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APOLLO EXPERIENCE REPORT - COMMAND AND SERVICE MODULE CONTROLS AND DISPLAYS SUBSYSTEM

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16. Abstract

A review of the command and service module controls and displays subsystem is presented. The subsystem is described, and operational requirements, component history, problems and solutions, and conclusions and recommendations for the subsystem are included.

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Electroluminescent light source		

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COMMAND AND SERVICE MODULE
CONTROLS AND DISPLAYS SUBSYSTEM

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SUMMARY

The command and service module controls and displays subsystem provided the interface between the crew and the spacecraft. This interface allowed the crew to manually operate the spacecraft under normal or contingency conditions, to safely shut down all equipment, to monitor subsystem conditions and energy resources, to recognize malfunctions and imminent hazardous conditions, and to adjust or select alternate subsystem operating modes. The equipment required to accomplish these functions is discussed in this report.

INTRODUCTION

The command and service module (CSM) controls and displays subsystem (CDS) was composed of various electrical controls and electrically powered displays. The control types were on/off (switch) and variable (potentiometer and rheostat). Display types included the electrical meter and the flag, or lighted, indicator that displayed the information in an EVENT, an ANALOG, or a DIGITAL format. Excluding the interior floodlighting, the exterior lights, and the caution detection unit (CDU), all controls and displays (C&D) were mounted on panels inside the spacecraft. This report discusses the equipment that comprised the subsystem and presents the major problems that occurred during the Apollo Program.

As an aid to the reader, where necessary the original units of measure have been converted to the equivalent value in the Système International d'Unités (SI). The SI units are written first, and the original units are written parenthetically thereafter.

DEVELOPMENT

When the requirements for the CSM CDS had been finalized and the desired methods of control and display had been decided, specifications were formulated for distribution to industry to obtain the hardware to fulfill the requirements. To qualify, each component had to undergo the following tests: design feasibility (breadboard model), design verification (engineering model), qualification (flight item), acceptance (flight item), and preinstallation (flight item). The qualification test levels varied, depending on the physical location of the component in the vehicle. Several items underwent additional qualification testing as late as 1968-1969 because the initial test levels were found to be inadequate.

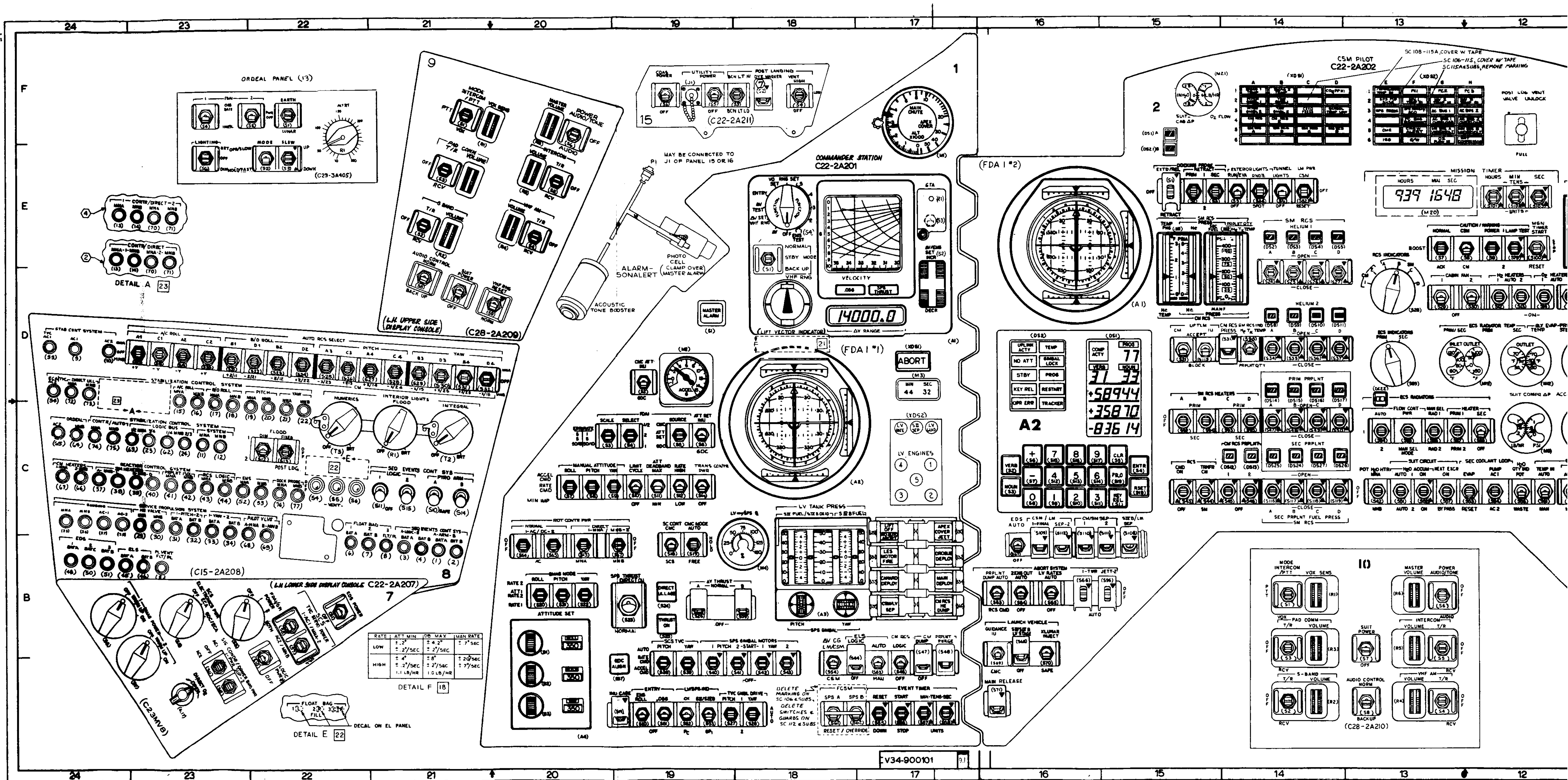
The basic C&D operational requirements for block II vehicles were very similar to those for block I vehicles. The block II vehicle was a redesigned block I vehicle with changes affecting spacecraft wiring, wiring harnesses, and internal structure. The additional C&D functions needed to accommodate the block II mission requirements mainly affected the lighting and the panels. Block I interior lighting consisted entirely of floodlights, whereas the block II spacecraft included integral lighting, backlighted panel components, and electroluminescent (EL) lighting for panel nomenclature and digital displays. Block I main display consoles (panels) consisted of several assorted small panels that were consolidated into three large display panels in the block II spacecraft (fig. 1).

COMPONENT HISTORY

The component, the quantity used per vehicle, the part number, and the certification test report number of the components used in the block II C&D subsystem are listed in table I. Most of the items are identical to their block I counterparts; e.g., the digital event timer (DET) is the same as the block I unit except for the addition of EL lighting to the indicator window. Some items, such as the mission timers, toggle switches, and EL panels, were completely new on block II spacecraft.

Lighting

The CSM lighting system (figs. 2 to 5) provided general interior lighting, panel instrumentation and nomenclature illumination, tunnel lights, exterior orientation lights, illumination for docking and extravehicular activity (EVA), and a flashing beacon for visual recognition at distances as great as 297 kilometers (160 nautical miles). The general interior lighting was provided by six fluorescent floodlights, each containing two separately controlled lamps (one primary and one secondary). The floodlight fixtures were mounted on the head position at each side of the center couch, on the bulkhead at the head position of each side couch, and on the couch struts at the foot position of the center couch to light the lower equipment bay (LEB). The control for each floodlight consisted of an on/off switch, a primary/secondary selection switch, and a rheostat for varying the lamp input voltage. The lamps operated on direct current (dc) voltage that was converted to alternating current (ac) voltage by a converter inside the lamp fixture. Because of the numerous failures and failure modes involved, the floodlight circuit design was progressively changed.


ORDEAL PANEL (13)
(C29-3A405)
DETAIL A 23
(L.H. UPPER SIDE DISPLAY CONSOLE)
(C28-2A209)
(C22-2A211)
MAY BE CONNECTED TO J1 OF PANEL 15 OR 16
ALARM-SONALERT
PHOTO CELL (CLAMP OVER MASTER ALARM)
ACOUSTIC TONE BOOSTER
COMMANDER STATION C22-2A201
VELOCITY
14000.0
ΔV RANGE
LIFT VECTOR INDICATOR
(FDA 1 #1)
(FDA 1 #2)
ABORT
MASTER ALARM
CSM PILOT C22-2A202
SC 108-115A, COVER W TAPE
SC 106-115, COVER W/ TAPE
SC 115A & SUBS, REMOVE MARKING
MISSION TIMER
939 1648
A2
77
3173
+58944
+35870
-83614
STAB CONT SYSTEM
AUTO RCS SELECT
STABILIZATION CONTROL SYSTEM
INTERIOR LIGHTS
NUMERICS
FLOOD
INTEGRAL
POST LDG
SERVICE PROPULSION SYSTEM
(C15-2A208)
(L.H. LOWER SIDE DISPLAY CONSOLE C22-2A207)
(C23MVB)
DETAIL F 18
DETAIL E 22
DECAL ON EL PANEL
LV TANK PRESS
SPS GIMBAL MOTORS
EVENT TIMER
SM RCS
HELIUM 1
HELIUM 2
PRIM PRPLNT
SEC PRPLNT
SEC PRPLNT FUEL PRESS
MAIN RELEASE
LAUNCH VEHICLE
ABORT SYSTEM
(C28-2A210)
DELETE MARKING ON SC 106 & SUBS. DELETE SWITCHES & GUARDS ON SC 112 & SUBS
V34-900101

FOLD OUT #1

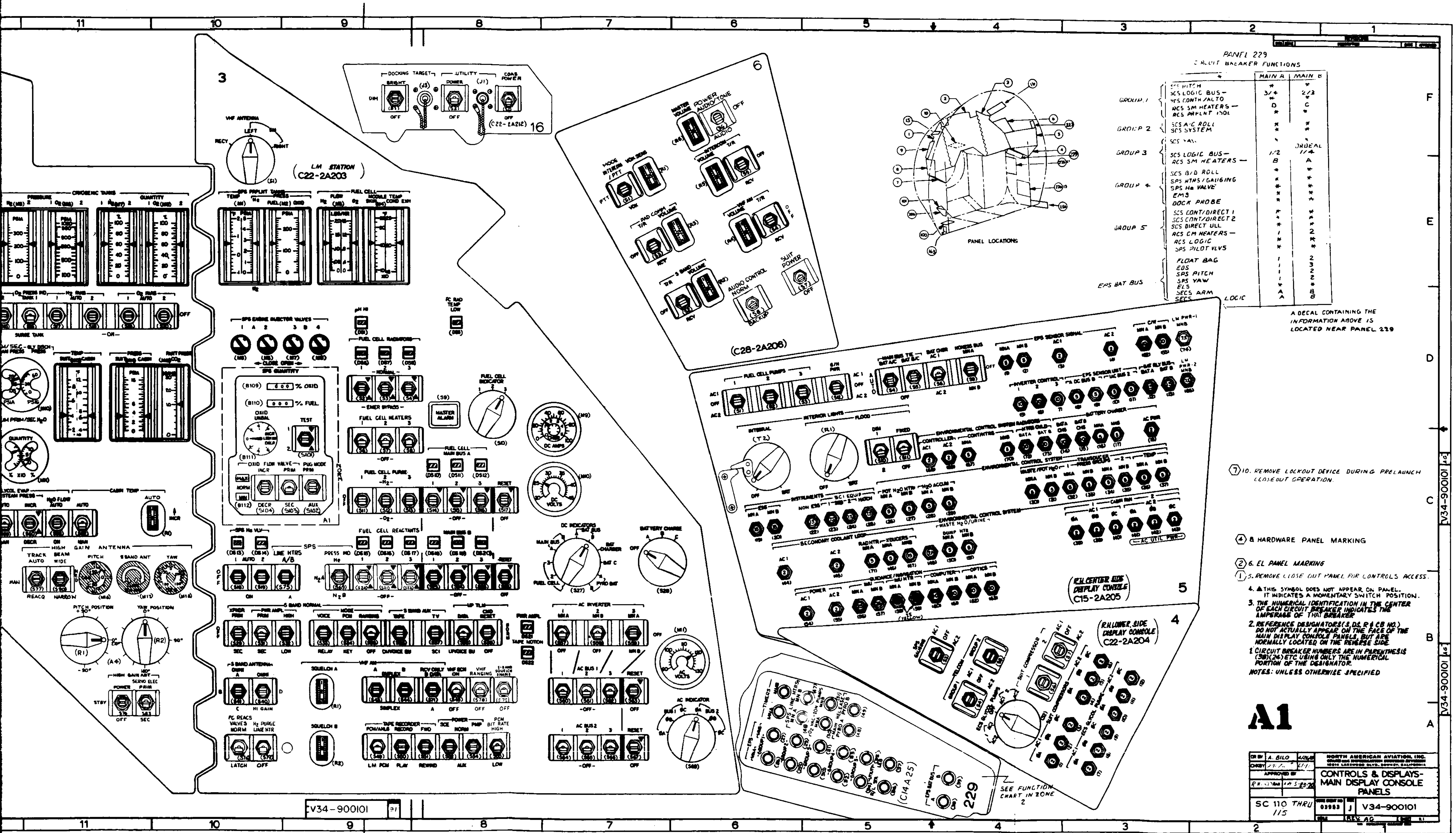
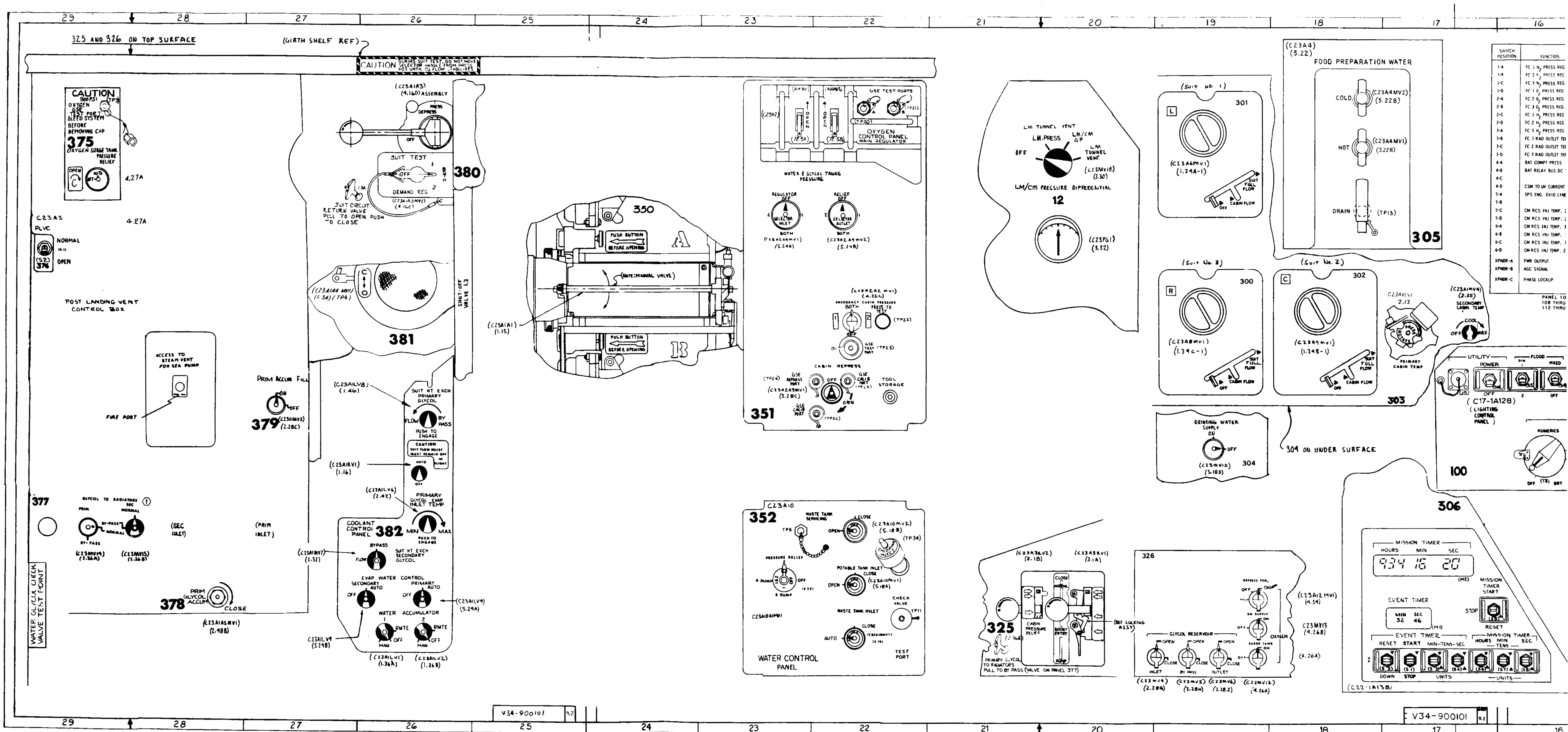


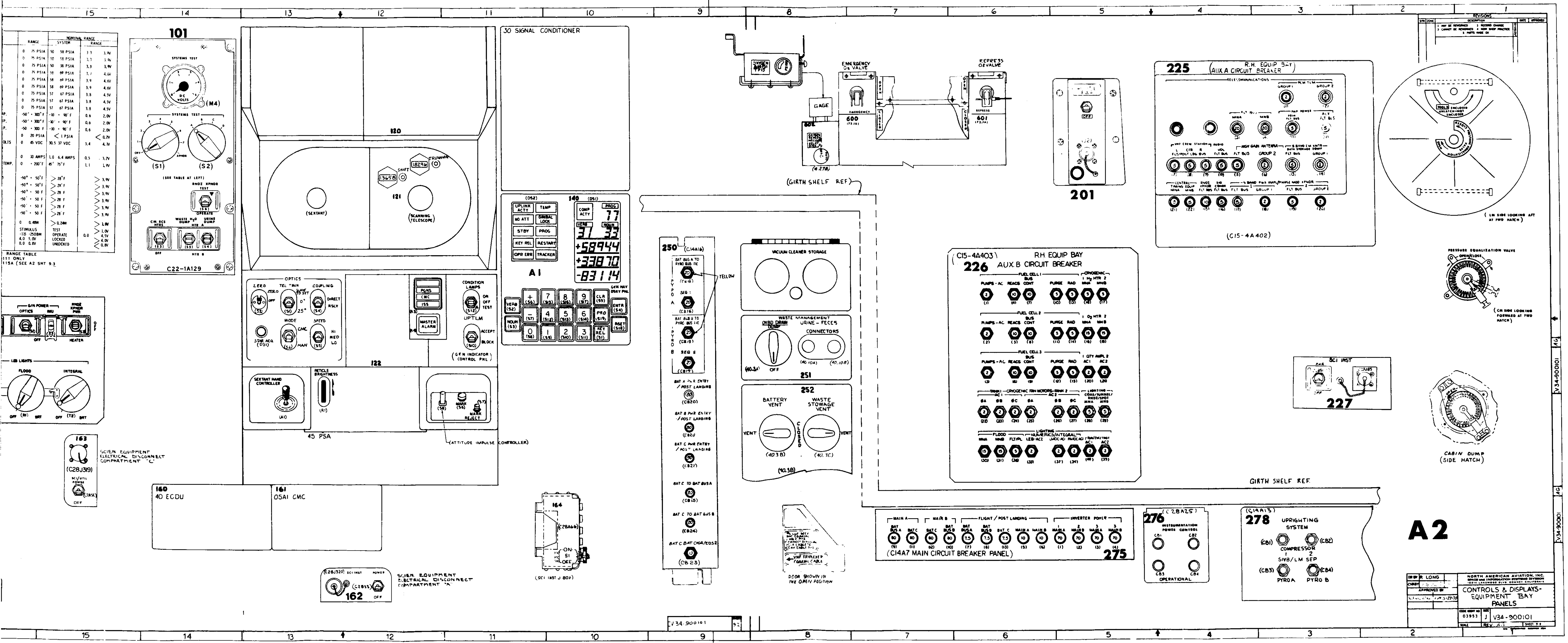
Figure 1.- The CDS panels.

FOLD-OUT #2



325 AND 326 ON TOP SURFACE
(GIRTH SHELF REF)
CAUTION DURING SUIT TEST, DO NOT MOVE SELECTOR HANDLE FROM PRESS POS UNTIL O2 FLOW STABILIZES
CAUTION 900 PSI OXYGEN GSE TEST PORT BLEED SYSTEM BEFORE REMOVING CAP
375
OXYGEN SURGE TANK PRESSURE RELIEF
4.27A
C23A3
PLVC
NORMAL
OPEN
376
POST LANDING VENT CONTROL BOX
ACCESS TO STEAM VENT FOR SEA PUMP
FIRE PORT
PRIM ACCUM FILL
379
(C23A1MV3) (2.28C)
377
GLYCOL TO RADIATORS
(SEC INLET)
(PRIM INLET)
378
PRIM GLYCOL ACCUM
CLOSE
(C23A1A5MV1) (2.48B)
WATER GLYCOL CHECK VALVE TEST POINT
(C23A1A3) (4.16D) ASSEMBLY
SUIT TEST
DEMAND REG
SUIT CIRCUIT RETURN VALVE PULL TO OPEN PUSH TO CLOSE
380
381
(C23A1A4 MV1) (1.3A) (TP4)
SHUT-OFF VALVE 1.3
(C23A1LV8) (1.46)
SUIT HT EXCH PRIMARY GLYCOL
FLOW
BY PASS
PUSH TO ENGAGE
(C23A1RV1) (1.16)
PRIMARY GLYCOL EVAP INLET TEMP
MIN
MAX
COOLANT CONTROL PANEL
382
SUIT HT EXCH SECONDARY GLYCOL
EVAP WATER CONTROL
SECONDARY
PRIMARY
AUTO
OFF
WATER ACCUMULATOR
RMTE
MAN
(C23A1LV4) (5.29A)
C23A1LV9 (5.29B)
350
PUSH BUTTON BEFORE OPENING
A
B
(C23A1A1) (1.15)
(NOTE: MANUAL VALVE)
GSE TEST PORTS
OXYGEN CONTROL PANEL MAIN REGULATOR
WATER & GLYCOL TANKS PRESSURE
REGULATOR
RELIEF
SELECTOR INLET
SELECTOR OUTLET
BOTH
(C23A2A4MV1) (5.24A)
(C23A2A4MV2) (5.24B)
EMERGENCY CABIN PRESSURE
PRESS TO TEST
CABIN REPRESS
GSE REPRESS PORT
GSE CALIB PORT
TOOL STORAGE
351
LM TUNNEL VENT
LM PRESS
LM/CM ΔP
OFF
LM TUNNEL VENT
(C23MV18) (3.30)
LM/CM PRESSURE DIFFERENTIAL
12
(C23PG1) (3.32)
(C23A4) (5.22)
FOOD PREPARATION WATER
COLD
HOT
(C23A4MV2) (5.22B)
(C23A4MV1) (5.22B)
DRAIN
(TP13)
305
(SUIT No. 1)
301
(C23A6MV1) (1.34A-1)
SUIT FULL FLOW
CABIN FLOW
OFF
(SUIT No. 3)
300
(C23A8MV1) (1.34C-1)
(SUIT No. 2)
302
(C23A7MV1) (1.34B-1)
303
PRIMARY CABIN TEMP
(C23A1MV4) (2.35)
SECONDARY CABIN TEMP
COOL
MAX
DRINKING WATER SUPPLY
ON
OFF
304
(C23MV10) (5.18D)
304 ON UNDER SURFACE
UTILITY
POWER
FLOOD
FIXED
(C17-1A128)
(LIGHTING CONTROL PANEL)
100
NUMERICS
306
MISSION TIMER
HOURS MIN SEC
EVENT TIMER
MIN SEC
MISSION TIMER START
STOP
RESET
EVENT TIMER
MISSION TIMER
RESET START MIN-TENS-SEC
DOWN STOP UNITS
(C22-1A138)
C23A10
352
WASTE TANK SERVICING
CLOSE
OPEN
(C23A10MV2) (5.18B)
(TP34)
PRESSURE RELIEF
POTABLE TANK INLET
WASTE TANK INLET
CHECK VALVE
TEST PORT
AUTO
WATER CONTROL PANEL
(C23A3LV2) (2.1B)
(C23A3RV1) (3.1A)
325
CABIN PRESSURE RELIEF
BOOST ENTRY
(REF LOCKING ASSY)
PRIMARY GLYCOL TO RADIATORS PULL TO BY PASS (VALVE ON PANEL 377)
326
GLYCOL RESERVOIR
INLET
BY PASS
OUTLET
(C23MV4) (2.28G)
(C23MV5) (2.28H)
(C23MV6) (2.28I)
(C23MV12) (4.26A)
(C23A12MV1) (4.34)
C23MV13 (4.26B)
OXYGEN
(4.26A)
PANEL 108 THRU 112 THRU
V34-900101

FOLD OUT #1



FOLD OUT #2

Figure 1.- Concluded.

TABLE I. - BLOCK II CDS

Component	Quantity per vehicle	Part no.	Certification test report
Accelerometer indicator	1	ME 106-0023	26456001
EL panel	23	ME 181-0XXX	26456002, 027
CDU	1	ME 430-0006	26456003, 01226301
Electrical indicating meter	31	ME 430-0170	26456004, 01226312, 315
Event indicator	51	ME 432-0172	26456005
Barometric-pressure indicator	1	ME 432-0173	26456006
Caution and warning subsystem event annunciator	2	ME 434-0020	01226319
Abort annunciator	1	ME 434-0021	01226318
Saturn V launch vehicle engine annunciator	1	ME 434-0043	01226317
External light	10	ME 434-0041	26456008, 022
Interior floodlight	6	ME 434-0045	26456019, 01226316
Miniature and dual tunnel lights	6	ME 434-0051	26456009, 01226319
Potentiometer	21	ME 444-0033	01226310
Rheostat	3	ME 444-0049	01226311
Variable power transformer	3	ME 446-0034	26456011
Variable transformer-rheostat	2	ME 446-0036	26456012, 021, 026
Pushbutton switch	2	ME 452-0060	01126303, 01226303, 313
Pushbutton switch	1	ME 452-0061	01126303, 01226303, 313
EL pushbutton switch	10	ME 452-0092	26456013

TABLE I. - Concluded

Component	Quantity per vehicle	Part no.	Certification test report
Rotary switch	15	ME 452-0093	01126302, 309, 26456014
Toggle switch	340	ME 452-0102	26456020
Digital event timer	2	ME 456-0044	01226306, 26456015
Electrical converter	1	ME 476-0049	26456018
Light-flashing beacon	1	ME 434-0055	26456023
Filter-flashing beacon	1	ME 447-0043	26456024
Docking light	1	ME 434-0054	26456025
Mission timer	2	LSC-355-31600-9 or LSC-355-31600-11	Government-furnished equipment
Panel assemblies 1 to 10 15, 16, 100, 101, 225, 226, 229, 306, 181, 162, 163, 164, 230, 604, 201, 227, 250, 275, 276, 277, and 278	31	V36	--

Triggers. - Most floodlight failures were attributed to the overloading of a semiconductor diode. Because of excessive current, the unit overheated the encapsulated circuitry and caused emission of smoke and noxious odors. The failure was traced to the thermal sensitivity of the three-layered semiconductor diode used as a trigger. The supplier confirmed that the three-layered diode could not be manufactured with adequate parameter controls.

The three-layered diode trigger was replaced with a four-layered one that could be manufactured with adequate controls. The new trigger maintained the required switching speed over the required environmental range of vibration, temperature, pressure, and humidity. An improved parts screening procedure was implemented on the new triggers. A summary of the screening tests includes the following.

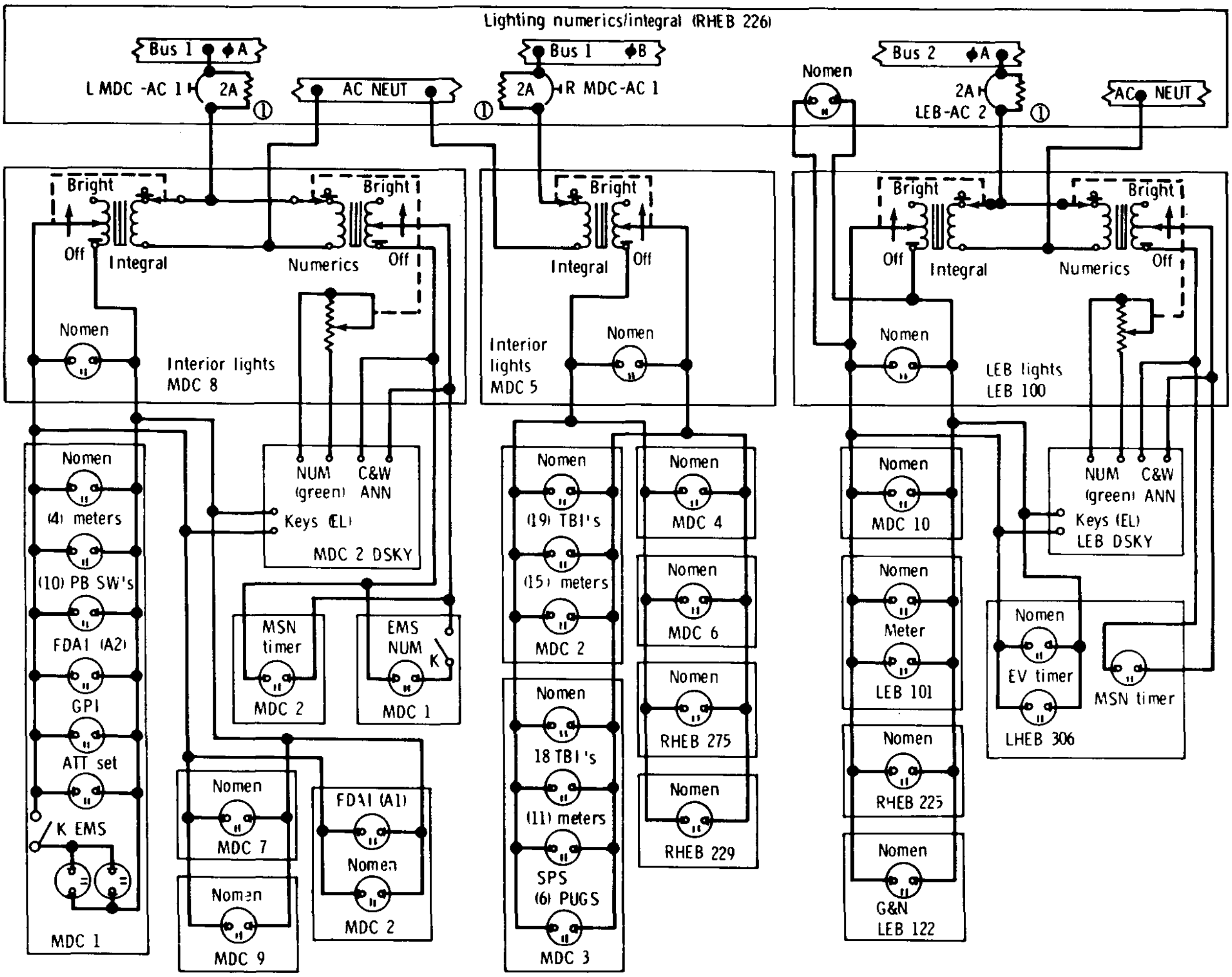
1. A functional test of the trigger at its highest power setting

2. A burn-in for 100 hours at 373 K (100° C) with 30 milliamperes of current

3. A thermal stress of the unit by varying the temperature from 218 to 398 K (-55° to 125° C) for 4 hours

4. A repeated functional test of the trigger

Transient susceptibility. - In July and August 1966, floodlight failures were attributed to transient susceptibility of the lamp circuitry, which created conditions that allowed the lamp assembly to draw excessive current that, in turn, opened the fuse. The circuitry was redesigned to limit the current flow by the addition of an input filter, a time-delay network, and a zener diode.



① Resistors on S/C 102, 106, and subsequent

ANN	annunciator	G&N	guidance and navigation	NUM	numerics
ATT	attitude	GPI	gimbal position indicator	PB	pushbutton
C&W	caution and warning	LHEB	left-hand equipment bay	PUGS	propellant utilization and gaging system
DSKY	display and keyboard	MDC	main display console	RHEB	right-hand equipment bay
EMS	entry monitor system	MSN	mission	S/C	spacecraft
EV	event	NEUT	neutral	SPS	service propulsion system
FDAI	flight director attitude indicator	Nomen	nomenclature	SW	switch
				TBI	talkback indicator

Figure 2. - Panel lighting schematic (integral and numerics).

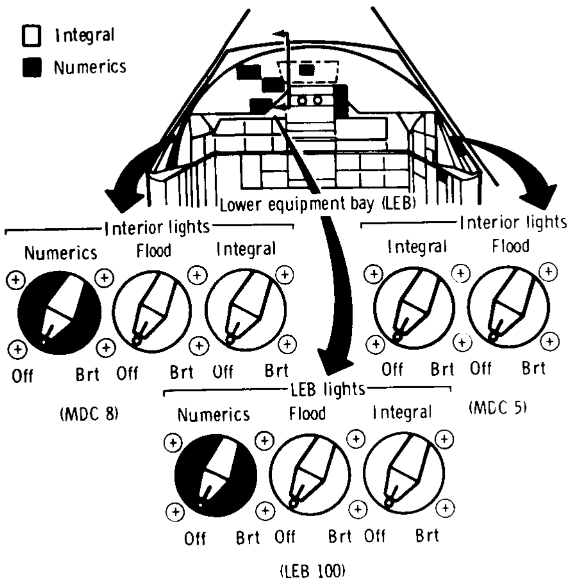


Figure 3. - The command module integral and numerics illumination system.

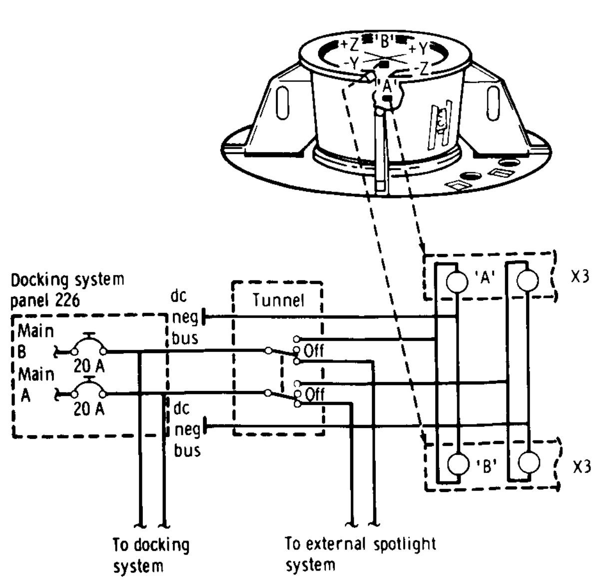


Figure 4. - Tunnel lighting schematic.

Lamp tube. - Other than problems caused by poor handling, the greatest cause of tube failure was an inadequate amount of mercury injected into the tube. Manufacturing procedures were changed to prevent recurrence of this discrepancy.

Lamp moisture absorption. - To prevent failures caused by moisture absorption, improved procedures were implemented for the application of the potting sealant to tube ends and epoxy surfaces.

Silicon-controlled rectifier. - Although no failure was directly attributed to the failure of a silicon-controlled rectifier, the following additional screening was implemented to preclude this possibility.

1. A burn-in for 100 hours at 1 ampere at 373 K (100° C)

2. A test for the minimum gate turn-on current

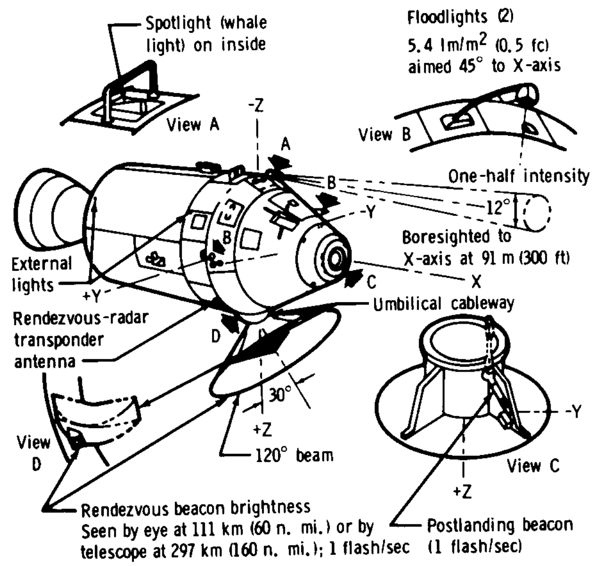


Figure 5. - The CSM docking, extravehicular activity, and external lights.

Circuit protection. - The power distribution circuit was examined to determine the feasibility of improving the overcurrent protection by reducing circuit breaker ratings. It was determined that the existing breakers were sized to protect the power distribution wiring and that the reduction of these ratings to provide protection for each light assembly would result in the loss of other lights or system elements on that same circuit breaker. The final corrective action adopted was to add a fuse in each power input circuit of each light assembly.

Transient protection. - To protect against the effects of switch contact bounce, the input filter chokes were replaced with 0.25-ohm resistors, and the 6.8-microfarad input filter capacitance was increased to 68 microfarads. Also, a 102-millisecond time-constant delay network and a zener diode were added to limit the maximum voltage that could be applied to the trigger circuit.

Lens strength. - During spacecraft checkout, the plastic lenses of the floodlights were found to be susceptible to breakage and dislodgment with normal checkout handling. This susceptibility disclosed the possibility that mercury could be released in toxic amounts into the command module (CM) interior if fluorescent lamps inside the floodlight should become cracked or broken. An investigation revealed that this susceptibility was caused by inadequate strength of the lenses and lens attachment to the floodlight cases. Lenses and lens-case attachments were redesigned. Thicker, more rugged lenses of high-impact plastic were used. Lenses were produced by machining and optical polishing. The previously used adhesive-bonding technique for lens attachment to the case was replaced by a mechanical fastening. These changes provided greater lens-case strength and, consequently, improved lamp protection. The redesigned lenses met the following requirements: stress of 890 newtons (200 pounds) applied on 6.45 square centimeters (1 square inch) of the lens surface; stress of 1334 newtons (300 pounds) applied on 25.8 square centimeters (4 square inches) of the lens surface; and impact by a 7-kilogram (15 pound) disk (38 centimeters (15 inches) in diameter by 1.3 centimeters (0.5 inch) thick) moving at 0.6 m/sec (2 ft/sec).

To make the CM interior more fireproof, the high-impact plastic lens was replaced by a lens of laminated glass. Because the glass lens could splinter on hard impact, it was overlaid with a thin (approximately 0.008 centimeter (0.003 inch)) coating of fluorinated ethylene propylene Teflon to prevent injury to any crewmember should splintering occur.

Fluorescent tube protection. - To prevent leakage of mercury vapor in case of fluorescent tube failure (crack or break), a protective coating of Teflon was added to the tube, and the tube ends were encapsulated to provide a seal.

Noise. - Rotation of the floodlight-brightness-control rheostat for the primary floodlights caused a high-pitched noise. The rheostat was encapsulated in a new configuration, using a hard-shelled room-temperature-vulcanizing (RTV) adhesive.

Electroluminescent lighting. - The panel nomenclature and instrumentation digital readouts were illuminated by EL panels. The EL lighting operated on 115-volt ac, 400-hertz power. The EL lighting was relatively trouble free throughout the program.

Other lighting. - Lighting for the tunnel area was provided by six incandescent fixtures, each containing two miniature 28-volt dc lamps (fig. 4). Exterior lighting (fig. 5) consisted of a docking spotlight, eight orientation or identification lights (four amber, two red, and two green), a rendezvous flashing beacon, and a white floodlight for EVA's. The docking spotlight was electrically deployed on a spring-mounted door and projected a light beam 21.3 meters (70 feet) in diameter at a range of 61 meters (200 feet). The orientation lights were small lamps spaced around the service module (SM) for position identification by other vehicles. The rendezvous flashing beacon, located in the forward section of the SM, used a xenon tube that flashed at a rate of 1 flash/sec for a duration of 20 milliseconds and was visible by telescope for 297 kilometers (160 nautical miles). The beacon was used for long-distance identification by the lunar module (LM) crew when approaching the CM. The white floodlight (EVA light), also located in the forward section of the SM, was spring loaded to deploy when the boost protective cover was jettisoned. The floodlight was used to illuminate the hatch area and the side of the CM during EVA. The tunnel lighting and all exterior lighting were relatively trouble free during the entire program.

Caution and Warning Subsystem

The caution and warning subsystem (C&WS) (figs. 6 and 7) monitored critical parameters of most of the spacecraft operational systems and alerted the crewmembers to malfunctions or out-of-tolerance conditions when they occurred in any of these systems. The C&WS was composed of the CDU, system status light matrices, master alarm (MA) switches, and associated control switches. The CDU, the major component in the C&WS, was located behind panel 3. The CDU contained two redundant regulated ±12-volt dc power supplies, high- and low-limit comparators, logic circuitry, level detectors, lamp drivers, and a two-level-output tone generator. When a malfunction or out-of-tolerance condition occurred, the alarm limits (reference voltages) were exceeded. These alarm-limit signals triggered voltage comparators that activated logic and lamp driver circuits. These circuits activated the MA, the tone generator, and the appropriate system status light on panel 2. The MA light and the bilevel (750 to 2000 hertz) tone were reset when the MA pushbutton switch was depressed. However, the status lights on panel 2 remained lighted as long as the malfunction or out-of-tolerance condition existed. The status lights could be prevented from lighting when there was an out-of-tolerance condition by positioning the "normal/boost/acknowledge" switch to "acknowledge." When a malfunction occurred with the switch in this position, only the MA and the tone generator were energized, but the status light was lighted only while the MA was depressed. When the switch was placed in the "boost" position, the MA light adjacent to the red "abort" light on panel 1 was disabled to prevent possible confusion between the two lights. Thirty-three of the available 48 matrix status lights on panel 2 were used as system inputs; some of the lights had as many as 7 separate inputs to indicate malfunctions. If there was more than one parameter to a status light, the first malfunction to occur triggered the tone, the MA light, and the status light. As long as the malfunction existed, the tone and MA circuits could not be reset for that particular channel, and any of the other parameters in the channel that went out of tolerance were unable to indicate the out-of-tolerance situation through the C&WS. This condition, known as masking, was very undesirable. However, by the time this condition was recognized as a problem, correction would have required a major redesign and redevelopment program on the CM C&WS. The main reason for not correcting the masking problem was that all parameters in question could be monitored elsewhere during this period.

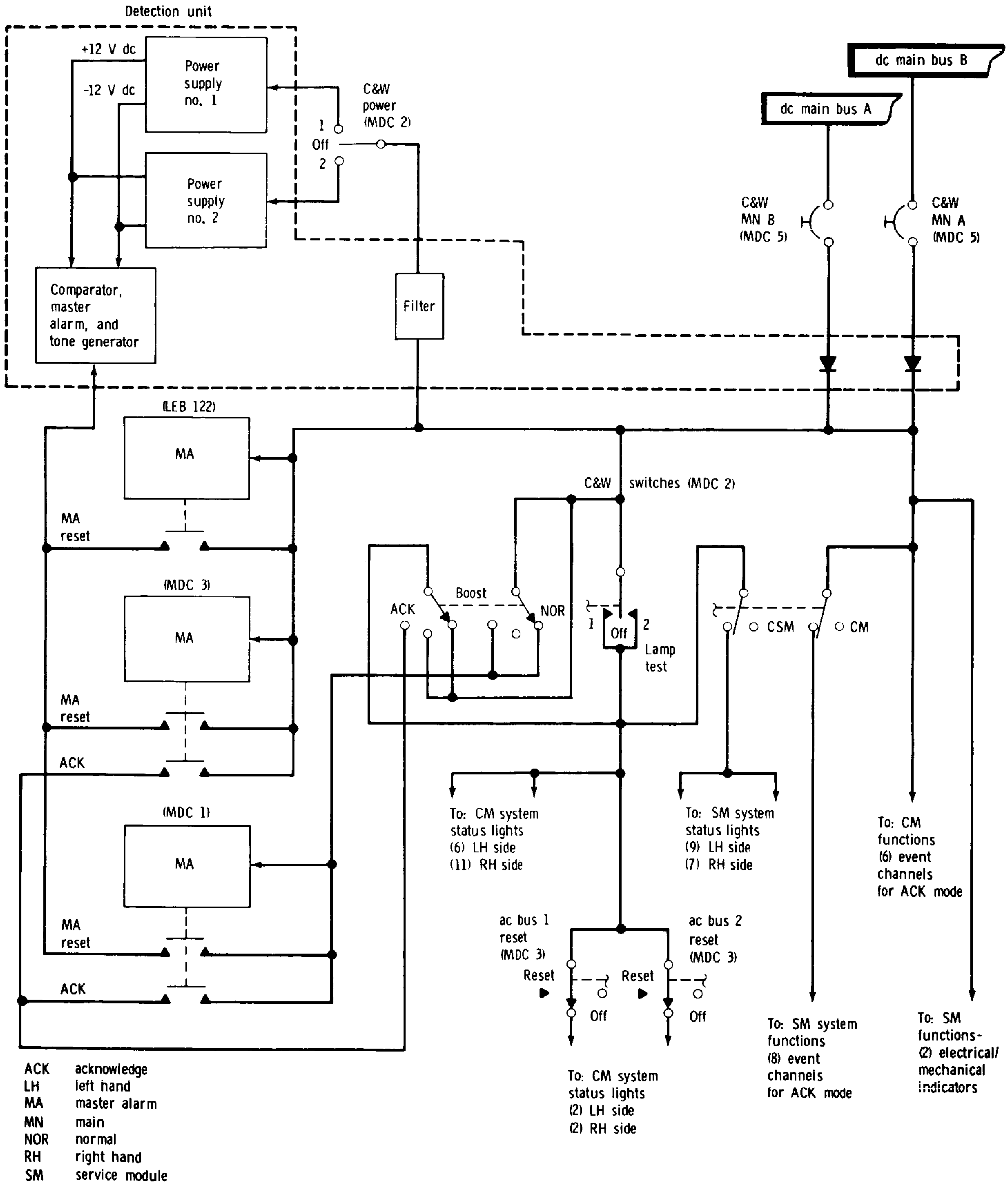


Figure 6. - Schematic of C&WS power distribution.

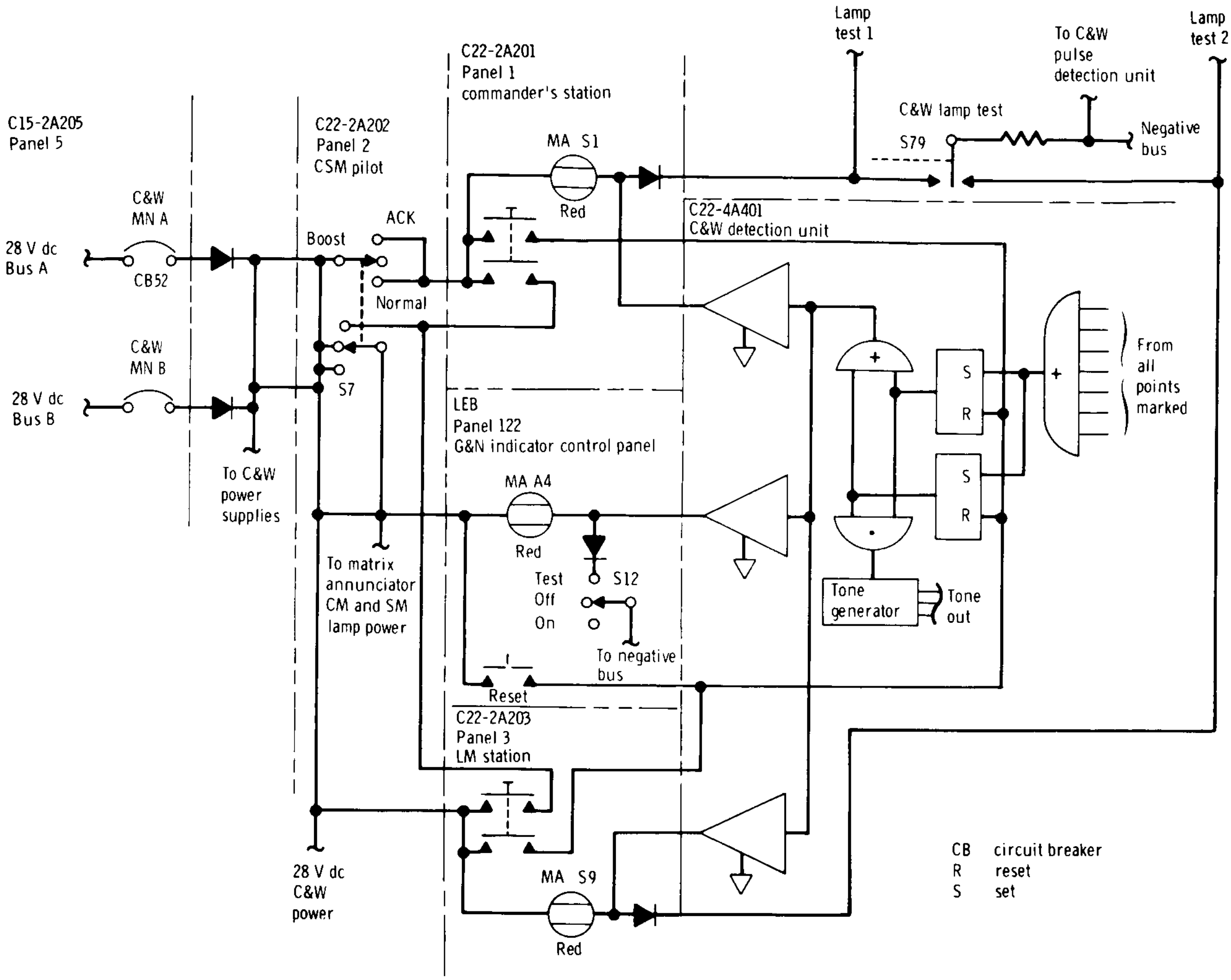


Figure 7.- The C&WS MA schematic.

No known vehicle test failures and only one reported flight anomaly occurred in the C&WS. The reported anomaly occurred approximately 0.45 second after final ring-latch closure during the initial transposition and docking maneuver of the Apollo 9 (CSM 104) mission. Analog and bilevel data did not indicate any out-of-tolerance condition at the time. The fact that the alarm occurred during the hard docking rather than at physical contact ruled out static discharge between the two vehicles and indicated a shock-sensitive condition. The C&WS MA circuit was a latching system that could respond to transients of 10 microseconds in duration, and a transient on any of the 69 inputs could trigger the alarm. Analysis of CSM 104 flight data revealed no system anomaly; thus, the specific cause remains unknown.

Most of the C&D components mounted in the panels had a relatively low or insignificant failure record; however, some had numerous failures and a few underwent a redesign and requalification phase. The following discussion describes the C&D components, emphasizing the ones with the most recorded failures.

Mission Timer

The electronic mission timer indicated time in hours, minutes, and seconds on a seven-digit EL panel. Maximum accumulative time was 999 hours 59 minutes 59 seconds. The timer operated on 28-volt dc and 115-volt ac, 400-hertz power, and it used the 10-pulse/sec signal from the central timing equipment (CTE) or its own internal tuning fork as a time reference.

The internal tuning fork time reference was effective when the CTE became inoperative. This time was indicated by a small lighted tuning fork symbol to the left of the seven-digit panel. Stop, start, reset, and slew controls were adjacent to each of the two mission timers in the CM (one each on panels 2 and 306). The mission timer was supplied to the CSM contractor as Government-furnished equipment and was identical to the one used on the LM. The initial recorded failure on a CM was in February 1967 on CSM 101. Between February 1967 and November 1969, there were 12 failures, 3 of which occurred during flight. An investigation revealed that the failures were traceable to cracked solder joints, which were caused by internal differential thermal expansion between the end terminal boards of the cordwood construction and the encapsulating material used to protect the internal electronic components against shock and vibration. Unsuccessful attempts were made to open the hermetically sealed units, to resolder the terminals, to reseal the units, and to reuse them. Finally, in 1969, the decision was made to redesign the units, using integrated circuits and welded connections instead of the cordwood construction and soldering, which had been dictated by the use of discrete components.

Several problems were encountered with the redesigned timer. The EL digital panels, reused from earlier timers made by a previous vendor, were not as bright as desired. However, the lack of light intensity was not as great a problem as a defect in the original EL panels made by the new vendor; these panels would not withstand the voltage (115-volt ac, 400 hertz). Thus, a second new vendor for the EL panels had to be found. The later EL units were relatively trouble free.

While testing the timer, it was found that the combination of electrical parameters at several temperature values reduced the gain to a point where the new oscillator would not restart every time it was stopped. Circuit analysis indicated that the problem was in the integrated-circuit logic gates. Substitution of transistors for the logic gates solved the problem.

Several procurement problems were experienced with printed circuit boards and integrated circuits, but these problems were solved by locating new supply sources. The insulation resistance of the input filter inductor was a problem on the earlier redesigned timers, but a better filter inductor was found after several months of searching. The new mission timer experienced three failures, one light input filter capacitor on CSM 112 (Apollo 15) and an EL segment on each of the timers on CSM 114 (Apollo 17). To correct these problems, the timer ac input circuits were fused, and the capacitor rated voltage was increased.

Delivery delays were caused by additional paperwork required because one initial prime contractor did not have direct access to the vendor but instead had to deal directly with a second prime contractor.

Digital Event Timer

The DET was an electromechanical device that counted up or down in minutes and seconds to a maximum of 59 minutes 59 seconds. There were two DET's on each CM: one on panel 1 for flight control and one on panel 306 for navigational purposes. The DET would reset to zero and begin counting up when a lift-off or an abort signal occurred. Timer controls to stop, start, count up and down, and slew each digit were on the same panel with the timer. Of the 22 failures that occurred, 5 were electrical, and the remaining 17 were caused by mechanical problems in the counting-wheel mechanism. Only four of these failures occurred during flight (one each on CSM 106 and CSM 113 and two on CSM 112). Although the electrical problems were varied (EL lighting, timer failing to stop on "reset," and broken wire), they were not difficult to correct.

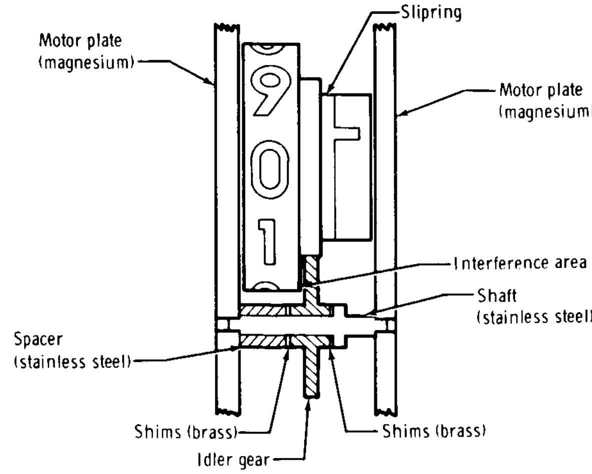


Figure 8.- Apollo 15 DET anomaly.

The mechanical problems consisted of interference between the gear and the counter wheels and contamination of the brushes and the sliprings. These problems were first attributed to the looseness of the dimensional tolerances, which allowed the idler gear (figs. 8 and 9) to rub against the number wheel and dislodge paint particles from it. Being nonconductive, the paint particles would adhere to the slipring and prevent the brush from making contact, which, in turn, would prevent the counting pulse from being transmitted to the next counter wheel. When brass shims, inserted between the idler gear and the spacers on the shaft to provide more clearance, did not correct the problem, a more thorough investigation was made. During this investigation, it was found that the stainless-steel shaft on which the idler gear turned was not fastened securely to the magnesium motor plates. This slack allowed the shaft to turn, and the small moment applied to the end of the shaft by the idler gear eroded the shaft hole to an oval shape. The end-plate erosion was sufficient to allow the shaft to move and to cause the idler gear to rub the paint off the number wheel. Because the timer was not mission critical and because only one flight remained in which the timer would be used very often, the decision was made to visually inspect the timers before flight rather than to attempt a redesign and refurbishment program.

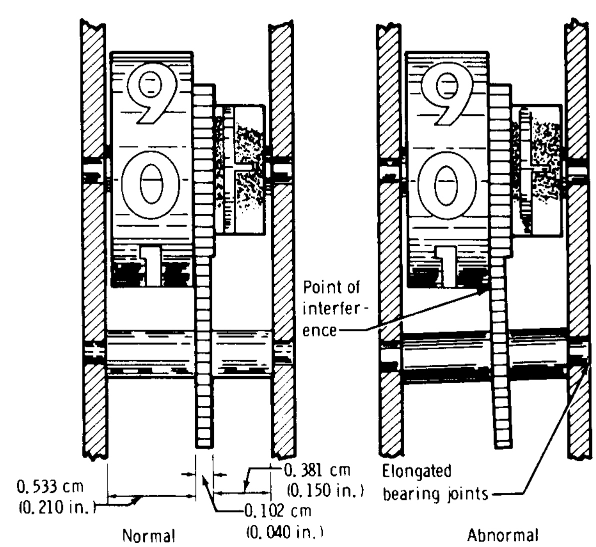


Figure 9.- Idler gear.

Electrical Indicating Meters

The panel meters were of the D'Arsonval or moving-coil type and varied in scale types from single to dual and from circular to elongated. No in-flight panel-meter failures occurred. Most failures occurred during vendor acceptance checkout and were caused by out-of-tolerance pressure leakage, by inadequate or excessive play in jewel-pivot end joints, by inaccurate calibrations, or by foreign particles interfering with coil rotation. The high leakage rate was corrected by adding a secondary seal of epoxy cement overlaying the solder seals. The inadequate or excessive play in the jewel-pivot end joints, the inaccurate calibrations, and the interior contamination problems were corrected by readjustment, recalibration, and an internal cleaning process in the vendor's facility before retest and shipment.

Event Indicators

The event indicator is an electromagnetic device that drives a drum or shutter-type display. The manner of display depends on the type of sensor driving it. In a valve position, if the valve-actuating device gives a proportional signal to the indicator, the varying position of the valve is shown by the indicator. No known flight failures of the event indicator occurred. Failures in the vendor acceptance and vehicle test programs were primarily caused by mishandling or by poor workmanship. These failures consisted of foreign particles, out-of-adjustment or out-of-socket jewel and pivot, and excessive flag excursion. The excessive flag excursion was caused by improper jewel and pivot adjustment and was the result of poor workmanship. To correct these problems, the vendor reworked and reevaluated each unit before delivery.

Potentiometer

The potentiometer, a hermetically sealed unit with a carbon element and a brush, was used in the CM for volume and squelch control in the communication system and for cabin heat control in the environmental control system. The shaft was connected to the internal element through a sealed bellows assembly. The potentiometer was mounted behind the panel so that the shaft was parallel to the panel surface and the thumbwheel control protruded through the panel surface. Because of the side loading on the shaft, the brush elements were overriding the stop and breaking inside the hermetically sealed unit. This problem was solved by installing a bearing support for the outer end of the shaft with an external mechanical stop. Some problems with shaft binding in the bearing surface still remained; however, these problems were solved with better alinement and some dry lubricant. No known flight failures occurred, and almost all the reported failures in the vendor acceptance, the panel checkout, and the spacecraft test phases were attributable to side loads on the shaft and overriding of the internal stop as a result of excessive loads.

Rotary Switches

Rotary switches were used as selector switches when there were more than two required selections. The switches used on the CM varied in capacity from 2 to 10 poles, from 2 to 12 positions, and from 1 to 5 decks. The switches had a very successful history with few vehicle test failures and only one reported flight anomaly. This anomaly occurred on the Apollo 15 (CSM 112) mission when the battery relay bus voltage on the system/test meter (panel 101) indicated approximately 14 volts instead of 32 volts. Panel 101 contained two rotary switches. Postflight tests conducted in the CM at the contractor's plant and at the vendor's facility revealed no conditions that supported the reported flight anomaly.

The "extra" detent indication was a development problem first noticed on spacecraft 008 and 012 (both block I vehicles). The irregularity had been previously noticed but no corrective action was taken until a short circuit occurred between contacts. When the short circuit occurred, the switch was in the extra or "false" detent position because assurance had been given that no current flow would occur when the switch was in that position. As a result, a redesign and reevaluation effort was made, and the switches delivered for spacecraft 012 changeout were of the redesigned configuration. The extra detent difficulty is explained in the following discussion.

The span between positions was 30° (fig. 10). During the first 15° of rotation, the spring-loaded star ball was climbing the ramp of the star wheel, and the spring-loaded contact ball was going down the ramp of the contact button. During the last 15° of rotation, the spring-loaded star ball was going down the ramp of the star wheel, generating enough force to permit the spring-loaded contact ball to climb the ramp of the contact button. Extra detent was felt at the 15° point because the spring-loaded contact ball was starting to climb the contact button ramp and the spring-loaded star ball was not yet going down the ramp of the star wheel. Because the star ball was at the point of the star, no force was being generated for climbing the ramp of the contact button. This effect was more noticeable on multiple-pole, multiple-deck switches where there were many contact button ramps to climb. Because of the tolerance buildup at this midway position, the number of contacts made could not be readily determined.

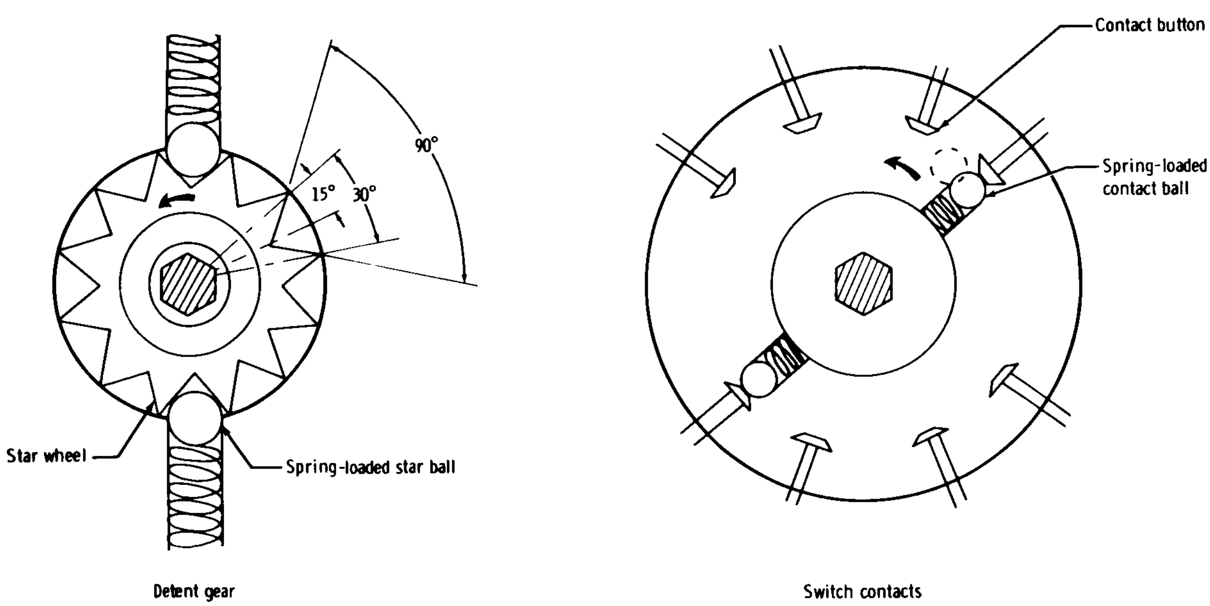


Figure 10. - Rotary switch.

To remedy the extra detent problem, the vendor lowered the ramp angle on the contact buttons and improved tolerances by changing the tooling jigs. The spacecraft 012 deliveries were the first to have these modifications.

Pushbutton Switches

Pushbutton switches were used as backup controls for the initiation of events that would normally occur as automatic functions and as the C&WS MA reset. No flight failures were recorded, but several failures occurred during vehicle testing. One of these failures, which forced the removal of approximately 57 switches from vehicles produced after CSM 107, resulted from the vendor's use of a paraffin-type material in a grinding operation to avoid the entry of ground-off particles around the actuation buttons on the contact portions of the switch. The material hardened and prevented switch actuation. Another vehicle failure resulted from a switch missing one of the manufacturing process steps, which was to have crimped the pushbutton plunger base over the insulation board of the EL lamp lead. The problem occurred in the environmentally sealed portion of the switch. When the external ambient pressure was reduced in the altitude chamber, the plunger, which had been held in by the strength of the EL lead wires as long as the pressures remained equal, pushed the face of the switch out so that it was protruding from the surface of panel 1 (fig. 11).

Toggle Switches

Toggle switches, used as manual control elements on the CM, were configured in one to four poles and in two or three positions and were of a maintained, momentary, or lock type, depending on the use. The block I toggle switch was an environmentally sealed unit with a low failure rate. After the block II requirements were established for a hermetically sealed switch, it was determined that the block I vendor did not manufacture one and did not choose to manufacture one. Consequently, the initial block II toggle switch was a hermetically sealed plunger-type unit with an unsealed toggle mechanism for switch actuation. Early in the block II program, switch problems became apparent. Failures caused by high contact resistance, by mechanical lockup, and by nonfunction at high or low pressure resulted in a change to the present completely hermetically sealed switch. This switch had many serious problems (figs. 12 to 14), such as internal solder balls, extra parts and pieces inside switches, contact buttons welded to a terminal post in an inverted position, defective contact-button welds, auxiliary contact-plate welds, momentary shorting of contact arm with spring, cracked toggle block assembly and bellows, annealing and displacement

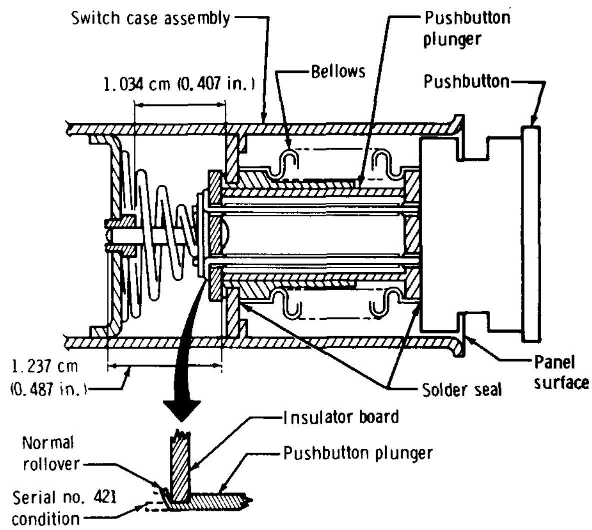


Figure 11.- Pushbutton switch plunger cutaway.

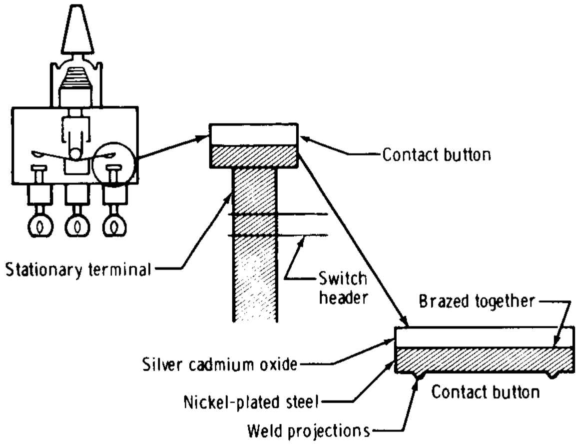


Figure 12.- Toggle-switch problem.

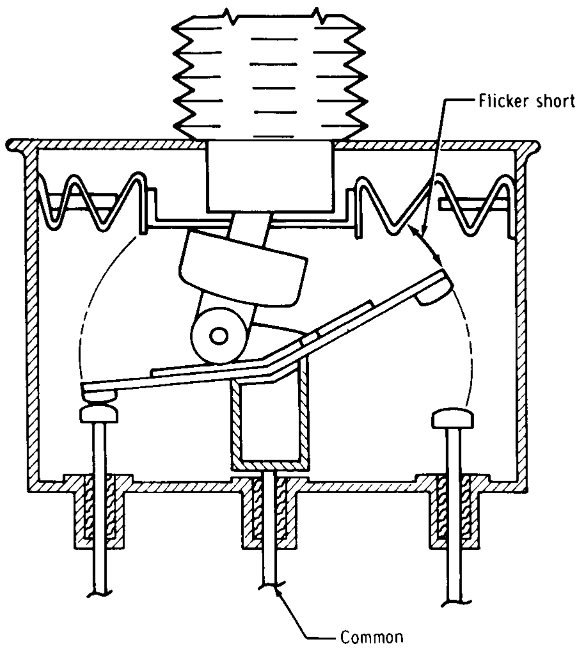
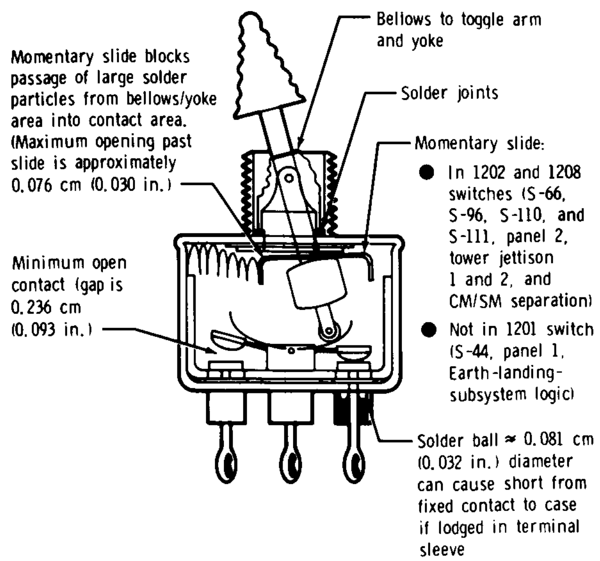
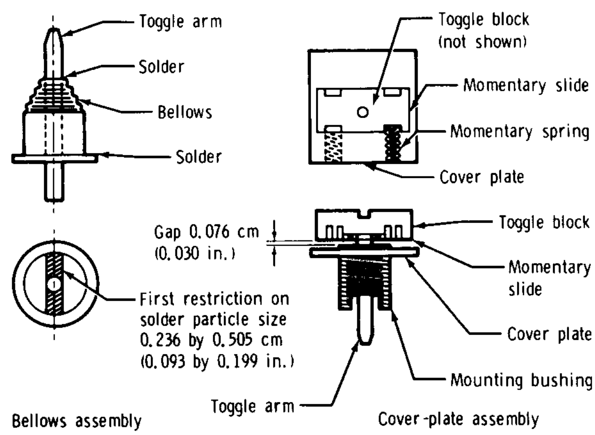


Figure 13.- Momentary switches.



(a) Cutaway view.



(b) Detailed view.

Figure 14.- Toggle switch.

of the momentary return spring, spring shorting to contact arm during sinusoidal vibration, and high contact resistance.

Most of these problems were corrected by the addition of inspection stations during manufacturing, by better workmanship, by altering some of the weld processes, and by adding several additional steps to the acceptance test procedure (ATP). One of the most valuable additions was the requirement for multiaxis X-ray exposures. Despite the poor performance during ATP and vehicle test, only one flight failure was recorded. On the Apollo 15 (CSM 112) mission, the differential-velocity thrust switch provided a groundpath through a small strand of wire from the switch internal braid and lighted the entry-monitor thrust lamp. The fault could have been serious had it been on the engine-valve power input of the service propulsion system solenoids.

Other Components

The variable transformer, variable transformer-rheostat, rheostat, accelerometer, barometric-pressure indicator, and engine annunciator each had a low failure rate during vehicle testing and had no flight failures.

CONCLUDING REMARKS AND RECOMMENDATIONS

The mission requirements for the command and service module controls and displays subsystem were satisfied very well during the Apollo Program. Failures were numerous and costly, but a very thorough qualification and test program prevented any catastrophic incidents. The failures were costly because the time required for component removal and replacement in the panel was long and the retest was extensive.

The following recommendations should be considered for future spacecraft.

1. Display panels containing control elements should be simple to remove as a unit or, whenever possible, the individual control elements should be modular plug-ins.

2. When there is a choice between the electrical or mechanical accomplishment of a function, the electrical method has been shown to be superior in this subsystem.

3. When the same component is used by more than one prime contractor or on more than one vehicle, the component should be procured from a single vendor using identical specifications, if at all possible.

4. Government-furnished equipment should be delivered directly to the user from the Government installation or from the vendor rather than from one prime contractor to another.

5. Crewmembers have on several occasions expressed a desire for switches that are lighted in some manner to indicate location and status during darkened operations and sleep periods.

Lyndon B. Johnson Space Center
National Aeronautics and Space Administration
Houston, Texas, July 2, 1976
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