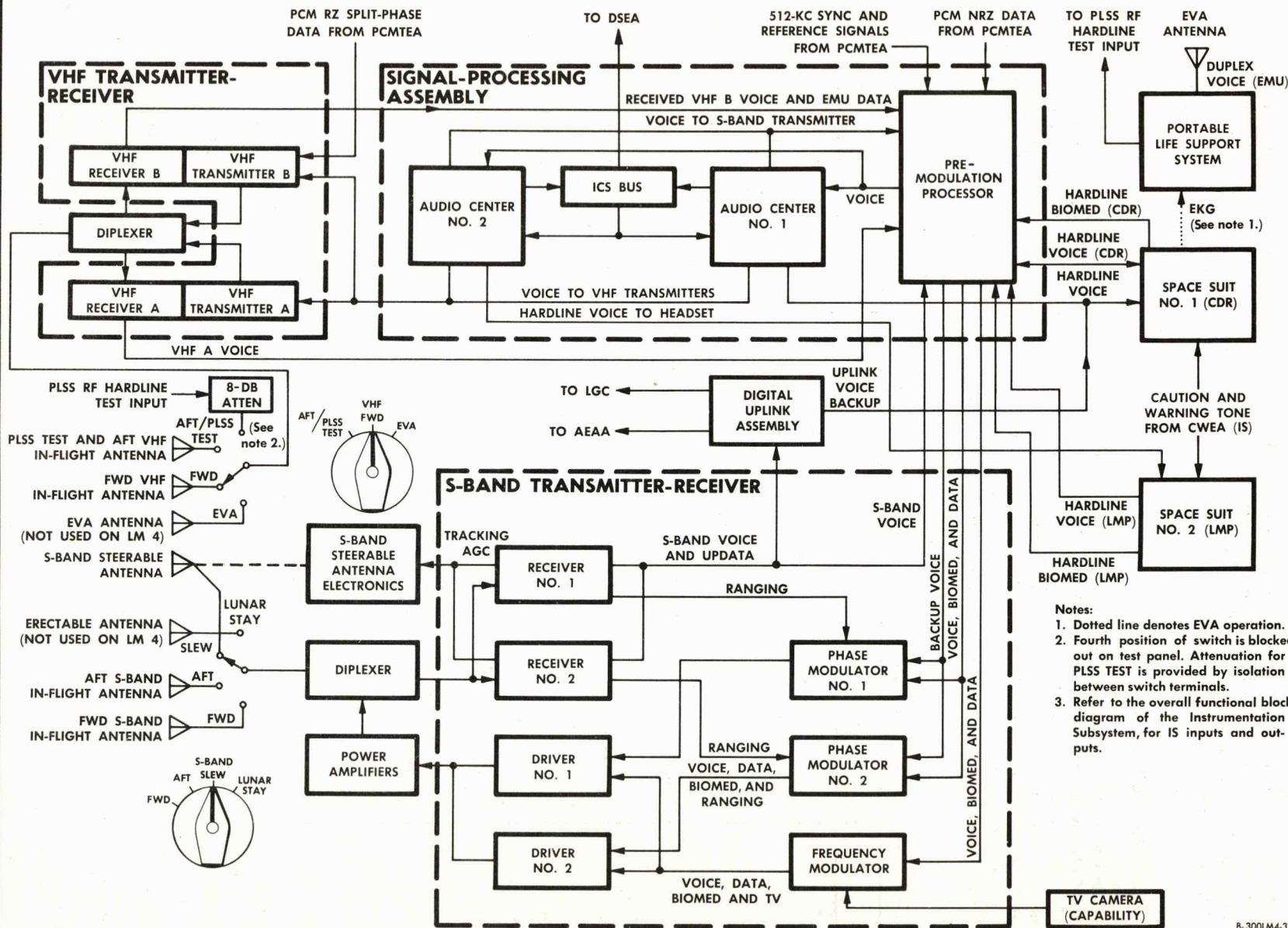


Figure 2.7-2. Communications Subsystem - Simplified Block Diagram

LMA 790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The Digital Uplink Assembly (DUA) updates the LM Guidance Computer (LGC) and sends a real-time command to the Ascent Engine Arming Assembly (AEAA). It receives data in a serial digit format, decodes the data, and transfers the decoded data to the LGC and the AEAA. The DUA also provides S-band uplink voice backup capability of the received S-band audio circuits in the PMP fail.

2.7.3.2 Power Distribution.

The CS receives primary electrical power from the Electrical Power Subsystem (EPS): d-c power from the Commander's and LM Pilot's buses; a-c power from the a-c bus. The normal d-c voltage requirement for all major CS assemblies is 28 volts, two-wire, negative ground. The required steady-state d-c range is 25 to 31.5 volts; the off-nominal range, 20 to 32 volts for periods not exceeding 5 seconds. The CS requires 115-volt, 400-cps power for the S-band steerable antenna electronics package. All operating power is supplied to the CS through circuit breakers. Figure 2.7-3 shows all EPS circuit breakers relevant to the CS, major CS assemblies, and related power switches (panel 12).

Table 2.7-1. Communications Links

Link	Mode	Band	Purpose
MSFN-LM-MSFN	Pseudorandom noise	S-band	Ranging and tracking
LM-MSFN	Voice	S-band, VHF (optional)	In flight communications
LM-CSM	Voice	VHF simplex	In-flight communications
CSM-LM-MSFN	Voice	VHF and S-band	Conference (with LM as relay)
LM-CSM	Low-bit-rate telemetry	VHF (one way)	Record and retransmit to earth
MSFN-LM	Voice	S-band, VHF (optional)	In-flight communications
MSFN-LM	Uplink data or uplink voice backup	S-band	Update LGC or voice backup for in-flight communications
LM-MSFN	Biomed-PCM telemetry	S-band	In-flight communications
LM-MSFN-CSM	Voice	S-band or VHF	Conference (with earth as relay)
LM-EVA-LM*	Voice and data	VHF duplex	EVA direct communication
EVA-LM-MSFN*	Voice and data	VHF, S-band	Conference (with LM as relay)
EVA-LM-CSM*	Voice and data	VHF duplex	Conference (with LM as relay)
CSM-MSFN-LM-EVA*	Voice and data	S-band, VHF	Conference (via MSFN-LM relay)

* Communications links not planned for in LM 4 mission. Two types of voice communication links are used in the CS: S-band and VHF. All voice links are line-of-sight (LOS); both ends of the communications path must be in unobstructed line of sight to each other.

The S-band transponder receives coded PRN signals from MSFN. These signals are routed to the selected PM transmitter for in-phase-coherence transmission back to MSFN. S-band transmission is at 2282.5 mc; reception at 2101.8 mc.

The tracking and ranging function uses the Doppler frequency shift caused by LM high velocity rates, to track the LM. The coded PRN signal transmitted by MSFN is received by a receiver with a phase-locked loop, and retransmitted to MSFN. The retransmitted signal received by MSFN is compared with the originally transmitted signal. Range is determined by the time it takes for the signals to travel from MSFN to the LM and back to MSFN. Once the range has been established accurately by the PRN code, the code is discontinued and range is updated continuously by the Doppler technique. The phase-locked loop in the LM receiver ensures accuracy of the tracking signals (Doppler) by comparing the phase of the received signal with the phase of a local-oscillator output in the detector stage. (See figure 2.7-7.) The output voltage of the detector, which is proportional to the phase difference between the two signals, is applied to the local oscillator. This signal is applied as a control voltage to alter the local-oscillator frequency to bring it in phase with the received MSFN signal.

2.7.3.6 Subsystem Operation.

The normal switch and circuit breaker positions for the communications modes are listed in table 2.7-2. If a switch or circuit breaker is not listed for a particular mode, that mode is not influenced by that switch or circuit breaker. The switch positions listed are for prime positions. In some cases, more than one position functions as a prime position; all such positions are indicated.

2.7.4 MAJOR COMPONENT/FUNCTIONAL DESCRIPTION.

The CS is divided into three groups: RF electronic equipment, signal processing equipment, and antenna equipment. The items included in each group are listed in table 2.7-3.

2.7.4.1 RF Electronic Equipment.

The RF electronic equipment includes all subsystem equipment required for transmission and reception of audio, PRN, and telemetry data. The antennas mentioned in the discussion of the RF electronic equipment are covered in paragraph 2.7.4.3. The CS can transmit video, but this function is not used in LM 4.

2.7.4.1.1 S-Band Transmitter-Receiver Assembly. (See figure 2.7-5.)

The S-band transmitter-receiver assembly provides deep-space communications between the LM and MSFN. S-band communications consist of voice and PRN transmission from MSFN to the LM, and voice, PRN turnaround, biomed, and subsystem data transmission from the LM to MSFN. The S-band communications capabilities are listed in table 2.7-4.

The S-band transmitter-receiver assembly consists of two identical phase-locked receivers, two phase modulators with driver and multiplier chains, and a frequency modulator. The receivers and phase modulators provide the ranging, voice, emergency-keying, and telemetry transmit-receive functions. The frequency modulator provides limited backup for both phase modulators. The operating frequencies of the S-band equipment are 2282.5 mc (transmit) and 2101.8 mc (receive).

The frequency modulator (figure 2.7-6) uses a voltage-controlled oscillator (VCO), operating at 75 mc, as the basis of its RF circuit. The FM transmitter is modulated by an input within a 10-cps to 1.5-mc range. A portion of the VCO output is routed through a series of frequency divider networks that reduce the frequency to 9.2 kc. This frequency is compared to the output (also reduced to 9.2 kc) of a 4.7-mc crystal oscillator, in a phase detector. The two inputs to the phase detector are normally 90° out of phase. The phase difference is represented as a voltage at the phase detector output and is used to control VCO frequency stability. The frequency modulator output is then supplied to the selected driver and multiplier chain for transmission through the S-band power amplifier, diplexer, and selected S-band antenna to MSFN.

COMMUNICATIONS SUBSYSTEM

Table 2.7-2. Communications Subsystem - General Switch and Circuit Breaker Configuration

Switches and Circuit Breakers	S-Band Voice	PRN Ranging	S-Band Telemetry	Hard-line-Biomed	S-Band Voice BU (Downlink)	Emergency Key	TV	VHF A Sim-plex	VHF B Sim-plex	VHF B Telemetry	VHF A/B Duplex (LM/CSM)
Panel 8 (AUDIO)											
S-BAND T/R switch	S-BAND T/R	OFF	OFF	OFF	S-BAND T/R	OFF	OFF	OFF	OFF	OFF	OFF
ICS T/R switch	ICS T/R or OFF	OFF	OFF	OFF	ICS T/R or OFF	OFF	OFF	ICS T/R or OFF	ICS T/R or OFF	OFF	ICS T/R or OFF
VHF A switch	OFF	OFF	OFF	OFF	OFF	OFF	OFF	T/R	OFF	OFF	T/R
VHF B switch	OFF	OFF	OFF	OFF	OFF	OFF	OFF	OFF	T/R	OFF	RCV
RELAY ON switch (See note 1.)	RELAY OFF	OFF	(See note 1. Only one RELAY ON switch can be set to RELAY ON at a time.)					OFF	OFF	OFF	OFF
MODE switch (See note 2.)	VOX	OFF	OFF	OFF	OFF	OFF	OFF	VOX	VOX	OFF	VOX
AUDIO CONT switch	NORM	OFF	OFF	OFF	NORM	OFF	OFF	NORM	NORM	OFF	NORM
Panel 12 (AUDIO)											
S-BAND T/R switch	S-BAND T/R	OFF	OFF	OFF	S-BAND T/R	OFF	OFF			OFF	
ICS T/R switch	ICS T/R or OFF	OFF	OFF	OFF	ICS T/R or OFF	OFF	OFF	ICS T/R or OFF	ICS T/R or OFF	OFF	ICS T/R or OFF
VHF A switch	OFF	OFF	OFF	OFF	OFF	OFF	OFF	T/R	OFF	OFF	OFF
VHF B switch	OFF	OFF	OFF	OFF	OFF	OFF	OFF	OFF	T/R	OFF	RCV
RELAY ON switch (See note 1.)	RELAY OFF	OFF	(See note 1. Only one RELAY ON switch can be set to RELAY ON at a time.)					OFF	OFF	OFF	OFF
MODE switch (See note 2.)	VOX	OFF	OFF	OFF	VOX or PTT	OFF	OFF	VOX	VOX	OFF	VOX
AUDIO CONT switch	NORM	OFF	OFF	OFF	NORM	OFF	OFF	NORM	NORM	NORM	NORM

Table 2.7-2. Communications Subsystem - General Switch and Circuit Breaker Configuration (cont)

Switches and Circuit Breakers	S-Band Voice	PRN Ranging	S-Band Telemetry	Hard-line Biomed	S-Band Voice BU (Downlink)	Emergency Key	TV	VHF A Sim-plex	VHF B Sim-plex	VHF B Telemetry	VHF A/B Duplex (LM/CSM)
Circuit Breaker Panel 16											
COMM: SE AUDIO	Closed	Open	Open	Closed	Closed	Open	Open	Closed	Closed	Open	Closed
COMM: VHF A XMTR		Open	Open			Open	Open	Closed	Open	Open	Closed
COMM: VHF B RCVR		Open	Open			Open	Open	Open	Closed	Open	Closed
COMM: PRIM S-BD PWR AMPL	Open	Open	Open	Open	Open	Open	Closed	Open	Open	Open	
COMM: PRIM S-BD XMTR/RCVR	Closed	Closed	Closed	Closed	Closed	Closed	Closed	Open	Open	Open	Open
COMM: S-BD ANT	Open	Open	Open	Open	Open	Open	Open	Open	Open	Open	Open
COMM: PMP	Closed	Open	Closed	Closed	Closed or Open	Open	Closed or Open	Closed	Closed	Open	Closed
COMM: TV	Open	Open	Open	Open	Open	Open	Closed	Open	Open	Open	Open
COMM: DISP	Closed	Closed	Closed	Closed	Closed	Closed	Closed	Closed	Closed	Closed	Closed
Push-to-Talk (PTT) Switches (See note 4.)											
Commander's attitude controller assembly PTT switch											
LM Pilot's attitude controller assembly PTT switch											
Commander's electrical umbilical PTT switch						PTT switch used as keyer					
LM Pilot's electrical umbilical PTT switch						PTT switch used as keyer					

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-2 Communications Subsystem - General Switch and Circuit Breaker Configuration (cont)

NOTES:

1. The normal positions for switches on the AUDIO portion of panels 8 and 12 for relay modes, are as follows:

- CSM-LM VHF A simplex, relayed to MSFN via S-band

Panel 8:

RELAY ON switch - RELAY ON
VHF A switch - T/R
VHF B switch - OFF
S-BAND T/R switch - OFF or S-BAND RCV
MODE switch - VOX

Panel 12:

RELAY ON switch - RELAY OFF
VHF A switch - OFF or RCV
VHF B switch - OFF
S-BAND T/R switch - S-BAND T/R
MODE switch - VOX

- or -

Panel 8:

RELAY ON switch - RELAY OFF
VHF A switch - OFF or RCV
VHF B switch - OFF
S-BAND T/R switch - S-BAND T/R
MODE switch - VOX

Panel 12:

RELAY ON switch - RELAY ON
VHF A switch - T/R
VHF B switch - OFF
S-BAND T/R switch - OFF or S-BAND RCV
MODE switch - VOX

- or -

- LM, CSM, MSFN voice conference
with LM relaying CSM voice to MSFN

Normal

Panel 8:

AUDIO CONT switch - NORM
RELAY ON switch - RELAY OFF
VHF A switch - T/R
VHF B switch - RCV
S-BAND T/R switch - S-BAND T/R
MODE switch - VOX

COMMUNICATIONS SUBSYSTEM

LMA 790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-4. S-Band Communications Capabilities

Information	Frequency or Rate	Subcarrier Modulation	Subcarrier Frequency	RF Carrier Modulation
UPLINK: 2101.8 mc				
Voice	300 to 3,000 cps	FM	30 kc	PM
Voice backup	300 to 3,000 cps	FM	70 kc	PM
PRN ranging code	990.6 kilobits/sec			PM
Digital data	1.0 kilobits/sec	FM	70 kc	PM
DOWNLINK: 2282.5 mc				
Voice	300 to 3,000 cps	FM	1.25 mc	PM or FM
Biomed	14.5 kc	FM	1.25 mc	PM or FM
EMU	4 to 12.4 kc (7 sub-carriers)	FM	1.25 mc	PM or FM
Voice	300 to 3000 cps	None	None	Direct PM baseband modulation
Biomed	14.5 kc	None	None	Direct PM baseband modulation
EMU	4 to 12.4 kc (7 sub-carriers)	None	None	Direct PM baseband modulation
Voice Backup	300 to 3,000 cps	None	None	Direct PM baseband modulation
PRN ranging code (turnaround)	990.6 kilobits/sec			PM
Emergency Keying	Morse code	AM	512 kc	PM
PCM NRZ Data	High bit rate: 512 kilobits/sec Low bit rate: 1.6 kilobits/sec	Phase Shift Keying (PSK)	1.024 mc	PM or FM
TV (not used during the LM 4 mission)	10 cps to 500 kc			FM baseband modulation

CS

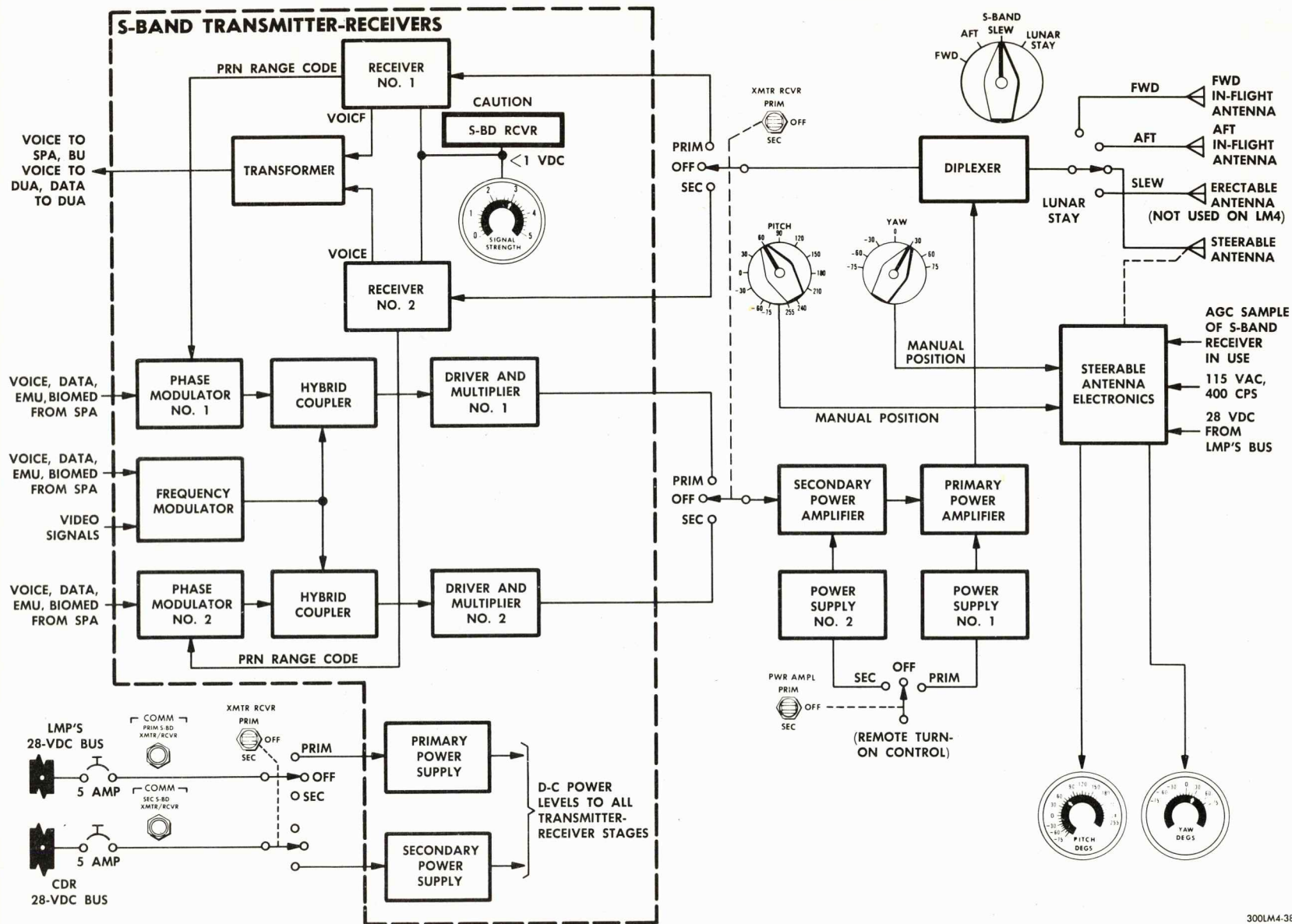


Figure 2.7-5. S-Band Transmitter-Receiver Assembly-Block Diagram

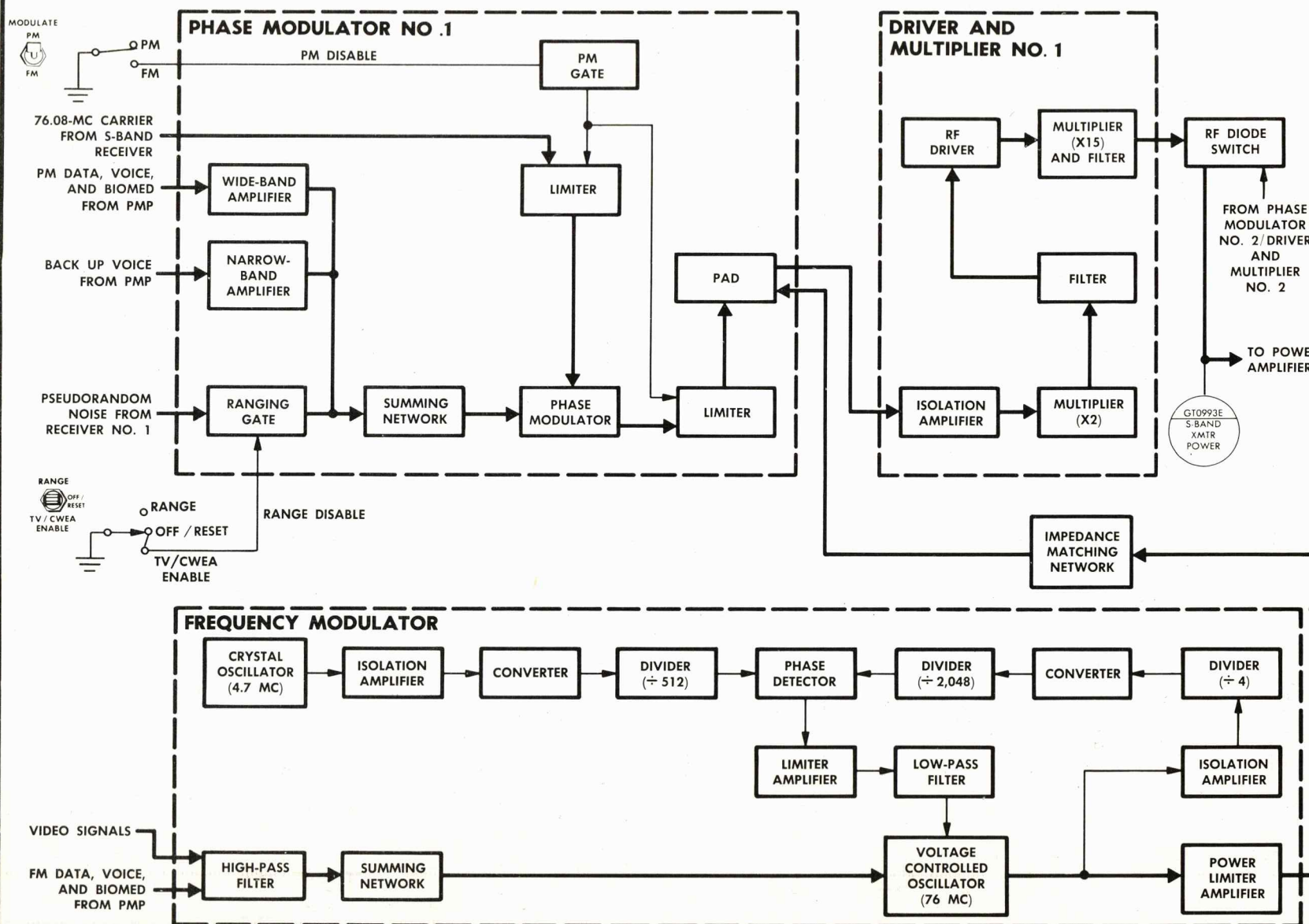


Figure 2.7-6. S-Band Transmitter-Block Diagram

300LM4-39

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The prime purpose of the phase modulator (figure 2.7-6) during deep-space missions is to provide all in-flight and lunar-stay communications with MSFN.

The phase modulator uses the output of the 19-mc VCO in the S-band receiver as the basis of its RF circuit. The VCO output is multiplied to approximately 76 mc in the receiver and routed to the phase modulator, where it is modulated by wide- or narrow-band information supplied from the SPA or by the PRN signal received from MSFN. The 76-mc carrier is then routed through a series of multipliers and drivers, where it is multiplied to the transmit carrier frequency (2282.5 mc).

The S-band receivers (figure 2.7-7) receive voice uplink data and coded PRN signals from MSFN. One receiver serves as backup for the normally used unit. The S-band receivers receive a 2101.8-mc carrier which is phase-modulated by the 30- and 70-kc subcarriers and the PRN ranging code. The output of the 19-mc VCO is routed through a chain of multiplier stages, where it is multiplied to 2054.3 mc, and heterodyned with the input carrier to obtain an output frequency of 47.5 mc from the first mixer. The first mixer output is mixed with 38 mc, provided by the VCO through a multiplier stage, yielding a second intermediate frequency of 9.5 mc. The second IF provides several narrow-band outputs, including automatic gain control (AGC) to the first IF, tracking AGC (used to position the steerable antenna), a signal-strength signal to the SIGNAL STRENGTH indicator (panel 12), and AGC-monitoring and static phase error signals that are supplied to the IS, where they are processed for telemetering. The wide-band signal outputs include the 30-kc voice subcarrier (routed to the PMP), the 70-kc data subcarrier (routed to the DUA) and the PRN ranging signal (routed to the S-band phase modulator).

Another function of the S-band receivers is to provide an output for use as the exciter frequency for the phase modulators. The 19-mc VCO output is routed through a chain of multiplier stages, where it is multiplied to 76 mc for use by the phase modulators as the base of the phase-coherent downlink carrier.

2.7.4.1.2 S-Band Power Amplifier. (See figure 2.7-8.)

The S-band power amplifier amplifies the S-band transmitter output when additional transmitted power is required. This assembly consists of two amplitrons, an input and an output isolator (ferrite circulators), and two power supplies, all mounted on a common chassis. The RF circuit is a series interconnection of the isolators and amplitrons. The amplitrons (which are characteristic of saturated, rather than linear, amplifiers) have broad bandwidth, high efficiency, high peak and average power output, but low gain. The isolators protect both amplitrons and both S-band transmitter driver and multiplier chains. The isolators exhibit a minimum isolation of 20 db and a maximum insertion loss of 0.6 db.

Each amplitron is supplied by its own power supply. One amplitron is designated primary; the other, secondary. Only one amplitron can be activated at a time. Selection is made with the PWR AMPL switch (panel 12). When neither amplitron is selected, a feedthrough path through the power amplifier is provided, with an insertion loss of 3.2 db, maximum (feedthrough mode).

When an amplitron is selected, 28 volts dc is supplied to its associated power supply. When 0.75-watt RF power is supplied to the power amplifier input from an S-band transmitter driver and multiplier stage, the minimum output RF power is 18.6 watts through the primary stage or 14.8 watts through the secondary stage at the 2282.5-mc transmit frequency into a 50 ohm resistive load with a VSWR of 1.1 to 1 or better. The slightly degraded power output through the secondary stage is caused by the added attenuation of the primary stage in series with the secondary stage.

2.7.4.1.3 S-Band Diplexer and RF Selector Switch.

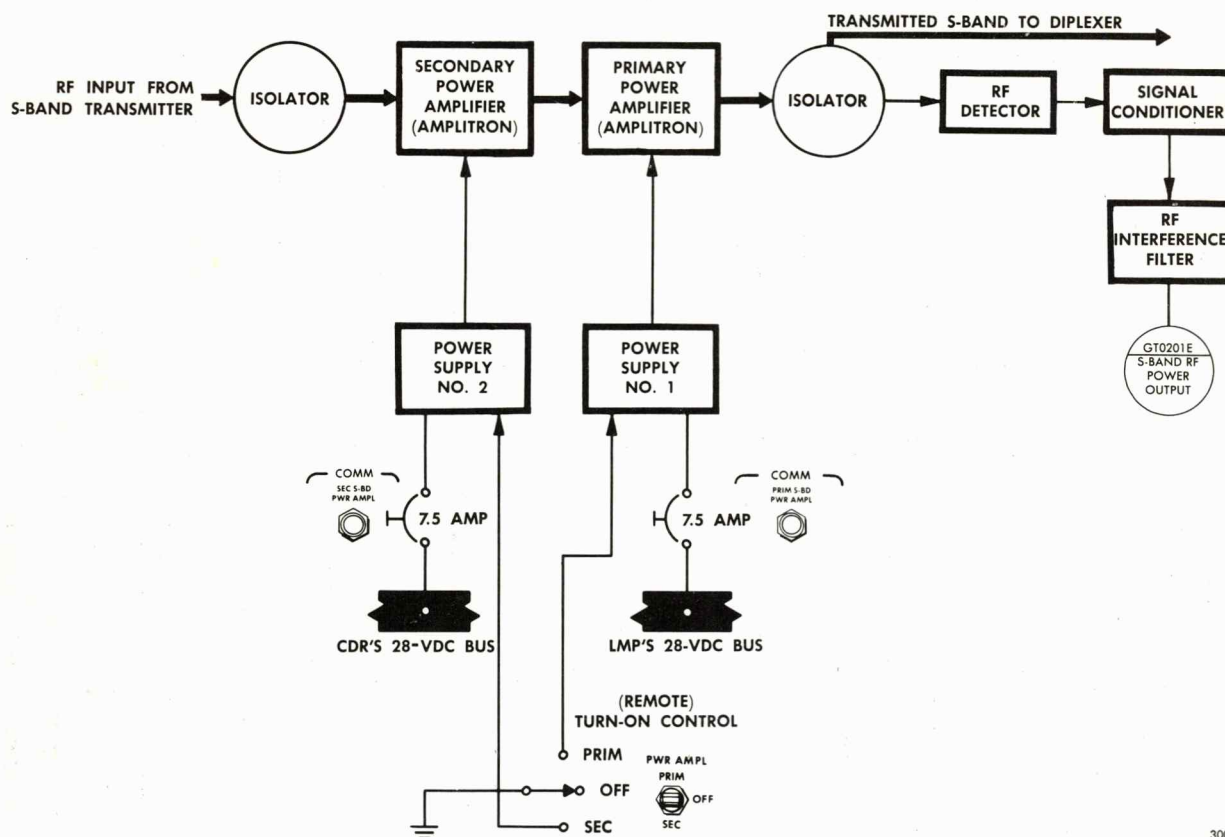
The S-band diplexer, connected directly to the output of the S-band power amplifier, permits simultaneous duplex transmission and reception, using one antenna. Insertion loss is 0.9 db (transmit) and 1.3 db (receive).

The S-BAND selector switch on the COMMUNICATIONS ANTENNAS portion of panel 12 enables selection of either S-band in-flight antenna (omnidirectional), the steerable antenna, or the erectable antenna (during lunar stay only). The insertion loss of the switch, in the transmit mode of operation (2282.5 mc), is less than 0.2 db.

COMMUNICATIONS SUBSYSTEM

Figure 2.7-7. S-Band Receiver - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



300LM4-41

Figure 2.7-8. S-Band Power Amplifier - Block Diagram

2.7.4.1.4 VHF Transmitter-Receiver Assembly. (See figure 2.7-9.)

The primary functions of the VHF transmitter-receiver assembly are to provide voice communications between the LM and the CSM and, during blackout of transmission to MSFN, to provide low-bit telemetry transmission from the LM to the CSM. When the LM mission profile includes extra vehicular activity, this equipment also provides EVA-LM voice communications, and reception of EVA biomed and suit data for transmission to MSFN over the S-band.

The VHF transmitter-receiver assembly consists of two solid-state superheterodyne receivers, two transmitters, a VHF diplexer, RF antenna selector switch, two in-flight antennas, and an EVA antenna (for lunar missions). One transmitter-receiver combination provides a 296.8-mc channel (channel A); the other, a 259.7-mc channel (channel B), for simplex or duplex voice communications. Channel B may also be used to transmit PCM data from the IS at the low bit rate and to receive biomed and suit data from the EVA during EVA-programmed missions.

The VHF transmitter (figure 2.7-10) derives voice modulation and carrier turn-on control signals from the audio centers in the SPA. Carrier turn-on is initiated by the VOX circuit or by operation of the PTT switches. One audio center is connected to the Commander's microphone circuit and to the modulation inputs of the VHF transmitters; the other audio center is similarly connected for the LM Pilot. The input to the channel B transmitter can be switched from voice to PCM transmission by setting the VHF B XMTR switch (panel 12) to DATA. The PCM signals are generated in the PCMTEA. Transmitter outputs are routed to the VHF selector switch (panel 12) and to the antenna via a diplexer that permits simultaneous operation of all VHF transmitters and receivers on a single antenna.

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Notes:

1. Fourth position of switch blocked out on panel. Attenuation for PLSS test is provided by isolation between switch terminals.
2. Refer to the overall functional block diagram of the Instrumentation Subsystem for IS inputs and outputs.

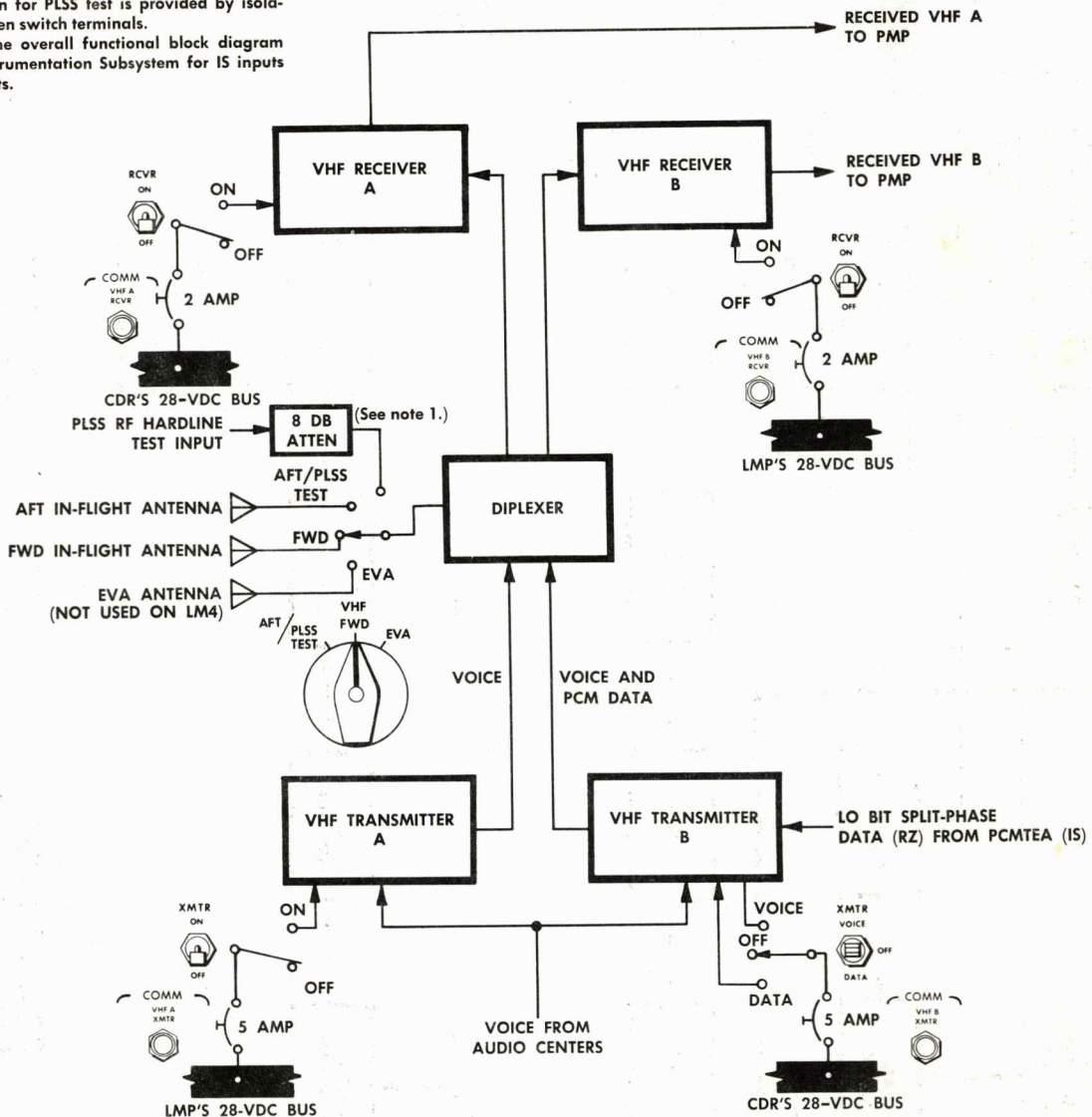


Figure 2.7-9. VHF Transmitter-Receiver Assembly - Block Diagram

The transmitter uses a keyed on-off carrier-type amplitude modulation and delivers 10 watts peak and 2.5 watts average RF power to the diplexer. Audio input is clipped and amplified by the chain of limiters and amplifiers in the modulator circuit. After clipping, the input signals appear as square-wave signals, which pulse the key amplifier circuit on and off. The resultant transmitter output is a pulse-shaped RF signal with 5 watts peak power. The 30-kc oscillator in the modulator continuously modulates the transmitter signal. This continuous modulation sustains the peak RF power output during pauses in speech, thereby maintaining the LM-CSM link so that the CSM receiver continues to hold its AGC level (remains quiet) during speech-pause periods. A 98.93-mc crystal oscillator is used as the basis of the RF circuit in transmitter A; an 86.56-mc crystal oscillator is used in transmitter B. The oscillator output frequency is tripled to the carrier frequency and is modulated by the output of the keyed amplifier.

COMMUNICATIONS SUBSYSTEM

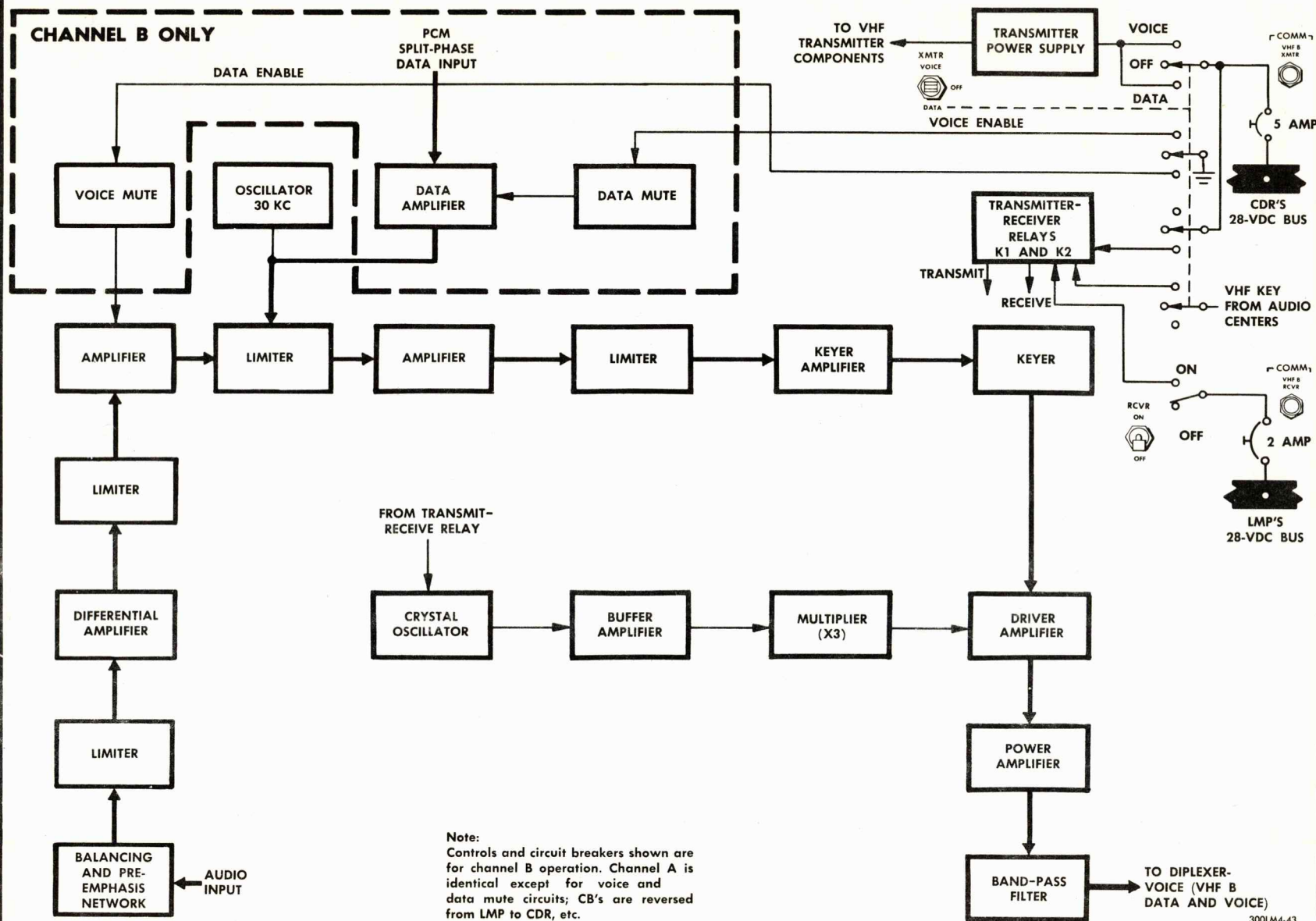


Figure 2.7-10. VHF Transmitter - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Channel B also provides PCM data transmission. Data inputs are clipped and amplified in the modulator the same as audio signals. This transmitter has a voice-data mute circuit, which is controlled with the VHF B XMTR switch.

The VHF receiver (figure 2.7-11) is an amplitude-modulation, fixed-tuned, single-conversion superheterodyne receiver that uses carrier-controlled squelch. There are two VHF receivers in the CS. The receivers can receive voice signals at 296.8 and 259.7 mc (channels A and B, respectively) and during EVA-programmed missions, biomed and suit data from the EVA via channel B. The frequency of a crystal-controlled-oscillator output is multiplied, and mixed with the received VHF signal in the mixer. The heterodyned output is a 30-mc IF signal, which is fed through a 30-mc crystal filter. The filter output is supplied to the AGC-controlled IF amplifiers, amplified, and routed to the PMP in the SPA. Squelch is carrier controlled and adjusted by the SQUELCH thumbwheels (panel 12).

2.7.4.1.5 VHF Diplexer and RF Selector Switch.

The VHF diplexer permits simultaneous transmission and reception, using one antenna at two frequencies. During in-flight phases of the mission, the diplexer permits voice transmission over one channel while reception is accommodated over the other channel. The normal LM-CSM communications mode is simplex over channel A; channel B is used for data transmission and voice backup.

The VHF selector switch on the COMMUNICATIONS ANTENNAS portion of panel 12 enables selection of either VHF in-flight antenna (omnidirectional) or the EVA antenna (during lunar stay). The insertion loss of the switch, in the transmit mode of operation (296.8 and 259.7 mc), is less than 0.2 db.

2.7.4.2 Signal-Processing Assembly. (See figure 2.7-12.)

The SPA is the common acquisition and distribution point for all CS received and transmitted data, except that low-bit-rate, split-phase data are coupled directly to VHF Channel B and TV signals are coupled directly to the S-band transmitter. The SPA processes voice and biomed signals and provides the interface between the RF electronics, DSEA, and PCMTEA. The SPA consists of an audio center for each astronaut and a PMP. The SPA does not handle PRN ranging and 70-kc uplink signals.

2.7.4.2.1 Premodulation Processor. (See figure 2.7-13.)

The PMP provides signal modulation, mixing, and switching in accordance with the selected mode of operation. The PMP also permits the LM to be used as a relay station between the CSM and MSFN, and for EVA-programmed missions, between EVA and MSFN.

The PMP accepts PCM NRZ data from the PCMTEA at 51,200 bits per second (high bit rate) or 1,600 bits per second (low bit rate). The data are routed to a biphase modulator, where the phase of a telemetry subcarrier frequency is controlled. Each logic-level change of the PCM data changes the biphase-modulator output by 180°. The biphase-modulator output is supplied to the PM or FM mixing network, depending on the selected mode, where it is combined with other signals processed in the PMP.

Two 512-kc clock signals are provided by the IS to the CS. A 1.024-mc telemetry subcarrier is generated from a 512-kc square wave (reference signal) from the pulse-code-modulation equipment of the PCMTEA. This 512-kc square wave is routed to a frequency doubler whose output is a 1.024-mc sine wave, phase-referenced to the PCM NRZ data. A 512-kc square wave (sync signal) received by the PMP, from the TE of the PCMTEA, is supplied to a tuned amplifier that supplies a 512-kc sine wave. This 512-kc signal is added to a 113-kc signal and the result is doubled to provide the 1.25-mc subcarrier. The biphase modulator accepts PCM NRZ data from the IS at the high or low bit rate. The input level of the data is 6 ± 0.5 volts for logic 1; 0 ± 0.5 volt for logic 0. The data modulates the 1.024-mc subcarrier. The biphase-modulator output reflects the change in logic level by a 180° change in subcarrier phase. The modulated subcarrier is routed through a 1.024-mc band-pass filter to the FM or PM mixing network.

Emergency keying is provided in the PMP by gating the 512-kc sync signal generated from the 512-kc square-wave sync input from the TE directly on the input line of the S-band phase modulator. The PTT switch on each electrical umbilical serves as the manually operated keyer to gate the output of the key gate circuit on and off. No 400 cycle keying sidetone is provided in the LM although such provision is made in the CSM.

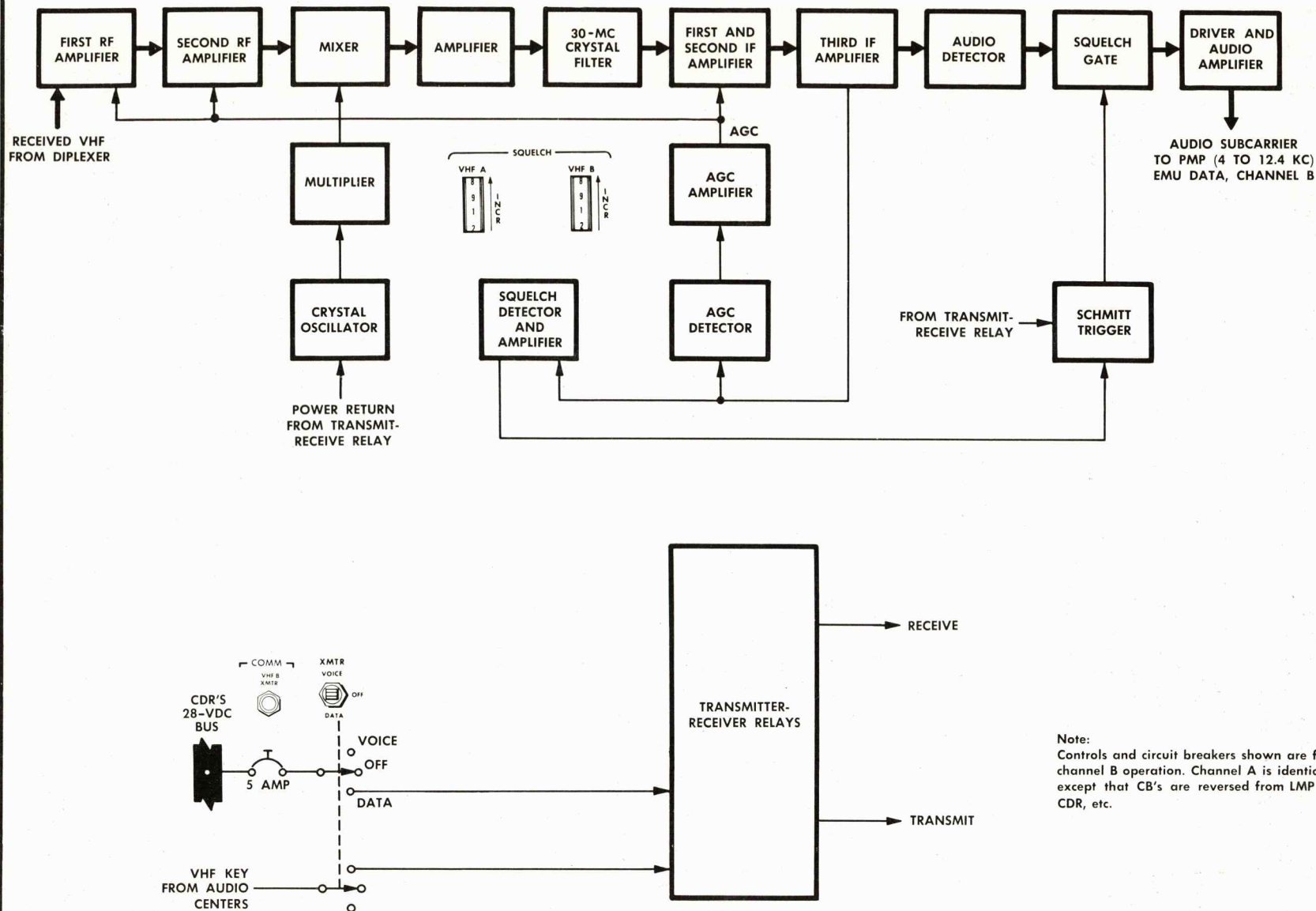


Figure 2.7-11. VHF Receiver - Block Diagram

Note:
Controls and circuit breakers shown are for
channel B operation. Channel A is identical
except that CB's are reversed from LMP to
CDR, etc.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Voice for S-band transmission is accepted by the PMP via a low-pass filter, which passes 300- to 3,000-cps signals. These voice signals are passed within ± 1 db referenced to 1,000 cps; signals above 4,000 cps are attenuated at least 20 db. The low-pass filter output modulates the 1.25-mc subcarrier or is routed directly to the S-band phase modulator for direct baseband modulation. The modulated 1.25-mc subcarrier is routed through tuned, isolation, and AGC amplifiers and supplied to the PM mixer or FM mixer.

Backup voice transmission is accommodated via S-band by routing the low-pass filter output directly to a speech-processing network. This network provides 24 db of speech clipping, referenced to 3,000 cps, and voice preemphasis of 6 db per octave between 300 and 3,000 cps. The speech-processing network output is supplied directly to the narrow-band PM input of the S-band equipment. This mode is attained by setting the VOICE FUNCTIONS switch (panel 12) to DN VOICE BU and the BIOMED TELEMETRY switch (panel 12) to OFF.

The PMP accepts hardline EKG biomed data from each astronaut and, by external control, selects one of these data for S-band transmission. The input is a 0- to 5-volt signal, at 0 to 30 cps. It frequency-modulates a 14.5-kc subcarrier oscillator, with a maximum deviation of 1 kc. The oscillator output is mixed with voice and the composite signal modulates the 1.25-mc subcarrier in the same manner as voice for S-band transmission, or is routed directly to the PM or FM mixer for direct baseband modulation.

Received S-band voice signals are routed through the PMP to the microphone and headset volume control circuits in the audio centers. A squelch control gate in the received S-band audio circuits establishes the threshold level of the incoming audio at a preset level. During normal voice communications the squelch control gate is turned on by setting the COMM: UPLINK SQUELCH switch (panel 14) to ENABLE. This allows incoming audio signals with an amplitude in excess of 0.25 volts rms to be amplified and monitored by the astronauts. Setting the COMM: UPLINK SQUELCH switch to OFF disables the squelch circuit. The VHF channel B input has high-pass and low-pass filters to separate voice and extravehicular mobility unit (EMU) data. The PMP has a discriminator to demodulate the S-band subcarrier. The discriminator output, which has a bandwidth of 300 to 3,000 cps ± 3 db referenced to 1,000 cps, is routed to the audio centers via a low-pass filter and mixer network and relay amplifiers.

The PM mixing network can process the outputs of the 1.25- and 1.024-mc channels (a composite of voice, biomed data, telemetry data, and EMU data). These outputs are then supplied to the selected S-band phase modulator for transmission to MSFN. The FM mixing network can process the outputs of the 1.024-mc data channel and a composite of voice, biomed data and EMU data, all on the 1.25-mc subcarrier. These outputs are then supplied to the S-band frequency modulator for transmission to MSFN. FM video (not processed during the LM 4 mission), PM ranging, backup voice, and emergency key signals are not processed through the mixing networks; they are supplied directly to the selected phase modulator for baseband modulation.

2.7.4.2.2 Audio Centers. (See figure 2.7-14.)

Two identical audio centers (one for each astronaut) in the SPA provide individual selection, isolation, and amplification of audio signals received by the CS receivers and of those which are to be transmitted by the CS transmitters. The audio centers are controlled from the AUDIO portion of panels 8 and 12. Each audio center contains a microphone amplifier, headset amplifier, VOX circuit, diode switches, volume control circuits, and isolation pads. The VOX circuit controls the microphone amplifier by activating it only when required for voice transmission.

Audio signals are routed to and from the VHF A, VHF B, and S-band equipments and the intercom bus via the audio centers. The intercom bus, common to both audio centers, provides hardline communications between the astronauts. Voice signals to be recorded by the DSEA are taken from the intercom bus.

Inputs and outputs are controlled by the VHF A, VHF B, S-BAND T/R, and ICS T/R (intercom) switches on panels 8 and 12. An associated thumbwheel (volume control) for each switch provides continuous adjustment of the received audio signal. Panels 8 and 12 also have VOX SENS and MASTER VOL thumbwheels. The VOX SENS thumbwheel permits continuous adjustment of the VOX control circuit threshold; the MASTER VOL thumbwheel, continuous adjustment of all audio outputs from the headset amplifier.

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Received audio signals are isolated and routed through the volume control circuits, which provide a 30-db range at the output of the headset amplifier. The four inputs (S-band, VHF A, VHF B, and intercom) routed through the headset circuit have minimum isolation of 40 db between any two inputs.

If one audio center fails, the astronaut affected by the malfunction can switch to the other audio center, using the AUDIO CONT switch (panels 8 and 12). (See figure 2.7-15.) With the switch set to NORM, the astronaut is using his own audio center; the BU position switches him to the other audio center.

2.7.4.2.3 Digital Uplink Assembly. (See figure 2.7-16.)

The DUA decodes S-band 70-kc digital uplink commands from MSFN and routes the data to the LGC and the AEAA. The LGC processes the data and routes a verification signal to the PCMTEA for transmission to MSFN, to indicate that the uplink commands have been processed by the LGC. The DUA also routes a verification signal to the PCMTEA for transmission to MSFN, to indicate that the uplink commands are properly coded and have been routed to selected LM equipment. The digital uplink commands addressed to the LGC parallel those inputs available to the LGC via the display and keyboard (DSKY). A real-time command from the DUA to the AEAA arms the ascent engine and switches control from primary guidance to abort guidance. The DUA also provides a voice backup capability if the received S-band audio circuits in the PMP fail.

The DUA consists essentially of a 70-kc discriminator, two audio amplifiers, a phase-shift-keyed (PSK) demodulator, two redundant decoders, and a power supply. The power supply converts the Commander's 28-volt d-c bus voltage to the regulated voltages required by the DUA. The power supply contains a dc-to-dc converter with regulators that provide 4.5 and 24 volts dc.

The DUA provides two modes of operation for uplink communications: data or voice backup. Selection of either mode is accomplished by the UP DATA LINK switch (panel 12).

Data Mode. Processing of digital uplinks commands is initiated by closing the COMM: UP DATA LINK circuit breaker (panel 11) and setting the UP DATA LINK switch to DATA. The 70-kc discriminator then receives the 70-kc subcarrier digital uplink commands from the S-band receiver. Each uplink command is composed of a 22-bit, serially coded message comprising three vehicle address bits, three system address bits (message identification), and 16 information bits.

The discriminator consists of two amplifier stages tuned to approximately 62 to 78 kc. These frequencies are the extremes of the FM deviation of the 70-kc digital uplink subcarrier. The demodulated output of the discriminator is a two-tone composite modulating signal consisting of a 2-kc subcarrier tone and a 1-kc coherent synchronization tone. The signal frequency-modulates the 70-kc subcarrier. The demodulated output is amplified by the data audio amplifier and applied to the PSK demodulator.

The PSK demodulator converts the signals from the 70-kc discriminator into a series of digital 1's and 0's for application to the two redundant decoders. The 2-kc subcarrier (which contains the uplink data) is phase-shift-keyed at a 1-kc binary rate. A subcarrier phase shift of 180° denotes a change in logic level. The synchronization tone is the same frequency as the binary bit command rate; zero crossings are coincident within ± 6 microseconds. The demodulator separates the two-tone composite signal. It detects the phase of the 2-kc subcarrier and provides data pulses, corresponding to 1 and 0 data bits, to the decoders. The two tones are used to time sequential clock pulses, which are used to strobe information into the decoders.

The decoders process the uplink data and at the direction of the uplink data, route information via an interface unit to the LGC. The decoders also generate an eight-bit verification word and route it via an interface unit to the PCMTEA, for telemetry, when command or data words are processed in the DUA and routed to the LGC. If the three vehicle address bits (of the 22-bit message) are not 011 or the system address bits are not recognizable, the DUA neither processes the uplink data nor generates the verification word.

Voice Backup Mode. The voice backup capability of the DUA is used if the PMP circuits that process the received S-band 30-kc audio subcarrier malfunction. If such failure occurs, MSFN can transmit S-band audio on the 70-kc uplink subcarrier. The DUA operates in the voice backup mode when the UP DATA LINK switch is set to VOICE BU. This supplies power to the voice backup circuits, enabling the 70-kc

COMMUNICATIONS SUBSYSTEM

B-300LM4-241

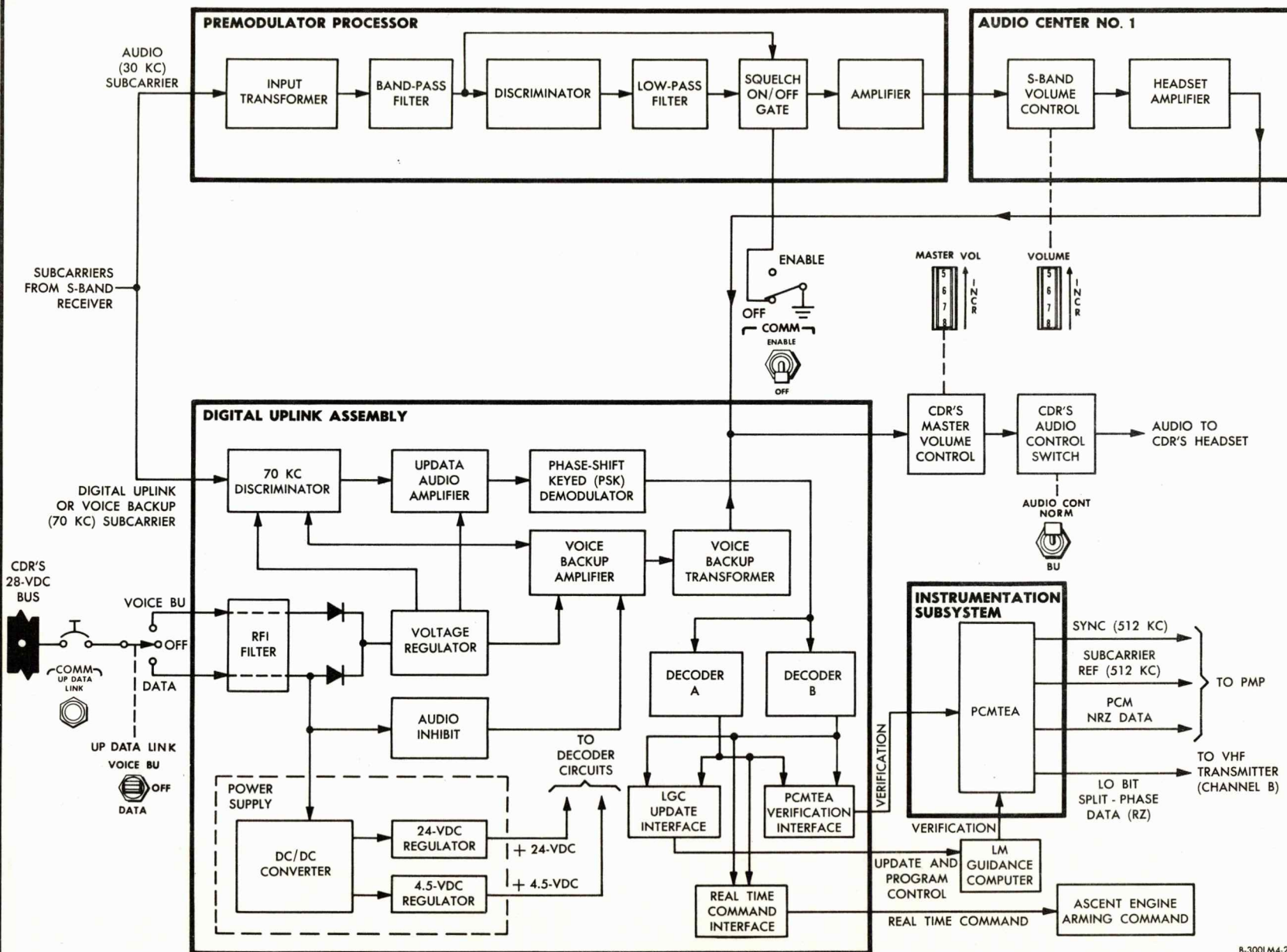


Figure 2.7-16. Digital Uplink Assembly - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

discriminator to demodulate the audio on the 70-kc subcarrier and to route the audio through the voice amplifier and transformer to each astronaut's headset, via the Commander's MASTER VOL control. When the DUA is in the data mode, an audio inhibit circuit supplies an inhibit signal to the voice amplifier. This prevents extraneous noise that may appear on the uplink subcarrier from being heard in the headsets.

2.7.4.3 Antenna Equipment. (See figure 2.7-17.)

The CS antenna equipment consists of two S-band in-flight antennas, an S-band steerable antenna, and two VHF in-flight antennas. For lunar missions, a high-gain, lunar-stay antenna for S-band communications and an EVA antenna for EVA-LM VHF communications will be provided.

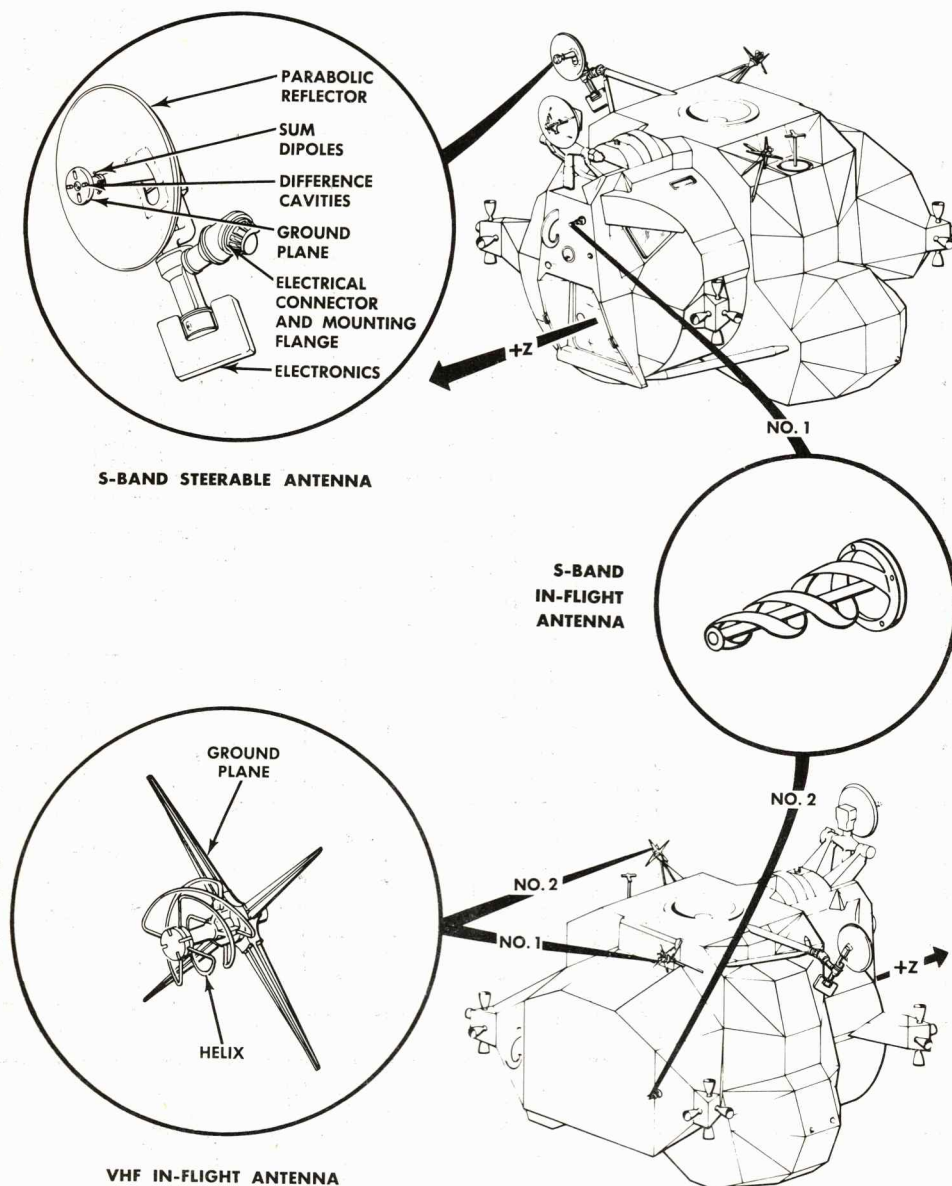


Figure 2.7-17. Communications Subsystem - Antenna Locations

COMMUNICATIONS SUBSYSTEM

2.7.4.3.1 S-Band In-Flight Antennas.

The S-band in-flight antennas are omnidirectional; one is forward and one is aft of the LM. The antennas are right-hand circularly polarized radiators that collectively cover 90% of the sphere at -3 db or better. They operate at 2282.5 ± 5 mc (transmit) and 2101.8 mc (receive). These antennas are the primary S-band antennas for LM 4.

2.7.4.3.2 S-Band Steerable Antenna. (See figures 2.7-18 and 2.7-19.)

The S-band steerable antenna is a 26-inch-diameter parabolic reflector with a point source feed that consists of a pair of cross-sleeved dipoles over a ground plane. The prime purpose of this antenna is to provide deep-space voice and telemetry communications and deep-space tracking and ranging. During earth-orbital missions, the steerable antenna will be checked and will serve as backup for the S-band in-flight antennas.

The S-band steerable antenna provides 174° azimuth coverage and 330° elevation coverage. The antenna can be operated manually or automatically. The manual mode is used for initial positioning of the antenna to orient it within $\pm 12.5^\circ$ (capture angle) of the LOS signal received from the MSFN station. Once the antenna is positioned within the capture angle, it can operate in the automatic mode. Switches and indicators on the COMMUNICATIONS ANTENNAS portion of panel 12 are used to operate the antenna.

When the S-BAND selector switch (panel 12) is set to SLEW (S-band steerable antenna selected) and the TRACK MODE switch (panel 12) is set to AUTO, the antenna tracks automatically. When in the automatic mode, signals received from a MSFN station cause the steerable antenna electronics to generate error information relative to the angular displacement and direction of the MSFN station LOS with respect to the existing antenna radiation axis. This information is processed and ultimately used to operate two servomotors that drive the antenna. One servomotor is mounted on the antenna X-axis; another, on the antenna Y-axis. Angular displacement data are generated by shifting the received signal pattern off the radiation axis in the X-direction and Y-direction, causing a change in strength of the received signal. If signal strength is increased when the antenna pattern is shifted, the MSFN LOS is displaced in the shifted direction; if signal strength is decreased, the MSFN LOS is displaced in the opposite direction. This periodic shifting of the antenna pattern permits generation of servo error signals by sampling the S-band receiver AGC level. The sampled AGC signal is supplied to the steerable antenna electronics subassembly and, through X- and Y-axis logic circuits, drives d-c servomotors to position the antenna to obtain the maximum received-signal level.

S-Band Steerable Antenna Heaters. Eight heater assemblies, each consisting of a heater and two (redundant) thermostats, are used to maintain the S-band steerable antenna temperature between -60° and $+155^\circ$ F. Four heater assemblies are installed in the electronics packages, and one each in the microwave network, X-axis and Y-axis gimbal housings, and X-axis gimbal spindle. The S-band steerable antenna heaters receive their power through the HEATERS. S-BD ANT circuit breaker (panel 16).

The temperature sensor in the S-band steerable antenna assembly routes the temperature signals to ERA-2 in the SCEA. The ERA-2 output is routed to the PCMTEA, CWEA, and the TEMP indicator (panel 3). The S-band steerable antenna temperature is monitored by setting the TEMP MONITOR selector switch to S-BAND and observing the TEMP indicator.

2.7.4.3.3 VHF In-Flight Antennas.

Two VHF in-flight antennas are used for VHF/AM communications. These are right-hand circularly polarized antennas that operate in the 259.7- to 296.8-mc range. One of these antennas is on each side of the LM, near the top of the structure.

COMMUNICATIONS SUBSYSTEM

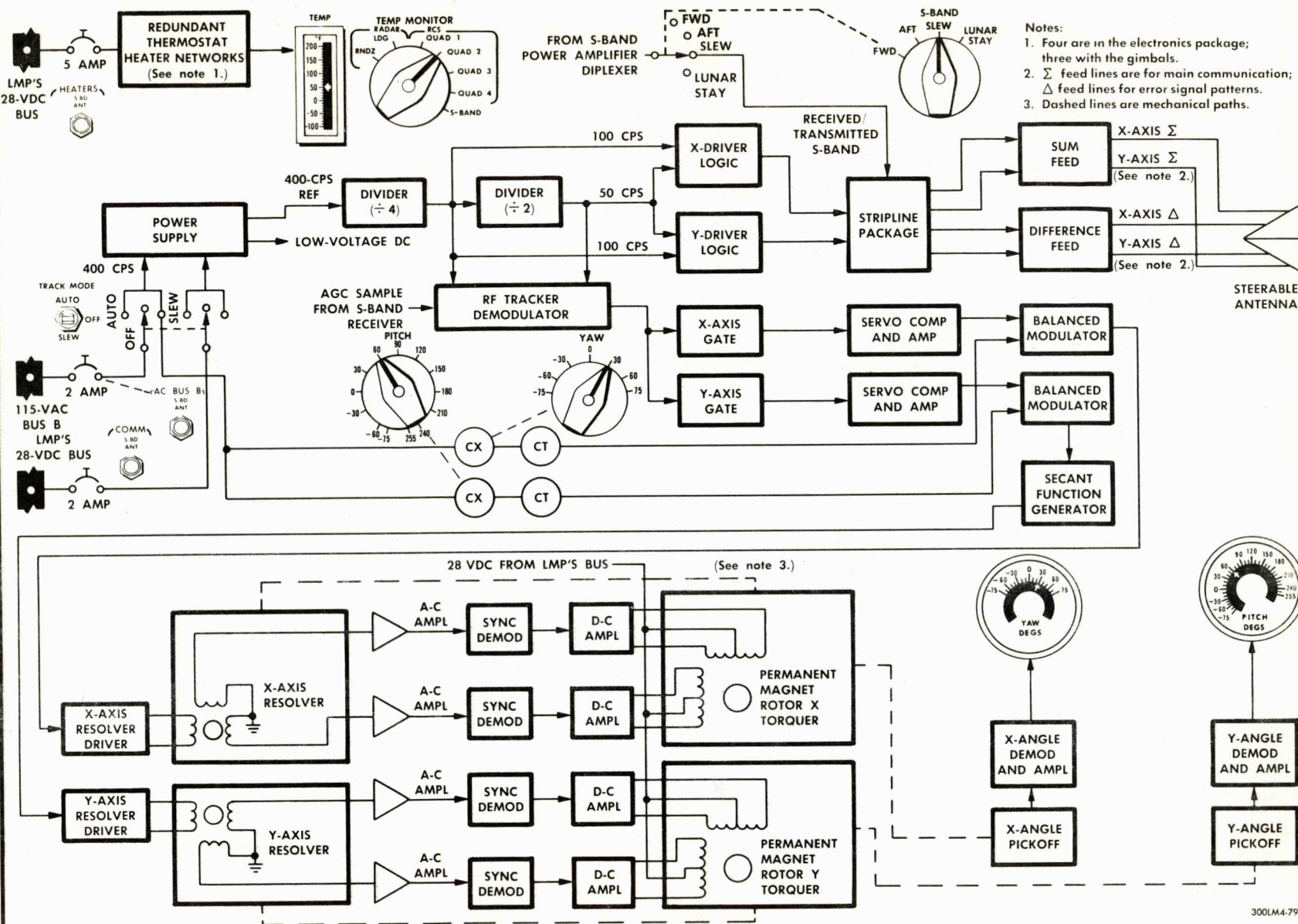


Figure 2.7-18. S-Band Steerable Antenna - Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEM DATA

2.7.5 FAILURE MODES.

The CS failure modes and the effect on mission capability are given in table 2.7-5.

2.7.6 PERFORMANCE AND DESIGN DATA.

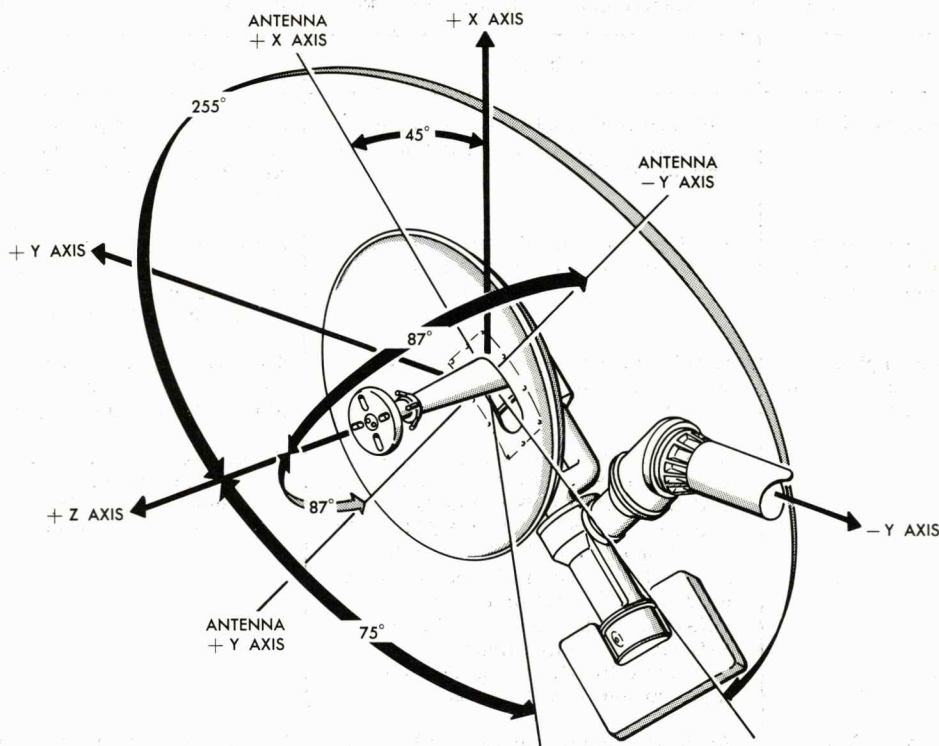
The performance and design data for the CS are given in table 2.7-6.

2.7.7 OPERATIONAL LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions for the CS are covered in paragraphs 2.7.7.1 and 2.7.7.2.

CS

COMMUNICATIONS SUBSYSTEM



300LM4-65

Figure 2.7-19. S-Band Steerable Antenna Gimballing

2.7.7.1 Antennas.

The S-band steerable antenna is mounted on a two-degree-of-freedom gimbal. Elevation (pitch) and azimuth (yaw) are zero when the antenna LOS is parallel to the LM Z-axis, in a positive direction. The antenna is tilted 45° in the LM X- and Y-axis. (See figure 2.7-19.) Slewing the antenna toward the antenna + X-axis is positive elevation; slewing the antenna toward the antenna +Y axis is positive azimuth. The angles of elevation and azimuth coverage are shown in figure 2.7-19. The restrictions in elevations and azimuth are shown in Figure 2.7-20.

The S-band in-flight antennas cover equivalent overlapping regions. The forward in-flight antenna covers a 170° cone about a line in the X-Z plane 20° above the +Z-axis. The rear antenna covers a 170° cone about a line in the X-Z plane 20° above the -Z-axis.

2.7.7.2 General.

The limitations and restrictions imposed on the CS during the mission are as follows:

- The maximum range of telemetry transmission between the LM and the CSM is 320 nautical miles. Greater ranges will cause loss of telemetry transmission.
- The LOS range of the VHF transmitter is limited to 740 nautical miles. Greater ranges will cause loss of VHF communications.
- Once the S-band steerable antenna is unstowed, it cannot be stowed for the remainder of the flight. If the steerable antenna is locked on the uplink signal, RCS thruster impingement will have no effect on antenna orientation. In the absence of the uplink signal, effect of LM or CSM plume can cause the steerable antenna to be reoriented.

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-5. Communications Subsystem - Failure Modes

Failure	Indication	Vehicle Capability (See note 3.)
Primary S-band transceiver failed.	Absence of S-band received voice at LM and/or received S-band voice and data at MSFN.	Secondary (backup) transceiver must be selected by setting XMTR/RCVR switch (panel 12) to SEC.
S-band transceiver nonredundant elements (power combiner diode switch, PM control lines) failed.	All S-band communication is lost, regardless of transceiver in use.	Evaluation of S-band link is impossible. Voice and data communications to CSM or MSFN may be continued via VHF section.
Primary S-band power amplifier failed (during mission phases when additional power is required).	Received S-band signal at MSFN is attenuated or absent.	Secondary (backup) power amplifier must be selected by setting PWR AMPL switch (panel 12) to SEC.
S-band diplexer failed.	S-band communications is lost, regardless of S-band antenna in use. If RANGE FUNCTIONS switch is set to RANGE or OFF, SIGNAL STRENGTH indicator on COMMUNICATIONS ANTENNAS portion of panel 12 reads 0. If switch is set to TV/CWEA ENABLE, indicator reads 0 and S-BD RCVR caution light (panel 2) and MASTER ALARM pushbutton lights (panel 1 and 2) go on.	Communication using the S-band link is impossible. Voice and data communications to CSM or MSFN may be continued via VHF section.
S-band steerable antenna failed.	All S-band communication is lost. If RANGE FUNCTIONS switch is set to RANGE or OFF, SIGNAL STRENGTH indicator on panel 12 reads 0 and HEATER caution light (panel 2) may go on. If switch is set to TV/CWEA ENABLE, indicator reads 0, HEATER caution light may go on, and S-BD RCVR caution light and MASTER ALARM pushbutton lights go on.	Either in-flight backup antenna (FWD or AFT) (relative to existing vehicle orientation and MSFN LOS) must be selected by setting S-BAND selector switch (COMMUNICATIONS ANTENNAS portion of panel 12) to FWD or AFT.
S-band in-flight antenna failed (steerable antenna inoperable).	All S-band communication is lost. If RANGE FUNCTIONS switch is set to RANGE or OFF, SIGNAL STRENGTH indicator on panel 12 reads 0. If switch is set to TV/CWEA ENABLE, indicator reads 0 and S-BD RCVR caution light and MASTER ALARM pushbutton lights go on.	Other in-flight backup antenna must be selected by setting S-BAND selector switch to FWD or AFT. Mission is constrained to vehicle orientation affording LOS with MSFN. (See note 1.)
S-band steerable antenna heaters.	HEATER caution light goes on. Temperature is displayed on TEMP indicator (panel 3) when TEMP MONITOR selector switch is set to S-BAND.	S-band data becomes unreliable.
VHF transmitter and receiver A (normal simplex operation) failed.	Voice communication between LM and CSM is lost.	Transmitter and receiver B must substituted for transmitter and receiver A by setting VHF A switches to OFF and VHF B switches to ON (COMMUNICATIONS portion of panel 12). Data and voice cannot be transmitted simultaneously on VHF B.

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-5. Communications Subsystem - Failure Modes (cont)

Failure	Indication	Vehicle Capability (See note 3.)
VHF A or VHF B transmitter (normal duplex operation) failed.	Voice communication from LM to EVA is lost.	To retain voice communications, transmitter B must be substituted for transmitter A, and receiver A substituted for receiver B with VHF A and VHF B switches (COMMUNICATIONS portion of panel 12). EMU data is thereby lost, and EVA must return to LM.

CS

COMMUNICATIONS SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.7-40A/2.7-40B

LMA 790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-5. Communications Subsystem - Failure Modes (cont)

Failure	Indication	Vehicle Capability (See note 3.)
VHF A or VHF B receiver (normal duplex operation) failed.	Voice and EMU data from EVA to LM is lost.	To retain voice communications, receiver A must be substituted for receiver B, and transmitter B substituted for transmitter A, with VHF A and VHF B switches (COMMUNICATIONS portion of panel 12). EMU data is thereby lost, and EVA must return to LM.
VHF diplexer failed.	All VHF communication is lost, regardless of antenna used.	Mission must be constrained, LM must communicate with CSM via MSFN on S-band.
VHF in-flight antenna failed.	LM-CSM or LM-EVA communication is lost.	Other in-flight antenna must be selected by setting VHF selector switch (COMMUNICATIONS ANTENNAS portion of panel 12) to AFT/PLSS TEST or FWD. Mission is constrained to vehicle orientation that affords LOS with CSM.
Audio center No. 1 or 2 (part of SPA) failed.	VHF, S-Band, and intercom audio communications control is lost. (See note 2.)	Other audio center must be selected by setting related AUDIO CONT switch (AUDIO portion of panel 8 and 12) to BU.
Premodulation processor (PMP) failed.	All or part of S-band and/or VHF communications capability is lost. (See note 2.)	Mission may be constrained severely. (See note 1.) Clipped-speech S-band voice transmission is available as backup mode. Received S-band voice backup from MSFN can be substituted by setting UP DATA LINK switch (panel 12) to VOICE BU.
<p>NOTES:</p> <ol style="list-style-type: none"> 1. Circumstances are dependent on mission phase at time of unit failure. 2. Circumstances are dependent on particular component failures within unit. 3. Where vehicle capability permits redundant equipment to replace malfunctioning equipment, the operating power to the malfunctioning unit must first be removed by opening the related circuit breaker (panels 11 and 16). 		

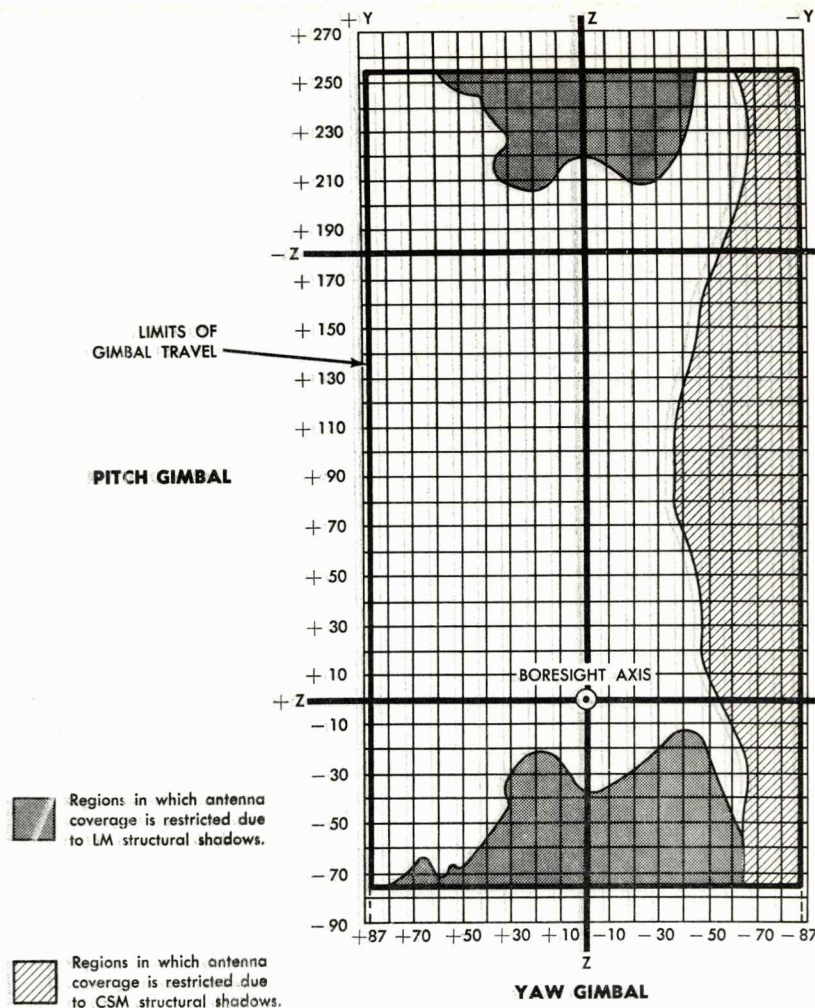
LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.7-6. Communications Subsystem - Performance and Design Data

Equipment	Mode (Status)	Power Requirements		Operating Frequency
		D-C Watts	A-C Watts	
S-band transmitter-receiver	Primary or secondary	38.0	0	
Receiver				2101.8 mc
Transmitter				2282.5 mc
S-band power amplifier and diplexer	Primary or secondary	73.5	0	
S-band steerable antenna				
Electronics	Slew	13.0	27.9	
	Auto	13.0	4.0	
Heaters	On	55.0	0	
VHF A transmitter	Voice	30.0	0	296.8 mc
	Not transmitting	3.5	0	
VHF A receiver	On	1.2	0	296.8 mc
VHF B transmitter	Voice	28.9	0	259.7 mc
	Data	31.7	0	
	Not transmitting	3.5	0	
VHF B receiver	On	1.2	0	259.7 mc
Signal-processing assembly				
Premodulation processor	On	4.7	0	
Audio center (one)	On	5.0	0	
Digital uplink assembly	Uplink data	12.5	0	
	Voice backup	2.5	0	
Biomedical sensors	On	3.0	0	
CS displays	On	3.0	0	
TV camera	On	7.5	0	

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



A-300LM4-244

Figure 2.7-20. S-Band Steerable Antenna Coverage Restrictions

- The S-band steerable antenna provides automatic tracking when its RF axis is aligned to within $\pm 12.5^\circ$ (capture angle) of the MSFN station LOS. Exceeding $\pm 12.5^\circ$ results in inability to acquire the MSFN signal.
- The maximum S-band steerable antenna tracking velocity is $20^\circ/\text{second}$.
- The maximum S-band steerable antenna tracking acceleration is $60^\circ/\text{second}/\text{second}$.
- The activation time for the S-band transmitter-receiver is 30 seconds from application of power; 5 seconds, after a change of modulation mode (FM to PM, or PM to FM). No S-band communication will result until warmup is completed.
- The warmup time for the S-band power amplifier is 60 seconds. The recycle time after RF interruption is 20 seconds. If this restriction is not followed, unpredictable communications will result.
- The warmup time for the S-band steerable antenna electronics is 60 seconds. No S-band communications will result before warmup is completed.
- S-band steerable antenna operating temperature must be maintained at -60° to $+155^\circ$ F. If these temperature limits are exceeded, S-band data are unreliable.

COMMUNICATIONS SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.7.8 TELEMETRY MEASUREMENTS.

The CS telemetry data are given in table 2.7-7. "(R)" in the "Telemetry Range" column signifies a real-time telemetry measurement. Real-time telemetry points are shown in figures 2.7-6 through 2.7-8.

Table 2.7-7. Communications Subsystem - Telemetry Measurements

Code	Description	Telemetry Range (Nominal)	Display Range	Crew Display
GTO201E	S-band power amplifier RF power output	0 to 25 watts (18 watts) (R)		None
GTO992B	S-band transponder static phase error	-15° to +15° (R)		None
GTO993E	S-band transmitter RF power	0.5 to 1.5 watt (0.75 watt) (R)		None
GTO994V	S-band receiver AGC voltage	0 to 5.0 vdc (R)		SIGNAL STRENGTH indicator (panel 12)
GTO454T	S-band steerable antenna temperature	-200° to +200°F	-100° to +200°F	TEMP indicator (panel 3)
GTO441X	DUA verification	Eight bit digital readout		None

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.8 EXPLOSIVE DEVICES SUBSYSTEM.

2.8.1 INTRODUCTION.

The Explosive Devices Subsystem (EDS) permits the astronauts to operate or enable various LM equipment, using explosive devices. The EDS controls deployment of the landing gear; pressurization of the descent, ascent, and reaction control propellant tanks; venting of descent propellant tanks; and separation of the ascent and descent stages.

There are two separate systems (A and B) in the EDS. (See figure 2.8-1.) The systems operate in parallel and provide complete redundant circuitry; each system has its own 37.1-volt d-c battery, relays, time-delay circuits (used in staging), fuse-resistors, buses, and explosive cartridges. Two separate cartridges are provided for each EDS function. Each cartridge uses power from a separate ED bus and can perform its task without the other cartridge.

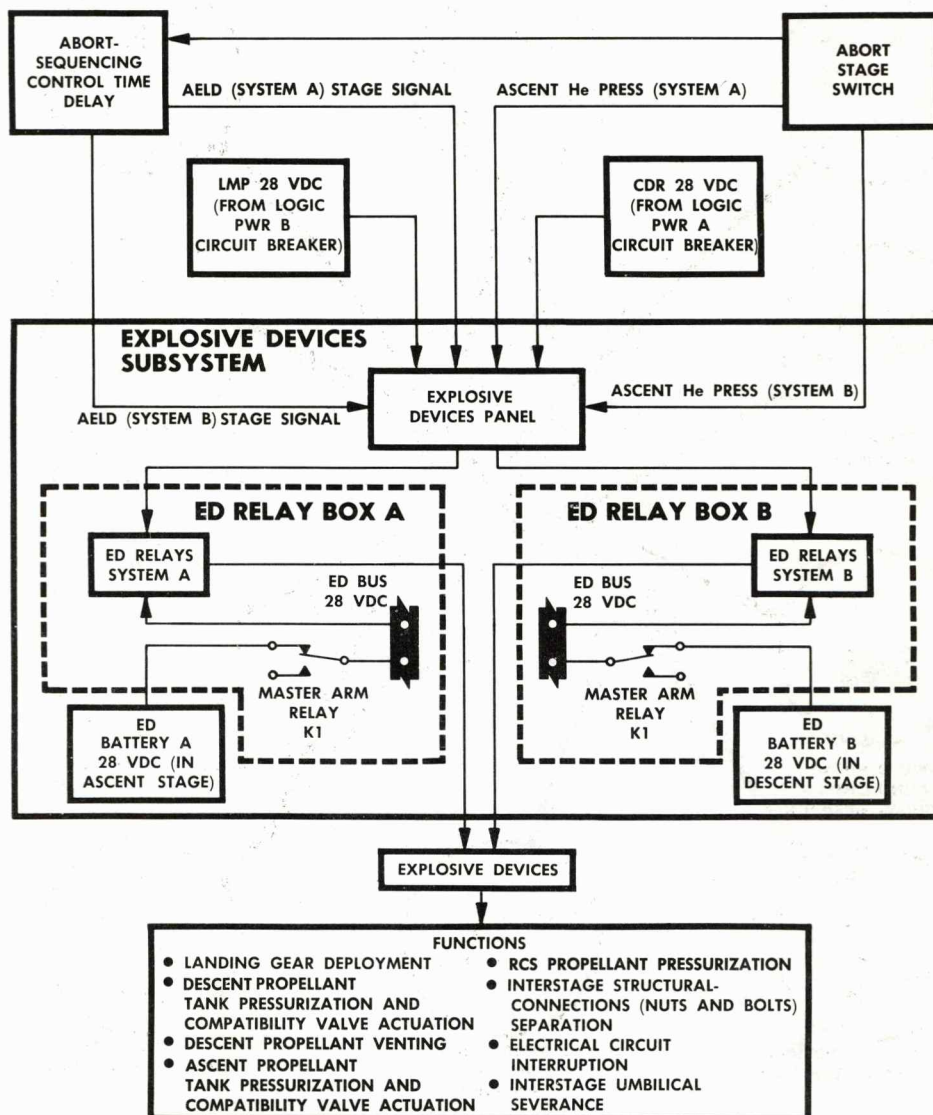


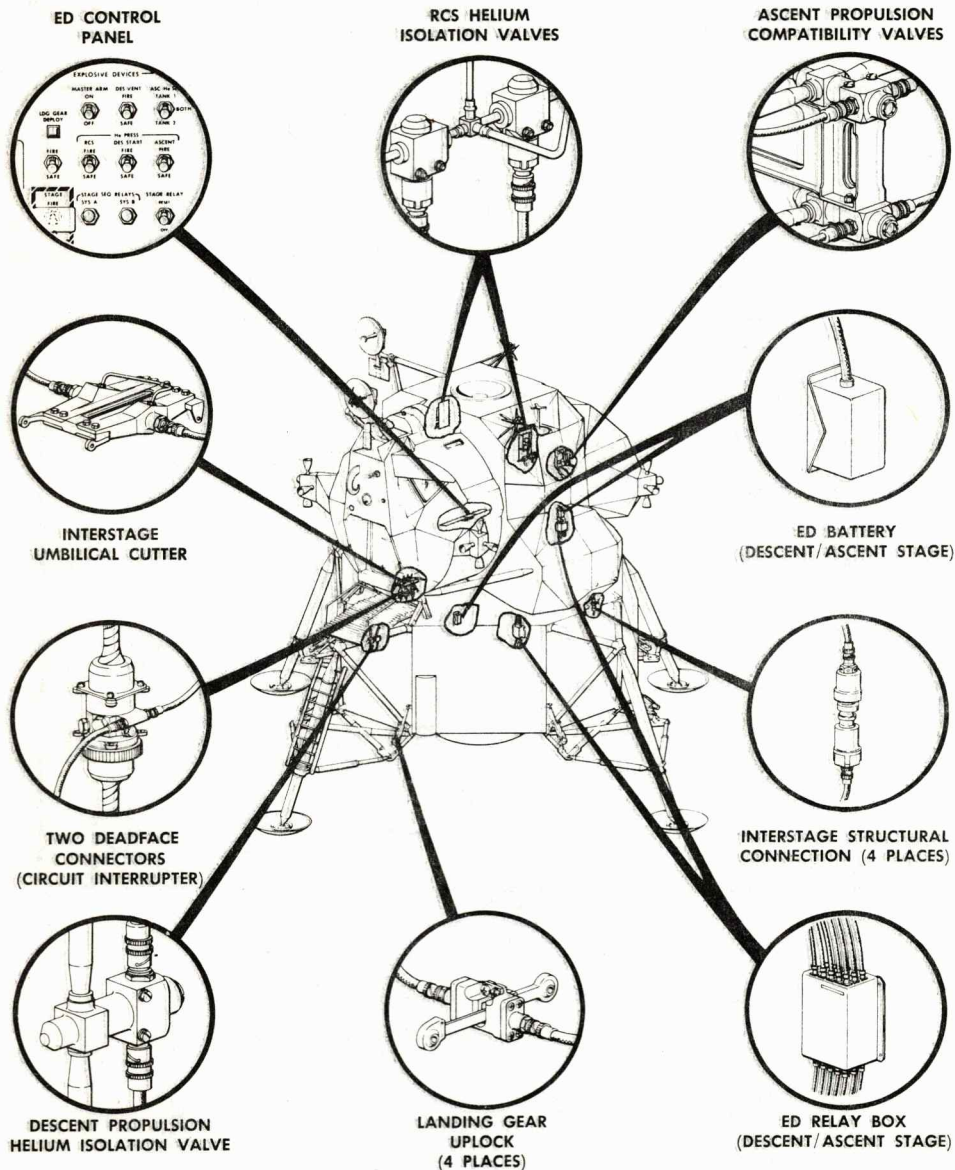
Figure 2.8-1. Explosive Devices Subsystem - Overall Block Diagram

EXPLOSIVE DEVICES SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The general location of all components of the EDS is shown in figure 2.8-2. Electrical power to activate the components in both systems is supplied through circuit breakers on the Commander's circuit breaker panel (11) and the LM Pilot's circuit breaker panel (16). Switches on the EXPLOSIVE DEVICES portion of panel 8 route this power to relays in the ED relay boxes. The relay contacts route ED battery current to explosive cartridges located in LM subsystems. The explosive devices actuated by the cartridges perform the EDS functions.

Ascent propellant tank pressurization and staging (separation of nuts and bolts, electrical circuit interruption, interstage umbilical cable severance) are performed automatically if the ABORT STAGE pushbutton (panel 1) is pressed.



300LM4-18

Figure 2.8-2. Explosive Devices - Component Location

EXPLOSIVE DEVICES SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2. 8. 1A SUBSYSTEM INTERFACES.

EDS relays are actuated by 28 volts dc received through the Commander's and LM Pilot's buses of the Electrical Power Subsystem (EPS). Contacts of these relays route ED battery current to associated explosive devices in the Main Propulsion Subsystem (MPS) and the Reaction Control Subsystem (RCS) and to explosive devices that enable landing gear deployment, interstage structural connection separation, electrical circuit interruption and interstage umbilical severance. In the MPS, explosive devices enable descent propellant tank pressurization and venting, and ascent propellant tank pressurization. In the RCS, explosive devices enable RCS propellant tank pressurization.

The EDS receives two commands from the control electronics section (CES) of the Guidance, Navigation, and Control Subsystem (GN&CS). After the descent engine is fired, a delayed descent engine on command is received from the CES. This command actuates the ED relay associated with the explosive device that enables descent propellant tank pressurization (with supercritical helium). An automatic stage command is received from the CES when the ABORT STAGE pushbutton (panel 1) is pressed. This command actuates the ED staging sequence relays.

ED relay status and malfunction signals are routed to the Instrumentation Subsystem (IS) which processes them for telemetering to MSFN. If the master arm relay or the staging sequence relays malfunction, a signal is routed to the failure detection circuits in the caution and warning electronics assembly (CWEA) of the IS, which causes the ED RELAYS caution light (panel 2) and the appropriate STAGE SEQ RELAYS SYS A or SYS B component caution light (panel 8) to go on. Landing gear deployment is monitored by sensors in the landing gear assemblies. A composite signal from these sensors is sent to the IS, where it is processed for telemetering to MSFN. The signal causes a gray LDG GEAR DEPLOY talkback display (panel 8).

ED

EXPLOSIVE DEVICES SUBSYSTEM

2.8.2 FUNCTIONAL DESCRIPTION. (See figure 2.8-3.)

The redundant circuits and components of the EDS operate simultaneously, as two independent systems (A and B). Either system can perform the EDS functions. Each function is completed before initiation of the next function. The EDS functions are performed in the following sequence:

- Reaction Control Subsystem (RCS) propellant tank pressurization
- Landing gear deployment
- Descent propellant tank prepressurization (ambient helium)
- Descent propellant tank pressurization (supercritical helium)
- Descent propellant tank venting
- Ascent propellant tank pressurization
- Interstage nut-and-bolt separation and ascent stage deadfacing (occur simultaneously)
- Interstage umbilical severance

EDS systems A and B are operated from the EXPLOSIVE DEVICES portion of panel 8. Power (28 volts dc) from the Electrical Power Subsystem (EPS), to actuate the relays in the system A and system B relay boxes, is routed through the ED: LOGIC PWR A circuit breaker (panel 11) and the ED: LOGIC PWR B circuit breaker (panel 16), respectively. Firing power for the cartridges is supplied by the ED batteries. When the MASTER ARM switch is set to ON, each battery supplies power to an identical redundant bus in the ED relay boxes, arming the firing circuits and enabling all explosive devices of the LM. Firing of any explosive device is controlled by its respective firing switch on panel 8. For descent tank venting, the DES VENT switch is used in conjunction with the FUEL VENT and OXID VENT switches on the DES PROPULSION portion of panel 8. The cryogenic (supercritical helium) pressurization is performed automatically when the descent engine-on command is present. For a detailed discussion of descent propellant tank pressurization refer to paragraph 2.8.2.4.

When the MASTER ARM switch is set to ON, the coils of relays K1 are energized by the Commander's and LM Pilot's bus. Contacts of relays K1 in systems A and B provide a path between each ED battery and its bus. The STAGE SEQ RELAYS SYS A and SYS B component caution lights on the EXPLOSIVE DEVICES portion of panel 8 go on to indicate that systems A and B, respectively, are armed. Telemetry signals advise MSFN that systems A and B are armed. Once systems A and B are armed, the firing switches can be used to initiate EDS functions. Each switch routes EPS bus voltage to the coils of two relays, one in each relay box. Contacts of these relays route 7.5 to 15 amperes from the ED battery to an associated explosive device. The staging operation is unique to other EDS functions in that it uses a time-delay circuit to perform a definite relay sequence operation. Contacts of staging relays connect the ED bus to an RC time-delay circuit in both ED relay boxes and provide a direct path to relays that operate to perform electrical circuit interruption (deadfacing) and separation of the nuts and bolts that secure the ascent stage to the descent stage.

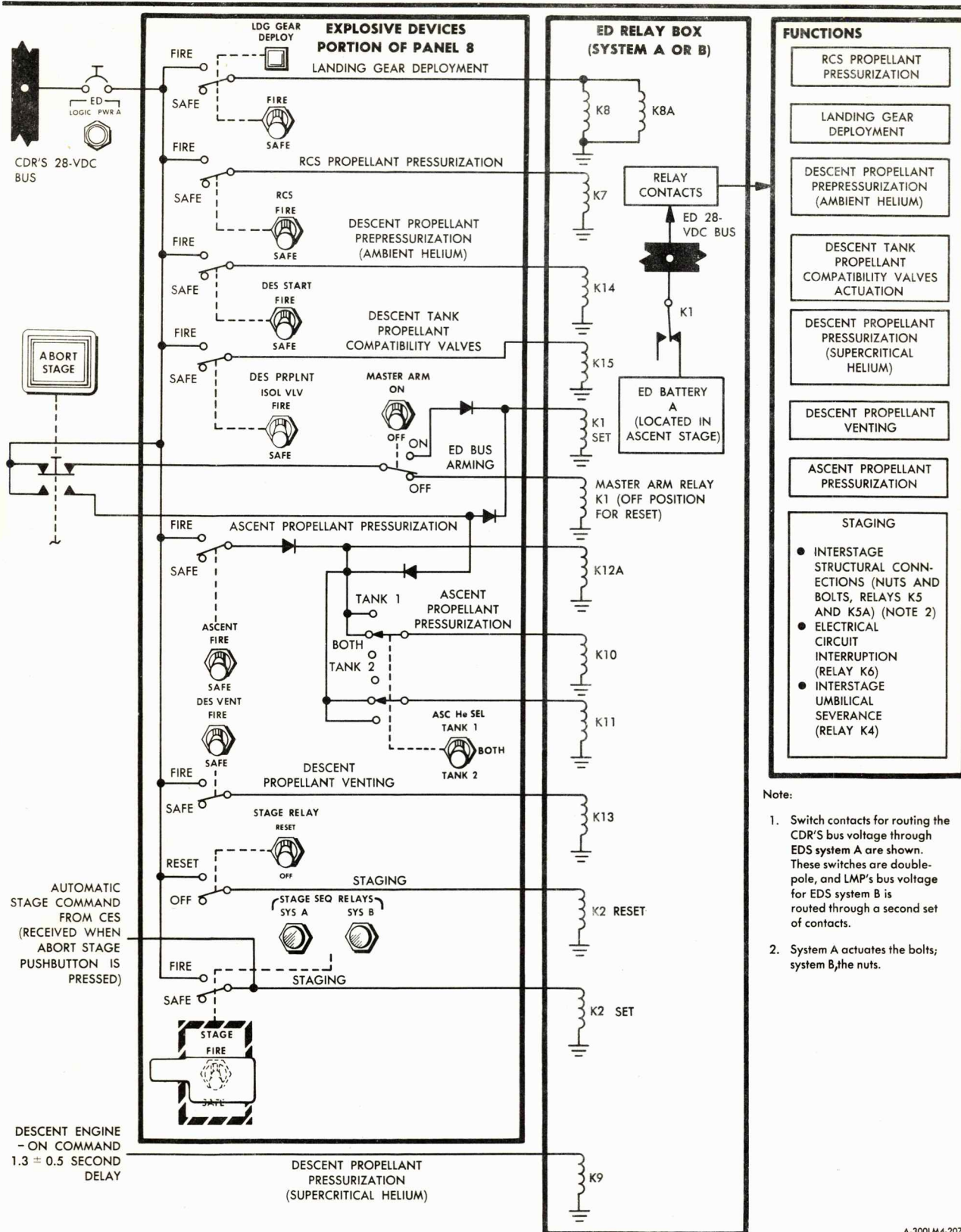
The outputs of the time-delay circuit actuate relays to perform interstage umbilical severance. If any of the relays used in the staging operation, or master arm relay K1, operates inadvertently, the SYS A and SYS B component caution lights and the ED RELAYS caution light (panel 2) go on. (When a caution or warning light goes on, the MASTER ALARM pushbutton/light also goes on and a tone is generated. Pressing the pushbutton/light causes the light to go off and terminates the tone.) Contacts of these relays provide an output (figure 2.8-4) that is routed to the signal-conditioning electronics assembly (SCEA) in the IS to provide MSFN with telemetry signals for monitoring the staging sequence.

Contacts of the remaining ED relays (K7 through K15) form a series output that provides MSFN with telemetry signals for constant monitoring of the status of these relays. If any one, or combination, of these relays operates inadvertently, their telemetry signal does not distinguish which relay or relays are at fault.

EXPLOSIVE DEVICES SUBSYSTEM

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.8-3

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA



A-300LM4-207

Figure 2.8-3. Explosive Devices Subsystem - Overall Functional Diagram

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

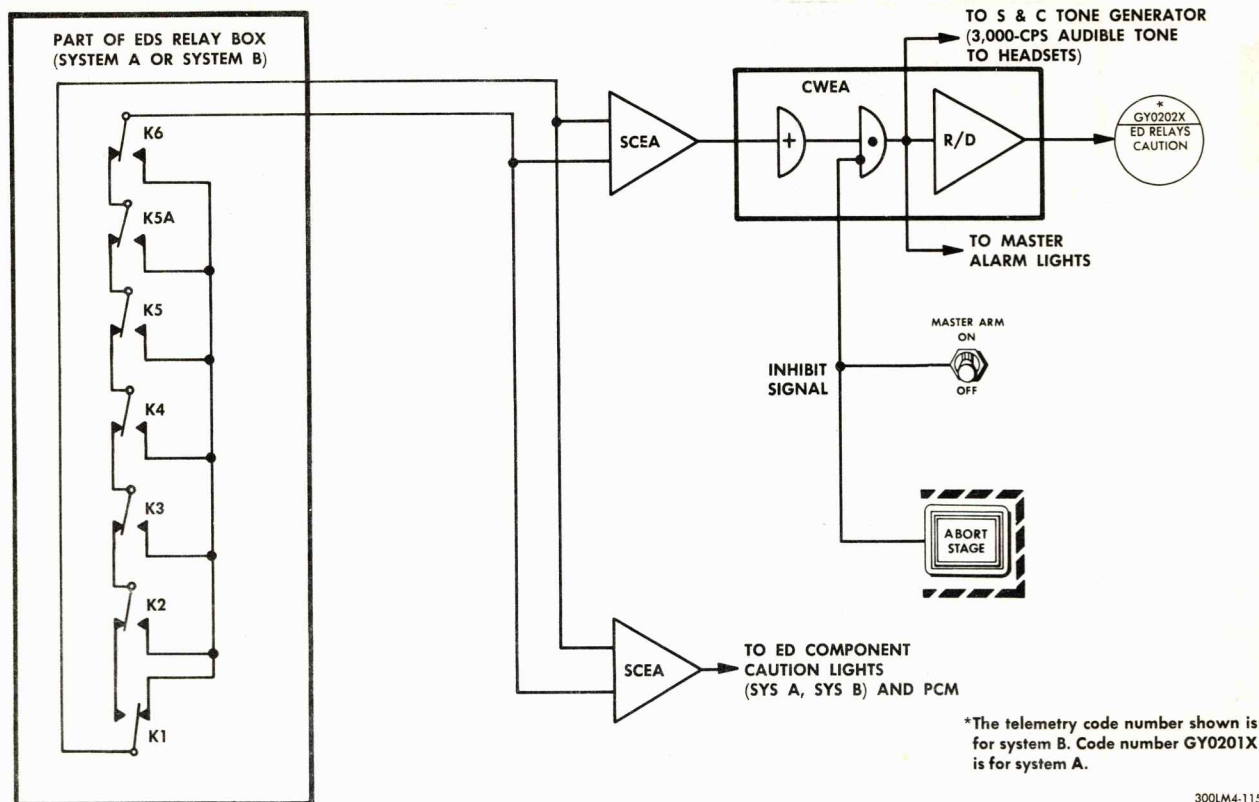


Figure 2.8-4. Stage Sequence Monitoring (Relays K1 through K6) - Simplified Schematic Diagram

The outputs from the stage sequence relays to the SCEA units are in parallel. The SCEA outputs are fed to the CWEA and to the pulse-code-modulation and timing electronics assembly (PCMTEA) in the IS. If one of the SCEA outputs fails, the remaining output is not affected and allows the stage sequence relays to be continuously monitored.

When the MASTER ARM switch or the ABORT STAGE pushbutton (panel 1) is used, an inhibit signal is routed to a CWEA AND gate; the ED RELAYS caution light remains off.

2.8.2.1 Circuitry.

The ED logic circuits are protected by circuit breakers; the firing circuits, by current-limiting fuses. Circuit breakers route the power needed to energize the ED relays. Additionally, they protect wiring against deterioration caused by faults or overloads.

ED circuits are the only electrical load connected to the ED power bus; they do not receive power from the EPS bus. The EPS bus energizes the ED relay coils. The explosive devices switch (panel 8) used determines which relay coil is energized. Relay contacts then route ED battery power (battery A and battery B) from the redundant ED buses to the devices (or cartridges) to perform a specific function.

The time delays preclude the possibility of initiating a function out of sequence. All ED firing circuits and the EPS circuits are shielded. Both systems are electrically and physically isolated from one another and the wiring is routed separately wherever possible. The circuits are not grounded; EPS and instrumentation ground returns are separated and in no way cause the ED circuits to become grounded.

EXPLOSIVE DEVICES SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

EDS design is fail-safe in all respects. There are no mechanical or electrical cross-overs, except for the mechanical actuation in the ED switches. Failure to arm or operate one EDS system (A or B) does not prevent performance of a function. Redundant wiring is used where loss of a single lead could prevent initiation or control of a function.

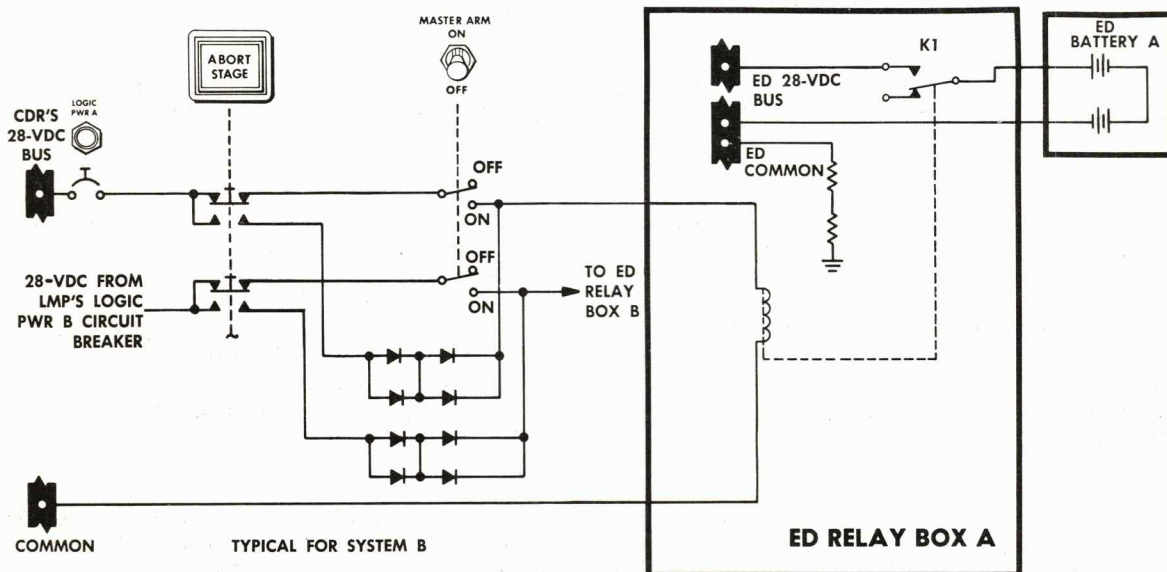
All initiators are protected against static electricity by means of a shorting circuit. The cartridge assemblies are connected to the ED negative bus at all times. When the circuit is fired, the control relay is energized and ED bus voltage is routed to the initiators in the cartridge assemblies. A fuse-resistor, in series with the circuit (between the pyro bus and control relay), protects the initiator bridgewire circuit and ED batteries against overloads. The fuse-resistors are rated at 20 amperes for 100 milliseconds with 1-ohm resistance.

2.8.2.2 ED Bus-Arming.

The MASTER ARM switch is a triple-pole double-throw switch, with a common two-position, lever-locking toggle mechanism for the OFF and ON positions. Before performing any EDS functions, this switch must be set to ON to arm the ED bus. If the ED RELAYS caution light and the SYS A or SYS B component caution light go on due to the MASTER ARM switch failing in the ON position or its relay K1 for one system (A or B) closing inadvertently, the affected EDS system should be deactivated by opening the ED: LOGIC PWR A or ED: LOGIC PWR B circuit breaker. A subsequent EDS failure may jeopardize crew safety.

Setting the MASTER ARM switch to ON applies 28 volts dc from EPS systems A and B to the set coils of latching control relays K1 (in EDS systems A and B). Contacts of K1 connect the ED batteries (A and B) to both ED buses, arming the EDS and enabling performance of any function. (See figure 2.8-5.) The buses are also armed by pressing the ABORT STAGE pushbutton, when staging is to be performed with automatic ascent engine firing. The ABORT STAGE switch bypasses the MASTER ARM switch.

Setting the MASTER ARM switch to OFF, deenergizes the ED buses. This position applies 28 volts dc from EPS systems A and B to the reset coils of relays K1 and breaks the circuit between the ED batteries and ED buses.



300LM4-22

Figure 2.8-5. Bus-Arming Control - Simplified Schematic Diagram

EXPLOSIVE DEVICES SUBSYSTEM

in propellant tank pressure is observed on the FUEL and OXID PRESS indicator (panel 1); the drop in ambient helium tank pressure, on the HELIUM indicator (panel 1). Ambient helium prepressurization permits fuel to flow through the fuel/helium heat exchanger before cryogenic helium flow through the helium/helium heat exchanger is initiated.

With the MASTER ARM switch set to ON and the START pushbutton (panel 5) pressed to command the descent engine to fire, relay K9 in both ED relay boxes receives a signal, after a 1.3-second delay, from the control electronic section (CES) of the Guidance, Navigation, and Control Subsystem. Contacts of K9 route ED battery power to the cryogenic helium isolation valve, opening the valve. Helium from the cryogenic helium storage tank pressurizes the descent propellant tanks in place of the expended ambient helium. The astronauts are alerted to descent helium pressure malfunctions by the DES REG warning light (panel 1). (Refer to table 2.9-3.)

2.8.2.5 Staging.

Stage separation involves simultaneous operation of the circuit interrupters (dead-facing) and separation of the interstage structural connections (nuts and bolts), then umbilical severance (cable cutting). This is accomplished by cartridges fired by operation of the STAGE switch or the ABORT STAGE pushbutton. The ABORT STAGE pushbutton provides automatic ascent engine firing after separation. The STAGE switch (not nominally used) initiates the same staging sequence, but does not provide automatic ascent engine firing. The time-delay circuits, which provide proper sequencing of the staging operation, use transistorized RC time-delay networks. The circuits are in ED relay boxes A and B.

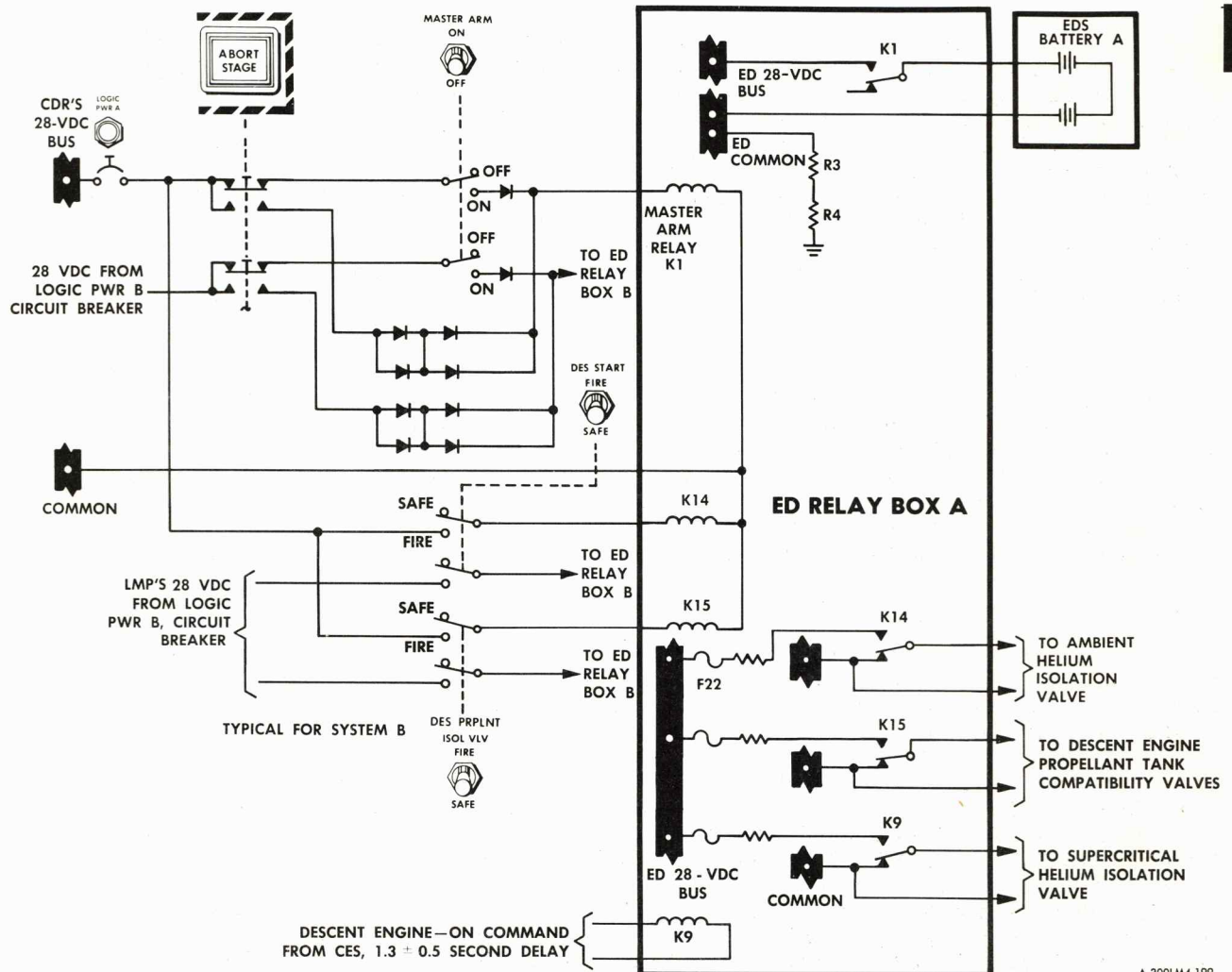


Figure 2.8-8. Descent Propellant Tank Prepressurization - Simplified Schematic Diagram

EXPLOSIVE DEVICES SUBSYSTEM

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Setting the MASTER ARM switch to ON arm the ED buses and setting the STAGE switch to FIRE applies 28 volts dc from EPS systems A and B to the set coils of latching relays K2. (See figure 2.8-9.) Contacts of K2 connect the ED stage-sequencing circuits (RC time-delay circuits) to the ED buses, passing current through closed contacts of relay K3. This energizes relays K5 and K5A (separation of nuts and bolts) and relays K6 (deadfacing). Relays K3 and K4 do not operate immediately, due to the time-delay circuitry. A time delay is necessary to ensure removal of all power from the interstage umbilical before it is cut. After approximately a 15- to 20-millisecond time delay (during which the nuts and bolts separate), transistor Q2 applies ED bus voltage to energize K3. Contacts of K3 close and the bus voltage is applied to another time-delay circuit (transistors Q3 and Q4). This circuit has the same RC time constant; Q4 applies bus voltage to the coil of K4 (guillotine relay), completing stage separation. The staging sequence can also be initiated with a 28-volt d-c signal from the ascent engine latching device in the CES, by pressing the ABORT STAGE pushbutton. This signal bypasses the STAGE switch. Contacts of K3 ensure that the deadface relay has removed electrical power from the umbilical before the guillotine detonators are fired.

2.8.2.6 Ascent Propellant Tank Pressurization. (See figure 2.8-10.)

Two helium isolation valves, each with two cartridges and two sets of parallel redundant fuel and oxidizer compatibility valves are used when pressurizing the ascent propellant tanks. One fuel compatibility valve and one oxidizer compatibility valve uses two cartridges; the other two compatibility valves, a single cartridge each. The ASC He SEL switch provides the astronauts with the option of using either one or both helium tanks to pressurize the system.

The valves are normally closed; they open when the cartridges are fired, depending on the setting of the ASC He SEL switch. In addition, the control relays for the helium isolation valves and the compatibility valves can be energized with a direct signal from the ABORT STAGE pushbutton, which bypasses the ASCENT He PRESS switch. The ASCENT He PRESS switch is a double-pole double-throw switch module, with a lever-locking toggle mechanism for the SAFE position. Normally, the ASC He SEL switch is set to BOTH so that both helium tanks are used for ascent propellant tank pressurization. Setting the MASTER ARM switch to ON arms the ED buses and momentarily holding the ASCENT He PRESS switch to FIRE routes 28 volts dc from EPS systems A and B to the coils of control relays K10 and/or K11, depending upon the setting of the ASC He SEL switch, and to relays K12A. Contacts of these relays route ED battery power to the helium isolation valve cartridges and compatibility valve cartridges, opening the valves to permit helium to pressurize the ascent fuel and oxidizer tanks.

The ASC He SEL switch is a three-position switch, with a lever-locking mechanism in all positions. Setting the switch to TANK 1 routes 28 volts dc to the coils of control relays K10. Contacts of K10, which are in series with cartridges used for helium tank No. 1, close. Setting the switch to BOTH routes 28 volts dc to the coils of control relays K10 and K11. The relay contacts close to complete the circuits to fire cartridges for helium tanks No. 1 and 2. Setting the switch to TANK 2 routes 28 volts dc to the coils of control relays K11. Contacts of K11, which are in series with cartridges used for helium tank No. 2, close. The astronauts are alerted to ascent helium pressure malfunctions by the ASC PRESS warning light (panel 1). (Refer to table 2.9-3.)

2.8.2.7 RCS Propellant Tank Pressurization. (See figure 2.8-11.)

The RCS has two helium, fuel, and oxidizer tanks. Each helium tank pressurizes one fuel tank and one oxidizer tank. Two helium isolation valves, in parallel, are used in conjunction with each helium tank when pressurizing the RCS propellant tanks. Each valve has one cartridge, which is fired by one of the EDS systems.

The valves are normally closed; their firing circuits are controlled by relays actuated by the RCS He PRESS switch. The switch is double-pole double-throw with a lever-locking toggle mechanism for the SAFE position. Setting the MASTER ARM switch to ON arms the ED buses. Momentarily holding the RCS He PRESS switch to FIRE routes 28 volts dc from EPS systems A and B to the coils of relays K7. Contacts of K7 complete the circuit between the cartridges in the helium isolation valves and the ED buses. The contacts of K7 in relay box A route ED bus voltage to helium isolation valves in RCS systems A and B. The other helium isolation valve in RCS systems A and B receives its ED bus voltage through contacts of K7 in relay box B. Opening either valve in system A

EXPLOSIVE DEVICES SUBSYSTEM

2.8.3.1.1 Cartridges.

The SBASI's (figure 2.8-12) in the EDS cartridges provide a high power-to-weight ratio, require low energy to initiate, are compact, and have a high degree of reliability. Most EDS functions use two cartridges; one is redundant. The ascent propulsion compatibility valves use single cartridges and dual cartridges. (Refer to paragraph 2.8.2.6.) When 3.5 amperes (minimum) is applied to the initiator bridgewire for a few milliseconds, heat is generated. The heat fires the initiator. Each initiator is controlled by separate, independent contacts of the firing relays. Power to fire the initiators is supplied by the ED batteries.

2.8.3.1.2 Interstage Structural Connections. (See figure 2.8-12.)

The LM uses four nuts and bolts as interstage structural connections. These nuts and bolts are separated explosively for stage separation. Each nut and bolt has its own cartridge; firing either cartridge separates the stages. The nut cartridges are fired by EDS system B; the bolt cartridges by EDS system A. When one or both cartridges are fired, gases that drive a piston within the nut and/or bolt assemblies are generated. This results in bolt fracture and nut separation. If the nut does not separate from the bolt, bolt fracture, alone, results in stage separation. Reaction time for complete nut and bolt separation is approximately 15 milliseconds. Power for the firing control circuit is controlled by the stage-sequencing circuitry (ED time-delay circuit) in the ascent and descent stages. This circuitry provides proper sequencing of the LM staging operations.

2.8.3.1.3 Interstage Umbilical Cutters (Guillotine). (See figure 2.8-12.)

The guillotine consists of two cutter subassemblies that sever the interstage electrical umbilical during stage separation. Each cutter assembly consists of an EDC, a transfer booster, a blade charge, and a blade. The transfer booster is connected by two manifold crossovers (shielded mild detonating cord), one on each side of the guillotine housing. The manifold crossovers provide for sympathetic ignition needed to operate both blades if one EDC does not fire.

The cutter assembly blades are held safe with shear pins that ensure that the interstage umbilical will not be cut without a positive fire signal to the control circuit. Firing the detonator and explosive blade charge generates a shock wave that drives the cutting blades through the interstage umbilical. The blades sever the umbilical, cutting all electrical wires and the descent water tank supply hardline between the ascent and descent stages. Loss of the ascent water tank supply is prevented by check valves in the ascent stage plumbing. Reaction time for cutter assembly operation is approximately 5 milliseconds, after 3.5 amperes (minimum) of direct current is applied to the initiator bridgewire.

2.8.3.1.4 Landing Gear Uplock and Cutter Assemblies. (See figure 2.8-12.)

Each landing gear assembly has a landing gear uplock and cutter assembly, which contains two EDC's. Firing an EDC generates a shock wave that acts on the base end of the two uplock cutting blades. The blades (driven from opposite directions) sever a strap that connects the landing gear strut with the LM descent stage, permitting the landing gear assembly to deploy. Either blade can sever a strap 35% greater in cross section than the strap used. The operational response time of the uplock is approximately 20 milliseconds, after 3.5 amperes (minimum) of direct current is applied to the EDC.

2.8.3.1.5 Helium Isolation Valves and Compatibility Valves. (See figure 2.8-12.)

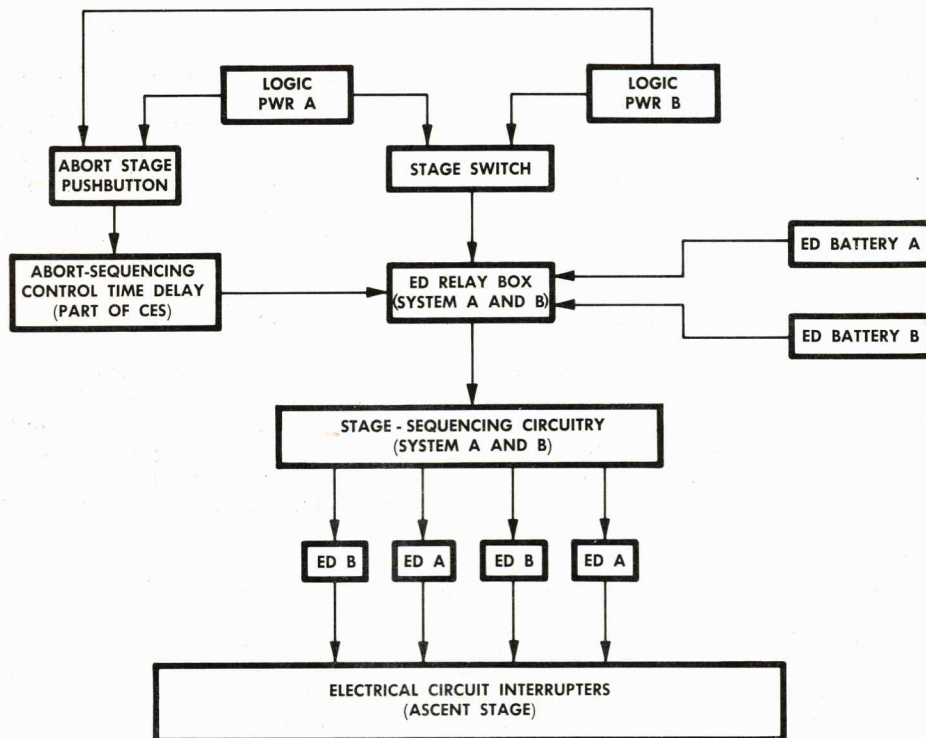
The helium isolation valves and compatibility valves operate in the same manner. They use either a single cartridge or two cartridges, depending on the function being performed. (Refer to paragraph 2.8.3.1.) Firing a cartridge in a valve generates gases in the valve explosion chamber at extremely high rates. The gases drive a piston to shear a closure disk in the pressure line. This opens the valves. At the end of its stroke, the piston is forced into the valve housing and the piston port in each valve is permanently aligned with the pressure line plumbing.

2.8.3.1.6 Circuit Interrupters (Deadface Connectors). (See figure 2.8-12.)

Two circuit interrupters are in the ascent stage. (See figure 2.8-2.) They are in front of the ascent stage tank thermal shielding, immediately above quadrant 4 of the descent stage. Each circuit interrupter has two cartridges. Each cartridge can operate the circuit interrupter. Using two cartridges for each circuit interrupter provides redundancy to ensure that the electrical power between the ascent and descent stages is terminated before umbilical severance.

The circuit interrupter cartridges are energized during the staging operation. When energized, the cartridges cause a pressure buildup in the circuit interrupter combustion chamber. As pressure builds up in the chamber, two initial lock detents move to their open positions, freeing the piston assembly and permitting its movement when its actuation pressure is reached. When the pressure in the combustion chamber is sufficient to overcome pin and O-ring friction, the piston assembly moves to the open-circuit position, disconnecting the pins of the electrical connectors. Spring-loaded final lock detent pins automatically drop into slots in the piston and retain the piston in the open-circuit position. Circuit interruption removes all electrical power from the interstage umbilical lines before the umbilical is severed. (See figure 2.8-13 for a functional block diagram of circuit interruption.)

The stage-sequencing circuits (paragraph 2.8.2.5) control circuit-interrupter operation. If one cartridge fired by the control circuit malfunctions, the other cartridge, fired by the redundant control circuit performs the function; normally, both cartridges fire. Approximately 240 electrical conductors are routed through each circuit interrupter. The interrupter pin connections are normally closed; they are opened by firing the cartridge. Operation response time of the circuit interrupters is approximately 20 milliseconds.



A-300LM4:21

Figure 2.8-13. Electrical Circuit Interrupter - Functional Block Diagram

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.10 LIGHTING.

2.10.1 INTRODUCTION.

Exterior and interior lighting aids in the performance of crew visual tasks and lessens astronaut fatigue and interior-exterior glare effects. Exterior lighting is used for LM and CSM tracking and docking maneuvers. Interior lighting illuminates the cabin and the controls and displays on the Commander's and LM Pilot's panels. Power for exterior and interior lighting is routed through the 28-volt d-c LTG circuit breakers on panels 11 and 16 and the 115-volt AC BUS A and AC BUS B lighting circuit breakers on panel 11.

2.10.2 FUNCTIONAL DESCRIPTION.

LM lighting is provided by exterior and interior lights and lighting control equipment. (See figure 2.10-1.) The exterior lighting enables the astronauts to visually guide and orient the LM and CSM to achieve successful tracking or docking. Interior lighting is divided into seven categories: incandescent annunciators, component caution lights, floodlights, DSKY and DEDA condition lights, integral electro-luminescent (EL) lighting, numeric EL lighting, and incandescently illuminated pushbuttons.

2.10.2.1 Exterior Lighting. (See figure 2.10-2.)

Exterior lighting includes a radioluminescent docking target, five docking lights, and a high-intensity tracking light. The color, brightness and lighting method of the exterior lights are listed in table 2.10-1. The EXTERIOR LTG switch on the LIGHTING portion of panel 3 controls the five docking lights and the high-intensity tracking light on the exterior of the ascent stage.

Table 2.10-1. Exterior Lights - Color, Brightness, and Lighting Method

Component	Lighting Method	Color	Brightness at Rated Capacity	Brightness Adjustment
Docking target	Radioluminescent materials	Green	0, 3 foot-lambert minimum (at time of launch)	Fixed
Docking lights				
Port	Incandescent	Aviation red	0, 15 candlepower minimum	Fixed
Starboard	Incandescent	Aviation green	0, 15 candlepower minimum	Fixed
Forward (2 lights)	Incandescent	Aviation white (+Y) Yellow (-Y)	0, 23 candlepower minimum	Fixed
Aft	Incandescent	Aviation white	0, 23 candlepower minimum	Fixed
Tracking light	Gaseous discharge	White	9,000 effective beam candlepower	One step to 1/150 of initial value

LTG

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.10.2.1.1 Docking Lights. (See figure 2.10-3.)

Five running lights mounted on the exterior of the LM provide visual orientation and permit gross attitude determination relative to a line of sight through the CSM windows during rendezvous and docking. The lights are controlled with the EXTERIOR LTG switch. Before earth launch, this switch is set to DOCK to provide for subsequent automatic turn-on of the docking lights. For transposition and docking, the docking lights are turned on by the normally open SLA/LM antenna transfer switch (located at SLA attachment points). This switch is automatically released to the closed position upon deployment of the spacecraft launch-vehicle adapter (SLA) panels. Upon completing the docking maneuver, LM power is turned off and the docking lights go off. Setting the EXTERIOR LTG switch to OFF interrupts the input power to the docking and tracking lights. The lights are visible, and their color recognizable, at a maximum distance of 1,000 feet.

2.10.2.1.2 Tracking Light.

The tracking light permits visual tracking of the LM by the CSM. It is turned on by setting the EXTERIOR LTG switch to TRACK. A flash tube in the tracking light electronics assembly causes the light, which has a 60° beam spread, to flash at a rate of 60 flashes per minute.

2.10.2.1.3 Docking Target.

The docking target is a three-dimensional, self-illuminated standoff cross that is separated by a shaft from a circular base. Illumination is provided by radioluminescent disks, which outline the cross and circular base markings. The docking target provides for visual alignment of the LM to the CSM during final docking maneuvers. (Refer to paragraph 2.11.6.2.)

2.10.2.2 Interior Lighting. (See figure 2.10-2.)

Interior lighting consists of integral panel and display lighting, backup floodlighting, electroluminescent lighting, and general ambient lighting. The color, brightness, and lighting method of the interior lights are listed in table 2.10-2. Interior lighting controls are located on the LIGHTING portion of panels 3 and 5. The SIDE PANELS switches (panels 3 and 5) control lighting for the LM Pilot's and Commander's side panels. The INTEGRAL control (panel 5) adjusts the light intensity for the side panels. Interior side panel floodlighting is provided by setting the FLOOD switch (panel 3) to ALL. In this position, the switch also turns the Commander's and LM Pilot's forward and overhead floodlights on. The light intensity of the forward and overhead floodlights is controlled with the OVHD/FWD FLOOD controls (panels 3 and 5). The light intensity of the side panel floodlights cannot be adjusted.

Table 2.10-2. Interior Lights - Color, Brightness, and Lighting Method

Component	Primary Lighting Method	Color	Brightness Adjustment	Color Under Incident Illumination
Control panels and pushbuttons (illuminated-character type)	Integral (EL) (primary)	White	Continuous from zero foot-lamberts	Control panels: gray Pushbuttons Background: black Characters: white
Floodlights	Flood (incandescent)	White (unfiltered)	Continuous from less than 0.1 foot-candle	
Circuit breakers	Flood (EL)	White	Continuous from zero foot-lamberts	Background: black Characters: white
Numeric displays	Integral (EL)	Green	Continuous from zero foot-lamberts	Background: gray
LUNAR CONTACT lights	Integral (incandescent)	Blue	Fixed	Background: non-specular gray

LIGHTING

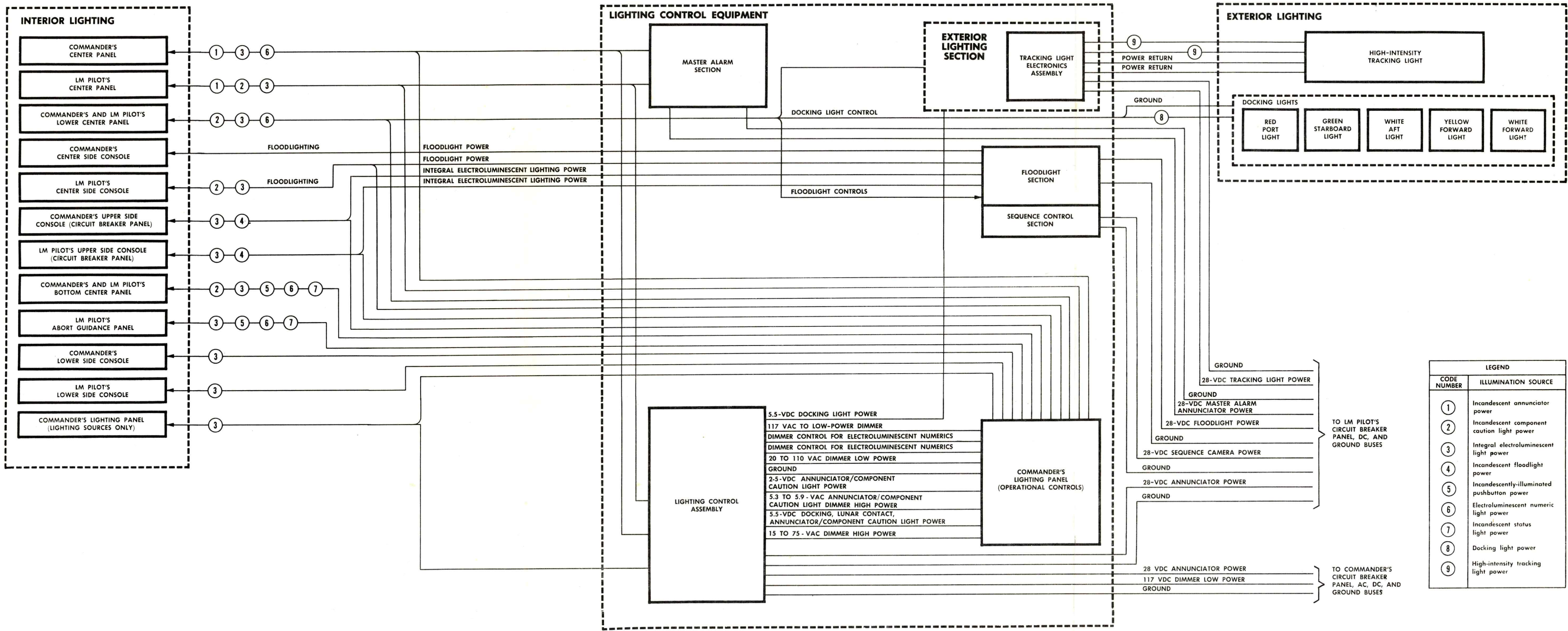


Figure 2.10-1. Lighting Control Equipment

300LM4-208

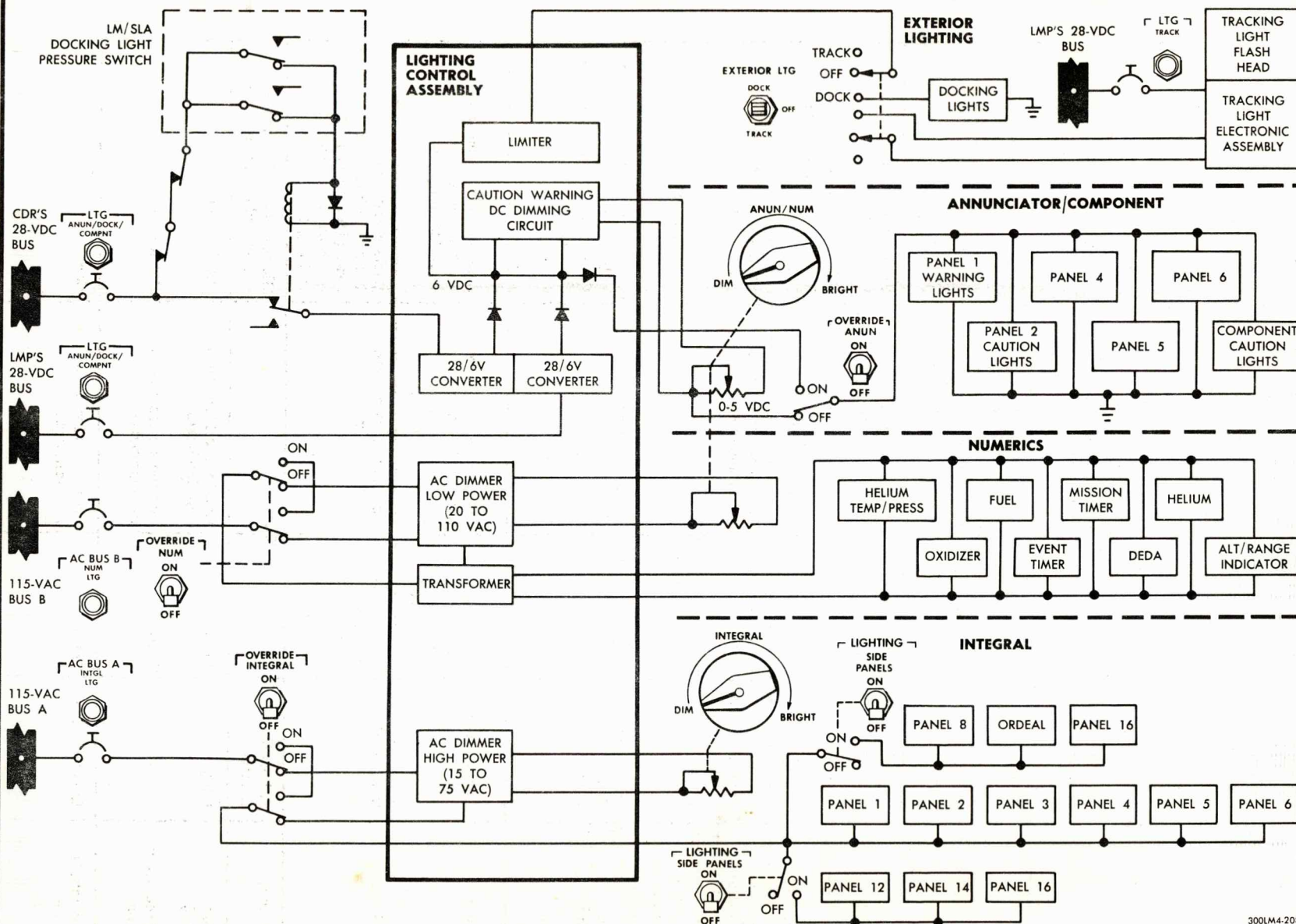


Figure 2.10-2. Lighting Circuits

LTG

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.10-2. Interior Lights - Color, Brightness, and Lighting Method (cont)

Component	Primary Lighting Method	Color	Brightness Adjustment	Color Under Incident Illumination
MASTER ALARM pushbutton/lights	Integral (incandescent)	Aviation red	Fixed (press to turn off)	Background: translucent gray/white Legend: black
Caution lights	Integral (incandescent)	Aviation yellow	Continuous from 1.5±0.5 foot-lamberts	Background: translucent gray/white Legend: black
Warning lights		Aviation red		
Indicator power failure lights	Integral	Aviation red	Continuous from 0.005±0.02 foot-lambert	Translucent gray/white
Component caution lights	Integral (incandescent)	Aviation yellow	Continuous from 0.5±0.2 foot-lambert	Background: translucent gray/white Legend: black
Engine START pushbutton/light	Integral (incandescent)	Aviation red	Continuous from 0.5±0.2 foot-lambert	Background: translucent gray/white Legend: black
Engine stop pushbutton/lights	Integral (EL)		Continuous from zero foot-lamberts	Background: translucent gray/white
Computer status condition indicator (DSKY and DEDA)	Integral (incandescent)	Aviation white	Continuous from zero foot-lamberts	Background: translucent gray/white
Computer caution condition indicator (DSKY)		Aviation yellow		Legends: black
Self-luminous devices (toggle) switch tips, landing-point designator)	Radio-luminescent	Green	Fixed	Pale green Pale yellow for landing point designator inner reticle
Talkbacks Two-position	Integral (EL)	White background and black striping (energized)	Continuous from zero foot-lamberts	Alternate black and white striping (energized) Gray (deenergized)
Three-position	Integral (EL)	White background and black	Continuous from zero foot-lamberts	Background: red Letters: black on gray Gray (deenergized) Alternate black and white striping (energized)
Interior pointers	Silhouette of EL flood			Black-yellow "Switzer"-rocket red or fire orange
Indicia (arrows, symbols, etc)	Integral (EL)	White	Continuous from zero foot-lamberts	Black - white
Characters (numbers and letters on displays)	Integral (EL)	White	Continuous from zero foot-lamberts	Black - white
Time-shared labels and multipliers (X10 indicator)	Integral (EL)	Green	Continuous from zero foot-lamberts	None

LIGHTING

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2.10-2. Interior Lights - Color, Brightness, and Lighting Method (cont)

Component	Primary Lighting Method	Color	Brightness Adjustment	Color Under Incident Illumination
Range marking (color bands, normal mode)	Integral (EL)	Green	Continuous from zero foot-lamberts	Green
Immediate or emergency action controls	Integral (EL)	Yellow	Continuous from zero foot-lamberts	Alternate yellow and black striping, at 45° (yellow is 3 times thickness of black)

2.10.2.2.1 Integrally Lighted Components.

There are three types of integrally lighted components: panel areas, displays, and caution and warning annunciators. The integrally lighted components use EL or incandescent devices that are controlled by on-off switches and potentiometer-type dimming controls. All panel placards are integrally lighted by white EL lamps with overlays. The displays have EL lamps within their enclosures. The numeric displays show green or white illuminated digits on a nonilluminated background, displays with pointers show a nonilluminated pointer travelling over an illuminated background. The brightness of the EL displays can be controlled manually by potentiometer-type dimming controls. Each EL dimming control can be bypassed by a related override switch, so that full brightness will be maintained should a dimming control fail.

With the ANNUN OVERRIDE switch (panel 5) set at OFF, the ANNUN potentiometer of the dual-potentiometer ANUN/NUM dimming control adjusts the light intensity of the caution and warning annunciators (except the C/W PWR caution light), the component caution lights (except the DC BUS FAULT light), the incandescent pushbuttons, and the indicator power failure lights at the top of indicators on panels 1 and 2. The C/W PWR caution light goes on if a CWEA power supply failure is detected. For this reason, it is not affected by the ANUN/NUM dimming control. The DC BUS FAULT component caution light goes on if an EPS bus voltage failure is detected. For this reason, it, too, is not controlled by the ANUN/NUM dimming control. If the ANUN potentiometer of the dimming control fails, setting the ANNUN OVERRIDE switch to ON routes 5-volt dc power directly to the lights.

When the NUM OVERRIDE switch (panel 5) is set to OFF, power is provided to the NUM potentiometer of the ANUN/NUM dimming control. This potentiometer adjusts the light intensity of the EL numeric displays. If the NUM potentiometer fails, setting the NUM OVERRIDE switch to ON bypasses the NUM potentiometer and routes 115-volt a-c bus voltage directly to the numeric displays.

The SIDE PANELS switches (panels 3 and 5) turn the EL lighting for the Commander's and LM Pilot's side panels on and off. With these switches set to ON, the integral side panel lights are connected to the INTEGRAL dimming control (panel 5) and the INTEGRAL OVERRIDE switch (panel 5). The INTEGRAL dimming control adjusts the light intensity of the side panels when the INTEGRAL OVERRIDE switch is set to OFF. If the INTEGRAL dimming control fails, setting the INTEGRAL OVERRIDE switch to ON, bypasses the INTEGRAL dimming control and routes 115-volt a-c bus voltage directly to the side panel lights.

Two MASTER ALARM pushbutton/lights (panels 1 and 2) go on to alert the astronauts when a failure or out-of-tolerance condition is sensed by the CWEA. The signal that causes the MASTER ALARM pushbutton/lights to go on also activates an audio tone (3 kc) generator. Pressing either MASTER ALARM pushbutton/light extinguishes the MASTER ALARM lights and deactivates the tone generator.

2.10.2.2.2 Lunar Contact Lights.

Two LUNAR CONTACT lights (panels 1 and 3) go on during a lunar-landing mission when the probes on the landing gear contact the lunar surface. These lights are not used for the LM-5 mission.

LIGHTING

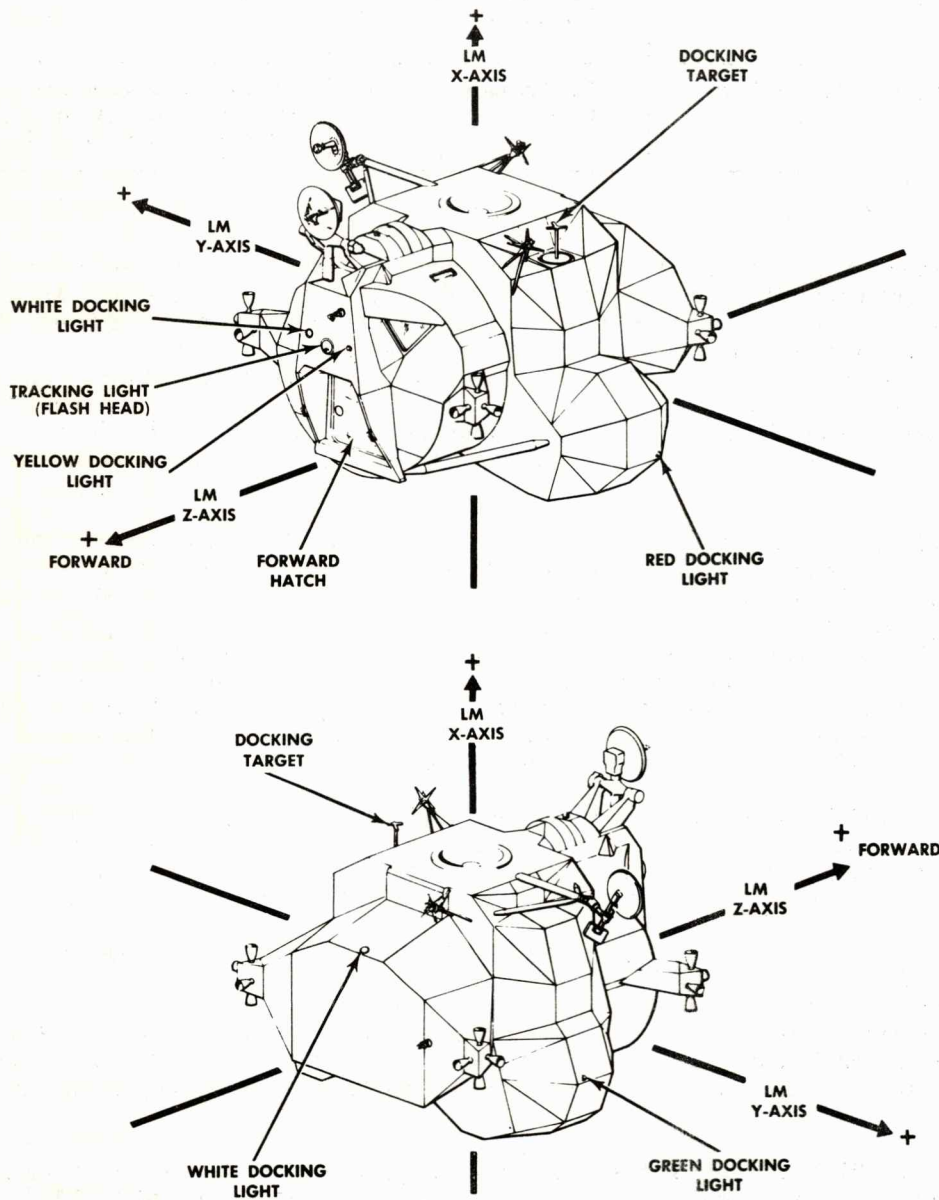


Figure 2.10-3. Exterior Lighting

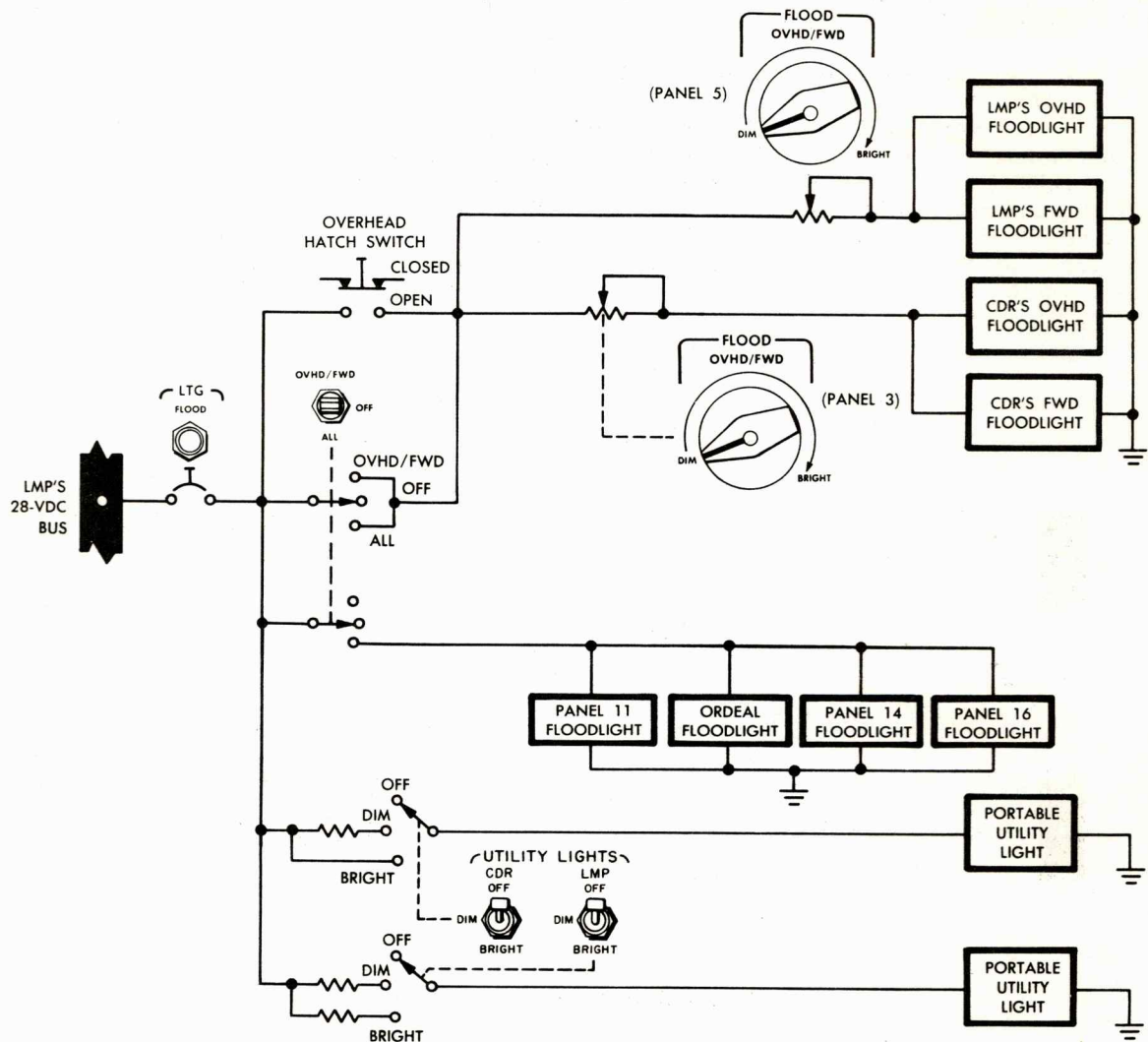
LIGHTING

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.10.2.2.3 Floodlighting. (See figure 2.10-4.)

Floodlighting is used for general cabin illumination and as a secondary source of illumination for the control and display panels. Floodlighting is provided by the Commander's overhead and forward floodlights, the LM Pilot's overhead and forward floodlights, and recessed floodlights in the bottom of extending side panels. These floodlight fixtures provide an even distribution of light with minimum reflection. Every panel area has more than one lamp.

When the SIDE PANELS switches are set to OFF, the COMMANDER's and LM pilot's side panel integral lighting is interrupted. Setting the FLOOD toggle switch (panel 3) to ALL routes 28 volts dc to the Commander's and LM Pilot's overhead and forward floodlights, and to their side panel floodlights. The OVHD/FWD FLOOD dimming controls (panels 3 and 5) adjust the intensity of floodlight illumination on the LM Pilot's and Commander's side of the cabin, respectively. In this mode of light, there is no intensity control for the side panels. Setting the FLOOD toggle switch to OVHD/FWD routes 28 volts dc to the Commander's and LM Pilot's overhead and forward floodlights only. With the FLOOD toggle switch set to OFF, all floodlights are off, unless the overhead hatch is open. The overhead hatch contains a micro-switch that overrides the FLOOD toggle switch and routes the 28-volt d-c bus voltage to limit the overhead and forward floodlights.



300LM4-195

Figure 2.10-4. Floodlighting and Utility Lights-Simplified Diagram

LIGHTING

2.10.2.2.4 Portable Utility Lights. (See figure 2.10-4.)

Two portable utility lights are used, when necessary, to supplement the cabin interior lighting. The lights, when removed from the flight data file container, connect to the overhead utility light panel. Power is applied to the connector from the LMP's 28-volt d-c bus through the LTG:FLOOD circuit breaker (panel 16) and the CDR and LMP UTILITY LIGHT switches on the overhead lighting panel. The switches provide one-step dimming for the light-intensity control.

2.10.2.2.5 Optical Sight Reticule Light. (See figure 2.10-5.)

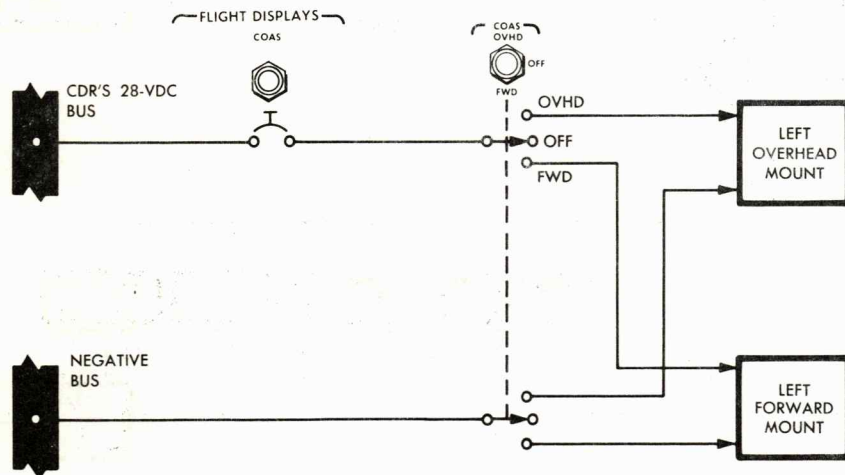
The crewman's optical alignment sight (COAS), used to sight the docking target on the CSM, has a reticle that is illuminated by a 28-volt d-c lamp. Power to the lamp, supplied by the CDR's 28-volt d-c bus, is routed through the FLIGHT DISPLAYS: COAS circuit breaker (panel 11) and the COAS switch (panel 8). (Refer to paragraph 2.11.6.1 for details.)

2.10.2.2.6 Alignment Optical Telescope Lights.

A thumbwheel on the computer control and reticle dimmer assembly controls the brightness of the AOT reticle. The lamp voltage is adjustable from 0.2 ± 0.1 to 4.0 ± 0.5 volts rms. The lamps edge-light the reticle with incandescent red light.

2.10.2.2.7 Lamp and Tone Test. (See figure 2.10-6.)

The LAMP/TONE TEST selector switch (panel 3) is used to perform a functional test of lights and the master alarm tone. The ALARM TONE position tests the MASTER ALARM pushbutton/lights and related tone. The C/W 1, C/W 3, and C/W 4 positions test the lamps of some of the caution and warning annunciators (panels 1 and 2). The ENG PB-C/W 2 position tests the lamps in the engine stop and START pushbutton/lights (panels 3 and 5) and the remainder of the caution and warning annunciators. The COMPNT position tests the component caution lights and the LUNAR CONTACT lights (panels 1 and 3).



300LM4-226

Figure 2.10-5. COAS Lighting

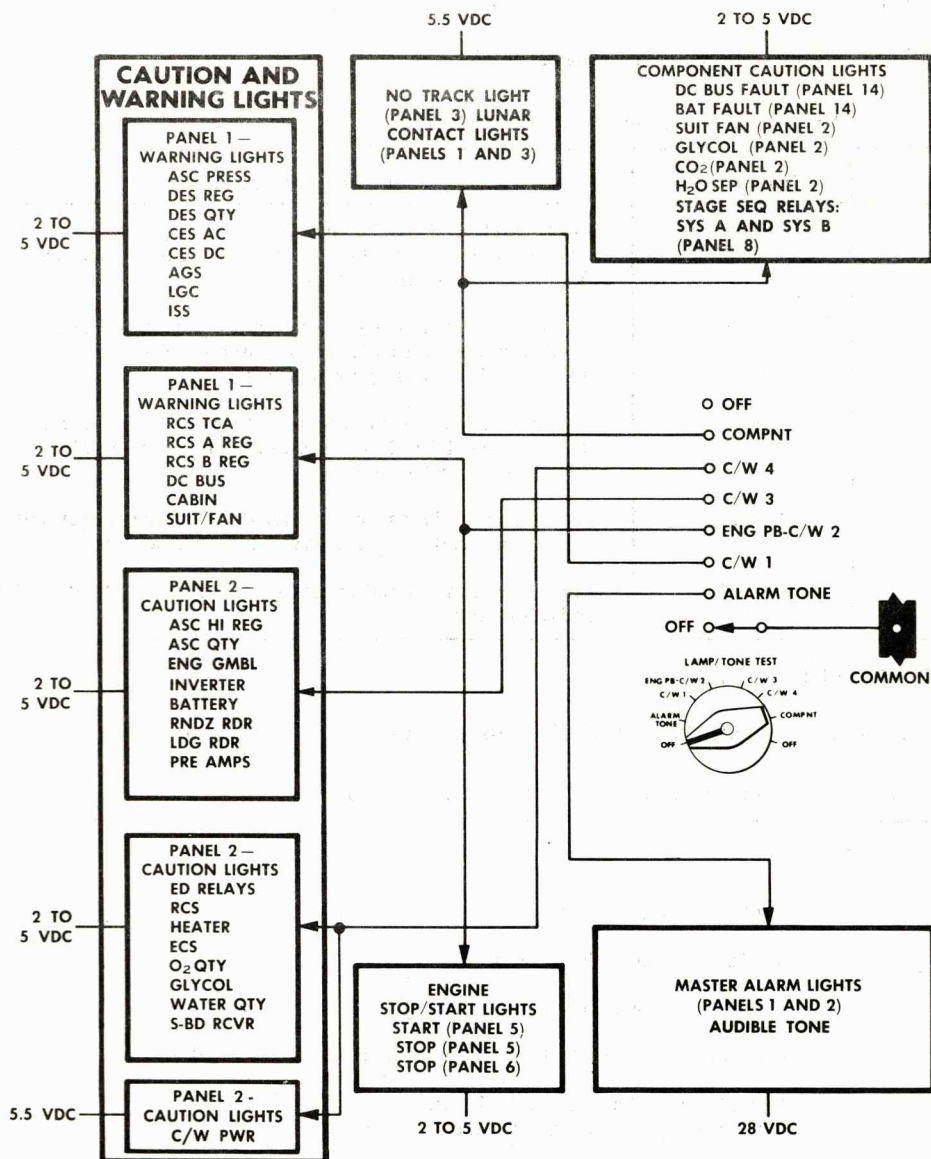


Figure 2.10-6. Lamp and Tone Test Circuits

LIGHTING

LMA 790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.10.3 FAILURE MODES.

The lighting failure modes are given in table 2.10-3.

Table 2.10-3. Lighting - Failure Modes

Failure	Indication	Vehicle Capability
Commander's and LM Pilot's side panel lighting	Visual	INTEGRAL OVERRIDE switch (panel 5) provides direct input to panels if INTEGRAL control failed. Side panel floodlighting is also provided.
Caution and warning annunciator lighting	Visual	ANNUN OVERRIDE switch (panel 5) provides direct input to annunciators if ANUN potentiometer of ANUN/NUM dimming control failed.
EL numeric lighting	Visual	NUM OVERRIDE switch (panel 5) provides direct input to numerics if NUM potentiometer of ANUN/NUM dimming control failed.
Docking lights	Visual	Docking procedures are degraded. Docking target and rendezvous radar is primary means for docking.
Tracking lights	Visual	Visual tracking of LM by CSM is degraded. Rendezvous radar is used.

2.10.4 PERFORMANCE AND DESIGN DATA.

The performance and design data for the LM lighting are given in tables 2.10-4 and 2.10-5.

Table 2.10-4. Exterior Lights - Performance and Design Data

Tracking Light	
Type	High intensity, flashing
Dimming	1/150 of initial value (one step)
Intensity	9,000 beam candlepower (minimum)
Visibility	Cone centered on LM +Z-axis, with semivertex angle of 30°
Dim	Visual: less than 10 nautical miles CSM sextant: less than 30 nautical miles
Bright	Visual: 10 to 130 nautical miles CSM sextant: 30 to 400 nautical miles
Docking lights	
Type	Incandescent
Intensity	Fixed
Visibility	1,000 feet
Docking target	
Type	Self-luminous
Intensity	Fixed

LIGHTING

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Table 2-10-5. Interior Lights - Performance and Design Data

	Intensity	Power
Control panels and pushbuttons		
Primary	0.5±0.2 foot-lambert	75 volts, 400 cps
Backup	0.2 foot-candle (minimum)	24 volts dc
Circuit breakers	0.5±0.2 foot-lambert	75 volts, 400 cps
EL numeric readout	8 foot-lamberts	115 volts, 400 cps (DSKY: 250 volts, 800 cps)
LUNAR CONTACT lights, and caution and warning annunciators	50±10 foot-lamberts	5 volts dc
MASTER ALARM pushbutton/lights	150±30 foot-lamberts	28 volts dc
COMPONENT caution lights	15±3 foot-lamberts	5 volts dc
Engine START and stop pushbutton/lights		
Incandescent	15±3 foot-lamberts	5 volts dc
EL	0.5±0.2 foot-lambert	75 volts, 400 cps
Computer status condition indicators (DSKY and DEDA)	10+8-2 foot-lamberts	5 volts dc
Self-luminous devices	0.05±0.02 foot-lambert	
Talkbacks (two- and three-position)	0.5±0.2 foot-lambert	75 volts, 400 cps
Displays		
Characters and indicia	0.5±0.2 foot-lambert	75 volts, 400 cps
Labels and multipliers	8 foot-lamberts (minimum)	115 volts, 400 cps
Range markings	0.4±0.1 foot-lambert	75 volts, 400 cps
Immediate or emergency controls	0.1 foot-lambert (minimum)	75 volts, 400 cps
Indicator power failure lights	20±5 foot-lamberts	5 volts dc

2.10.5 OPERATIONAL LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions for LM lighting are as follows:

- The tracking light should be turned off when the docking lights are visible. When docking, the tracking light is not visible, therefore, unnecessary power is used if it is left on.
- The docking lights shall not be used at distances in excess of 1,000 feet. Beyond 1,000 feet, these lights cannot be distinguished from stars.

LIGHTING

2.11 CREW PERSONAL EQUIPMENT.

2.11.1 INTRODUCTION.

This subsection covers crew personal equipment, which includes equipment required for life support and astronaut safety, and accessories related to successful completion of the mission. The portable life support system (PLSS) oxygen and water recharge connectors interface with the Environmental Control Subsystem (ECS) for refills of oxygen and water. The pressure garment assembly (PGA) interfaces with the ECS for conditioned oxygen, through oxygen umbilicals, and with the Communications and Instrumentation Subsystems for communications and bioinstrumentation, through the electrical umbilical. Power to the crewman optical alignment sight is supplied by the Electrical Power Subsystem. For stowage locations of crew personal equipment, refer to paragraph 2.11.8.

2.11.2 EXTRAVEHICULAR MOBILITY UNIT. (See figures 2.11-1, 2.11-2, and 2.11-3.)

The extravehicular mobility unit (EMU) provides life support in an unpressurized cabin, and up to four hours (at 4,800 Btu) of extravehicular life support.

In its extravehicular configuration the EMU is a closed-circuit pressure vessel that envelops the astronaut. The environment inside the pressure vessel consists of 100% oxygen at a nominal pressure of 3.75 psia. The oxygen is provided at a flow rate of 6 cfm. The extravehicular life support equipment configuration includes the following:

- Liquid cooling garment
- Pressure garment assembly
- Integrated thermal micrometeoroid garment
- Portable life support system
- Oxygen purge system
- Communications carrier
- EMU waste management system
- EMU maintenance kit
- PLSS Remote Control Unit
- Extravehicular visor assembly
- Biomedical belt

During intravehicular phases of the mission, several configurations are possible. The basic configuration provides intravehicular life support independent of the ECS, in an unpressurized cabin. It includes the following:

- Pressure garment assembly (with PLSS)
- Integrated thermal micrometeoroid garment
- Communications carrier
- Constant-wear garment (not used in LM)
- Liquid cooling garment

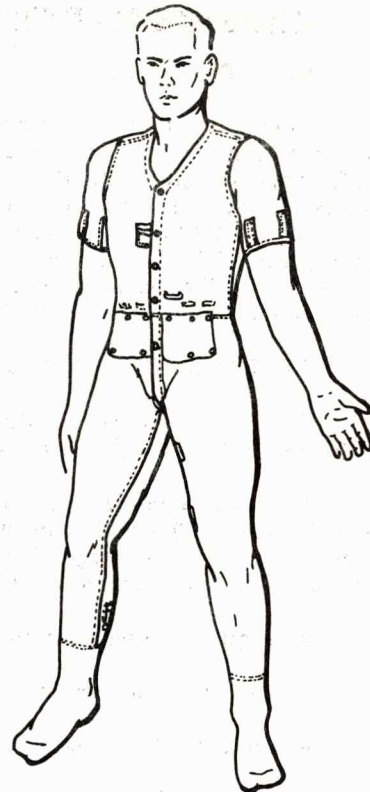
CREW PERSONAL EQUIPMENT



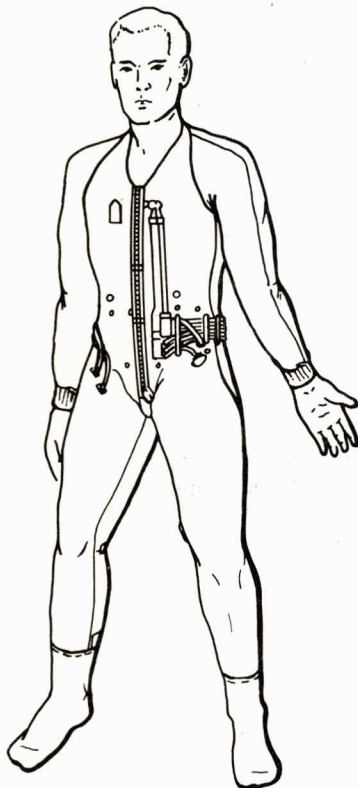
**FECAL
CONTAINMENT
SYSTEM**



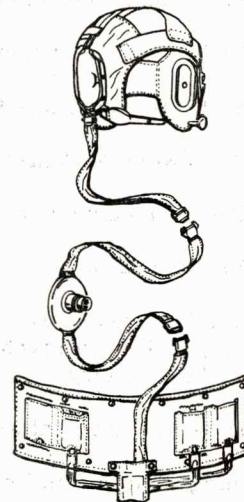
**URINE COLLECTION AND
TRANSFER ASSEMBLY**



CONSTANT WEAR GARMENT



LIQUID COOLED GARMENT



**BIOINSTRUMENTATION HARNESS
AND COMMUNICATIONS SOFT HAT**

Figure 2.11-1. Extravehicular Mobility Unit

300LM4-205

CREW PERSONAL EQUIPMENT

- Urine collection and transfer assembly
- Auxiliary equipment.

2.11.2.1 Liquid Cooling Garment. (See figure 2.11-1.)

The liquid cooling garment (LCG) is worn by the astronauts while in the LM and during all extravehicular activity. It cools the astronaut's body during extravehicular activity by absorbing body heat and transferring excessive heat to the sublimator in the PLSS. The LCG is a one-piece, long-sleeved, integrated-stockings undergarment of netting material. It consists of an inner liner of nylon chiffon, to facilitate donning, and an outer layer of nylon Spandex into which a network of Tygon tubing is woven. The tubing does not pass through the stocking area. A double connector for incoming and outgoing water is located on the front of the garment. Cooled water, supplied from the PLSS, is pumped through the tubing. Pockets for bioinstrumentation signal conditioners are located around the waist. A zipper that runs up the front is used for donning and doffing the LCG; an opening at the crotch is used for urinating. Dosimeter pockets and snaps for attaching a biomedical belt are part of the LCG.

2.11.2.2 Pressure Garment Assembly. (See figures 2.11-2 and 2.11-3.)

The pressure garment assembly (PGA) is the basic pressure vessel of the EMU. It provides a mobile life-support chamber if cabin pressure is lost due to leaks or puncture of the vehicle. The PGA consists of a helmet, torso and limb suit, intravehicular gloves, and various controls and instrumentation to provide the crewman with a controlled environment. The PGA is designed to be worn for 18 hours at a regulated pressure of 3.75 ± 0.25 psig, in conjunction with the LCG, and backpack.

2.11.2.2.1 Torso and Limb Suit.

The torso and limb suit is a flexible pressure garment that encompasses the entire body, except the head and hands. It has four gas connectors, a PGA multiple water receptacle, a PGA electrical connector, and a PGA urine transfer connector for the PLSS/PGA and ECS/PGA interface. The PGA connectors have positive locking devices and can be connected and disconnected without assistance. The gas connectors comprise an oxygen inlet and outlet connector, on each side of the suit front torso. Each oxygen inlet connector has an integral ventilation diverter valve. The PGA multiple water receptacle, mounted on the suit torso, serves as the interface between the LCG multiple water connector and PLSS multiple water connector. A protective external cover provides PGA pressure integrity when the LCG multiple water connector is removed from the PGA water receptacle. The PGA electrical connector, mated with the vehicle or PLSS electrical umbilical, provides a communications, instrumentation, and power interface to the PGA. The PGA urine transfer connector is used to transfer urine from the urine collection transfer assembly (UCTA) to the waste management system.

The PGA urine transfer connector on the suit right leg, permits dumping the urine collection bag without depressurizing the PGA. A PGA pressure relief valve on the suit sleeve, near the wrist ring, vents the suit in the event of overpressurization. The valve opens at approximately 4.8 psig and reseats at 4.7 psig. If the valve does not open, it can be manually overridden. A pressure gage on the other sleeve indicates suit pressure.

A low-pressure warning-tone switch and pressure transducer are part of the PLSS equipment.

2.11.2.2.2 Helmet.

The helmet is a Lexan (polycarbonate) shell with a bubble-type visor, a vent-pad assembly, and a helmet attaching ring. The vent-pad assembly permits a constant flow of oxygen over the inner front surface of the helmet. The astronaut can turn his head within the helmet neck-ring area. The helmet does not turn independently of the torso and limb suit. The helmet has provisions on each side for mounting an extravehicular visor assembly (EVVA). When the LM is unoccupied the helmet bags are stowed on the cabin floor at the crew flight stations. Each bag has a hollow-shell plastic base with a circular channel for the helmet and the EVVA, two recessed holes for glove connector rings, and a slot for the EMU maintenance kit. The bag is made of cloth, with a circumferential zipper; it folds toward the plastic base when empty.

CREW PERSONAL EQUIPMENT

2.11.2.2.3 Intravehicular Gloves.

The intravehicular gloves are worn during operations in the LM cabin. The gloves are secured to the wrist rings of the torso and limb suit with a slide lock; they rotate by means of a ball-bearing race. Freedom of rotation, along with convoluted bladders at the wrists and adjustable antiballooning restraints on the knuckle areas, permits manual operations while wearing the gloves.

2.11.2.2.4 Controls.

All PGA controls are accessible to the crewman during intravehicular and extravehicular operations. The PGA controls comprise two ventilation diverter valves, a pressure relief valve with manual override, and a manual purge valve. For intravehicular operations, the ventilation diverter valves are open, dividing the PGA inlet oxygen flow equally between the torso and helmet of the PGA. During extravehicular operation, the ventilation diverter valves are closed and the entire 6.0-cfm oxygen flow enters the helmet. The valve accommodates flow from a failed-open primary oxygen pressure regulator. If the valve fails open, it may be manually closed. The purge valve interfaces with the PGA through the PGA oxygen outlet connector. Manual operation of this valve initiates an 8-lb/hr purge flow, providing CO₂ washout and minimum cooling during contingency or emergency operations.

2.11.2.2.5 Instrumentation.

A pressure transducer on the right cuff indicates pressure within the PGA. Biomedical instrumentation comprises an EKG (heart) sensor, ZPN (respiration rate) sensor, dc-to-dc converter, and wiring harness. The ZPN sensor is only operational with spacecraft telemetry system. A personal radiation dosimeter (active) is attached to the integrated thermal micrometeoroid garment for continuous accumulative radiation readout. A chronograph wristwatch (elapsed-time indicator), is readily accessible to the crewman for monitoring.

2.11.2.3 Communications Carrier.

The communications carrier (cap) is a polyurethane-foam headpiece with two independent earphones and microphones, which are connected to the suit 21-pin communications electrical connector. The communications carrier is worn with or without the helmet during intravehicular operations. It is worn with the helmet during extravehicular operations.

2.11.2.4 Integrated Thermal Micrometeoroid Garment.

The ITMG worn over the PGA, and protects the astronaut from harmful radiation, heat transfer, and micrometeoroid activity.

The ITMG is a one-piece, form-fitting multilayered garment that is laced over the PGA and remains with it. The EVVA, gloves, and boots are donned separately. From the outer layer in, the ITMG is made of a protective cover, a micrometeoroid-shielding layer, a thermal-barrier blanket (multiple layers of aluminized mylar), and a protective liner. Zippers on the ITMG permits connecting or disconnecting umbilical hoses.

2.11.2.4.1 Extravehicular Gloves.

For extravehicular activity, the PGA gloves are replaced with the extravehicular gloves. The extravehicular gloves are made of the same material as the ITMG to permit handling intensely hot or cold objects outside the cabin and for protection against lunar temperatures.

2.11.2.4.2 Extravehicular Boots.

The extravehicular boots are worn over the PGA boots for extravehicular activity. They are made of the same material as the ITMG. The soles have additional insulation for protection against intense temperatures.

2.11.2.4.3 Extravehicular Visor Assembly. (See figure 2.11-4.)

The EVVA provides protection against solar heat, space particles, and radiation, and helps to maintain thermal balance. The two pivotal visors of the EVVA may be attached to a pivot mounting on the PGA helmet. The lightly tinted (inner) visor reduces fogging in the helmet. The outer visor has a vacuum deposited, gold-film reflective surface, which provides protection against solar radiation and space particles. The EVVA is held snug to the PGA helmet by a tab-and-strap arrangement that allows the visors to be rotated approximately 90° up or down, as desired. An EVVA for each astronaut is stowed on the floor at his station during launch. When the LM is occupied, the EVVA's are stowed in the PGA helmet bags secured on the ascent engine cover.

2.11.2.5 Portable Life Support System. (See figures 2.11-5 and 2.11-6.)

The PLSS is a self-contained, self-powered, rechargeable environmental control system. In the extravehicular configuration of the EMU, the PLSS is worn on the astronaut's back. The PLSS supplies pressurized oxygen to the PGA, cleans, cools, removes CO₂ and the expired gas, circulates cool liquid in the LCG through the liquid transport loop, transmits astronaut biomedical data, and functions as a dual VHF transceiver for communication.

The PLSS has a contoured fiberglass shell to fit the back, and a thermal micrometeoroid protective cover. It has three control valves, and, on a separate control box, two control switches, a volume control, and a five-position switch for the dual VHF transceiver. The remote control unit is set on the chest below the neck ring.

Fully charged, the equipment can assimilate an average astronaut output of 1,600 Btu per hour with peak rates of 2,000 Btu per hour. The PLSS can operate for 4 hours with a maximum of 4,800 Btu in space environment before oxygen and feedwater must be replenished and the battery replaced. The basic systems and loops of the PLSS are: primary oxygen subsystem, oxygen ventilation loop, feedwater loop, liquid transport loop, and electrical system.

The PLSS attaches to the astronaut's back, over the ITMG; it is connected by a shoulder harness assembly. When not in use, it is stowed in the midsection, on the floor. To don the PLSS the astronaut backs against the pack, makes PGA and harness connections, and unhooks the PLSS straps from the overhead attachment.

2.11.2.5.1 Primary Oxygen Subsystem. (See figure 2.11-6.)

The primary oxygen subsystem is rechargeable. It provides regulated oxygen for the oxygen ventilation loop. The subsystem consists of a primary oxygen bottle, recharge connector, primary oxygen regulator assembly, and associated instrumentation.

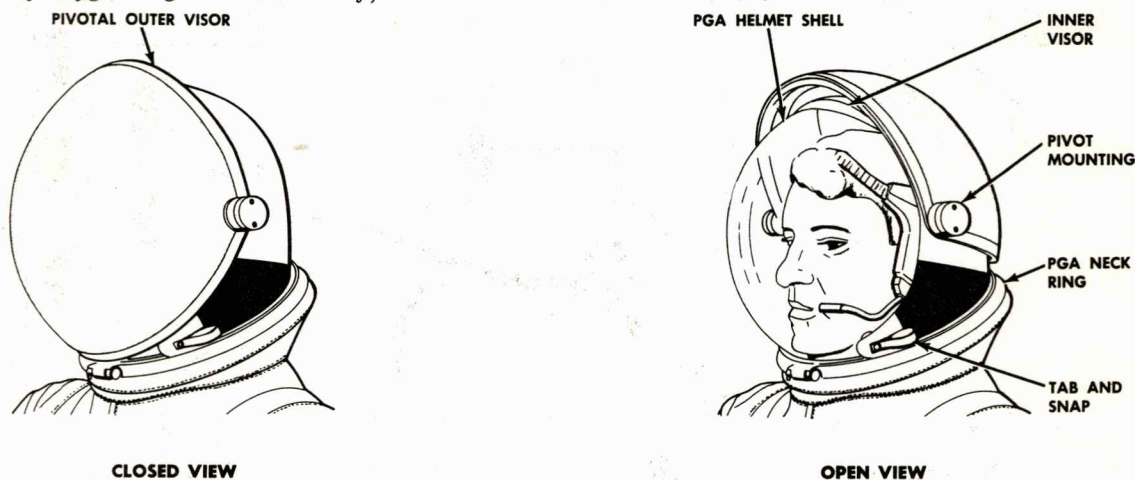


Figure 2.11-4. Extravehicular Visor Assembly

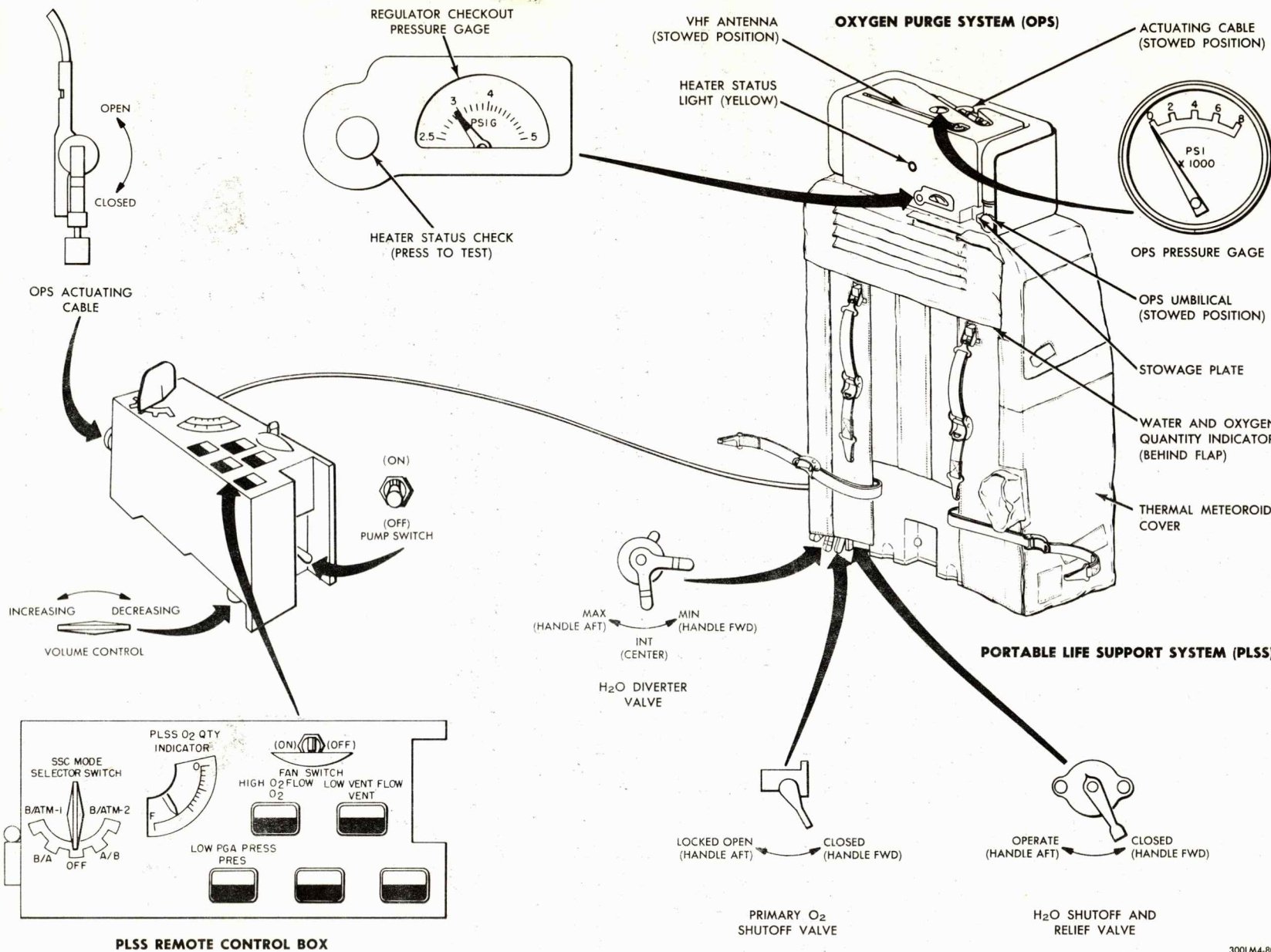


Figure 2.11-5. PLSS Controls

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

Instrumentation. The battery current and voltage transducers share the same telemetry frequency. Manual actuation of the mode selector switch on the PLSS control box determines which transducer output is being telemetered.

2.11.2.5.6 Space Suit Communicator. (See figure 2.11-6.)

The space suit communicator (SCC) provides primary and secondary duplex voice communication and physiological and environmental telemetry. The SSC consists of primary and secondary duplex VHF-AM transceivers, transducers, seven voltage-conditioning oscillators (VCO's), a mixer for VCO coupling to the primary transmitter, two tone generators and associated controls, and a VHF antenna switch to connect the antenna to the primary or secondary transceiver.

Primary and Secondary Duplex VHF-AM Transceivers and Antenna. The primary duplex VHF-AM transceiver can transmit telemetry (table 2.11-1) and voice at 259.7 mc to MSFN and receive at 296.8 mc. The secondary transceiver is identical with the primary transceiver, except that it transmits on 296.8 mc, receives on 259.8 mc, and has no telemetry capability. Each transceiver has an independently regulated power supply of 13.5 volts dc. The power output of the primary and secondary transceivers is 75 milliwatts at the antenna terminals. All EMU data and voice must be relayed through the LM and CSM and transmitted to MSFN via S-band. The VHF antenna is permanently mounted on the oxygen purge system (OPS); absence of the OPS results in loss of EMU communications.

Voltage-Conditioning Oscillators and Mixer. The VCO's have an independently regulated power supply of 10 volts dc. The center frequency deviations of the VCO's are proportional to the changes in the d-c output of the transducers. Each VCO output is coupled at the mixer for telemetry transmission.

Controls. Both primary and secondary volume control potentiometers on the PLSS remote control unit are mechanically coupled to a common shaft. SSC modes are selected with the five-position mode selector switch on the PLSS remote control unit. The functions of the switch positions are as follows:

- The B/A TM-1 position, used for the normal mode of operation, permits voice and telemetry transceiver operation. This position allows battery current to be telemetered by positioning the mode selector switch for telemetry transmission; the voice-operated switch (VOX) is bypassed, establishing a hot-mike mode.

Table 2.11-1. Extravehicular Mobility Unit - Telemetry Measurements

Code	Description	Telemetry Range	Display Range	Crew Display
GT8110P	Feedwater pressure	0 to 5.0 psig		
GT8124J	EKG	0 to 5.0 volts		
GT8140C	PLSS battery current	0 to 10 ampere		
GT8141V	PLSS battery voltage	12 to 22 volts		
GT8154T	LCG H ₂ O inlet temperature	+40° to +90° F		
GT8168P	PGA pressure	2.5 to 5.0 psia		PLSS O ₂ QTY
GT8182P	PLSS O ₂ pressure	0 to 1,110 psia		
GT8196T	LCG H ₂ O ΔT	0° to +15° F		
GT9991U	Space suit/PLSS MPX output	Multiplexed		

CPE

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

- The B/A position energizes the primary transceiver. The primary receiver is continuously operative; the primary transmitter is inoperative unless activated by the VOX. This mode has no telemetry.
- The OFF position removes power from all communications equipment and instrumentation.
- The A/B position energizes the secondary transceiver. The secondary receiver is continuously operative; the secondary transmitter is inoperative unless activated by the VOX. This mode has no telemetry.
- The B/A TM-2 position is functionally identical with the B/A TM-1 position, except that battery voltage is telemetered instead of battery current.

Tone Generators. Both tone generators and the microphones are common to all active mode selector switch positions. The generators provide audible 3- and 1.5-kc warning tones. The generator outputs are coupled directly to the communications cap receiver and isolation networks; they are turned on automatically by high oxygen flow, low vent flow, or low PGA pressure. Both tones are readily distinguishable when activated simultaneously.

Instrumentation. The PLSS O₂ QTY IND receives power from a 10-volt d-c regulator in all active mode selector switch positions.

2.11.2.5.7 PLSS Control Box.

The PLSS control box is a chest-mounted instrumentation and controls unit. It has a fan switch, pump switch, SSC mode selector switch, volume control, PLSS oxygen quantity indicator, five status indicators, and an interface for the OPS actuator.

2.11.2.6 Oxygen Purge System.

The OPS is a self-contained, independently powered, high-pressure, nonrechargeable emergency oxygen system that provides 30 minutes of regulated purge flow. The OPS consists of two interconnected spherical high-pressure oxygen bottles, an automatic temperature control module, an oxygen pressure regulator assembly, a battery, an oxygen connector, and checkout instrumentation. In the normal extravehicular configuration, the OPS is mounted on top of the PLSS and is used with PLSS systems during emergency operations. In the contingency extravehicular configuration, the OPS is attached to the PGA front lower torso and functions independently of the PLSS. The OPS has no communications capability, but provides a hard mount for the SSC VHF antenna. Two OPS's are stowed in the LM.

2.11.2.6.1 OPS Oxygen Bottles.

The OPS oxygen bottles contain a total of 4 pounds of usable oxygen at $7,500 \pm 750$ psig. This supply provides a 30-minute operational lifetime at a purge flow rate of 8 lb/hr.

2.11.2.6.2 OPS Oxygen Pressure Regulator Assembly.

This assembly is a single-stage, variable-orifice regulator that regulates PGA inlet oxygen pressure to 3.45 ± 0.25 psig. The regulator has an inline heater and automatic temperature control module to offset the adiabatic expansion cooling of the oxygen and to maintain PGA inlet oxygen temperature between $+50^\circ$ and $+80^\circ$ F. The manually operated OPS oxygen shutoff valve is mechanically coupled to the heater switch.

2.11.2.6.3 OPS Battery.

The battery supplies 27 volts dc to the 116-watt heater and automatic temperature control module. The battery is rated at 2.5 ampere-hours.

CREW PERSONAL EQUIPMENT

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.11.2.6.4 OPS Oxygen Connectors.

The OPS connector interfaces with the PGA inlet gas connector. The GSE fill connector is used only for initial ground charge.

2.11.2.6.5 OPS Instrumentation.

The regulator and heater have preoperational checkout systems. The OPS pressure gage monitors the OPS oxygen bottles and is used for ground fill and preoperational checkout. The gage verifies regulated flow (8 lb/hr) through an orifice mounted on the stowage plate. The heater circuitry is checked before egress, with a momentary-contact switch which, when actuated, causes a yellow light to go on to indicate circuit continuity.

2.11.2.7 EMU Maintenance Kit.

The EMU maintenance kit consists of an accumulator key for the PLSS, lubricant, spare O-rings, and patching tape for repair of the EMU. The kit is stowed in the helmet stowage bag until needed for extravehicular activity.

2.11.3 UMBILICAL ASSEMBLY.

The umbilical assembly consists of hoses and connectors for securing the PGA to the ECS, Communications Subsystem (CS), and Instrumentation Subsystem (IS). Separate oxygen and electrical umbilicals connect to each astronaut.

2.11.3.1 Oxygen Umbilical.

The oxygen umbilical consists of Fluorol hoses (1.25-inch inside diameter) with wire reinforcement. The connectors are of the quick-disconnect type, with a 1.25-inch, 70° elbow at the PGA end. Each assembly is made up of two hoses and a dual-passage connector at the ECS end and two separate hoses (supply and exhaust) at the PGA end. When not connected to the PGA, the ECS connector end remains attached and the hoses stowed. The hoses are fastened together with keepers and Beta sleeve.

2.11.3.2 Electrical Umbilical.

The electrical umbilical carries voice communications and biomedical data, and electrical power for warning tone impulses.

2.11.4 CREW LIFE SUPPORT.

The crew life support equipment includes food and water, a waste management system, personal hygiene items, and medical equipment for in-flight emergencies. A potable-water unit and food packages contain sufficient life-sustaining supply for completion of the LM mission.

2.11.4.1 Crew Water System.

The water dispenser assembly consists of a Trevarno mounting bracket, a coiled hose, and a trigger-actuated, water dispenser. The hose and dispenser extend approximately 72 inches to dispense water from the ECS water feed control assembly. The ECS water feed control valve is opened to permit water flow. The dispenser assembly supplies water at +50° to +90° F for drinking or food preparation and fire extinguishing. The water dispenser is inserted directly into the mouth for drinking. Pressing the trigger-type control supplies water.

2.11.4.2 Food Preparation and Consumption.

The astronaut's food supply (3,200 calories per man per day) includes liquids and solids with adequate nutritional value and low waste content. Food packages are stowed in the LM midsection and the right hand stowage compartment, on the shelf above PLSS No. 1.

CREW PERSONAL EQUIPMENT

Mission LM 4 Basic Date 15 June 1968 Change Date 15 September 1968 Page 2.11-17

CPE

The food is vacuum packed in plastic bags that have one-way poppet valves into which the water dispenser can be inserted. Another valve allows food passage for eating. The food bags are packaged in aluminum-foil-backed plastic bags for stowage and are color coded: red (breakfast), white (lunch) and blue (snacks).

Food preparation involves reconstituting the food with water. The food bag poppet-valve cover is cut with scissors and pushed over the water dispenser nozzle after its protective cover is removed. Pressing the water dispenser trigger releases water. The desired consistency of the food determines the quantity of water added. After withdrawing the water dispenser nozzle, the protective cover is replaced and the dispenser returned to its stowage position. The food bag is kneaded for approximately 3 minutes, after which the food is considered reconstituted. After cutting off the neck of the food bag, food can be squeezed into the mouth through the food-passage valve. A germicide tablet, attached to the outside of the food bag, is inserted into the bag after food consumption, to prevent fermentation and gas formation. The bag is rolled to its smallest size, banded, and placed in the waste disposal compartment.

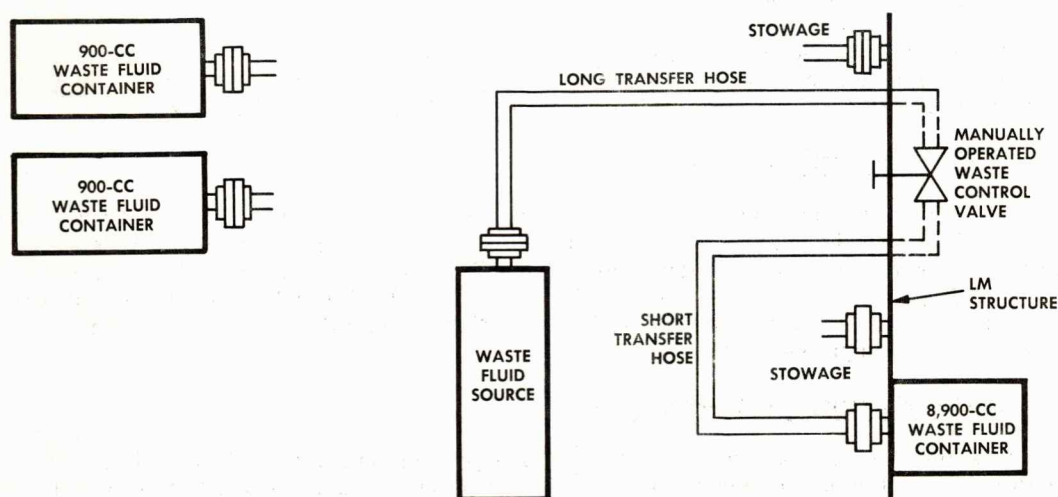
2.11.4.3 EMU Waste Management System (See figure 2.11-8.)

The EMU waste management system provides for the disposal of body waste through use of a fecal containment system, and a urine collection and transfer assembly, and for neutralizing odors. Personal hygiene items are stowed in the right-hand stowage compartment.

2.11.4.3.1 Waste Fluid Transfer.

Waste fluids are transferred to a waste fluid collector assembly by a controlled difference in pressures between the PGA and cabin (ambient). The primary waste fluid collector consists of a long transfer hose, control valve, short transfer hose, and a 8,900-cc multilaminate bag. The long transfer hose is stowed on a connector plate when not in use. To void his in-suit urine container, the astronaut attaches the hose to the PGA quick-disconnect, which has a visual flow indicator. Rotating the handle of the spring-loaded waste control valve controls passage of urine to the assembly. The 8,900-cc bag is in the PLSS LiOH storage unit; the short transfer hose is connected between the waste control valve and the bag.

With cabin pressure normal (4.8 psia), the long transfer hose is removed from the connector stowage plate and attached to the PGA male disconnect. The PGA is overpressurized by ± 2 psia and the waste control valve is opened. Urine flows from the PGA to the collector assembly at a rate of approximately 200 cc per minute. When bubbles appear in flow indicator, the valve handle is released and allowed to close.



300LM4-124

Figure 2.11-8. Waste Management System

A secondary waste fluid collector system provides 900-cc waste fluid containers which attach directly to the PGA. Urine is transferred directly from the PGA, through the connectors, to the bags. These bags can then be emptied into the 8,900-cc collector assembly.

2.11.4.3.2 Fecal Containment System.

The fecal system consists of an outer fecal/emesis bag (one layer of Aclar) and a smaller inner bag. The inner bag has waxed tissue on its inner surface. Polyethylene-backed toilet tissue and a disinfectant package are stored in the inner bag.

2.11.4.4 Personal Hygiene Items.

Personal hygiene items consist of wet and dry cleansing cloths, chemically treated and sealed in plastic covers, and packs of teeth-cleansing, ingestable chewing gum. The cloths measure 4 by 4 inches and are folded into 2-inch squares. They are stored in the food package container along with the chewing gum.

2.11.4.5 Medical Equipment.

The medical equipment consists of biomedical sensors, personal radiation dosimeters, and emergency medical equipment.

2.11.4.5.1 Bioinstrumentation.

Biomedical sensors gather physiological data for telemetry. Government-furnished impedance pneumographs continuously record heart beat (EKG) and respiration rate. Each assembly (one for each astronaut) has four electrodes which contain electrolyte paste; they are attached with tape to the astronaut's body.

2.11.4.5.2 Personal Radiation Dosimeters.

Six dosimeters are provided for each astronaut. They contain thermoluminescent powder, nuclear emulsions, and film that is sensitive to beta, gamma, and neutron radiation. They are placed on the forehead or right temple, chest, wrist, thigh, and ankle to detect radiation to eyes, bone marrow and skin. Serious, perhaps critical, damage results if radiation dosage exceeds a predetermined level. For quick, easy reference each astronaut has a dosimeter mounted on his EMU. Dosimeters are government-furnished equipment.

2.11.4.5.3 Emergency Medical Equipment.

The emergency medical equipment consists of a kit of six capsules: four pain killers (Darvon) and two pep pills (Dexedrine). The kit is attached to the interior of the flight data file, readily accessible to both astronauts. The emergency medical equipment is government furnished.

2.11.5 CREW SUPPORT AND RESTRAINT EQUIPMENT. (See figure 2.11-9.)

The crew support and restraint equipment includes armrests, handholds (grips), Velcro on the floor to interface with the boots, and a restraint assembly operated by a rope-and-pulley arrangement that secures the astronauts in an upright position under zero-g conditions.

2.11.5.1 Armrests.

The armrests, at each astronaut station, provide stability for operation of the thrust/translation controller assembly and the attitude controller assembly, and restrain the astronaut laterally.

CREW PERSONAL EQUIPMENT

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

The armrests are adjustable (four positions) to accommodate the astronaut; they also have stowed (fully up) and docking (fully down) positions. The armrests, held in position by spring-loaded detents, can be moved from the stowed position by grasping them and applying downward force. Other positions are selected by pressing latch buttons on the armrest forward area. Shock attenuators are built into the armrests for protection against positive-g forces (lunar landing). The maximum energy absorption of the armrest assembly is a 300-pound force, which will cause a 4-inch armrest deflection.

2.11.5.2 Handholds.

The handholds, at each astronaut station and at various locations around the cabin, provide support for the upper torso when activity involves turning, reaching, or bending; they attenuate movement in any direction. The forward panel handholds are single upright, peg-type, metal grips. They are fitted into the forward bulkhead, directly ahead of the astronauts, and can be grasped with the left or right hand.

2.11.5.3 Restraint Assembly.

The restraint assembly consists of ropes, restraint rings, and a constant-force reel system. The ropes attach to rings on the PGA sides, waist high. The constant-force reel control provides a downward force of approximately 30 pounds; it is locked during landing or docking operations. When the constant-force reel is locked, the ropes are free to reel in. A ratchet stop prevents paying out of the ropes and thus provides bonding restraint. During docking maneuvers, the Commander uses pin adjustments to enable him to use the crewman optical alignment sight (COAS) at the overhead (docking) window.

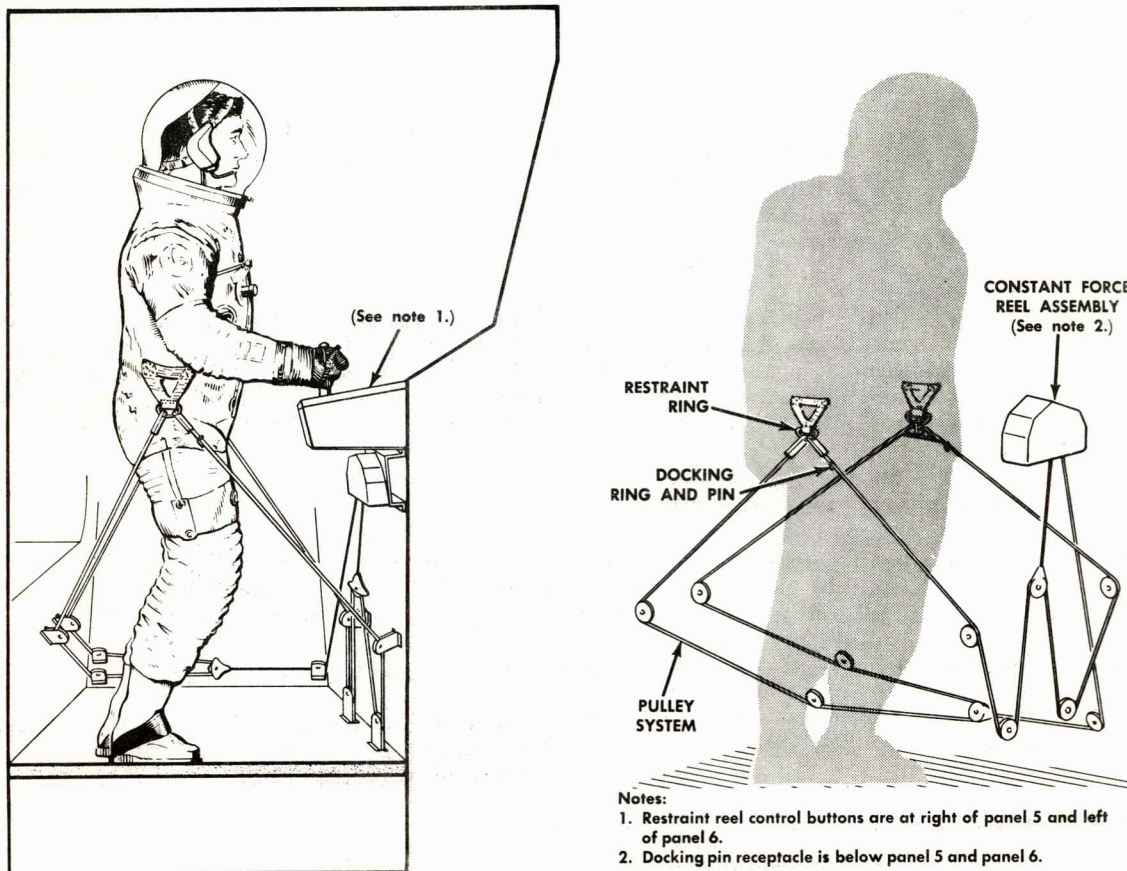


Figure 2.11-9. Restraint Equipment

CREW PERSONAL EQUIPMENT

2.11.6 DOCKING AIDS AND TUNNEL HARDWARE.

Docking operations require special equipment and tunnel hardware to effect linkup of the LM with the CSM. Docking equipment includes the COAS and a docking target. A drogue assembly, probe assembly, the SM forward hatch, and hardware inside the LM tunnel enable completion of the docking maneuver.

2.11.6.1 Crewman Optical Alignment Sight (See figure 2.11-10.)

The COAS provides the Commander with gross range cues and closing rate cues during the docking maneuver. The closing operation, from 150 feet to contact, is an ocular, kinesthetic coordination that requires control with minimal use of fuel and time. The COAS provides the Commander with a fixed line-of-sight attitude reference image, which appears to be the same distance away as the target.

The COAS is a collimating instrument. It weighs approximately 1.5 pounds, is 8 inches long, and operates from a 28-volt d-c power source. The COAS consists of a lamp with an intensity control, a reticle, a barrel-shaped housing and mounting track, and a combiner and power receptacle. The reticle has vertical and horizontal 10° gradations in a 10° segment of the circular combiner glass, on an elevation scale (right side) of -10° to +31.5°. The COAS is capped and secured to its mount above the left window (position No. 1).

To use the COAS, it is moved from position No. 1 to its mount on the overhead docking window frame (position No. 2) and the COAS switch (panel 8) is set from OFF to OVHD. The intensity control is turned clockwise until the reticle appears on the combiner glass; it is adjusted for required brightness.

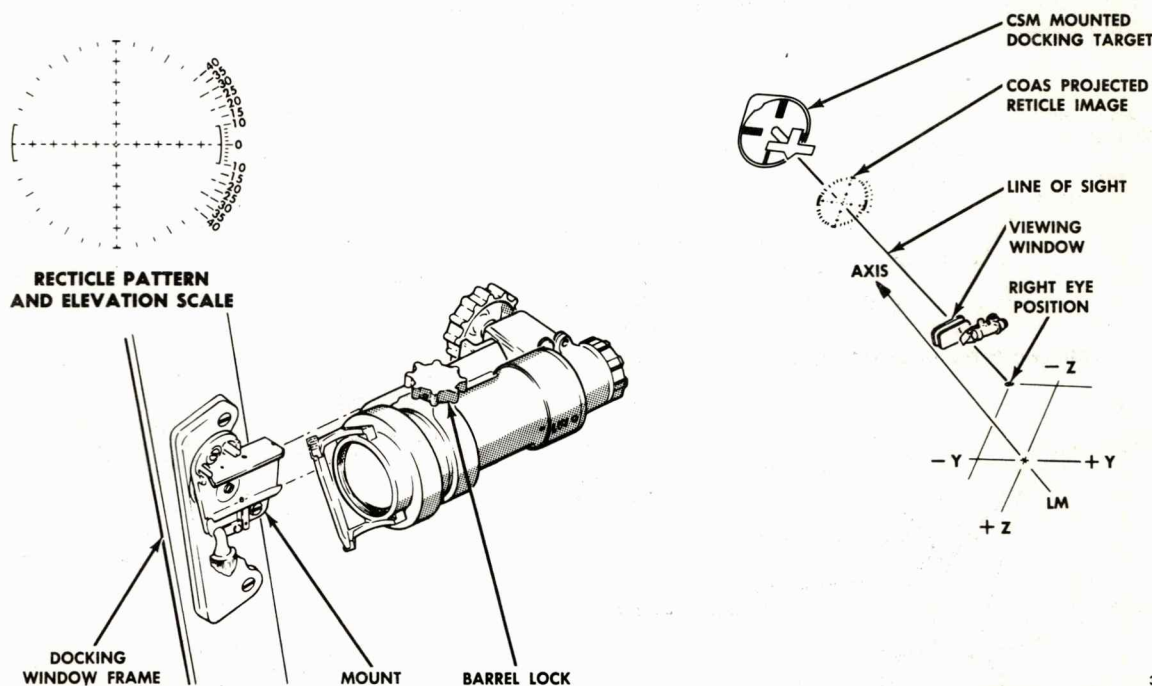


Figure 2.11-10. Crewman Optical Alignment Sight

2.11.6.2 Docking Target. (See figure 2.11-11.)

The docking target permits docking to be accomplished on a three-dimensional alignment basis. The target is located at the intersection of +Y46.3 and +Z203; it does not protrude beyond the LM launch configuration envelope or the maximum contact envelope. The target consists of an inner circle and a standoff cross of black within an outer circumference of white. The target-base diameter is 17.68 inches. The standoff cross is centered 15 inches higher than the base and, as seen at the intercept, is parallel to the X-axis and perpendicular to the Y-axis and the Z-axis.

2.11.6.3 Drogue Assembly.

The drogue assembly consists of a conical structure mounted within the LM docking tunnel. It is secured at three points on the periphery of the tunnel, below the LM docking ring. The LM docking ring is part of the LM midsection outer structure, concentric with the X-axis. The drogue assembly can be removed from the CSM end or LM end of the tunnel.

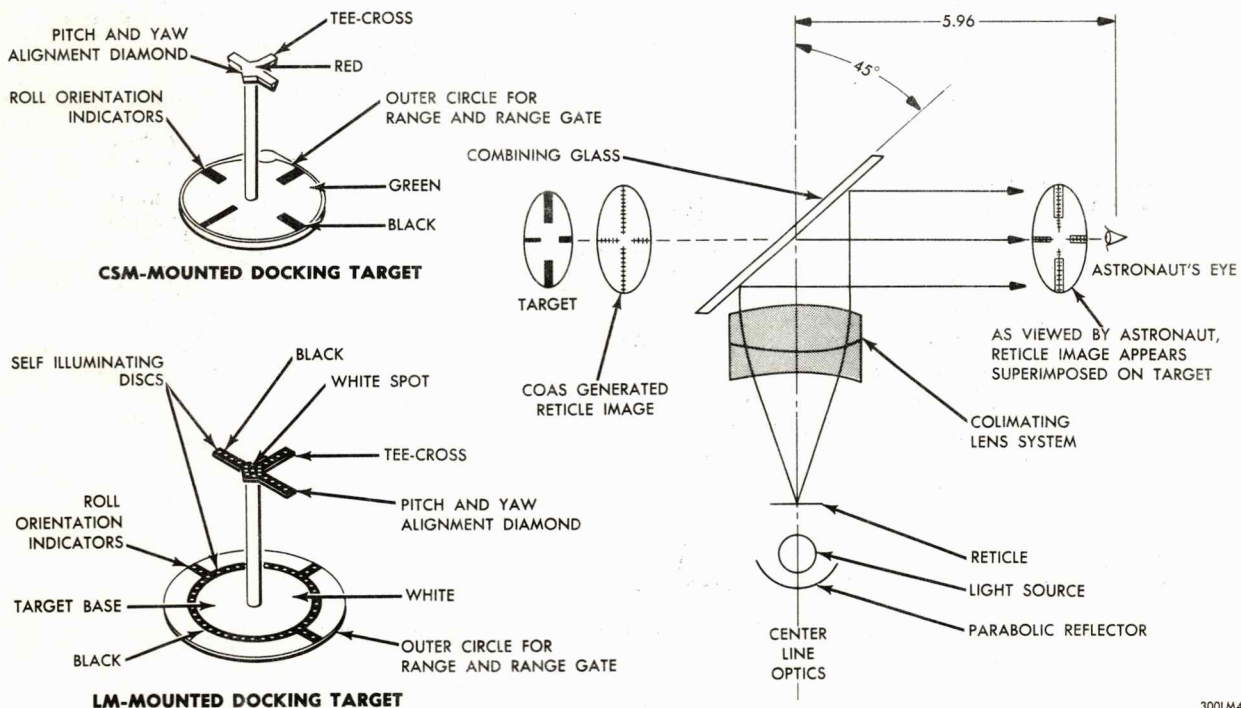
Basically, the assembly is a three-section aluminum cone that is secured with mounting lugs to the LM tunnel ring structure. A lock and release mechanism on the probe, controls capture of the CSM probe at CSM-LM contact. Handles are provided to release the drogue from its tunnel mounts.

2.11.6.4 Tunnel Hardware.

The tunnel contains hardware essential to final docking operations. This includes connectors for the electrical umbilicals, docking latches, probe-mounting lugs, tunnel lights, and deadfacing switches.

2.11.6.5 Probe Assembly.

The probe assembly provides initial CSM-LM coupling and attenuates impact energy imposed by vehicle contact. The probe assembly may be folded for removal and for stowage within either end of the CSM transfer tunnel.



300LM4-123

Figure 2.11-11. COAS Functional Schematic and Docking Targets

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.11.6.6 CSM Forward Hatch.

The CSM tunnel has two forward hatches: an inner (forward pressure) hatch and an outer (forward thermal) hatch. The forward pressure hatch can be removed into the CSM crew compartment. The hatch is retained at the aft end of the CSM tunnel by a latching mechanism whose linkage is driven from the crew compartment by a handle. A socket opposite the handle, on the tunnel side of the pressure hatch, permits hatch removal from the LM side.

The CSM forward thermal hatch can be removed from the tunnel, into the CSM crew compartment or into the LM. The hatch is retained at the forward end of the CSM tunnel by an expanding peripheral latching mechanism, which also serves as a water barrier. The linkage that operates the mechanism is driven by an off-center handle, from the CSM crew compartment side, and by a socket, using emergency tool B, on the LM side. The thermal hatch is reinstalled from the CSM side only; three of the 12 docking latches have expendable stops, which position the thermal hatch axially for proper engagement of the latching mechanism. The CSM thermal hatch is stowed in the CSM right equipment bay, within a thermal stowage bag.

2.11.7 CREW MISCELLANEOUS EQUIPMENT.

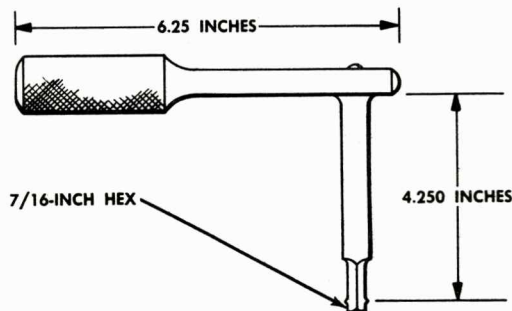
Miscellaneous equipment required for completion of crew operations consists of in-flight data with checklists, emergency tool B, and window shades.

2.11.7.1 In-flight Data.

The in-flight data are provided in three polyethylene-coated packages. The Commander's checklist is stowed at his station. The in-flight data kit is stowed in a stowage compartment. The packages include the flight plan, experiments data and checklist, mission log and data book, systems data book, and star charts.

2.11.7.2 Tool B. (See figure 2.11-12.)

Tool B (emergency wrench) is a modified Allen-head L-wrench. It is 6.25 inches long and has a 4.250-inch drive shaft with a 7/16-inch drive. The wrench can apply a torque of 1,475 inch-pounds; it has a ball-lock device to lock the head of the drive shaft. The wrench is stowed on the right side stowage area inside the cabin. It is a contingency tool for use with the probe and drogue, battery removal from descent stage, CM hatch from outside, and with the water dispenser. Tool B is also used for CM forward hatch operations from the LM side only.



300LM4-126

Figure 2.11-12. Tool B

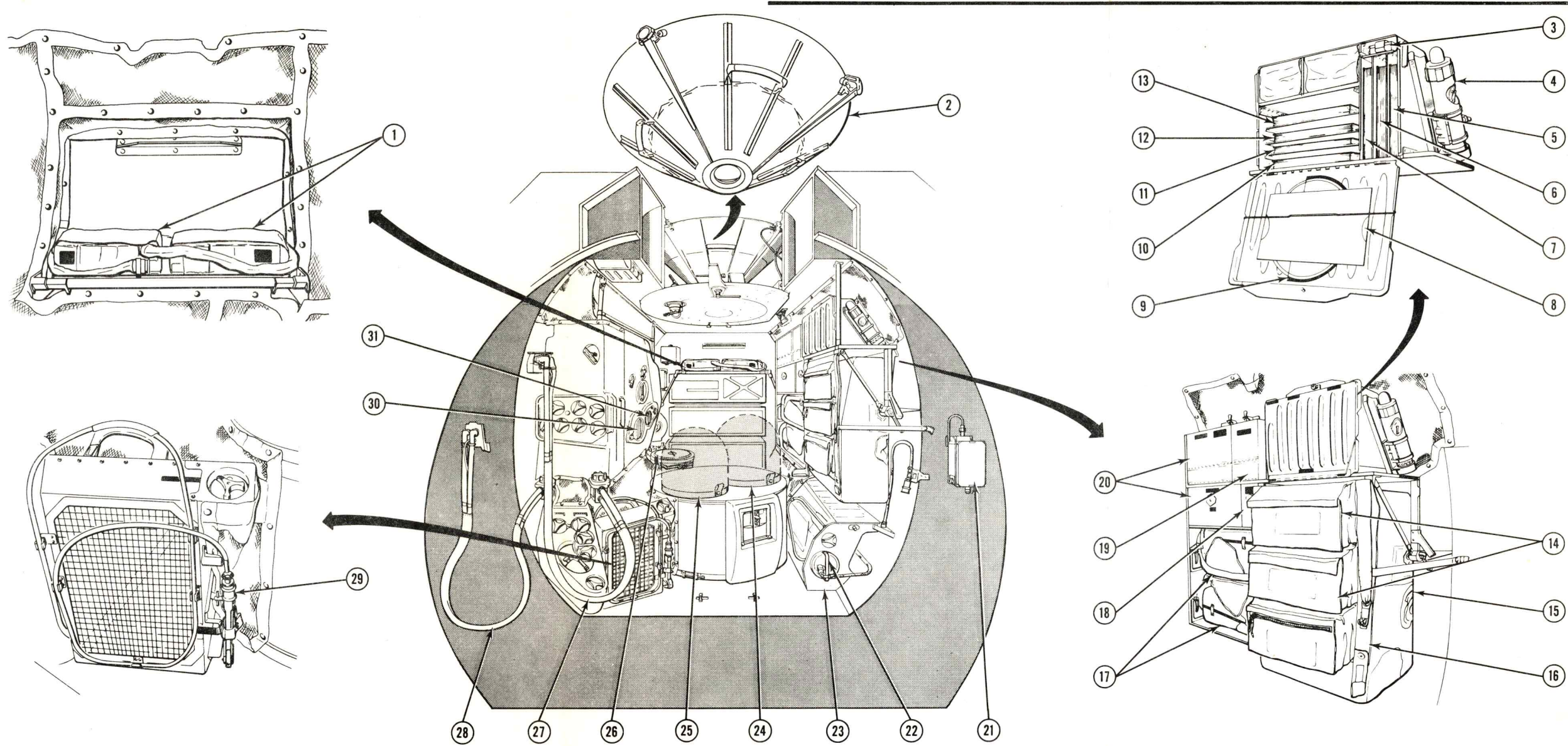
LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

2.11.7.3 Window Shades.

Window shades are used for the overhead docking window and forward windows. The window shade material is Aclar. The surface facing outside the cabin has a highly reflective metallic coating. The shade is secured at the bottom (rolled position). To cover the window, the shade is unrolled, flattened against the frame area and secured with velcro fasteners.

2.11.8 STOWAGE LOCATIONS.

Figure 2.11-14 shows actual stowage positions for crew personal equipment. Some items can be shown only in areas closest to actual stowage locations. The equipment is identified as contractor-furnished equipment (CFE), or government-furnished equipment (GFE).

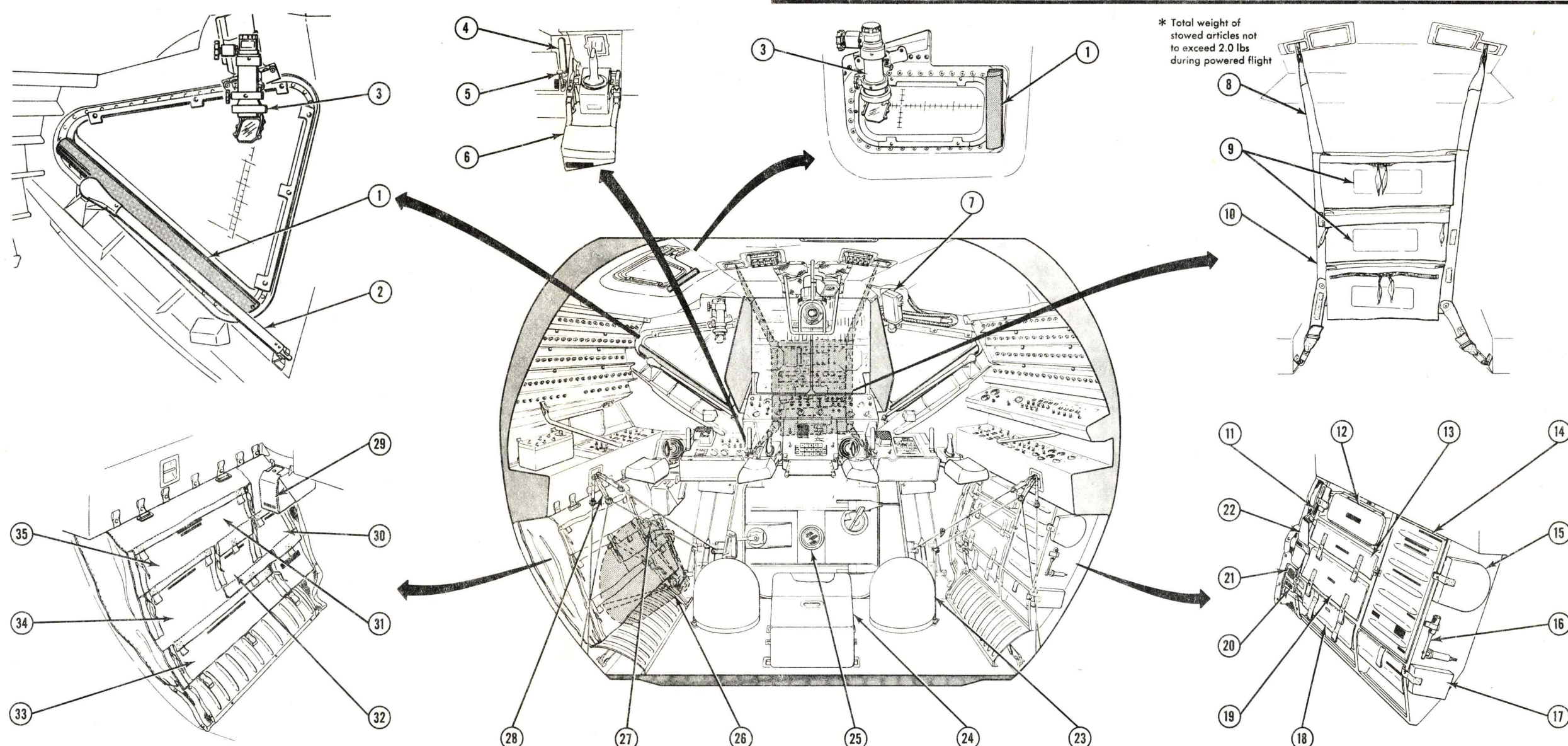


ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
1	PLSS remote control unit (2)	center — Z 27 bulkhead	114	GFE
2	Docking drogue	Inside docking tunnel (launch position)	112	CFE
3	LM medical pkg.	on board data file	135	
4	Radiation survey meter	on board data file	134	
5	LM hard suit handbook	on board data file	135	GFE
6	LM soft suit handbook	on board data file	135	GFE
7	LM procedures	on board data file	135	GFE
8	Mercator Lunar star chart	on board data file	135	GFE
9	Polar Lunar star chart	on board data file	135	GFE
10	Lunar descent aids	on board data file	135	GFE
11	LM systems data	on board data file	134	GFE
12	Flight plan	on board data file	134	GFE
13	Crew log	on board data file	134	GFE

ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
14	Utility light assy (2)	In interim stowage assy	134	CFE
15	PLSS	Recharge station	107	GFE
16	Interim stowage assy	Front of PLSS (Position No. 2)	109	CFE
17	Oxygen purge system (2) and SRC/OPS adapter	Aft of PLSS		GFE/CFE
18	Pilot's preference kit	Above OPS	104	GFE
19	Food (3 pkgs) 2 man days	LH midsection	105	GFE
20	Lunar overshoes (2 pair)	LH midsection	130	CFE
21	Data storage electronic assembly	+ Z 27 bulkhead	115	CFE
22	Urine Collection Assy	In waste management section	131	CFE
23	Waste management section	LH midsection		CFE
24	LMP helmet stowage bag, EVVA, EV gloves, purge fitting	Ascent engine cover (position 2)	129	GFE

ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
25	CDR helmet stowage bag, EVVA, EV gloves, EMU maintenance kit, purge fitting	Ascent engine cover (position 2)	128	GFE
26	LiOH canister	Midsection, aft of ascent engine cover	116	CFE
27	O ₂ umbilical assy (CDR)	RH midsection		CFE
28	O ₂ Umbilical assy (LMP)	RH side		CFE
29	Water dispenser fire extinguishing assy	Fwd of engine cover	113	CFE
30	LiOH cartridge (ECS)	RH aft midsection	190	CFE
31	LiOH cartridge (PLSS)	RH aft midsection	190	CFE

Figure 2.11-13. Stowage Locations, and Crew Personal Equipment



ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
1	Window shade (3)	Forward and overhead windows		GFE
2	Crash bar (2)	Forward windows		
3	COAS	Left window (position No. 1) Overhead window (position No. 2)		CFE
4	Handgrtp (4)	Forward of CDR's and LMP's stations		CFE
5	Restraint reel controls (2)	Lower right panel 5, lower left panel 6		CFE
6	Arm rests (4)	CDR's and LMP's stations		CFE
7	16 mm data acquisition camera/ 18 mm lens, 16 mm magazine, power cable	Above right window		GFE
8	Interim stowage assembly	(position No. 1)	109	CFE *
9	Utility light assy (2)	In interim stowage assy	134	CFE
10	Lunar orbital map & star chart	Interim stowage assy	109	GFE
11	Mirror	RHSSC	125	CFE
12	70 mm hasselblad camera, container, 70 mm film magazine (2)	RHSSC	127	GFE/CFE
13	Defecation collecting device, emesis bag (4)	RHSSC	123	CFE

ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
14	16 mm film magazine, S/W angle camera, 16 mm film magazine container, Hasselblad magazine EVA camera tether, polarizing filter, 16 mm magazine container	Camera compartment	117	GFE/CFE
15	PLSS checkout cable			CFE
16	Emergency tool B	RHSSC	118	GFE
17	LM utility towel assy		119	GFE
18	Food (3 pkg) 2 man days	RHSSC	120	GFE
19	Large & small urine collection assemblies (2 each)	RHSSC	121	GFE
20	COAS light bulb	RHSSC	122	GFE
21	AOT eyeguard assy, and high density filter	RHSSC	126	GFE
22	EVA waist tether assy & Lunar equip conv assy	RHSSC	124	GFE

ITEM	NOMENCLATURE	LOCATION	PKG. NO.	REMARKS
23	Helmet stowage bag (2)	CDR's and LMP's stations (position No. 1)	128	GFE
24	PLSS	Between crew flight stations		GFE
25	Anti bacteria filter stowage		108	GFE
26	Docking drogue (stowed)	Left of CDR's station (stowed position)	112	
27	Docking restraint rings	On restraint harness cable		CFE
28	Restraint cables and hooks	CDR's and LMP's stations		CFE
29	CDR's check list	Below TTC		
30	Disposable container assy	Disposal bag		CFE
31	ECS LiOH canister			These items are disposed of on lunar surface
32	Small urine collection assy (2)			
33	PLSS condensate container			
34	PLSS LiOH cartridge			
35	Large urine collection assy (2)			

A-300LM4-138-1

Figure 2.11-14. Stowage Locations, and Crew Personal Equipment

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2	CAUTION LIGHTS				
6DS21	ASC HI REG (MPS)	Goes on if helium pressure downstream of regulators in ascent helium lines exceeds 220 psia.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Goes off when pressure is reduced to nominal values.
6DS22	ASC QTY (MPS)	Goes on if fuel or oxidizer in ascent tank is sufficient for only 10-second burn.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Signal permits manual shutdown of ascent engine to prevent propellant depletion.
6DS23	ENG GMBL (MPS)	Lights if difference between gimbal drive signal and gimbal response from Y- or Z-axis trim function is sensed.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	GDA signal is enabled only when descent engine is armed and is disabled after separation of ascent and descent stages.
6DS26	INVERTER (EPS)	Goes on if a-c bus voltage is less than 112 volts, or if a-c bus frequency is less than 398 cps or exceeds 402 cps.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Light goes off when within-tolerance conditions are restored or when INVERTER switch is set to OFF.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks	
Panel 2 (cont)	6DS27	BATTERY (EPS)	Goes on if overcurrent, reverse current, or overtemperature occurs in any descent or ascent battery.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Light goes off when nominal conditions are restored, or if affected battery is turned off.
	6DS28	RNDZ RDR (GN&CS)	Goes on if RR data-not-good condition occurs.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Signal is enabled only when rendezvous radar selector switch is set to AUTO TRACK. Light goes off when data-good condition is restored or RR is turned off.
	6DS29	LDG RDR (GN&CS)	Goes on if LR range data are not good or if velocity data are not good.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Light goes off when range and velocity data-good condition is restored.
	6DS30	PRE AMPS (RCS)	Goes on if either of two parallel redundant -4.7-volt d-c regulated power supplies that provide power to RCS jet preamplifiers are out of tolerance.	LTG: ANUN/ DOCK/ COMPNT (panel 11 and/or panel 16)	CDR's 28-vdc bus LMP's 28-vdc bus	Light goes off when DES BATS switch is set to DEADFACE or ABORT pushbutton is pressed.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S12 8FL10	SYSTEM A THRUSTER PAIR QUAD 2 switch OPEN Center CLOSE	Selects inputs to two normally open latching, solenoid-operated, shutoff, isolation valves in propellant lines upstream of TCA's, to control propellant flow to TCA No. A2D and A2A. Energizes two solenoids that drive isolation valves to open position. Removes solenoid excitation. Energizes two solenoids that drive isolation valves to closed position and feeds signal to ACEA to inhibit automatic operation of TCA's.	RCS SYS A: ISOL VLV (panel 16)	CDR's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight. Gray talkback is displayed when valves are open. Isolation valves remain in last commanded position. Barber-pole talkback is displayed when isolation valves are closed.
	SYSTEM A THRUSTER PAIR QUAD 2 talkback Gray display Barber-pole display Red display	Indicates condition of two isolation valves in propellant lines upstream of TCA's. Indicates two isolation valves controlled by QUAD 2 switch directly below talkback are closed. Indicates isolation valves are closed. Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			LMP's 28-vdc bus Gray display can be obtained with only one isolation valve open. Both isolation valves must be closed to obtain a barber-pole display. If TCA fails, SYSTEM A THRUSTER PAIR QUAD 2 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed and RCS TCA warning light goes out.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont)	8S15	SYSTEM A THRUSTER PAIR QUAD 3 switch	RCS SYS A: ISOL VLV (panel 11)	CDR's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight.
	OPEN	Energizes two solenoids that drive isolation valves to open position.			Gray talkback is displayed when valves are open.
	Center	Removes solenoid excitation.			Isolation valves remain in last commanded position.
	CLOSE	Energizes two solenoids that drive isolation valves to closed position and feeds signal to ACEA to inhibit automatic operation of TCA's.			Barber-pole talkback is displayed when isolation valves are closed.
8FL13	SYSTEM A THRUSTER PAIR QUAD 3 talkback	Indicates condition of two isolation valves in propellant lines upstream of TCA's.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	
	Gray display	Indicates two isolation valves controlled by QUAD 3 switch directly below talkback are open.			Gray display can be obtained with only one isolation valve open.
	Barber-pole display	Indicates isolation valves are closed.			Both isolation valves must be closed to obtain a barber-pole display.
	Red display	Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			If TCA fails, SYSTEM A THRUSTER PAIR QUAD 3 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed and RCS TCA warning light goes out.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont)	8S16 SYSTEM A THRUSTER PAIR QUAD 4 switch	Selects inputs to two normally open, latching, solenoid-operated, shut-off, isolation valves in propellant lines upstream of TCA's, to control propellant flow to TCA No. A4D and A4R.	RCS SYS A: ISOL VLV (panel 11)	CDR's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight.
	OPEN Center CLOSE	Energizes two solenoids that drive isolation valves to open position. Removes solenoid excitation. Energizes two solenoids that drive isolation valves to closed position and feeds a signal to ACEA to inhibit automatic operation of TCA's.			Gray talkback is displayed when valves are open. Isolation valves remain in last commanded position. Barber-pole talkback is displayed when isolation valves are closed.
8FL14	SYSTEM A THRUSTER PAIR QUAD 4 talkback	Indicates condition of two isolation valves in propellant lines upstream of TCA's.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	
	Gray display	Indicates open condition of two isolation valves controlled by QUAD 4 switch directly below status flag.			Gray display can be obtained with only one isolation valve open.
	Barber-pole display	Indicates isolation valves are closed.			Both isolation valves must be closed to obtain a barber-pole display.
	Red display	Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			If TCA fails, SYSTEM A THRUSTER PAIR QUAD 4 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed and RCS TCA warning light goes out.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S10	SYSTEM B ASC FEED 1 switch	Selects inputs to two latch-type solenoid-operated interconnect valves in ascent propellant line, which control flow of propellant from ascent propellant tanks to RCS thrusters.	RCS SYS B: ASC FEED 1 (panel 16)	LMP's 28-vdc bus	Switch is spring-loaded to center position. In event of RCS malfunction, ascent propulsion section can supply propellant to 8 or 16 TCA's, (if LM is experiencing a force in +X-direction). This can be accomplished by setting ASC FEED 1 switch for system A or B, or both, to ASC FEED 1 and MAIN SOV switch for system A or B, or both, to CLOSE. (ASC FEED 2 switch must be set to ASC FEED 2.)
	ASC FEED 1	Energizes solenoids that drive ascent propellant interconnect valves to open position.			Gray talkback is displayed when interconnect valves are open.
	Center	Removes solenoid excitation.			Interconnect valves remain in last commanded position.
	CLOSE	Energizes solenoids that drive ascent propellant interconnect valves to closed position.			Barber-pole talkback is displayed when interconnect valves are closed.
8FL8	SYSTEM B ASC FUEL talkback		RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	
	Gray display	Indicates ascent fuel interconnect valves are open.			Both ascent fuel interconnect valves must be open to obtain a gray display.
	Barber-pole display	Indicates interconnect valves are closed.			Barber-pole display can be obtained with only one ascent fuel interconnect valve closed.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S23	SYSTEM B ASC FEED 2 switch	Selects inputs to two latch-type solenoid-operated interconnect valves in ascent propellant line, which control flow of propellant from ascent propellant tanks to RCS thrusters.	RCS SYS B: ASC FEED 2 (panel 16)	LMP's 28-vdc bus	Switch is spring-loaded to center position. In event of RCS malfunction, ascent propulsion section can supply propellant to 8 or 16 TCA's, (if LM is experiencing a force in +X-direction). This can be accomplished by setting ASC FEED 2 switch for system A or B, or both, to ASC FEED 2 and MAIN SOV switch for system A or B, or both, to CLOSE. (ASC FEED 1 switch must be set to ASC FEED 2.) Gray talkback is displayed when interconnect valves are open. Interconnect valves remain in last commanded position. Barber-pole talkback is displayed when interconnect valves are closed.
	ASC FEED 2	Energizes solenoids that drive ascent propellant interconnect valves to open position.			
	Center	Removes solenoid excitation.			
	CLOSE	Energizes solenoids that drive ascent propellant interconnect valves to closed position.			
	SYSTEM B ASC OXID talkback				
	Gray display	Indicates ascent oxidizer interconnect valves are open.			
8FL19	Barber-pole display	Indicates interconnect valves are closed.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	Both ascent oxidizer interconnect valves must be open to obtain a gray display. Barber-pole display can be obtained with only one ascent oxidizer interconnect valve closed.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S13	SYSTEM B THRUSTER PAIR QUAD 1 switch OPEN Center CLOSE	Selects inputs to two normally open, latching, solenoid-operated, shut-off, isolation valves in propellant lines upstream of TCA's, to control propellant flow to TCA No. B1D and B1L. Energizes two solenoids that drive isolation valves to open position. Removes solenoid excitation. Energizes two solenoids that drive isolation valves to closed position and feeds signal to ACEA to inhibit automatic firing of TCA's.	RCS SYS B: ISOL VLV (panel 16)	LMP's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight. Gray talkback is displayed when valves are open. Isolation valves remain in last commanded position. Barber-pole talkback is displayed when isolation valves are closed.
	8FL11 SYSTEM B THRUSTER PAIR QUAD 1 talkback Gray display Barber-pole display Red display	Indicates condition of two isolation valves in propellant lines upstream of TCA's. Indicates two isolation valves controlled by QUAD 1 switch directly below talkback are open. Indicates isolation valves are closed. Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			A gray display can be obtained with only one isolation valve open. Both isolation valves must be closed to obtain a barber-pole display. If TCA fails, SYSTEM A THRUSTER PAIR QUAD 1 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed, and RCS TCA warning light goes out.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S14	SYSTEM B THRUSTER PAIR QUAD 2 switch	Selects inputs to two normally open, latching, solenoid-operated, shut-off, isolation valves in propellant lines upstream of TCA's, to control propellant flow to TCA No. B2U and B2L.	RCS SYS B: ISOL VLV (panel 16)	LMP's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are closed during lunar-stay shut-down; they are used for emergency shutdown during flight.
	OPEN	Energizes two solenoids that drive isolation valves to open position.			Gray talkback is displayed when valves are open.
8FL12	Center	Removes solenoid excitation.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	Isolation valves remain in last commanded position.
	CLOSE	Energizes two solenoids that drive isolation valves to closed position and feeds signal to ACEA to inhibit automatic firing of TCA's.			Barber-pole talkback is displayed when isolation valves are closed.
8FL12	SYSTEM B THRUSTER PAIR QUAD 2 talkback	Indicates condition of two isolation valves in propellant lines upstream of TCA's.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	A gray display can be obtained with only one isolation valve open.
	Gray display	Indicates two isolation valves controlled by QUAD 2 switch directly below talkback are open.			Both isolation valves must be closed to obtain a barber-pole display.
8FL12	Barber-pole display	Indicates isolation valves are closed.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	If TCA fails, SYSTEM A THRUSTER PAIR QUAD 2 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed, and RCS TCA warning light goes out.
	Red display	Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont)	8S17 SYSTEM B THRUSTER PAIR QUAD 3 switch	Selects inputs to two normally open, latching, solenoid-operated, shut-off, isolation valves in propellant lines upstream of TCA's to control propellant flow to TCA No. B3D and B3A.	RCS SYS B: ISOL VLV (panel 16)	LMP's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight.
	OPEN Center CLOSE	Energizes two solenoids that drive isolation valves to open position. Removes solenoid excitation. Energizes two solenoids that drive isolation valves to closed position and feeds signal to ACEA to inhibit automatic firing of TCA's.			Gray talkback is displayed when valves are open. Isolation valves remain in last commanded position. Barber-pole talkback is displayed when isolation valves are closed.
8FL15	SYSTEM B THRUSTER PAIR QUAD 3 talkback	Indicates condition of two isolation valves in propellant lines upstream of TCA's.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	A gray display can be obtained with only one isolation valve open. Both isolation valves must be closed to obtain a barber-pole display. If TCA fails, SYSTEM A THRUSTER PAIR QUAD 3 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed, and RCS TCA warning light goes out.
	Gray display	Indicates two isolation valves controlled by QUAD 3 switch directly below talkback are open.			
	Barber-pole display	Indicates isolation valves are closed.			
	Red display	Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont)	8S18	SYSTEM B THRUSTER PAIR QUAD 4 switch	RCS SYS B: ISOL VLV (panel 16)	LMP's 28-vdc bus	Switch is spring loaded to center position. Isolation valves are used for emergency shutdown during flight.
	OPEN	Energizes two solenoids that drive isolation valves to open position.			Gray talkback is displayed when valves are open.
	Center	Removes solenoid excitation.			Isolation valves remain in last commanded position.
	CLOSE	Energizes two solenoids that drive isolation valves to closed position and feed signal to ACEA to inhibit automatic firing of TCA's.			Barber-pole talkback is displayed when isolation valves are closed.
	8FL16	SYSTEM B THRUSTER PAIR QUAD 4 talkback	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	A gray display can be obtained with only one isolation valve open.
	Gray display	Indicates two isolation valves controlled by QUAD 4 switch directly below talkback are open.			Both isolation valves must be closed to obtain a barber-pole display.
	Barber-pole display	Indicates isolation valves are closed.			If TCA fails, SYSTEM A THRUSTER PAIR QUAD 4 switch is set to CLOSE. This shuts down failed pair of TCA's; barber-pole is displayed, and RCS TCA warning light goes out.
	Red display	Indicates TCA fired without an on command. Indicates TCA did not fire upon command to TCA primary coil.			

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 2 (cont) 8S20	CRSFD switch	Selects inputs to two latching, solenoid-operated fuel and oxidizer crossfeed valves in crossfeed piping arrangement between systems A and B.	RCS SYS B: CRSFD (panel 16)	LMP's 28-vdc bus	Switch is spring loaded to center position. If system A or B malfunctions, the related MAIN SOV switch is set to CLOSE, and CRSFD switch is set to OPEN. This opens crossfeed valves and permits fuel and oxidizer to flow from operative system to thrust chambers of both systems. CRSFD talkback displays status of crossfeed valves.
	OPEN	Energizes two solenoids that open crossfeed valves.			Gray talkback is displayed when crossfeed valves are open.
	Center	Removes solenoid excitation.			Crossfeed valves remain in last commanded position.
	CLOSE	Energizes two solenoids that close crossfeed valves.			Barber-pole talkback is displayed when crossfeed valves are closed.
8FL17	CRSFD talkback		RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	
	Gray display	Indicates fuel and oxidizer crossfeed valves controlled by CRSFD switch are open.			Both crossfeed valves must be open to obtain gray display.
	Barber-pole display	Indicates fuel and oxidizer crossfeed valves are closed.			Barber-pole display can be obtained with only one crossfeed valve closed.
8S19	TEMP/PRESS MON selector switch	Selects inputs to propellant temperature- and pressure-indicating devices for systems A and B.	RCS SYS B: TEMP/ PRESS DISP-FLAGS (panel 16)	LMP's 28-vdc bus	Switch permits monitoring systems A and B propellant temperature and helium and propellant pressures.
	He	Connects outputs of systems A and B helium tank pressure transducers to PRESS indicator.			

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 6 1S4	Engine stop pushbutton	Provides discrete stop signals to descent and ascent engines, independent of setting of ENG ARM switch	STAB/CONT: ENG ARM (panel 16)	LMP's 28-vdc bus	Pushbutton is pressed on lunar landing, after LUNAR CONTACT lights go on. When pressed, pushbutton remains pressed and illuminated red. It is reset by pressing it again.
12S17	AGS STATUS switch OFF STAND BY OPERATE	Enables astronauts to place AGS in any one of three states. No power is supplied to AGS. Supplies power to ASA heaters. Applies full power to entire ASA and then places AEA in warmup mode. Applies full power to ASA and AEA.	STAB/CONT: AEA (panel 16) STAB/CONT: ASA (panel 16)	LMP's 28-vdc bus	
12DS1	OPR ERR condition indicator (white)	Goes on if astronaut presses incorrect number of pushbuttons for operation being performed, if + or - keys are not pressed first in entering data, and if "8" or "9" is used in entering address (which is limited to octal digits).	N/A	N/A	If too few pushbuttons are pressed, indicator goes on after ENTR or READOUT pushbutton is pressed. If too many pushbuttons are pressed, indicator goes on before ENTR or READOUT pushbutton is pressed. Indicator goes off when CLR pushbutton is pressed.
12M1	Address indicator	Displays digits that identify AEA computer address selected by astronaut via keyboard.	N/A	N/A	Three octal digits must be set on this indicator, using keyboard, before each DEDA operation.
12M2	Data indicator	Displays data to be entered into AEA computer address displayed on address indicator. Displays data selected for readout from AEA computer location identified on address indicator, if READOUT pushbutton is pressed.	N/A	N/A	Five-digit numeric indicator and + and - indicator. Data to be entered are selected by astronaut via keyboard. Updating rate of any changing data selected for readout is limited to two times per second.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
Panel 6 (cont)					
12S1 thru 12S12	Keyboard pushbuttons	Enable astronauts to load shift register with necessary values (0 through 9) and signs (+ and -) to be transferred to AEA.	STAB/CONT: AEA (panel 16)	LMP's 28-vdc bus	Pressing pushbuttons successively fills address and data indicators in order, from left to right.
12S13	CLR pushbutton (momentary)	Erases all numerals displayed on address and data indicators and cancels effect of HOLD switch.	N/A	N/A	When pressed, switch extinguishes OPR ERR indicator lamp.
12S15	READOUT pushbutton (momentary)	Enables astronauts to call up AEA computer data from address set up on address indicator, for display on data indicator.	N/A	N/A	
12S14	ENTR pushbutton (momentary)	Enables astronauts to enter data (set up on data indicator) into AEA computer at address set up on address indicator.	N/A	N/A	Address and data indicators are cleared for immediate reuse when AEA computer has accepted data.
12S16	HOLD pushbutton (momentary)	Enables astronauts to interrupt updating of any continuously changing data displayed on data indicator.	N/A	N/A	Information displayed on data indicator remains unchanged until pushbutton is pressed again, enabling display of present value of data and continuance of updating.

Location/ Ref Des	Name and Position	Function	Circuit Breaker	Power Source	Remarks
ORDEAL (cont) 9S27	MODE switch	Inserts and maintains proper angular offset into FDAI's.	N/A	N/A	
	OPR/SLOW	When used with UP and DOWN positions of the SLEW switch, permits slewing the FDAI at 10X orbital rate in plus or minus direction. Used for fine adjustment.			
	HOLD/FAST	When used with UP and DOWN positions of SLEW switch, permits slewing FDAI at 200X orbital rate in plus or minus direction. Used for coarse adjustment.			
9S28	SLEW switch	Slews pitch indication on FDAI in plus (UP position) or minus (DOWN position) at rate dependent on setting of MODE switch.			

MISC

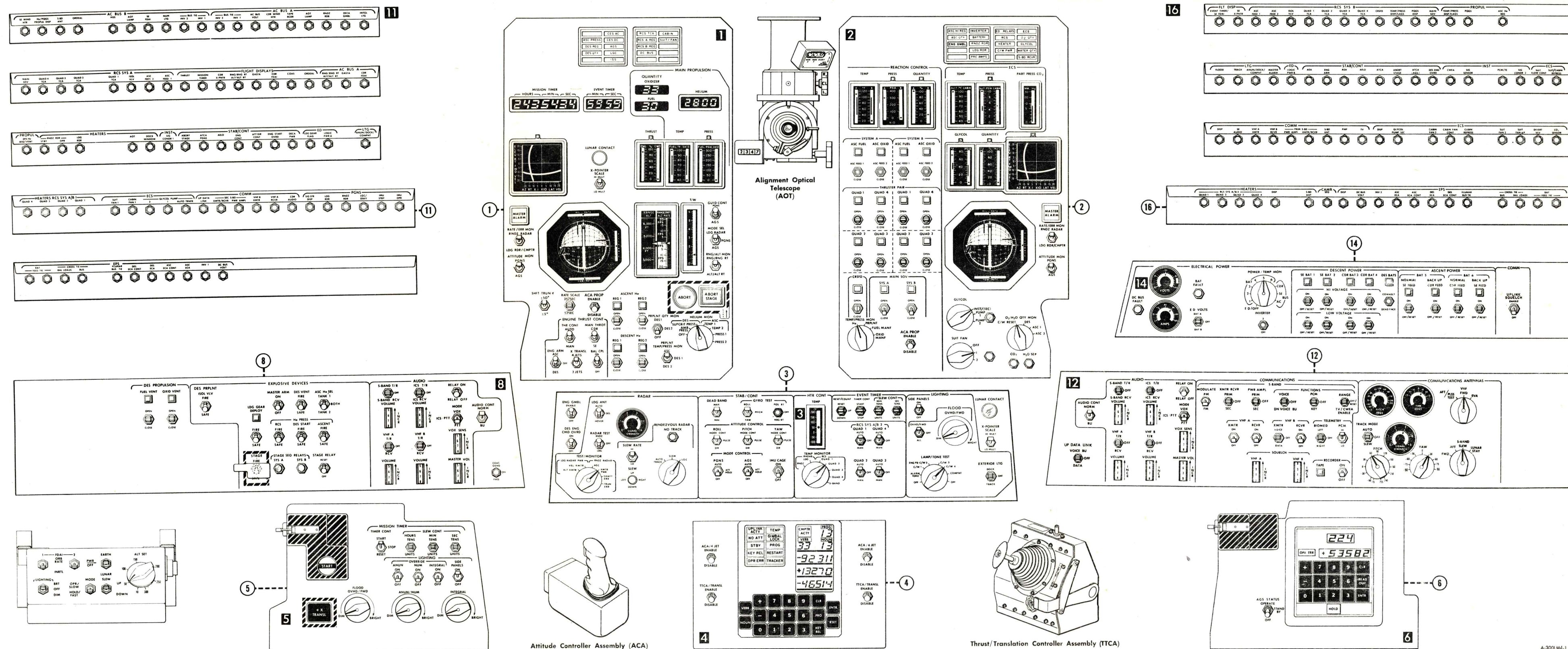


Figure 3-1. Main Display Console

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ABBREVIATION LIST

A		C (cont)	
A/D	Analog-to-digital	CPS	Cold plate section
AAP	Abort autopilot	CRSFD	Crossfeed
abs	Absolute	CS	Communications Subsystem
ac	Alternating current (a-c, u. m.)	CSI	Coelliptic sequence initiation maneuver
ACA	Attitude controller assembly	CSM	Command and Service Modules
AEA	Abort electronics assembly	CSS	Computer subsection
AEAA	Ascent engine arming assembly	CT	Control transformer
AELD	Ascent engine latching device	CW	Continuous-wave
AFC	Automatic frequency control	C/W	Caution and warning
AGC	Automatic gain control	CWEA	Caution and warning electronics assembly
AGS	Abort guidance section	CX	Control transmitter
ALT	Altitude		
amp	Ampere(s)		
AMPL	Amplifier		
ANT	Antenna		
ANUN	Annunciator		
AOT	Alignment optical telescope		
APS	Ascent propulsion section		
ARS	Atmosphere revitalization section		
ASA	Abort sensor assembly		
ASC	Ascent		
ASD	Apollo standard detonator		
ASI	Apollo standard initiator		
ASSY	Assembly		
ATA	Abort timing assembly		
ATCA	Attitude and translation control assembly		
ATM	Altimeter transmitter multiplier		
ATT	Attitude		
AUTO	Automatic		
AUX	Auxiliary		
B		D	
BAL	Balance	D/A	Digital-to-analog
BAT	Battery	DAP	Digital autopilot
BD	Band	db	Decibel
biomed	Biomedical	dc	Direct current (d-c, u. m.)
BIPRPLNT	Bipropellant	DCA	Digital command assembly
BPF	Band-pass filter	DECA	Descent engine control assembly
BU	Backup	DECR	Decrease
C		DEDA	Data entry and display assembly
cb	Circuit breaker	DEG	Degree(s)
CBW	Constant bandwidth	DEMOMD	Demodulator
CBX	C-band transponder	DES	Descent
CDH	Constant delta altitude maneuver	DET	Detector
CDR	Commander	DFI	Developmental Flight Instrumentation
CDU	Coupling data unit	DFR	Deadface relay
CES	Control electronics section	DID	Display inertial data (discrete)
cg	Center of gravity	DIF	Differential
CKT	Circuit	DISP	Display
CLR	Clear	DIST	Distribution
CMPTR	Computer	DN	Down
COAS	Crewman optical alignment sight	DPS	Descent propulsion section
COMM	Communications	DRB	Deadface relay box
COMP	Comparator	DSEA	Data storage electronics assembly
CONDR	Conditioner	DSKY	Display and keyboard
CONT	Control		
cont	Continued		
cos	Cosine		
CO ₂	Carbon dioxide		
CPL	Couple		
cps	Cycle(s) per second		
		E	
		ECA	Electronic control assembly
		ECI	Electrical circuit interrupter
		ECS	Environmental Control Subsystem
		ED	Explosive device
		EDC	End detonator cartridge
		EDS	Explosive Devices Subsystem
		EKG	Electrocardiograph
		EL	Electroluminescent
		EMI	Electromagnetic interference
		EMP	Emphasis
		EMU	Extravehicular mobility unit
		ENG	Engine
		ENTR	Enter
		EOS	Emergency oxygen system
		EPS	Electrical Power Subsystem
		ERA	Electronic replaceable assembly
		ERR	Error
		EVA	Extravehicular astronaut
		EVVA	Extravehicular visor assembly
		E _x	X-component of attitude error
		E _y	Y-component of attitude error
		E _z	Z-component of attitude error

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ABBREVIATION LIST (cont)

F		K (cont)	
F	Fahrenheit	kpps	Kilopulse(s) per second
FDAI	Flight director attitude indicator		
FDBK	Feedback		
FF	Flip-flop		L
FITH	Fire in the hole	LAT	Lateral
FM	Frequency modulation	LCA	Lighting control assembly
FM/CW	Frequency-modulated continuous-wave	LCG	Liquid-cooled garment
FOV	Field of View	LDG	Landing
fps	Foot (feet) per second	LDR	LUT deadface relay
ft	Foot (feet)	LGC	LM guidance computer
FWD	Forward	LH	Left-hand
		LiOH	Lithium hydroxide
G		LLC	Low-level commutators
g	Gravity	LLS	Low-level sensor
GASTA	Gimbal angle sequencing transformation assembly	LM	Lunar Module
gc	Gigacycle(s)	LMP	LM mission programmer; LM Pilot
GDA	Gimbal drive actuator	LOR	Lunar orbital rendezvous
GET	Ground elapsed time	LOS	Line of sight
GETI	Ground elapsed time of ignition	LPD	Landing point designator
GMBL	Gimbal	LPF	Low-pass filter
GN&CS	Guidance, Navigation, and Control Subsystem	LR	Landing radar
GOX	Gaseous oxygen	LSD	Least significant digit
GSE	Ground support equipment	LTG	Lighting
GUID	Guidance	LUT	Launch umbilical tower
		LV	Low voltage
H			M
H/X	Heat exchanger	MAN	Manual
h	Altitude	MANF	Manifold
h	Altitude rate	max	Maximum
He	Helium	mc	Megacycle(s)
HEA	High-efficiency antireflection	MDF	Mild detonating fuse
Hg	Mercury	MFC	Main feed contactor
HPF	High-pass filter	MGA	Middle gimbal axis
HTR	Heater	min	Minimum
HTS	Heat transport section	MISC	Miscellaneous
HV	High voltage	mm	Millimeter(s)
H ₂ O	Water	MOD	Modulation
		MON	Monitor
I		MPS	Main Propulsion Subsystem
IAM	Incidental amplitude modulation	MPX	Multiplexer
ICS	Intercommunication system	MSD	Most significant digit
ID	Identification	msec	Millisecond(s)
IF	Intermediate frequency	MSFN	Manned Space Flight Network
IGA	Inner gimbal axis	mv	Millivolt(s)
IMU	Inertial measurement unit	mw	Milliwatt(s)
INCR	Increase		N
INV	Inverter	N/A	Not applicable
IOPS	Interim oxygen purge system	NB	Navigation base
IRIG	Inertial reference integrating gyro	NC	Normally closed
IS	Instrumentation Subsystem	nm	Nautical mile(s)
ISOL	Isolation	NO	Normally open
ISS	Inertial subsection	No.	Number
ITMG	Integrated thermal micrometeoroid garment	NORM	Normal
		NRZ	Nonreturn-to-zero
K		N ₂ H ₄	Hydrazine
kc	Kilocycle(s)	N ₂ O ₄	Nitrogen tetroxide
kmc	Kilomegacycle(s)		O
		O/C	Overcurrent

ABBREVIATION LIST (cont)

Q

QI	Quantity indicator
QTY	Quantity
quad	Quadrant

R

R	Rankine
R/C	Reverse current
RC	Resistance-capacitance
RCCA	Rough combustion cutoff assembly
RCR	Reverse-current relay
RCS	Reaction Control Subsystem
RCVR	Receiver
RD	Relay driver
RDR	Radar
REF	Reference
REG	Regulator
RES	Resolver
RF	Radio frequency
RGA	Rate gyro assembly
RH	Right-hand
RJB	Relay junction box
rms	Root mean square
RNDZ	Rendezvous
RNG	Range
RR	Rendezvous radar
RRE	Rendezvous radar electronics
RT	Resistance thermometer
RZ	Return-to-zero

S

S&C	Stabilization and control
S/S	Subsystem
SBASI	Single bridgewire Apollo standard initiator
SBPA	S-band power amplifier
SBX	S-band transponder
SC	Signal conditioner
SCEA	Signal-conditioning electronics assembly
SCERA	Signal conditioner electronic replaceable assembly
SCO	Signal-controlled oscillator
SE	Systems Engineer (superseded by LM Pilot)
sec	Second(s); secondary
SEL	Select
SENS	Sensitivity
SEP	Separator
SEQ	Sequence
SERVO-AMPL	Servoamplifier
She	Supercritical helium
SG	Signal generator
SHFT	Shaft
SIG	Signal
Σ	Sum (summing)
sin	Sine
SLA	Spacecraft launch-vehicle adapter
SOL	Solenoid
SMRD	Spin motor rotation detector
SOV	Shutoff valve
SP	Static pressure
SPA	Signal-processing assembly
SPDT	Single-pole double-throw
SS	Speed sensor

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ABBREVIATION LIST (cont)

S (cont)		V (cont)	
ST	Strain/temperature signal conditioner	VCO	Voltage-controlled oscillator
STBY	Standby	vdc	Volts, direct-current
SYS	System	VEL	Velocity
T		VG	Magnitude of velocity to be gained
TAI	Absolute time	VGPS	Vehicle ground power supply
Tig	Time of ignition	VHF	Very high frequency
TN	Trim negative	VLV	Valve
TP	Trim positive	VOL	Volume
TC	Thermocouple	VOX	Voice-operated relay
TCA	Thrust chamber assembly	VPI	Valve position indicator
TE	Timing electronics equipment	vrms	Volts root mean square
TEMP	Temperature	V _x	Altitude rate
TFF	Time of free fall to 300,000 ft	V _y	Lateral velocity
TFI	Time from Tig	V _z	Forward velocity
THR	Thrust	W	
TLE	Tracking light electronics	W/B	Water boiler
TM	Telemetry	WC	Weighted current
TPF	Transfer phase final	WCG	Weighted-current gate
TPI	Transfer phase initiation	WMS	Water management section
TPM	Transfer phase midcourse	X	
T/R	Transmitter-receiver	XMTR	Transmitter
TRANSL	Translation	XPNDR	Transponder
TRUN	Trunnion	XTAL	Crystal
TS	Temperature sensor	∠	Angle
TT	Temperature transducer	Δh	Altitude differential
TTCA	Thrust/translation controller assembly	ΔP	Pressure differential
TTI	Time to initiate	Δro	Differential altitude in coelliptic orbit
TTIg	Time to ignition	ΔV	Velocity change (differential)
TV	Television	ΔV _m	Measured ΔV magnitude
TX	Telemetry transmitter	ΔR	Magnitude of difference between position state vectors before and after incorporation of mark data
U		Δθg	Change in gimbal angle
UDMH	Unsymmetrical dimethylhydrazine	Δθc	Gimbal angle change command
UHF	Ultrahigh Frequency	1X	One-speed resolver
usec	Microsecond(s)	16X	Sixteen-speed resolver
V			
V _{cx}	X-component of CSM velocity		
V _{cy}	Y-component of CSM velocity		
V _{cz}	Z-component of CSM velocity		
vac	Volts, alternating-current		

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX

1-KC Ramp Signal	2.2-162
1-KC Utility Pulse	2.2-162A
400-PPS Signals	2.2-162A
64-KPPS, 1-Microsecond Signal	2.2-162
8-KPPS Synchronizing Signal	2.2-162A
800-cps Synchronization Loop	2.2-187
Abort Electronics Assembly	
Functional Description	2.2-57
Major Component/Functional Description	2.2-162B
Abort Guidance Section	
Descent or Ascent Engine and Thrust Chamber Assembly Burn	2.2-80
Functional Description	2.2-54
Guidance Routines	2.2-80
Introduction	2.2-2
Initialization Function	2.2-76
Major Component/Functional Description	2.2-159
Modes of Operation	2.2-78
Self-Test Routine	2.2-76
Thrust Termination	2.2-80
Abort Sensor Assembly	
Functional Description	2.2-54
Major Component/Functional Description	2.2-159
Accelerometers	2.2-159
Accelerometer Loops	2.2-19
Accumulator	2.2-59
A-C Section	2.5-4
AC-to-DC Converters	2.9-30
Adder	2.2-162E
Address Register	2.2-58
Aft Equipment Bay	1-14
AGS Power Distribution	2.2-124
AGS Warning Light	2.9-56
Alarm-Detection Network	2.2-156
Alignment Mode	2.2-80B
Alignment Optical Telescope	
Axes	2.2-10
Functional Description	2.2-24
Lights	2.10-10
Major Component/Functional Description	2.2-140
Altimeter Transmitter	2.2-171
Altitude-Altitude Rate Register	2.2-162E
Analog Multiplexer Gate Drivers	2.9-33
Analog Signal Isolating Buffers	2.9-30
Analog-to-Digital Conversion	2.2-134
Analog-to-Digital Converter	2.9-33
Antenna Assembly (LR)	2.2-171
Antenna Assembly (RR)	2.2-164
Antenna Assembly (RRT)	2.2-168
Antenna Control Amplifier Subassembly	2.2-166
Antenna Equipment	2.7-30
Antenna Heaters and Temperature Control (LR)	2.2-172
Antenna Orientation No. I	2.2-90
Antenna Orientation No. II	2.2-90
Antenna Orientation, RR	2.2-90
Antenna Pedestal Tilt Mechanism	2.2-173
Antenna Positions	2.2-104

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Antenna Supports	1-15
Antennas	2.7-39
Armrests	2.11-19
ASC HI REG Caution Light	2.9-54
ASC PRESS Warning Light	2.9-54
ASC QTY Caution Light	2.9-54
Ascent Engine Control	2.2-123
Ascent Propellant Tank Pressurization	2.8-10
Ascent Propulsion Section	
Failure Modes	2.3-48
Functional Description	2.3-34
Interfaces	2.3-34
Major Component/Functional Description	2.3-42
Operational Limitations and Restrictions	2.3-51
Performance and Design Data	2.3-48
Telemetry Measurements	2.3-51
Ascent Oxygen Tanks	2.6-30
Ascent Stage Batteries	2.5-23
Ascent Stage Electrical Control Assemblies	2.5-27
Ascent Stage General Description	1-1
Ascent Stage Power	2.5-23
Ascent Stage Structure	1-1
Ascent Stage Water Tanks	2.6-35
Ascent Water Valve	2.6-35
Atmosphere Revitalization Section	
Functional Description	2.6-10
Introduction	2.6-1
Major Component/Functional Description	2.6-23
Attenuators	2.9-30
Attitude and Translation Control Assembly	
Functional Description	2.2-107
Major Component/Functional Description	2.2-178
Attitude Control, CES	2.2-112
Attitude Control State	2.2-80A
Attitude Controller Assemblies	
Functional Description	2.2-107
Major Component/Functional Description	2.2-175
Attitude Error Display Mode	2.2-23
Attitude Error Signal Conditioner-Limiter (ATCA)	2.2-178
Attitude Hold	2.2-78
Attitude Hold Mode	2.2-112
Audio Centers	2.7-29
Automatic Mode	2.2-112
Automatic Tracking Mode, RR	2.2-86
Battery Voltage- and Current-Monitoring Circuits	2.5-15
BATTERY Caution Light	2.9-53
Battery Status-Monitoring Circuits	2.5-4
Beacon Mode	2.2-94
Bioinstrumentation	2.11-19
Bit, Digit, and Cycle Counters	2.2-163
Body-Axis Alignment Submode	2.2-80
Burst Diaphragm Assembly	2.6-30
Burst Disk Assemblies	2.3-20
Bypass Oxygen Pressure Relief Valve	2.6-27

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Cabin Air Recirculation Assembly	
Functional Description	2.6-14
Major Component/Functional Description	2.6-26
Cabin Fans	2.6-26
Cabin Gas Return Check Valve	2.6-24
Cabin Heat Exchanger	2.6-26
Cabin Pressure Switch	2.6-34
Cabin Relief and Dump Valves	2.6-33
Cabin Repressurization and Emergency Oxygen Valve	2.6-32
Cabin Temperature Control Valve	2.6-38
CABIN Warning Light	2.9-48
Calibration (AGS)	2.2-80
Calibrator	2.9-36
Carbon Dioxide Partial Pressure Sensor	2.6-25
Carrier Lock Loop Subassembly (RRT)	2.2-170
Cartridges	2.8-17
Caution and Warning Electronics Assembly	2.9-44
Central Computer	
Functional Description	2.2-58
Major Component/Functional Description	2.2-162F
Central Processor	2.2-148
CES AC Warning Light	2.9-54
CES DC Warning Light	2.9-55
CES Power Distribution	2.2-129
Check Valves	2.6-35
Circuit Breaker and EPS Control Panels	2.5-29
Circuit Interrupters (Deadface Connectors)	2.8-18
Circuitry (ED)	2.8-5
Clock Countdown Subassembly	2.2-162I
CO ₂ Canister Selector Valve and CO ₂ and Odor Removal Canisters	2.6-24
Coarse-Alignment Mode	2.2-20
Coelliptic Flight Plan Rendezvous Guidance	2.2-80
Coelliptic Sequence Initiation Routine	2.2-80
Cold Plates and Rails	2.6-39
Combustion Chamber and Nozzle (RCS)	2.4-19
Combustion Chamber and Nozzle Extension (DPS)	2.3-25
Combustion Chamber Assembly (APS)	2.3-47
Communications Subsystem	
Failure Modes	2.7-38A
Functional Description	2.7-2
Interfaces	2.7-1
Introduction	2.7-1
Major Component/Functional Description	2.7-7
Operational Limitations and Restrictions	2.7-38A
Performance and Design Data	2.7-38A
Telemetry Measurements	2.7-44
Complete Control State	2.2-80A
Computer Control and Reticle Dimmer Assembly	
Functional Description	2.2-24
Major Component/Functional Description	2.2-142
Computer Subsection	2.2-25
Communications Carrier	2.11-6
Constant Delta Altitude	2.2-80
Contaminant Control Assembly	2.11-11
Control Electronics Section	
Functional Description	2.2-104C
Introduction	2.2-5

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Control Electronics Section (cont)	
Major Component/Functional Description	2.2-175
Modes of Operation	2.2-112
Control and Display Panels	1-6
Controls (PGA)	2.11-6
Controls (Space Suit Communicator)	2.11-15
Coolant Accumulator	2.6-37
Coolant Pumps	2.6-37
Coolant Regenerative Heat Exchanger	2.6-38
Correcting Erroneous Data, DSKY	2.2-28
Coupling Data Unit	
Functional Description	2.2-14
Major Component Functional Description	2.2-134
Crew Compartment	1-6
Crew Life Support	2.11-17
Crew Miscellaneous Equipment	2.11-23
Crew Personal Equipment	2.11-1
Crew Support and Restraint Equipment	2.11-19
Crew Transfer Tunnel	1-19
Crew Water System	2.11-17
Crewman Optical Alignment Sight	2.11-21
Cryogenic Helium Storage Vessel	2.3-16
CSM Forward Hatch	2.11-23
Current Limiters	2.11-13
Cycle Counter	2.2-59
D-C Amplifiers	2.9-30
D-C Section	2.5-3
Data AEA Transfer	2.2-163
Data Bus	2.2-162E
Data Entry and Display Assembly	
Functional Description	2.2-61
Major Component/Functional Description	2.2-162I
Data Insertion, DEDA	2.2-62
Data Loading, DSKY	2.2-28
Data Mode	2.7-30
Data Operations	2.7-6
Data Readout, DEDA	2.2-62
Data Storage Electronics Assembly	2.9-75
Data Transfer Sequencer	2.2-163
DC BUS Warning Light	2.9-53
Deadband Circuits (ATCA)	2.2-178
Deadface Relay Box	2.5-27
Decimal Point Location, DEDA	2.2-63
Decision Network	2.9-44
DEDA Operation	2.2-61
DEDA Register	2.2-162I
Demodulator-Filter Network (ATCA)	2.2-178
DES QTY Warning Light	2.9-54
Descent Engine Control	2.2-113
Functional Description	2.3-15
Descent Engine Control Assembly	
Functional Description	2.2-112
Major Component/Functional Description	2.2-180
Descent Oxygen Tank	2.6-27
Descent Propulsion Section	
Failure Modes	2.3-26

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Descent Propulsion Section (cont)	
Functional Description	2.3-2
Interfaces	2.3-1
Major Component/Functional Description	2.3-16
Operational Limitations and Restrictions	2.3-30
Performance and Design Data	2.3-28
Telemetry Measurements	2.3-31
Descent Propellant Tank Prepressurization	2.8-8
Descent Propellant Tank Venting	2.8-14
Descent Stage Batteries	2.5-23
Descent Stage Electrical Control Assemblies	2.5-25
Descent Stage General Description	1-15
Descent Stage Power (High-Voltage Taps)	2.5-20A
Descent Stage Power (Low-Voltage Taps)	2.5-20
Descent Stage Structure	1-15
Descent Stage Water Tank	2.6-34
Descent Water Valve	2.6-35
Digital-to-Analog Conversion	2.2-137
Digital Multiplexer	2.9-40
Digital Uplink Assembly	2.7-30
Direct Mode	2.2-113
Discrete Signal Isolating Buffers	2.9-31
Display (DEDA)	2.2-162I
Display (DSKY)	2.2-30
Display and Keyboard	
Functional Description	2.2-25
Major Component/Functional Description	2.2-157
Display Inertial Data Mode	2.2-23
Displays and Indicators	2.2-157
Docking Aids	1-20
Docking Aids and Tunnel Hardware	2.11-21
Docking Drogue	1-20
Docking Hatches	1-19
Docking Lights	2.10-2
Docking Probe	1-20
Docking Target (Lighting)	2.10-2
Docking Target	2.11-22
Docking Tunnel	1-13
Downlink Telemetry Register	2.2-162E
Drive-Out and Downlock Mechanisms	1-18
Drogue Assembly	2.11-21
DSKY Operation Under LGC Control	2.2-30
Early Blanking Gate	2.2-162
Earth Orbit - CSM/LM Power	2.5-20
ECS Caution Light	2.9-50
ED Batteries	2.8-19
ED Bus-Arming	2.8-6
ED RELAYS Caution Light	2.9-56
Electrical Power Subsystem	
Failure Modes	2.5-30
Functional Description	2.5-3
Interfaces	2.5-1
Introduction	2.5-1
Major Component/Functional Description	2.5-23
Operational Limitations and Restrictions	2.5-35
Performance and Design Data	2.5-34

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Electrical Power Subsystem (cont)	
Telemetry Measurements	2.5-35
Electrical Power Subsystem (PLSS)	2.11-12
Electrical Umbilical	2.11-17
Electronics Assembly (LR)	2.2-173
Electronics Assembly (RR)	2.2-164
Electronics Assembly (RRT)	2.2-168
Emergency Medical Equipment.	2.11-19
EMU Electrical Connectors	2.11-13
EMU Maintenance Kit	2.11-17
EMU Waste Management System	2.11-18
ENG GMBL Caution Light	2.9-52
Engine Assembly (APS)	2.3-41
Engine Assembly (DPS)	2.3-13
Environmental Control Subsystem	
Failure Modes	2.6-39
Functional Description	2.6-10
Interfaces	2.6-9
Introduction	2.6-1
Major Component/Functional Description	2.6-23
Operational Limitations and Restrictions	2.6-48
Performance and Design Data	2.6-42
Telemetry Measurements	2.6-48
EPS Stages of Operation	2.5-20
Explosive Devices	2.8-15
Explosive Devices Relay Boxes	2.8-19
Explosive Devices Subsystem	
Failure Modes	2.8-19
Functional Description	2.8-3
Interfaces	2.8-2A
Introduction	2.8-1
Major Component/Functional Description	2.8-15
Operational Limitations and Restrictions	2.8-22
Performance and Design Data	2.8-21
Telemetry Measurements	2.8-22
Explosive Valves (APS)	2.3-42
Explosive Valves (DPS)	2.3-16
Exterior Lighting	2.10-1
External ΔV Routine	2.2-80
External Power and Signal Interface	2.5-1
Extravehicular Boots	2.11-6
Extravehicular Gloves	2.11-6
Extravehicular Mobility Unit	2.11-1
Extravehicular Visor Assembly	2.11-7
Failure Detection	2.2-138
Failure Modes	
Ascent Propulsion Section	2.3-48
Communications Subsystem	2.7-39
Descent Propulsion Section	2.3-26
Electrical Power Subsystem	2.5-30
Environmental Control Subsystem	2.6-39
Explosive Devices Subsystem	2.8-20
Guidance, Navigation, and Control Subsystem	2.2-191
Instrumentation Subsystem	2.9-75
Lighting	2.10-12
Reaction Control Subsystem	2.4-19

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Fan/Motor Assembly	2.11-11
Fast Warmup Controller	2.2-162A
Fecal Containment System	2.11-19
Feedwater Reservoir and Vent Connector	2.11-11
Fine-Alignment Mode	2.2-22A
Final Docking Latches	1-19
Fine Temperature Controller	2.2-162A
Floodlighting	2.10-9
Flow Control Valves	2.3-23
Followup State	2.2-80
Food Preparation and Consumption	2.11-17
Footpad	1-17
Forward Hatch	1-6A
Frequency Countdown Subassembly	2.2-161
Frequency Modulator	2.2-171
Frequency Synthesizer Subassembly	2.2-165
Frequency Tracker Subassembly	2.2-165
Frequency-to-DC Converters	2.9-31
Fuel and Oxidizer Shutoff Valves	2.3-24
Fuel and Oxidizer Tanks	2.4-17
Fuel and Oxidizer Valve Assemblies	2.4-19
Fuel/Helium Heat Exchanger	2.3-18
Functional Description	
Ascent Propulsion Section	2.3-34
Communications Subsystem	2.7-2
Descent Propulsion Section	2.3-2
Electrical Power Subsystem	2.5-3
Environmental Control Subsystem	2.6-10
Explosive Devices Subsystem	2.8-4
Guidance, Navigation, and Control Subsystem	2.2-7
Instrumentation Subsystem	2.9-2
Lighting	2.10-1
Reaction Control Subsystem	2.4-5
General Operation of Abort Guidance Section	2.2-80
General Operation of Primary Guidance and Navigation Section	2.2-31
General Operation, Landing Radar	2.2-100
General Operation, Rendezvous Radar	2.2-86
General Operation, RR Transponder	2.2-94
Gimbal Angle Sequencing Transformation Assembly	2.2-188
Gimbal Drive Actuators	
Functional Description	2.2-112
Major Component/Functional Description	2.2-186
Gimbal Lock Mode	2.2-24
Gimbal Ring and Gimbal Drive Actuators	2.3-25
GLYCOL Caution Light	2.9-51
GN&CS - CS Interface	2.2-7
GN&CS - ECS Interface	2.2-7
GN&CS - EDS Interface	2.2-7
GN&CS - EPS Interface	2.2-7
GN&CS - IS Interface	2.2-8
GN&CS - MPS Interfaces	2.2-6
GN&CS - RCS Interface	2.2-6
GN&CS Functional Description	2.2-7
GSE-Supplied Prelaunch Power	2.5-20
Guidance Steering	2.2-78

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Guidance, Navigation, and Control Subsystem	
Failure Modes	2.2-191
Functional Description	2.2-8
Interfaces	2.2-6
Introduction	2.2-1
Major Component/Functional Description	2.2-130
Operational Limitations and Restrictions	2.2-194
Performance and Design Data	2.2-203
Telemetry Measurements	2.2-205
Gyros	2.2-159
Gyro-Torquing Loop	2.2-17
Handholds	2.11-20
Hardover Mode	2.2-141
Heat Transport Section	
Functional Description	2.6-19
Introduction	2.6-9
Major Component/Functional Description	2.6-37
HEATER Caution Light	2.9-52
Heaters	2.4-15
Helium Isolation Solenoid Valves (APS)	2.3-44
Helium Isolation Solenoid Valves (DPS)	2.3-18
Helium Isolation Valves and Compatibility Valves	2.8-17
Helium Pressure Regulator Assemblies (APS)	2.3-44
Helium Pressure Regulator Assembly (RCS)	2.4-16
Helium Pressure Regulators (DPS)	2.3-19
Helium Pressurization Section	2.4-5
Helmet	2.11-5
High-Level Analog Multiplexer and High-Speed Gates	2.9-33
High-Pressure Oxygen Control Assembly	2.6-27
High-Pressure Oxygen Regulator	2.6-27
In Flight Accelerometer Calibration Only	2.2-80
Ignition and Shutdown	2.2-113
IMU Axes	2.2-9
IMU Case	2.2-132
IMU Cage Mode	2.2-23
IMU Failure-Detection Assembly	2.2-134
IMU Temperature Control Assembly	2.2-132
IMU Turn-On Mode	2.2-20
Index Register	2.2-58
In-Flight Data	2.11-23
In-Flight Sightings	2.2-24
Inertial Measurement Unit	
Functional Description	2.2-14
Major Component/Functional Description	2.2-130
Inertial Reference Integrating Gyros	2.2-130
Inertial Reference Integrating Gyro Axes	2.2-10
Inertial Reference Mode, AGS	2.2-78
Inertial Reference Mode, PGNS	2.2-24
Inertial Reference States	2.2-80
Inertial Subsection	
Functional Description	2.2-14
Functional Loops	2.2-16
Modes of Operation	2.2-19
Injector Assembly (APS)	2.3-46
Injector Head Assembly (RCS)	2.4-19

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Inner Gimbal Assembly	2.2-132
Input Counters	2.2-162E
Input-Output Interfaces	2.2-149
Input-Output Subassembly	
Functional Description	2.2-59
Major Component/Functional Description	2.2-162E
Instrumentation (PGA)	2.11-6
Instrumentation (PLSS)	2.11-11
Instrumentation (PLSS Electrical Power Subsystem)	2.11-15
Instrumentation (PLSS Feedwater Loop)	2.11-12
Instrumentation (PLSS Liquid Transport Loop)	2.11-12
Instrumentation (PLSS Oxygen Ventilation Loop)	2.11-11
Instrumentation Converter Subassembly	2.2-167
Instrumentation Subsystem	
Failure Modes	2.9-75
Functional Description	2.9-2
Interfaces	2.9-1
Introduction	2.9-1
Major Component/Functional Description	2.9-8
Operational Limitations and Restrictions	2.9-87
Performance and Design Data	2.9-78
Telemetry Measurements	2.9-87
Integrally Lighted Components	2.10-7
Integrated Thermal Micrometeoroid Garment	2.11-6
Integrator Registers	2.2-162F
Interior Lighting	2.10-2
Internal Power Interface	2.5-3
Interstage Structural Connections	2.8-17
Interstage Umbilical Cutters (Guillotine)	2.8-17
Intravehicular Gloves	2.11-6
Introduction	
Communications Subsystem	2.7-1
Crew Personal Equipment	2.11-1
Electrical Power Subsystem	2.5-1
Environmental Control Subsystem	2.6-1
Explosive Devices Subsystem	2.8-1
Guidance, Navigation, and Control Subsystem	2.2-1
Instrumentation Subsystem	2.9-1
Lighting	2.10-1
Main Propulsion Subsystem	2.3-1
Reaction Control Subsystem	2.4-1
Spacecraft	1-1
ISS Warning Light	2.9-55
INVERTER Caution Light	2.9-53
Inverters	2.5-27
Jet Select Logic (ATCA)	2.2-180
Keyboard	2.2-157
Ladder	1-19
Lamp and Tone Test	2.10-10
Landing Gear	1-17
Landing Gear Deployment	2.8-7
Landing Gear Uplock and Cutter Assemblies	2.8-17
Landing Radar Antenna Axes	2.2-12

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Landing Radar	
Functional Description	2.2-96
Introduction	2.2-5
Major Component/Functional Description	2.2-171
Late Blanking Gate	2.2-162
LDG RDR Caution Light	2.9-55
LGC Control Mode	2.2-86
LGC Warning Light	2.9-55
Lighting	
Failure Modes	2.10-12
Functional Description	2.10-1
Introduction	2.10-1
Operational Limitations and Restrictions	2.10-13
Performance and Design Data	2.10-12
Liquid Cooling Garment	2.11-5
Liquid Transport Loop	2.11-12
LM 4 Configuration	1-1
LM Guidance Computer	
Functional Description	2.2-25
Major Component/Functional Description	2.2-143
LM Vehicle, Guidance, Navigation and Control Subsystem Axes	2.2-9
LM Vehicle Axes	2.2-9
LM-CSM Interfaces	1-19
LM-SLA-S-IVB Connections	1-19
Lunar Alignment Submode	2.2-90
Lunar Contact Lights	2.10-7
Lunar Surface Sighting	2.2-24
Lunar-Surface Sensing Probe	1-18
Main Propulsion Subsystem	2.3-1
Major Component/Functional Description	
Ascent Propulsion Section	2.3-42
Communications Subsystem	2.7-7
Descent Propulsion Section	2.3-16
Electrical Power Subsystem	2.5-23
Environmental Control Subsystem	2.6-23
Explosive Devices Subsystem	2.8-16
Guidance, Navigation, and Control Subsystem	2.2-130
Instrumentation Subsystem	2.9-8
Reaction Control Subsystem	2.4-16
Malfunctioning-Battery Isolation Circuitry	2.5-17
Manual Operation of DSKY	2.2-27
MASTER ALARM Pushbutton/Lights and C/W PWR Caution Light	2.9-48
Medical Equipment	2.11-19
Memory (AEA)	
Functional Description	2.2-58
Major Component/Functional Description	2.2-162B
Memory (LGC)	2.2-156
Memory Register	2.2-58
Microwave Subassembly	2.2-168
Middle Gimbal Assembly	2.2-132
Midsection	1-13
Modes of Operation, Rendezvous Radar	2.2-86
Modes of Operation, RR Transponder	2.2-94
Moding	2.2-137
Multiplier-Quotient Register	2.2-59

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Navigation Base Axes	2.2-9
Navigation Base	2.2-130
Navigation Initialization	2.2-76
O ₂ QTY Caution Light	2.9-51
Operation Code Register	2.2-59
Operational Limitations and Restrictions	
Ascent Propulsion Section	2.3-51
Communications Subsystem	2.7-39
Descent Propulsion Section	2.3-30
Electrical Power Subsystem	2.5-35
Environmental Control Subsystem	2.6-48
Explosive Devices Subsystem	2.8-23
Guidance, Navigation, and Control Subsystem	2.2-203
Instrumentation Subsystem	2.9-87
Lighting	2.10-13
Reaction Control Subsystem	2.4-24
Operator Error, DEDA	2.2-63
Operator Error, DSKY	2.2-29
OPS Battery	2.11-16
OPS Instrumentation	2.11-17
OPS Oxygen Bottles	2.11-16
OPS Oxygen Connectors	2.11-17
OPS Oxygen Pressure Regulator Assembly	2.11-16
Optical Sight Reticle Light	2.10-10
Optical Subsection Functional Description	2.2-24
Optical Subsection Operation	2.2-24
Orbit Insertion Routine	2.2-80
Orbital Rate Display - Earth and Lunar	
Introduction	2.2-6
Major Component/Functional Description	2.2-188
ORDEAL Power Distribution	2.2-130
Oscillator Failure-Detection Circuits No. 1, 2, and 3	2.9-43
Outer Gimbal Assembly	2.2-132
Output Discretes	2.2-162E
Output Register and Data Transfer Buffers	2.9-42
Output Telemetry Register	2.2-162G
Overhead Hatch	1-14
Overboard Relief Valve	2.6-28
Oxygen Bacteria Filter	2.6-34
Oxygen Control Module	2.6-31
Oxygen Demand Regulators	2.6-31
Oxygen Filters	2.6-31
Oxygen Purge System	2.11-16
Oxygen Ventilation Loop (PLSS)	2.11-11
Oxygen Umbilical	2.11-17
Oxygen Supply and Cabin Pressure Control Section	
Functional Description	2.6-14
Introduction	2.6-9
Major Component/Functional Description	2.6-27
Performance and Design Data	
Ascent Propulsion Section	2.3-48
Communications Subsystem	2.7-39
Descent Propulsion Section	2.3-26
Electrical Power Subsystem	2.5-34
Environmental Control Subsystem	2.6-42

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Performance and Design Data (cont)	
Explosive Devices Subsystem	2.8-22
Guidance, Navigation, and Control Subsystem	2.2-194
Instrumentation Subsystem	2.9-78
Lighting	2.10-12
Reaction Control Subsystem	2.4-22
Personal Radiation Dosimeters	2.11-19
Personal Hygiene Items	2.11-19
PGNS Power Distribution	2.2-123
PGNS-to AGS Alignment Submode	2.2-80B
Phase-Locked Oscillator and Doublers	2.9-43
Phase-Sensitive Demodulators	2.9-32
Pilot Valves and Shutoff Valve Actuators	2.3-24
Platform	1-19
PLSS Control Box	2.11-16
PLSS Multiple Water Connector	2.11-12
PLSS and Oxygen Tank Shutoff Valves	2.6-31
PLSS Oxygen Connectors	2.11-11
Portable Life Support System	2.11-7
Portable Utility Lights	2.10-10
Power and Servo Assembly	
Functional Description	2.2-15
Major Component/Functional Description	2.2-138
Power Distribution	
Communications Subsystem	2.7-4
Guidance, Navigation, and Control Subsystem	2.2-123
Power Subassemblies (DEDA)	2.2-162I
Power Supply Subassembly (RRT)	2.2-171
Power Monitoring	2.5-4
Power Supply	2.9-42
Power Supply (ASA)	2.2-162A
Power Supply, DSEA	2.9-75
PRE AMPS Caution Light	2.9-55
Premodulation Processor	2.7-23
Pressure Garment Assembly	2.11-5
Pressurization Section (APS)	2.3-34
Pressurization Section (DPS)	2.3-2
Primary and Secondary Duplex VHF-AM Transceivers and Antenna	2.11-15
Primary Bus Voltage-Monitoring Circuits	2.5-17
Primary Coolant Loop Cold Plates and Rails	2.6-20
Primary Coolant Filter	2.6-37
Primary Coolant Loop	2.6-19
Primary Evaporator Flow Valve No. 1	2.6-36
Primary Evaporator Flow Valve No. 2	2.6-36
Primary Guidance and Navigation Section	
Functional Description	2.2-12
Introduction	2.2-2
Major Component/Functional Description	2.2-130
Modes of Operation	2.2-31
Primary Oxygen Bottle	2.11-11
Primary Oxygen Regulator Assembly	2.11-11
Primary Oxygen Subsystem (PLSS)	2.11-7
Primary Structure	2.2-159
Primary Strut	1-17
Priority Control	2.2-148
Probe Assembly	2.11-22
Program Counter	2.2-59

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETIC INDEX (cont)

Program Selection, DSKY	2.2-29
Programmer	2.9-36
Propellant Feed Section (APS)	2.3-37
Propellant Feed Section (DPS)	2.3-9
Propellant Section (RCS)	2.4-6
Propellant Quantity Gaging System (DPS)	2.3-10
Propellant Quantity Measuring Device	2.4-6
Propellant Shutoff Valve Assemblies (DPS)	2.3-24
Propellant Storage Tanks (APS)	2.3-45
Propellant Storage Tanks (DPS)	2.3-21
Pulse Integrating Pendulous Accelerometer	2.2-132
Pulse Integrating Pendulous Accelerometer Axes	2.2-11
Pulse Mode	2.2-113
Pulse Ratio Modulators (ATCA)	2.2-180
Pulse Torque Assembly	
Functional Description	2.2-15
Major Component/Functional Description	2.2-138
Pulse-Code-Modulation and Timing Electronics Assembly	2.9-32
Pump/Motor Assembly	2.11-12
Pushbutton Entry	2.2-163
Radar Power Distribution	2.2-124
Range Computer	2.2-173
Range Frequency Tracker	2.2-173
Range Tracker Subassembly	2.2-166
Ranging and Tracking	2.7-6
Rate Gyro Assembly	
Functional Description	2.2-112
Major Component/Functional Description	2.2-180
Rate Signal Summation Amplifier	2.2-178
RCS Caution Light	2.9-52
RCS Propellant Tank Pressurization	2.8-10
RCS TCA Warning Light	2.9-52
Reaction Control Subsystem	
Failure Modes	2.4-22
Functional Description	2.4-5
Interfaces	2.4-1
Introduction	2.4-1
Major Component/Functional Description	2.4-16
Operational Limitations and Restrictions	2.4-24
Performance and Design Data	2.4-22
Telemetry Measurements	2.4-25
Receiver Subassembly (RR)	2.2-163
Receiver Subassembly (RRT)	2.2-170
Relay Junction Box	2.5-27
Relay Package	2.2-157
Release of Display and Keyboard System	2.2-29
Relief Valve Assemblies (APS)	2.3-45
Relief Valve Assemblies (DPS)	2.3-20
Relief Valve Assemblies (RCS)	2.4-17
Rendezvous Radar	
Functional Description	2.2-81
Major Component/Functional Description	2.2-164
Rendezvous Radar and Transponder	2.2-2
Rendezvous Radar Antenna Axes	2.2-11
Rendezvous Radar Transponder Antenna Axes	2.2-11
Rendezvous Radar Transponder	2.2-168

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Resistance-to-DC Converters	2.9-31
Restraint Assembly	2.11-20
RF Electronic Equipment	2.7-7
RNDZ RDR Caution Light	2.9-56
RR Transponder	2.2-90
S-Band Diplexer and RF Selector Switch	2.7-18
S-Band In-Flight Antennas	2.7-37
S-Band Power Amplifier	2.7-18
S-Band Steerable Antenna	2.7-37
S-Band Steerable Antenna Heaters	2.7-37
S-Band Transmitter-Receiver Assembly	2.7-7
S-BD RCVR Caution Light	2.9-56
S&C Control Assemblies	2.2-187
Secondary Coolant Loop	2.6-20
Secondary Coolant Loop Cold Plates and Rails	2.6-23
Secondary Evaporator Flow Valve	2.6-36
Secondary Struts	1-17
Self-Test Mode	2.2-94
Self-Test Subassembly	2.2-166
Sensing Equipment	2.9-8
Sensor Power Fuse Assemblies	2.5-30
Sequence Generator	2.2-147
Shift Register	2.2-162I
Shutdown	2.2-162I
Signal Conditioner Assembly	
Functional Description	2.2-16
Major Component/Functional Description	2.2-139
Signal-Conditioning Electronic Assembly	2.9-8
Signal Data Converter Subassembly (RR)	2.2-167
Signal-Processing Assembly	2.7-23
Slew Mode	2.2-86
Solenoid Driver Preamplifiers and Jet Solenoid Drivers	2.2-180
Space Suit Communicator	2.11-15
Spin Motor Assembly	2.2-161
Stabilization Loops	2.2-17
STAGE SEQ RELAYS SYS A and SYS B Component Caution Lights	2.8-7
Staging - Normal or Abort	2.5-23
Staging	2.8-9
Startup	2.2-162I
Stowage Locations	2.11-24
Stowage Provisions	1-21
Sublimator (PLSS)	2.11-12
Sublimator (PLSS Liquid Transport Loop)	2.11-12
Sublimators	2.6-38
Subsystem Operation (CS)	2.7-7
Suit Circuit Fans	2.6-24
Suit Circuit Heat Exchanger	2.6-24
Suit Circuit Regenerative Heat Exchanger	2.6-25
Suit Circuit Relief Valve	2.6-23
Suit Circuit Sublimator	2.6-24
Suit Gas Diverter Valve	2.6-23
Suit Isolation Valves	2.6-25
Suit Temperature Control Valve	2.6-25
Suit Temperature Control Valve	2.6-38
SUIT/FAN Warning Light	2.9-51

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

TCA Cluster Supports	1-15
Telemetry Measurements	
Ascent Propulsion Section	2.3-51
Communications Subsystem	2.7-44
Descent Propulsion Section	2.3-31
Electrical Power Subsystem	2.5-35
Environmental Control Subsystem	2.6-49
Explosive Devices Subsystem	2.8-23
Guidance, Navigation, and Control Subsystem	2.2-35
Instrumentation Subsystem	2.9-87
Reaction Control Subsystem	2.4-25
Terminal Phase Initiation Routine	2.2-80
Thermal and Micrometeoroid Shield, Ascent Stage	1-15
Thermal and Micrometeoroid Shield, Descent Stage	1-17
Throttle Control	2.2-113
Throttle Valve Actuator	2.3-21
Thrust Chamber Assemblies	
Functional Description	2.4-10
Major Component/Functional Description	2.4-19
Thrust Chamber Assembly Failure-Detection System	2.4-13
Thrust/Translation Controller Assemblies	
Functional Description	2.2-107
Major Component/Functional Description	2.2-176
Thruster-Off Failures	2.4-15
Thruster-On Failures	2.4-14
Time Accumulator and Serial Time Code Generator	2.9-44
Time Initialization	2.2-78
Timer	2.2-147
Timing	2.2-59
Timing and Control Registers	2.2-162E
Timing Generator	2.9-44
Tone Generators	2.11-16
Tone Modulation Detector Subassembly	2.2-170
Tool B	2.11-23
Torso and Limb Suit	2.11-5
Tracking Light	2.10-2
Translation Control	2.2-113
Transponder Mode	2.2-94
Trim Control	2.2-123
Tunnel Hardware	2.11-22
Umbilical	1-19
Umbilical Assembly	2.11-17
Uplock Assembly	1-18
Valve Package Assembly	2.3-46
Variable-Area Injector	2.3-24
Velocity Frequency Trackers	2.2-173
Velocity Sensor Transmitters	2.2-171
VHF Diplexer and RF Selector Switch	2.7-23
VHF In-Flight Antennas	2.7-37
VHF Transmitter-Receiver Assembly	2.7-20
Voice Backup Mode	2.7-30
Voice Operation	2.7-6
Voltage-Conditioning Oscillators and Mixer	2.11-15

LMA790-3-LM 4
APOLLO OPERATIONS HANDBOOK
SUBSYSTEMS DATA

ALPHABETICAL INDEX (cont)

Waste Fluid Transfer	2.11-18
Water Accumulator and Fill Connector	2.11-12
Water Control Module	2.6-35
Water Diverter Valve	2.11-12
Water Fill and Drain Connectors	2.11-12
Water Management Section	
Functional Description	2.6-17
Introduction	2.6-9
Major Component/Functional Description	2.6-34
Water Pressure Regulators	2.6-36
Water Quantity Measuring Device	2.6-35
WATER QTY Caution Light	2.9-50
Water Separator Selector Valve	2.6-24
Water Separators	2.6-24
Water Shutoff and Relief Valve	2.11-11
Water Tank Selector Valve	2.6-35
Window Shades	2.11-24
Windows	1-11