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JSC-10607

(NASA-TM-X-74149) APOLLO SOYUZ, MISSION N76-30242
EVALUATION REPORT (NASA) 254 p HC \$9.00
CSCL 22A
Unclas
G3/12 15157

APOLLO SOYUZ
MISSION EVALUATION REPORT



National Aeronautics and Space Administration
LYNDON B. JOHNSON SPACE CENTER

Houston, Texas
December 1975

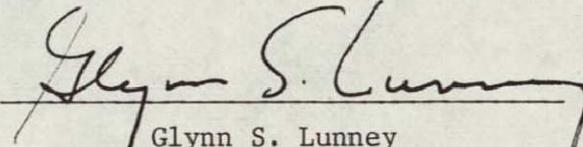


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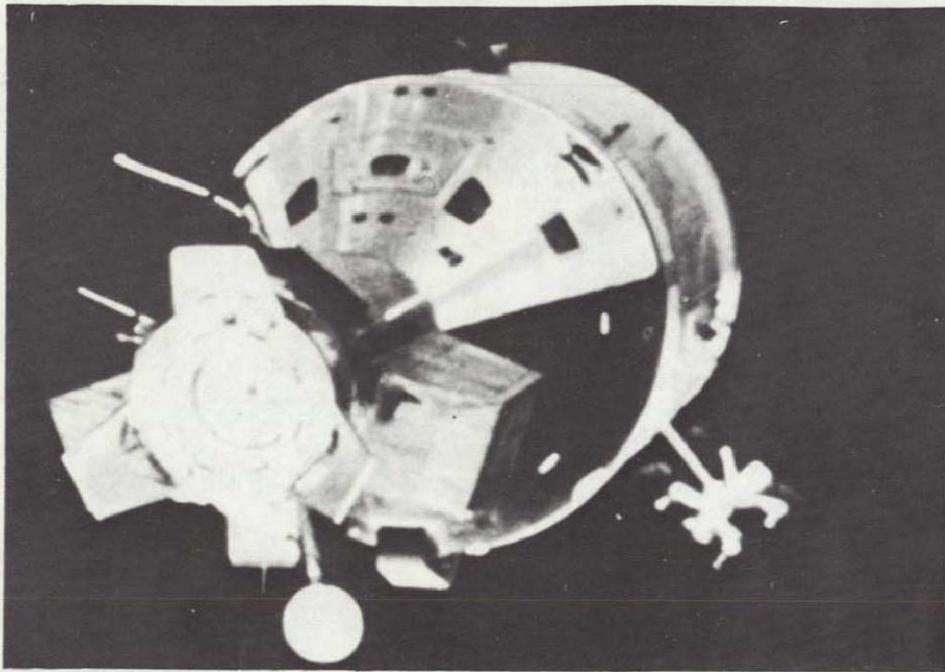
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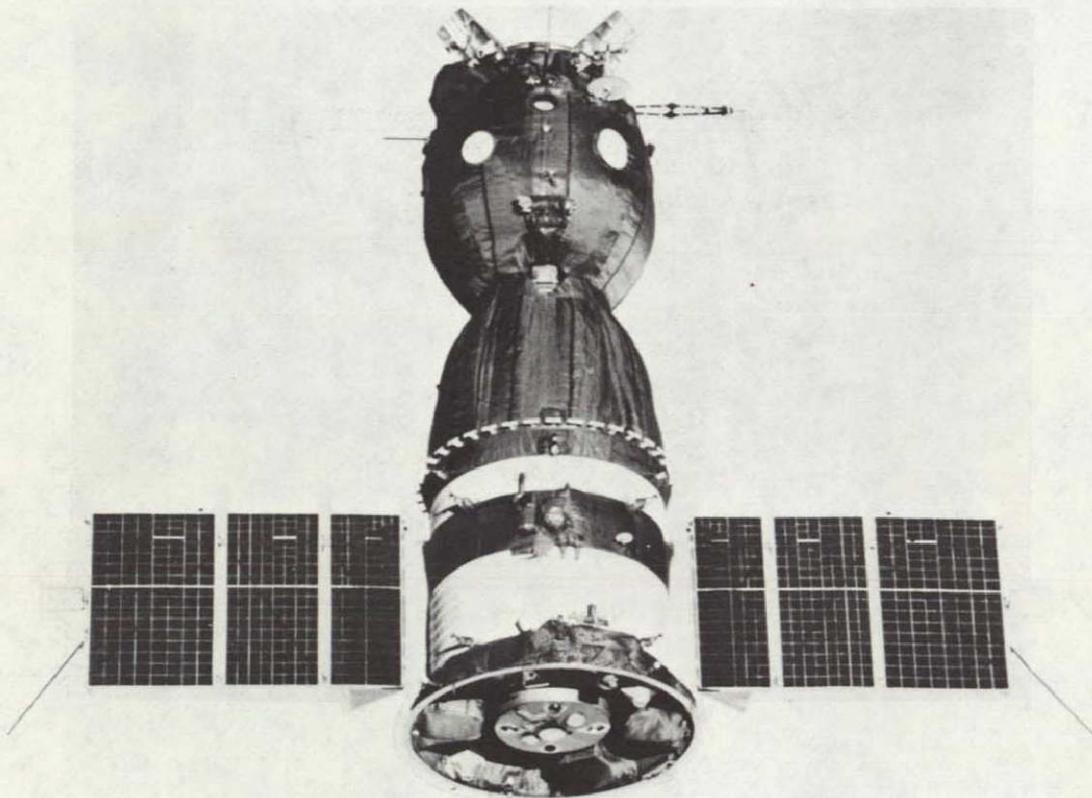
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS
December 1975



Apollo as seen from Soyuz



Soyuz as seen from Apollo

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1.0 INTRODUCTION

The Apollo Soyuz mission was the first manned space flight to be conducted jointly by two nations - the United States and the Union of Soviet Socialist Republics. The primary purpose of the mission was to test systems for rendezvous and docking of manned spacecraft that would be suitable for use as a standard international system, and to demonstrate crew transfer between spacecraft. The secondary purpose was to conduct a program of scientific and applications experimentation. With minor modifications, the Apollo and Soyuz spacecraft were like those flown on previous missions. However, a new module was built specifically for this mission - the docking module. It served as an airlock for crew transfer and as a structural base for the docking mechanism that interfaced with a similar mechanism on the Soyuz orbital module.

This report contains the postflight evaluation of the performance of the docking system and docking module, as well as the overall performance of the Apollo spacecraft and experiments. In addition, the mission is evaluated from the viewpoints of the flight crew, ground support operations, and biomedical operations. Descriptions of the docking mechanism, docking module, crew equipment and experiment hardware are given in appendix A. Other appendixes contain an as-flown flight plan as well as information on spacecraft manufacturing and checkout history, mass properties, and postflight testing.

The International System of Units (SI) is used throughout this report except for the biomedical evaluation where exceptions are made for physiological measurements. Scales for conversion of SI units to conventional units are contained in appendix G.

Unless otherwise indicated, all times given in this report are elapsed time referenced to the time of Soyuz lift-off, 12:20:00 Greenwich mean time (G.m.t.), July 15, 1975. References to mission days are based upon each day ending at 24:00:00 G.m.t.

2.0 MISSION SUMMARY

The Soyuz spacecraft, manned by Alexei A. Leonov, Commander, and Valeri N. Kubasov, Flight Engineer, was launched atop a Soyuz Rocket Booster from the Baikonur Cosmodrome in Kazakhstan on July 15, 1975, at 12:20:00 G.m.t. and the spacecraft was inserted into an orbit of 222 by 186 kilometers. Two maneuvers were performed to circularize the orbit, the first during the fourth orbit and the second during the 17th orbit. The second maneuver placed the Soyuz spacecraft in a near-circular orbit of about 223 kilometers in preparation for docking with the Apollo spacecraft.

The Apollo spacecraft, manned by Thomas P. Stafford, Commander, Vance D. Brand, Command Module Pilot, and Donald K. Slayton, Docking Module Pilot, was launched by a Saturn IB launch vehicle from Pad B of Complex 39 at the Kennedy Space Center 7 1/2 hours after the Soyuz launch. The time of launch was approximately 19:50 G.m.t. The spacecraft and S-IVB stage were inserted into a 168 by 149 kilometer orbit 9 minutes and 56 seconds later. All launch vehicle systems performed normally.

The Apollo spacecraft was separated from the S-IVB stage 1 hour and 14 minutes after lift-off. The spacecraft was then docked with the docking module which was attached to the S-IVB, and the docking module was extracted. These operations were normal; however, the removal of the docking probe was hindered by a misrouted pyrotechnic connector cable. The corrective procedure given to the crew was used successfully to remove the probe.

The Apollo circularization, phasing, and rendezvous maneuvers were performed as planned and the first docking was performed on July 17, 1975, at 16:09:09 G.m.t. (51:49:09). Upon opening hatch 1, the Apollo crew smelled an odor like "burnt glue" or "acetate" which, upon investigation, proved to be of no consequence. The first transfer between the American and Russian spacecraft was made by the Apollo Commander and the Docking Module Pilot. All planned activities were conducted including connection and checkout of communications cables, and experiment operations. Messages of congratulations were received by all crewmen from the heads of state of both nations. Second and third transfers were made so that each crewman could visit the other spacecraft. Activities during these periods included: television tours of the two spacecraft and of parts of the United States and the Soviet Union during overflight, a press conference in which all crewmen participated, experiment activities, eating, exercising, and exchanges of symbolic items to commemorate the successful docking and transfer. The fourth transfer returned all crewmen to their own spacecraft.

The first undocking was performed normally, with the Apollo docking system active, after the two spacecraft had been docked for almost 44 hours. A joint artificial solar eclipse experiment (MA-148) was performed following undocking and the spacecraft were docked a second time to test the docking mechanisms with the Soyuz docking system active. The docking was performed at 96:13:39. Final undocking was at 99:06:12 after which the ultraviolet absorption experiment (MA-059) was conducted. Unilateral experiments were performed upon completion of the joint activities.

The Soyuz deorbit maneuver was performed at 141:50:30 and the Soyuz reentry vehicle was brought to a safe landing at approximately 50 degrees 15 minutes north, 67 degrees 36 minutes east in Kazakhstan on July 21, 1975, at 10:50:54 G.m.t. after a flight of 142 hours 30 minutes and 54 seconds. The Apollo spacecraft continued in orbit for four additional days. Experiments accomplished during this time were conducted essentially according to plan. One finding of special significance was the first known detection of a cosmic source of extreme ultraviolet radiation. The source was found with the extreme ultraviolet telescope (MA-083). Some experiment hardware problems were experienced but, in general, the hardware performed very well and excellent scientific data were obtained.

The docking module was jettisoned at 199:27:00 in preparation for the doppler tracking experiment (MA-089) which required a 300-kilometer separation of the command and service module from the docking module. The maneuvers to accomplish the separation and to maintain a constant range were performed as planned and good data were received from the doppler measurements. All experiment operations were concluded during Apollo revolution 136 and preparations were begun for deorbit and landing.

The deorbit maneuver was performed at 224:17:47 and the command module landed about 1.3 kilometers from the target point on July 24, 1975, at 21:18:24 G.m.t. after a flight of 217 hours 28 minutes and 24 seconds. The coordinates of the landing point are 22 degrees 0 minutes 36 seconds north and 163 degrees 0 minutes 54 seconds west. The command module went to the stable II attitude for about 4 1/2 minutes after landing. The crew remained in the spacecraft during recovery and were aboard the *U.S.S. New Orleans* about 41 minutes after landing.

During shipboard ceremonies, the crew appeared to be in good physical condition but it was later learned that the crew had been exposed to oxidizer vapors in the cabin for several minutes. The situation arose when an arming function and a manual backup function were missed. The crew remained in Hawaii for 2 weeks for medical observation before departing from Hawaii.

3.0 TRAJECTORY

Soyuz lift-off occurred on time at 12:20:00 G.m.t. on July 15, 1975. The Soyuz ascent resulted in an insertion orbit of 221.5 by 186.3 kilometers. The inclination of the orbital plane was a near-nominal 0.90 radian. Soyuz performed a 3.1-meter-per-second maneuver on the fourth revolution at 5:31:40 to correct the phasing and altitude for the 17th revolution circularization maneuver. The circularization maneuver was performed at 24:23:35 and the 11.6-meter-per-second velocity change placed the Soyuz in a near-circular assembly orbit of 223.0 by 222.2 kilometers.

The Apollo lifted off at 19:50:00.6 G.m.t. (7:30:00.6 g.e.t.) on July 15. The ascent resulted in an insertion orbit of 168.2 by 148.9 kilometers. Table 3-I lists the insertion parameters. After the docking module was extracted on the second revolution, the Apollo orbit was circularized at 167.4 by 167.2 kilometers. From this orbit, the first Apollo phasing maneuver was executed to provide the proper catch-up rate so that docking with the Soyuz spacecraft could occur on the 36th Soyuz revolution. The 20.5-meter-per-second phasing maneuver at 13:08:30 placed the Apollo spacecraft in a 232.8 by 168.7 kilometer orbit. A small phase and plane correction maneuver of 2.7 meters per second was executed on the 16th revolution.

The final rendezvous sequence began at 48:31:00 with the Apollo spacecraft performing a series of maneuvers which placed it in a 224.8 by 205.2 kilometer intercept orbit with Soyuz. After braking and station-keeping, the docking (capture) was accomplished at 51:49:09, about 6 minutes earlier than the nominal time. The duration of the first docked phase was 43 hours 54 minutes 11 seconds. The orbit was 223.5 by 222.8 kilometers after first docking and 220.9 by 219.1 at first undocking.

First undocking occurred at 95:43:20, about a minute after sunrise. Fifteen seconds later, Apollo began the first of two reaction control system maneuvers to back away from the Soyuz spacecraft to provide an artificial eclipse of the sun. These maneuvers were executed as planned and the experiment was concluded at approximately 95 hours 48 minutes when the distance between the spacecraft was 220 meters. The Apollo spacecraft then reapproached the Soyuz spacecraft and the second docking (capture) occurred at 96:13:39. Final undocking occurred 2 hours 52 minutes 33 seconds later. The Apollo maintained a distance from the Soyuz of about 20 meters until the first maneuver for the ultraviolet absorption experiment was executed at approximately 99 hours 22 minutes. This began a series of Apollo out-of-plane maneuvers to accomplish 150-meter and 500-meter data-takes during the next 3 hours. The Apollo subsequently performed a small inplane separation maneuver of 0.6 meter per second at 102:22:27 which caused the Apollo to pass directly above the Soyuz at 102:49:00 for

TABLE 3-I.- APOLLO INSERTION PARAMETERS

Time elapsed from Soyuz lift-off, hr:min:sec	7:39:56
G.m.t. on July 15, hr:min:sec	19:59:56
Altitude, kilometers	150.0
Inertial velocity, meters/sec	7818.9
Inertial flight path angle, radian	0.0
Inertial azimuth, radian	0.934
Inclination, radian	0.904
Target node, radians	2.737
Eccentricity	0.001
Apogee, kilometers	168.2
Perigee, kilometers	148.9
Period, minutes	87.7

a 1000-meter ultraviolet data-take. The separation maneuver and 1000-meter data-take terminated the joint phase of the mission. The Apollo orbit at this time was 222.1 by 220.9 kilometers, which caused the Apollo to begin trailing the Soyuz at an increasing range rate of approximately 9 kilometers per revolution.

At 118 hours 21 minutes, the Soyuz spacecraft performed a 1.5-meter-per-second retrograde engine test firing which lowered the orbit to 215.8 by 213.5 kilometers and caused the Soyuz to move away from the Apollo, which was trailing by 209 kilometers, at a rate of approximately 50 kilometers per orbit. At the time of Soyuz deorbit, 141:50:30, the Apollo trailed the Soyuz by about 833 kilometers. The Soyuz landed at 142:30:54 in Kazakhstan. The approximate landing location was latitude 50 degrees 15 minutes north and longitude 67 degrees 36 minutes east.

An Apollo trim maneuver was executed at 146 hours 38 minutes to shift the end-of-mission ascending node 19.2 milliradians to the east to assure that the entry ground track would be near the nominal recovery area. The ground track had shifted to the west during the mission because of the initial Soyuz plane position and slightly higher than nominal orbits. The retrograde trim maneuver was 2.2 meters per second and placed the Apollo in a 219.1 by 211.3 kilometer orbit.

The docking module was jettisoned at 199 hours 27 minutes. Thirty-five minutes later, the command and service module performed a 9.5 meter-per-second service propulsion system maneuver, placing it in a 227.2 by 218.2 kilometer orbit. This was the first of two service propulsion system maneuvers to set up a 300-kilometer range between the docking module and the command and service module for the doppler tracking experiment. The second service propulsion system maneuver was executed at 204:13:42 and this retrograde maneuver of 8.1 meters per second placed the command and service module in a 219.6 by 206.5 kilometer orbit. The docking module remained in orbit for about 9 days 21 hours after having been jettisoned.

The Apollo deorbit maneuver of 57.9 meters per second occurred on revolution 138 at 224:17:47 and entry interface (122 000 m) was 21 minutes 13 seconds later. The command module landed at 21:18:24 G.m.t. on July 24 in the Pacific Ocean at 22 degrees 0 minutes 36 seconds north and 163 degrees 0 minutes 54 seconds west, which was only 1.3 kilometers from the target point.

The times of the major events are given in table 3-II. The major maneuvers are summarized in table 3-III. Definitions of events and trajectory terminology are given in tables 3-IV and 3-V.

TABLE 3-II.- SEQUENCE OF EVENTS

<u>Event</u>	<u>Time, g.e.t., Hr:min:sec</u>	<u>Time, G.m.t., Day:hr:min:sec</u>
Soyuz lift-off		196:12:20:00
Soyuz maneuver 1	5:31:40	196:17:51:40
Apollo lift-off	7:30:01	196:19:50:01 ^a
Earth orbit insertion	7:39:56	196:19:59:56
Command and service module/S-IVB separation	8:44:01	196:21:04:01
Command and service module/docking module docking	9:08:00	196:21:28:00
Docking module extraction	10:04:00	196:22:24:00
Apollo evasive maneuver	10:04:01	196:22:24:01
Apollo circularization maneuver	11:11:02	196:23:31:02
S-IVB deorbit	12:45:45	197:01:05:45
First phasing maneuver	13:08:30	197:01:28:30
Soyuz circularization	24:23:35	197:12:43:35
Phasing correction maneuver	31:58:00	197:20:18:00
Second phasing maneuver	48:31:00	198:12:51:00
Corrective combination	49:15:05	198:13:35:05
Coelliptic maneuver	49:52:05	198:14:12:05
Terminal phase initiation	50:56:42	198:15:16:42
Braking	51:26:00	198:15:46:00
Apollo/Soyuz docking no. 1	51:49:09	198:16:09:09
Apollo/Soyuz undocking no. 1	95:43:20	200:12:03:20
Apollo/Soyuz docking no. 2	96:13:39	200:12:33:39
Apollo/Soyuz final undocking	99:06:12	200:15:26:12
Separation from Soyuz maneuver	102:22:27	200:18:42:27
Soyuz test maneuver	118:21:00	201:10:41:00
Soyuz deorbit maneuver	141:50:30	202:10:10:30
Soyuz landing	142:30:54	202:10:50:54
Apollo trim maneuver	146:38:00	202:14:58:00
Docking module jettison	199:27:00	204:19:47:00
Apollo/docking module separation maneuver	200:02:00	204:20:22:00
Apollo stable orbit maneuver	204:13:42	205:00:33:42
Apollo deorbit maneuver	224:17:47	205:20:37:47
Command/service module separation	224:24:09	205:20:44:09
Entry interface	224:39:00	205:20:59:00
Begin blackout	224:45:16	205:21:05:16
End blackout	224:50:21	205:21:10:21
Drogue parachute deployment	224:54:37	205:21:14:37
Main parachute deployment	224:55:17	205:21:15:17
Landing	224:58:24	205:21:18:24

^aRange zero (the integral second prior to lift-off) occurred at 196:19:50:00 G.m.t.

TABLE 3-III.- MANEUVER SUMMARY

Maneuver	Time, g.e.t., hr:min:sec	System	Firing time, sec	Velocity change, m/sec	^a Resultant orbit, km	
					Apogee	Perigee
Soyuz maneuver 1	5:31:40	--	^b 05.7	^b 3.1	^b 226.3	^b 192.1
Apollo evasive maneuver	10:04:01	RCS	08.7	0.9	167.6	150.0
Apollo circularization	11:11:02	SPS	00.8	5.5	167.4	167.2
First phasing maneuver	13:08:30	SPS	03.2	20.5	232.8	168.7
Soyuz circularization	24:23:35	--	^b 21.0	^b 11.6	^b 223.0	^b 222.2
Apollo phase correction maneuver	31:58:00	RCS	23.7	2.7	226.1	168.2
Second phasing maneuver	48:31:00	SPS	01.1	7.7	196.7	165.6
Corrective combination maneuver	49:15:05	SPS	01.5	10.3	203.3	189.8
Coelliptic maneuver	49:52:05	SPS	00.9	6.9	205.4	205.0
Terminal phase initiation	50:56:42	SPS	00.9	6.6	224.8	205.2
Midcourse maneuver 1	51:08:42	RCS	01.6	0.2	225.2	205.0
Midcourse maneuver 2	51:20:42	RCS	03.7	0.4	225.6	205.0
Braking	51:26:00	RCS	176.2	20.1	221.7	221.3
Apollo/Soyuz first docking	51:49:09	RCS	10.0	0.9	223.5	222.8
Apollo/Soyuz first undocking	95:43:20	(c)	(c)	^b 0.1	^b 220.9	^b 219.1
Apollo/Soyuz second docking	96:13:39	RCS	5.0	0.5	219.8	219.1
Apollo/Soyuz final undocking	99:06:12	(c)	(c)	^b 0.4	^b 219.8	^b 217.2
Separation from Soyuz	102:22:27	RCS	05.8	0.6	222.1	220.9
Soyuz test maneuver	118:21:00	--	^b 02.7	^b 1.5	^b 215.8	^b 213.5
Soyuz deorbit maneuver	141:50:30	--	^b 194.9	^b 120.0	^b 214.1	--
Apollo trim maneuver	146:38:00	RCS	36.1	2.2	219.1	211.3
Docking module jettison	199:27:00	--	00.0	0.1	219.6	206.3
Apollo/docking module separation	200:02:00	SPS	01.0	9.5	227.2	218.2
Apollo stable orbit maneuver	204:13:42	SPS	00.9	8.1	219.6	206.5
Apollo deorbit maneuver	224:17:47	SPS	06.6	57.9	205.2	--

^aAbove mean equatorial radius.^bSoyuz parameters.^cData not available - small Soyuz thrusters.ORIGINAL PAGE IS
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TABLE 3-IV.- DEFINITIONS OF EVENTS

<u>Event</u>	<u>Definition</u>
Soyuz maneuver 1	Maneuver to provide nominal phasing and altitude at Soyuz circularization.
Apollo lift-off	Time of instrumentation umbilical disconnect as indicated by launch vehicle telemetry.
Command and service module/S-IVB separation	Physical separation of the command and service module from the S-IVB.
Earth orbit insertion	S-IVB engine cutoff plus 10 seconds.
Docking module extraction	Physical separation of the command and service module and docking module from the S-IVB.
S-IVB deorbit	Beginning of deorbit sequence at time base five initiation.
Apollo/Soyuz docking	Capture of the two vehicles as indicated by telemetry.
Apollo/Soyuz undocking	Physical separation of the two vehicles.
Separation from Soyuz	Final Apollo maneuver during the ultraviolet absorption experiment to set up the 1000-meter data-take and provide adequate separation distance between the Apollo and Soyuz.
Soyuz test maneuver	Maneuver to test the Soyuz propulsion system.
Apollo trim maneuver	Maneuver to adjust the orbital period for end-of-mission groundtrack control.
Docking module jettison	Physical separation of the docking module from the command and service module.
Apollo/docking module separation	Maneuver to provide the proper separation rate between the command and service module and docking module for the Apollo stable orbit maneuver.
Apollo stable orbit maneuver	Maneuver to maintain proper range between the command and service module and docking module for the Doppler experiment.
Command module landing	Time the spacecraft was observed to touch the water.

TABLE 3-V.- DEFINITIONS OF TRAJECTORY TERMINOLOGY

<u>Term</u>	<u>Definition</u>
Altitude	Altitude above an equatorial radius of 6378.2 kilometers.
Apogee	Apogee altitude above an equatorial radius of 6378.2 kilometers.
Eccentricity	Ratio between the center and focus of an ellipse to its semimajor axis.
Firing time	Main engine firing duration. Does not include ullage times for service propulsion system maneuvers.
Inclination	Angle between the orbital plane and the true of-date equatorial plane.
Inertial azimuth	Angle measured from north to the inertial velocity vector projected into the local horizontal plane.
Inertial flight path angle	Angle between the inertial velocity vector and the local horizontal plane. Positive if the velocity vector is above the horizontal plane.
Inertial velocity	Total inertial velocity of the spacecraft.
Perigee	Perigee altitude above an equatorial radius of 6378.2 kilometers.
Period	Time required for spacecraft to complete one orbit.
Phase angle	Angle from the Apollo position vector to the Soyuz position vector measured in the direction of motion.
Target node	Angle measured from launch pad meridian to descending node of the target orbit plane at the time of guidance reference release.
Velocity change	Differential inertial velocity imparted to the spacecraft due to thrusting.

4.0 SPACECRAFT SYSTEMS PERFORMANCE

This section contains the performance evaluation of the docking module and the command and service module.

4.1 DOCKING MODULE

4.1.1 Compatible Docking Systems

The docking systems used on the docking module and the Soyuz spacecraft performed properly. The Apollo spacecraft was the maneuvering vehicle for the two dockings.

4.1.1.1 Initial docking. The initial docking was successfully accomplished with the Apollo docking system in the active configuration (the guide ring extended). Telemetry data and motion picture photography indicate that the initial contact conditions between the two spacecraft were normal. The closing velocity of the Apollo spacecraft at contact was approximately 0.25 meter per second and the misalignment between the X axes of the two spacecraft was approximately 82 millimeters. The spacecraft appeared to have essentially no angular misalignment. The docking system attenuated the impact normally. Both crews received their respective capture signals when capture was achieved. The Apollo docking system guide ring retraction was manually initiated and the retraction was automatically terminated 35 seconds later. The closing of the structural ring latches was also manually initiated; the latches were completely closed in 8 seconds and the drive mechanism was automatically turned off. Subsequently, during the docking module checkout, the Docking Module Pilot verified that all eight structural ring latches were preloaded by checking the load cell reading for each latch. The pressure integrity of the interface seals was verified by the Soyuz crew and by subsequent tunnel pressurization.

4.1.1.2 Initial undocking.- After being docked for 43 hours and 54 minutes, the undocking was performed normally by using the Apollo docking system. Undocking was manually initiated with the opening of the structural ring latches. After 8 seconds, the latches were open and the Apollo crew released the capture latches. The two spacecraft separated with no apparent angular velocities between them.

4.1.1.3 Test docking.- The second docking was achieved utilizing the Soyuz docking system in the active configuration. The closing velocity of the Apollo spacecraft at contact was from 0.15 to 0.18 meter per second and the lateral misalignment between the spacecraft was from 70

to 100 millimeters. The spacecraft oscillated after capture (see sec. 4.2.8), but the Soyuz docking system aligned the spacecraft. The Soyuz guide ring retraction was automatically initiated and the structural ring latches automatically started closing just as the guide ring completed retraction. The Soyuz crew received a proper indication when the structural ring latches reached the fully closed position. The Soyuz crew then verified the integrity of the docking interface seals.

4.1.1.4 Test undocking.— The test undocking was executed normally. First, the Soyuz guide ring was extended just far enough to remove the preload on the Soyuz capture latches. The capture latches were then automatically released and the structural ring latches were automatically activated to open. The spacecraft separated properly and no angular velocities were discernable.

4.1.2 Structure

The docking module structure performed satisfactorily.

4.1.3 Thermal Control

The thermal protection was satisfactory. Docking module temperatures were monitored by two temperature sensors - one on the cylindrical wall and the other on the bulkhead near hatch 3. The bulkhead temperature varied between 282° and 295° K during the undocked phases and between 283° and 293° K during the docked operations. The wall temperature varied between 289° and 299° K during the undocked phases and between 291° and 300° K during docked operations. The average internal wall temperature was maintained within allowable limits of 286° to 300° K throughout the mission. It was not necessary to utilize the three electrical heaters.

The docking system was monitored by three temperature measurements; one each on attenuators 1, 3 and 5 (one sensor for each attenuator pair). The maximum temperatures were 319°, 321°, and 325° K, respectively, and the minimum temperatures were 265°, 271°, and 283° K, respectively, all of which remained within the design limit of 227° to 394° K. These sensors effectively measured the temperature effect of the space environment at these locations because they were covered only by a film of room-temperature vulcanizing material with a maximum thickness of 0.02 centimeter.

4.1.4 Communications

The docking module portion of the communications systems is discussed in conjunction with the command and service module communications system (sec. 4.2.5).

4.1.5 Instrumentation

The oxygen partial pressure sensors appeared to operate 0.07 to 0.13 newton/sq cm low following the last crew transfer. This effect was probably caused by condensation on the Teflon membrane resulting from high humidity associated with deactivation of the command module environmental control system evaporator. Otherwise, the 25 measurements in the docking module operated normally.

4.1.6 Displays and Controls

The docking module displays and controls performed normally during the mission.

4.1.7 Electrical Power Distribution

The docking module electrical power distribution system performed normally.

4.1.8 Environmental Control and Life Support

The docking module environmental control and life support system performance was normal and operations were such that the total and partial pressures were as predicted. No hardware problems were experienced with the exception of the low oxygen partial pressure readings discussed in section 4.1.5.

Typically, during transfers, the docking module pressure was raised with nitrogen from 3.32 to 6.56 newtons/sq cm. Per Soyuz crewmen request, nitrogen was added to decrease the percentage of oxygen in the Soyuz, raising the docking module/Soyuz combined volume total pressure by 0.27, 0.40, and 0.13 newton/sq cm during transfers 1, 2, and 4, respectively (a total of 1.3 kilograms of nitrogen). The oxygen purge before docking module depressurization was to within 0.01 newton/sq cm of the desired 4.27 newtons/sq cm oxygen partial pressure. All transfer operations, both manual and automatic, were accomplished such that no corrective actions were required. During the transfer operations, 8.34 kilograms of oxygen were used versus 7.71 kilograms predicted, and 5.96 kilograms of nitrogen were used versus 6.29 kilograms predicted.

At 52 hours 11 minutes, after reopening hatch 1 which had been closed for docking, the crew reported a strong odor in the docking module, but could not find the source. The crew performed normal procedures to verify the acceptability of the docking module atmosphere and subsequently

initiated mixing with the command module atmosphere in order to allow removal of the odor by the command module environmental control system. No subsequent reports of odor were made by the crew. Postflight odor testing of suspected materials and chemicals by the Apollo crew identified methyl ethyl ketone (MEK) and diluted methyl isobutyl ketone (MIBK) as the reported odor. MEK was used in the docking module as a cleaner, in adhesives, and in fabrication of furnace insulation. MIBK was used as a solvent for paint.

4.2 COMMAND AND SERVICE MODULES

4.2.1 Structure

The command and service module structure performed satisfactorily.

4.2.2 Mechanisms

All command and service module mechanisms functioned properly except that the crew had difficulty removing the docking probe from the command module tunnel after docking with the docking module. When the crew attempted to remove the docking probe, a special tool for releasing the probe capture latches from the drogue could not be installed in the aft end of the probe. The Command Module Pilot observed that one of the pyrotechnic electrical connectors was obstructing the tool entrance hole. The interfering connector was for the expended pyrotechnic initiator that had been used for probe retraction. Procedures were sent to the crew for removing the connector cover plate and the interfering connector. The connector was disengaged, the cover replaced, the loose connector taped to the cover, and the probe successfully removed in accordance with the checklist. The docking probe problem is discussed further in section 14.1.1.

4.2.3 Thermal Control

The thermal performance of the command and service modules was monitored by temperature sensors located on propellant tanks and lines, communications equipment, and experiments. All command and service module temperatures were maintained within allowable limits throughout the flight.

4.2.4 Sequencing and Electrical Power Distribution

During the landing phase, an abnormal sequence of events involving manual and automatic control of the earth landing system caused reaction control system combustion products and, primarily, a mixture of nitrogen tetroxide vapors and air to be drawn into the cabin for about 30 seconds commencing at an altitude of about 5650 meters. This sequence is discussed in detail in section 14.1.2. Functioning of the automatic and manual landing systems and associated hardware was normal.

The electrical power distribution system performed normally. Power switching and sequential functions were also normal.

4.2.5 Communications and Ranging

Except for the conditions discussed in the following paragraphs, the performance of the communications and ranging system for all phases of the mission was satisfactory.

One of the most significant accomplishments of the mission was the first use of a communications satellite to relay manned spacecraft signals to the ground. The satellite used was Applications Technology Satellite 6 (ATS-6). The ATS-6 system, including the power amplifier and high gain antenna in the service module, performed very well. Continuous coverage for approximately 50 minutes during each revolution was provided except for an occasional loss of signal because of spacecraft maneuvering past the antenna directional limits.

During the joint activities, two sources of external interference were encountered. The VHF/FM receiver frequency of 121.75 MHz is in the aircraft frequency band and received intermittent transmissions from various sources. The spacecraft was configured to relay the FM signals onto the S-band; therefore, these were heard on the downlink. During each pass over the USSR, periodic clicks occurred in both the voice and video signals. The most probable cause was interference from a ground radar.

Except for two apparent television camera failures, both real-time and played back television performed very well. During the joint activities, there was a periodic loss of color synchronization from one television camera. Color was lost from another camera about 28 hours after the joint activities were concluded. (These anomalies are discussed in sec. 14.1.3 and 14.1.4.)

The crew reported that squeals in the headsets caused considerable distraction during entry. Tapes from the onboard recorder (data recorder-reproducer), the Apollo range instrumented aircraft, the recovery ship, and the mission evaluation room air-to-ground loop recorder revealed that, at approximately 27 430 meters altitude, an intermittent warbling tone was received by the command module on the S-band link which lasted for a total of 8 to 10 seconds. This tone was apparently being transmitted by the Apollo range instrumented aircraft. No squeals were detected following this time period until landing. The data indicate that the communications system performed normally during the entry phase of the mission.

The command module went to the stable II position after landing and was uprighted about 4 1/2 minutes later. After uprighting, a squeal was present in the headsets and was also detected on VHF by the recovery ship and the Apollo range instrumented aircraft. The crew also reported later that they had not received any recovery communications while they were on the water. These conditions were duplicated in postflight testing by shorting the key line to ground and shorting the headset line to the microphone line. The three lines pass through a tunnel connector. Thus, collection of salt water in the tunnel connector during the time the command module was in the stable II position could have caused the conditions.

4.2.6 Instrumentation

The command and service module instrumentation system operated normally with the following exceptions. The quantity measurement for service module cryogenic oxygen tank 1 failed (sec. 14.1.5), the temperature measurement for the service module reaction control system quad A helium tank failed (sec. 14.1.6), the pressure measurement for service module cryogenic oxygen tank 2 was intermittent for about 25 minutes (sec. 14.1.7), and the command module primary coolant loop evaporator inlet temperature measurement became erratic and later failed (sec. 14.1.8).

4.2.7 Displays and Controls

The displays and controls system performed normally during the mission except for one minor problem. At approximately 154 hours, the crew reported a degradation in the caution and warning tone volume as heard in the headsets but there was no apparent degradation in the speaker box tone level. This problem is discussed further in section 14.1.9.

4.2.8 Guidance, Navigation and Control Systems

The guidance and navigation, entry monitor, and stabilization and control systems performed normally throughout the mission. Only two minor anomalies occurred. One of these was an intermittent failure in the guidance and navigation system alarm sensing circuitry. The other anomaly was that the orbital rate display gave intermittent angular read-outs when used with flight director attitude indicator 1.

The guidance and navigation system anomaly manifested itself as an inertial coupling data unit (ICDU) failure indication which resulted in an inertial subsystem warning and activated the master alarm. The analysis of available system data revealed that these spurious ICDU FAIL alarms were false and that the system performance was normal. In order to eliminate the nuisance which was created each time the alarm circuitry falsely triggered the master alarm, a bit was set in the system computer which prevented triggering for an ICDU FAIL indication; however, this bit which was set would not have prevented other system failures from triggering the master alarm had they occurred. The probable cause of the anomalous condition is discussed in section 14.1.10.

When the orbital rate display anomaly occurred, the crew switched the unit over to flight director attitude indicator 2 and obtained proper performance. The crew was advised not to take the time to determine the source of the problem because of a busy schedule and the very limited use of the display required by the mission. The crew used the orbital rate display only with flight director attitude indicator 2 for the remainder of the mission. Possible failure mechanisms are discussed in section 14.1.11.

Satisfactory tracking of the sunlit Soyuz was accomplished prior to the corrective combination maneuver during rendezvous and the required navigation update was accomplished. The Soyuz beacon was observed through the Apollo sextant shortly after the maneuver and VHF ranging data were taken beginning at a range of about 223 kilometers. The Apollo began braking at a range of approximately 1.6 kilometers.

For two orbits prior to docking, the Soyuz maintained automatic orbital orientation using the infrared sensor, the rate gyro unit in the instrument module, and the 1-kilogram thrusters. In preparation for docking, the Soyuz began automatic inertial attitude hold using the rate gyro unit in the instrument module with rate integration and the 1-kilogram thrusters. An automatic roll maneuver of 1.05 radians was performed to provide proper alignment for Apollo docking. The Soyuz primary docking target was deployed and provided satisfactory Apollo crew control cues.

The first docking can be analyzed only from a qualitative standpoint since high-bit-rate data were not obtained during the initial docking sequence. The docking was achieved using the stabilization and control system and the planned four-thruster plus-X translation to provide the latching thrust. The standby control mode was with the command-and-service-module-alone digital autopilot in the free mode. There were only slight angular disturbances at contact, as exhibited by television, and capture occurred at 51:49:09. At capture, the control switch was switched to the command module computer position, disabling the stabilization and control system. The four reaction control system thrusters which fire along the Apollo minus-X axis were disabled, and the automatic mode of the digital autopilot was selected, at which time active attitude stabilization and control were provided.

Retraction of the Apollo docking system was completed at 51:51:51. About 1 1/2 minutes later, Apollo control was transferred to the docked digital autopilot which was configured prior to launch for the force jet control mode (Z and/or Y translation thrusters used for yaw and pitch attitude control), 87.3-milliradian deadband, and attitude maneuver rate control of 8.7 milliradians per second. For the duration of the docked period, the docked digital autopilot maintained attitude control within the confines of the control phase planes. The Soyuz control system, which was automatically deactivated at capture, remained off throughout the docked operations.

Undocking (loss of capture) occurred at 95:43:20. Three seconds later, the Soyuz control system was activated in the manual orientation mode using the rate gyro unit in the instrument module with rate integration and the 1-kilogram thrusters. Automatic inertial orientation was then maintained throughout the artificial solar eclipse experiment (MA-148).

The stabilization and control system maintained command and service module/docking module attitude within the 3.5-milliradian deadband limits during the subsequent four-thruster minus-X translation which lasted for 3 seconds. A second minus-X translation of 4 seconds was applied using the translation hand controller. This separation maneuver provided satisfactory shading of the Soyuz vehicle for the solar eclipse experiment.

In preparation for the test docking, the automatic yaw (0.53-radian) and pitch (1.17-radian) maneuvers were initiated by the Soyuz. Automatic inertial hold was then maintained. The test docking activity was accomplished using a reference attitude which required the Apollo pilot to observe Soyuz with a sunlit earth background during the maneuver. This apparently caused poor docking target visibility and washout of the crewman optical alignment sight reticle. Prior to docking, the Soyuz attitude was verified to be within the specified limits of ± 140 milliradians in

each axis, based on the Apollo reference attitude at docking. After contact, a four-thruster plus-X translation command was applied for a period of approximately 5 seconds. A two-thruster minus-Y translation command was subsequently applied for a period of 3 1/2 seconds. Both commands terminated simultaneously. The stabilization and control system was then disabled and the command module computer-free mode was selected. The automatic reaction control system select switches were then activated to disable the Apollo minus-X thrusters. The command module computer-automatic mode was commanded after the docked spacecraft attitude had drifted in excess of 0.88 radian from the reference docking attitude. This orientation resulted in a gimbal lock alarm and loss of lockup of the high-gain antenna. Unlike the first docking, the digital autopilot had to contend with appreciable spacecraft rates at the time of its activation (6 and 7 milliradians/sec in the pitch and yaw axes, respectively) and the command module computer-automatic mode was essentially ineffective in maintaining attitude-hold. Because of the increasing attitude errors, the crew used the rotational hand controller to manually insert commands in the pitch and yaw axes. These commands created only slight alterations in the thruster firing and the crew placed the hand controller to the maximum negative pitch deflection. This action closed the emergency direct solenoids, bypassing the digital autopilot. The command resulted in a minus pitch acceleration. A similar but opposite command was initiated 10 seconds later in order to null the negative pitch rate. After the completion of retraction, the docked digital autopilot was initiated and it provided satisfactory control of the docked configuration.

The Soyuz initiated the second undocking at 99:03:00. Following release of the Apollo by the Soyuz docking mechanism at 99:06:12, an automatic program commanded 8 seconds of firing of the Soyuz 10-kilogram thrusters along the plus X_{C4} axis to achieve separation. Then the Soyuz established orbital orientation using the infrared sensor, the rate gyro unit in the instrument module, and the 1-kilogram thrusters in preparation for the ultraviolet absorption experiment (MA-059). Manual yaw commands were used to maintain visual contact with the Apollo vehicle.

At the time of undocking, Apollo attitude control was transferred to the stabilization and control system. The Apollo began stationkeeping on the Soyuz after separating approximately 18 meters and the digital autopilot was reestablished with the minimum-impulse translation-hand-controller erasable memory program in effect. The Apollo began translation thrusting for the ultraviolet absorption experiment 150-meter out-of-plane data-take at 99:18:31. Prior to the second out-of-plane maneuver, the Soyuz performed a manual yaw maneuver to point the reflector along the Soyuz minus X_{C4} axis toward the Apollo spacecraft during the data-take period. This maneuver was performed nominally and the Soyuz was returned to the nominal attitude prior to the inplane separation maneuver. At 100:31:20, the Apollo began translation thrusting for the 500-meter out-of-plane data-take. About 1 hour and 50 minutes later, the

Apollo performed a 5.8-second four-thruster reaction control system velocity change maneuver to provide an inplane separation from the Soyuz and permit the 1000-meter ultraviolet absorption data-take. The digital autopilot, utilizing the specially designed software for minimum-impulse translation and for pitch and yaw attitude control, provided acceptable manual maneuvering control for accomplishment of the planned ultraviolet absorption experiment measurements.

4.2.9 Propulsion

4.2.9.1 Service propulsion system.- The service propulsion system was used nine times for a total duration of 17 seconds. The first firing was for the Apollo circularization maneuver; the next five uses of the system were maneuvers for the Apollo Soyuz rendezvous. After separation from Soyuz, two maneuvers were performed to position the command and service module behind the docking module for the doppler tracking experiment (MA-089). The final use of the system was for deorbit of the command and service module. Propellant consumption data are given in section 4.3.1. System performance was normal.

Soyuz 16-mm photography of the Apollo spacecraft, taken during the Apollo pitch-around maneuver at approximately 102 hours (following the second undocking), showed one large and several small blisters on the exterior surface of the columbium portion (forward section) of the columbium/titanium nozzle extension (fig. 4-1). The columbium alloy was coated with an oxidation-protective fused-on aluminide coating which, in turn, was covered with a vinyl-type plastic film for handling protection. Rather than removing the protective plastic film before launch and risking last minute damage, the film has been left on. On lunar missions, the film was quickly burned off during the long (approximately 400 seconds) lunar orbit insertion maneuver. Ground test data have shown that the temperature of the columbium portion of the nozzle extension stabilized at approximately 1033° to 1422° K in about 60 seconds. With the short maneuvers of 0.8, 3.2, 1.1, 1.5, 0.9, and 0.9 seconds prior to pitch-around on the Apollo Soyuz mission, the temperature for the longest (3.2 seconds) would cause softening and outgassing (dehalogenation) of the plastic film, resulting in the blisters observed in the photographs. Normal performance of the engine was not affected by the blistering.

4.2.9.2 Service module reaction control system.- The performance of the service module reaction control system was normal throughout all phases of the mission. The total propellant consumed was 896 kilograms (sec. 4.3.2) versus a predicted quantity of 738 kilograms. The propellant usage was greater than planned for three reasons: firing of the roll engines to purge a bubble introduced in the propellant storage module manifold during an off-nominal activation prior to launch, higher than expected usage associated with the second docking, and the performance of

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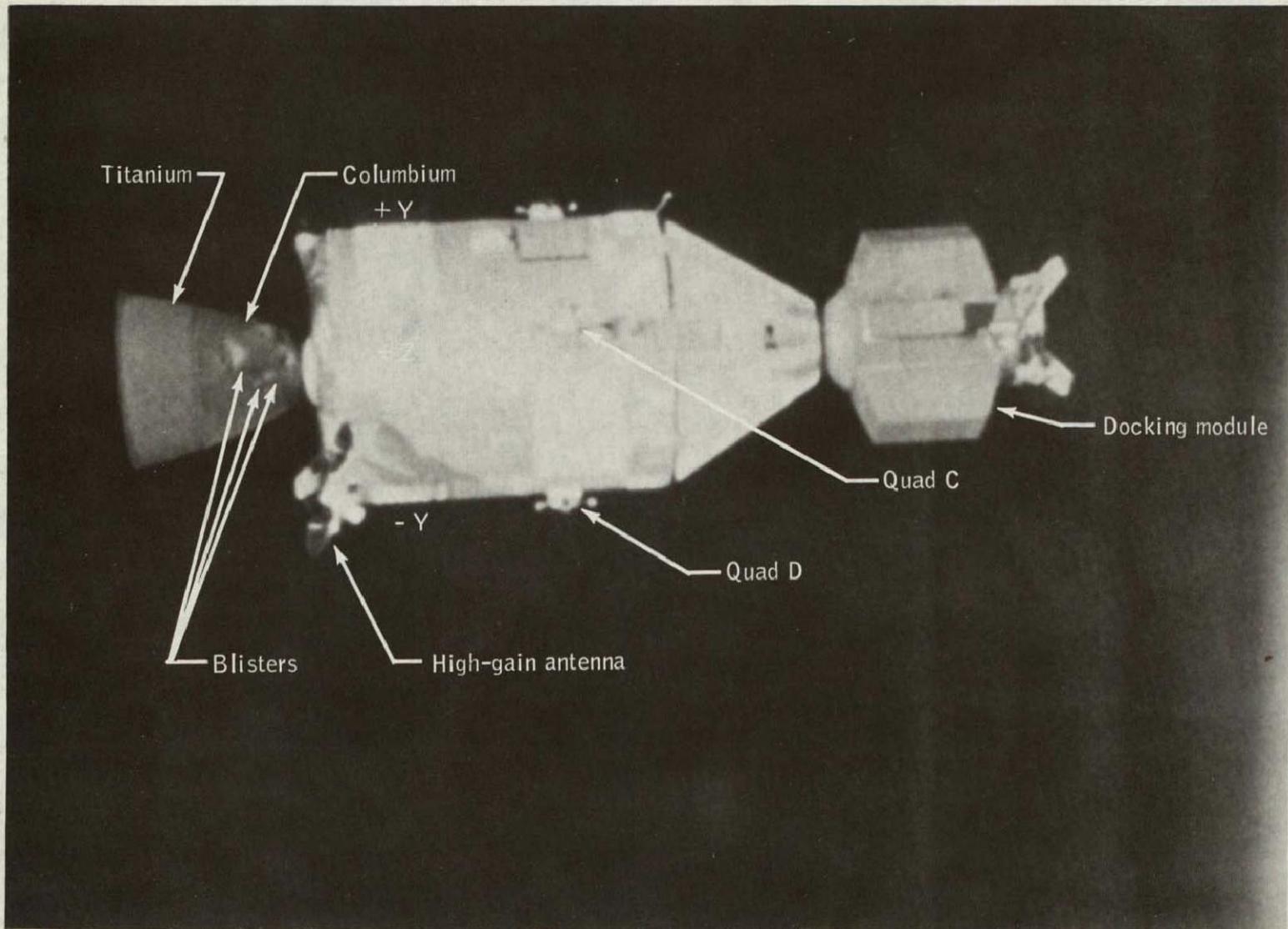


Figure 4-1.- Blisters on service propulsion system nozzle extension.

maneuvers required for the ultraviolet absorption experiment (MA-059). Pressurization and propellant feed systems operated properly and system temperatures were maintained within expected operating ranges throughout the mission. As discussed under Instrumentation (sec. 4.2.6), the quad A helium tank temperature was lost.

4.2.9.3 Command module reaction control system.- The performance of the command module reaction control system was normal throughout system activation and entry. Propellant consumption data are given in section 4.3.2.

4.2.10 Fuel Cells and Cryogenic Storage

The three Apollo fuel cells performed normally throughout the mission. Prior to lift-off, the fuel cells were configured with fuel cells 1 and 2 powering bus A and fuel cell 3 powering bus B. This configuration was maintained throughout the mission. From lift-off to command module/service module separation, the fuel cells provided 18 907 ampere-hours of energy at an average bus voltage of 28.5 volts.

The fuel cells consumed 22.9 kilograms of hydrogen during the mission, and 0.91 kilogram remained in each of the two tanks at command module/service module separation (sec. 4.3.3). The hydrogen storage system performance was normal.

Quantity remaining in oxygen tank 1 was determined after the failure of the tank quantity measurement (sec. 14.1.5) by calculations based on pressure and temperature data. The fuel cells consumed 176.8 kilograms of oxygen while 37.4 kilograms of oxygen were provided to the environmental control system, leaving 35.8 kilograms in each tank at command module/service module separation (sec. 4.3.3). The oxygen storage system performance was as predicted throughout the mission.

4.2.11 Batteries

The performance of the entry and postlanding batteries and the pyrotechnic batteries was normal.

4.2.12 Environmental Control

The environmental control system performed satisfactorily throughout the mission, providing an acceptable environment for the crew and adequate thermal control of the spacecraft equipment. System performance during the joint operations was normal.

maintained the pressure for about 6 minutes. The crew later indicated that the hatch 2 equalization valve (and subsequently hatch 2) were opened before the docking module vent valves were finally closed. The pressure was automatically raised to 3.33 newtons/sq cm by the command module pressure regulators.

In order to insure that the cabin oxygen partial pressure would be above the low-partial-pressure caution and warning system activation point (nominally 1.96 newtons/sq cm) during the sleep period, the crew initiated a procedure at 84 hours 20 minutes to raise the cabin oxygen partial pressure above 2.20 newtons/sq cm. This procedure consisted of opening the docking module low pressure relief valve to vent the cabin oxygen/nitrogen mixture while adding oxygen through the docking module oxygen purge valve. The procedure was completed at approximately 86 hours with the cabin oxygen partial pressure at 2.91 newtons/sq cm. The purge procedure resulted in decreasing docking module oxygen tank supplies from 50 percent to approximately 12 percent, corresponding to an oxygen use of 9.3 kilograms.

Prior to docking module jettison, the crew used the same procedure with the remaining docking module oxygen to raise the cabin oxygen partial pressure so that nitrogen boiloff from the electrophoresis experiment would not reduce the oxygen partial pressure to an unacceptable level before the end of the mission. At the completion of the procedure, the oxygen partial pressure was 3.07 newtons/sq cm and the total pressure was 3.73 newtons/sq cm.

The crew reported during postflight debriefing that when the post-landing ventilation system was activated after the command module was uprighted, the fan produced only a negligible amount of flow. Postflight testing showed that the two postlanding valves opened but the fan failed to operate. Electrical continuity tests did not reveal any problem. No further testing was performed because this fan will not be used on future missions.

4.3 COMMAND AND SERVICE MODULE CONSUMABLES

The command and service module consumable usage was maintained well within the redline limits. Specific system usage is discussed in the following paragraphs.

4.3.1 Service Propulsion System

The service propulsion system propellant and helium loadings and consumption values are listed in the following table. The loadings were calculated from gaging system readings and measured densities prior to lift-off.

Condition	Propellant, kilograms		
	Fuel	Oxidizer	Total
Loaded	483	784	1267
Consumed	209	339	548
Remaining	274	445	719

Condition	Helium, kilograms	
	Storage bottles	Propellant tanks
Loaded	19.2	13.9
Remaining	17.2	15.9

4.3.2 Reaction Control System Propellant

4.3.2.1 Service module.-- The propellant loading and utilization data for the service module reaction control system were as shown in the following table. Consumption was calculated from telemetered helium tank pressure histories and was based on pressure, volume, and temperature relationships.

Condition	Propellant, kilograms		
	Fuel	Oxidizer	Total
Loaded			
Quad A	50	102	152
Quad B	50	102	152
Quad C	50	102	152
Quad D	50	102	152
Propellant storage module	228	455	683
Total	428	863	1291
^a Usable loaded			1190
Consumed			896
Remaining at command module/service module separation			395

^aUsable propellant is the amount loaded minus the amount trapped with corrections made for gaging system errors.

4.3.2.2 Command module.— The loading and utilization of command module reaction control system propellant was as follows:

Condition	Propellant, kilograms		
	Fuel	Oxidizer	Total
Loaded			
System 1	20	41	61
System 2	20	41	61
Total	40	82	122
^a Consumed	9	18	27

^aBased on helium tank pressures and temperatures prior to landing and confirmed during postflight propellant deservicing.

4.3.3 Cryogenic Storage System

The total cryogenic hydrogen and oxygen quantities available at lift-off and consumed are as follows. Consumption values were based on quantity data transmitted by telemetry.

Condition	Hydrogen, kilograms	Oxygen, kilograms
Available at lift-off		
Tank 1	12.38	144.2
Tank 2	12.34	144.2
Total	24.72	288.4
Consumed		
Tank 1	11.47	108.4
Tank 2	11.43	108.4
Total	22.90	216.8
Remaining at command module/service module separation		
Tank 1	0.91	35.8
Tank 2	0.91	35.8
Total	1.82	71.6

4.3.4 Water

The water quantities loaded, produced, and expelled during the mission are shown in the following table.

Condition	Quantity, kilograms
Lift-off quantities	
Potable tank	8
Waste tank	27
Produced in flight	
Fuel cells	197
^a Lithium hydroxide canisters	10
^a Metabolic oxidation	10
Total (lift-off and produced)	253
Urine dumps	46
Waste water dumps	39
^b Evaporator usage	139
Remaining at command module landing	
Potable tank	9
Waste tank	15
Total utilized and remaining	253

^aCalculated prior to mission.

^bEstimated.

5.0 CREW STATION

This section contains discussions of crew provisions, crew equipment, and crew height measurements that were made to help define man-machine interfaces for the Space Shuttle and future designs.

5.1 CREW PROVISIONS

The crew provisions for this mission were similar to those used in Apollo and Skylab missions. Differences are noted in appendix A.

All photographic systems and ancillary equipment performed properly; however, some poor exposures and out-of-focus imagery were obtained. From low earth orbit, a viewing system with magnification power higher than 10 is difficult to use. The 10-power size is proper for altitudes between 220 to 280 kilometers. Under certain conditions of scene brightness and viewing geometry, reflex viewing with long lenses was difficult for earth-looking photography.

The emergency oxygen masks were used after landing when the Commander was able to reach the location in which they were stowed. The crewmen donned the masks to protect themselves against the contaminated cabin atmosphere as described in section 14.1.2.

5.2 SUITS

Suit performance was normal. Helmets and gloves were doffed after orbital insertion, and the suits were doffed and stowed at 11 hours 20 minutes, after extraction of the docking module. The suits remained stowed until 198 hours when they were again donned in preparation for docking module jettison. After docking module jettison, the suits were doffed and stowed for the remainder of the mission.

5.3 CREW HEIGHT MEASUREMENTS

From a human engineering viewpoint, the gross effects of zero gravity on man's anthropometry are known. For example, in space beyond the earth gravitational forces, a crewman "grows taller". Also, the earth-gravity erect body posture is altered at several skeletal joints, namely,

the neck, shoulder, elbow, hip, knee, and ankle. These phenomena are known to have occurred after an extended (i.e., 3-week) orbital flight period, as measured during the Skylab Program.

Quantifying these effects more specifically is important in order to optimize the interfaces between the crewman and his work stations and/or provisions. The Apollo Soyuz mission provided an opportunity to quantify easily and economically the effects of zero gravity on crewman height and determine the change as a function of time over a mission length comparable to the initial Space Shuttle operations.

Originally approved data requirements included preflight, inflight, and postflight crew measurements of maximum stature, sitting height, and eye height. One hundred percent of the preflight data were obtained during the routine crew physicals conducted 45, 30 and 15 days prior to launch. However, only 62 percent of the approved inflight measurements were obtained, primarily a result of scheduling problems during the mission. A minimal amount of postflight measurements were made because of crew health concerns.

The anticipated increase in stature due to spinal column expansion and decrease in eye height due to the "relaxation" in body posture were observed in cursory data analyses. The final results will be applied to improve the engineering of man-machine interfaces in the Space Shuttle and future spacecraft designs.

6.0 EXPERIMENTS

This section contains the performance results of the experiments conducted in conjunction with the Apollo-Soyuz mission. Descriptions of the experiment equipment are given in appendix A.

6.1 SPACE SCIENCES

6.1.1 Stratospheric Aerosol Measurement Experiment

The stratospheric aerosol measurement experiment (MA-007) was conducted to determine the feasibility of using a simple photometer (or set of photometers) in an earth satellite as a long-term monitor of atmospheric aerosols. Measurements of solar intensity were made with a window-mounted photometer during two sunset and two sunrise data-takes in accordance with the flight plan. Infrared photographs of the sun were obtained with the 70-mm reflex camera simultaneously with the photometer data-takes on the second sunset and both sunrise passes to record solar disc shape changes. (On the first sunset pass, the camera did not operate because of an improperly installed lens.)

Ground truth measurements using a balloon-borne instrument having capability similar to the one in the spacecraft, and a ground-based LIDAR (light detection and ranging) system were made at the Richards-Gebaur Air Force Base in support of the second sunset pass.

Results indicate that the data obtained are sufficient to determine, within the spectral resolution of the photometer, aerosol concentration versus altitude, and that the technique investigated is feasible for satellite applications.

6.1.2 Soft X-Ray Experiment

The soft X-ray experiment (MA-048) was conducted to observe low energy X-rays from celestial and earth atmospheric emissions. Specifically, the objectives were (1) to map celestial X-ray emission in the 0.1 to 1.0 keV energy range, (2) to observe discrete X-ray sources in the 0.1 to 10.0 keV energy range, and (3) to record spatial dependence and time variability of atmospheric X-ray emission in the range of 0.1 to 10.0 keV. The minimum objectives for the discrete targets were accomplished; however, the minimum scanning objectives were not accomplished.

The X-ray instrument exhibited two distinct and unrelated anomalies during the mission. The first was a high-voltage discharge which occurred each time the instrument was operated, usually after an initial period of correct operation which lasted from 1 to 20 minutes. This problem caused the loss of approximately 80 percent of the expected data. The second was positioning of one of the calibration sources in the field of view of the detector. This problem did not cause a serious degradation of the data. (See sec. 14.2.1 for further discussion.)

The experiment was initially operated during the activation of the three experiments mounted in service module bay 1 on the second day of flight. The detector was purged as scheduled and the background calibration procedure was performed. After approximately 10 minutes of normal operation, high count rates were observed. At the time, these were thought to be due to the spacecraft passing through the South Atlantic anomaly, an area of known large particle fluxes. It later became apparent that these high count rates were caused by the aforementioned detector malfunction.

The soft X-ray experiment was operated successfully for a total of approximately 1 hour during the mission. During this time, data were obtained on several discrete sources including the small and large Magellanic clouds, Cygnus X-2, Hercules X-1, and Vela-X. The experiment also obtained good low-energy data on the cosmic source of ultraviolet radiation that was observed by the extreme ultraviolet telescope (MA-083) as described in section 6.1.4.

6.1.3 Ultraviolet Absorption Experiment

The ultraviolet absorption experiment (MA-059) was performed to measure the concentration of atomic oxygen and nitrogen in the atmosphere using optical absorption spectroscopy. Prior to data collection operations, the crew optical alignment sight was calibrated to the ultraviolet spectrometer optical axis so that pointing error could be determined. Following separation of the Apollo and Soyuz, ultraviolet light was transmitted from the docking module to a retroreflector on the Soyuz and returned to a detector on the docking module.

The calibration of the crew optical alignment sight and the star tracker on the star Vega were successfully carried out on day 2. The crew noted that the star tracker field-of-view was wider than expected. Subsequent tests on the backup star tracker indicated that this resulted from a modification made when the star tracker linearity was adjusted prior to flight. The condition did not interfere with the experiment operations, however.

Prime data were collected on day 5. During the 150-meter data-take period, no reflected signal was detected by the spectrometer. Initial assessment of the problem by ground personnel indicated that the side retroreflector may have had low ultraviolet reflectivity because of contamination, or that the star tracker may have been locked on a different light source. The aft retroreflector was used for the 500-meter data-take and good data were obtained. Discussion with the crew after the mission and telemetry data indicated that the side retroreflector was not contaminated by Apollo thruster firings and that the retroreflector was exactly on the Vega calibration with the error nulled on the pitch pointing meter. To account for parallax, the retroreflector should have been aligned 12 milliradians below the zero calibration. The star tracker was probably biased by a light from the Soyuz window or a surface reflection.

During the lamp warmup period for the 1000-meter data-take, the closed-door ultraviolet calibration signal was significantly decreased from prior runs. The decrease occurred during a time for which no data were recorded. It was not possible to determine the cause of the problem. The 1000-meter data-take was performed as planned using the third (top) retroreflector and the background ultraviolet data indicate no decrease in spectrometer sensitivity. No problem is anticipated in reducing the data.

The nitrogen lamp intensity monitor output was erratic during the 1000-meter data-take (sec. 14.2.3). The malfunction did not affect the lamp output as indicated by closed-door calibration data obtained at the end of the 1000-meter data collection period. The out-of-plane fluorescence scans have yielded airglow data from which additional atomic oxygen densities can be derived. Excellent data were obtained showing gas pile-up during the roll maneuver.

6.1.4 Extreme Ultraviolet Survey Experiment

The primary purpose of the extreme ultraviolet survey experiment (MA-083) was to search for discrete sources of ultraviolet radiation in the region of the spectrum between 50 and 1000 angstroms. Instrument operation was normal and the data obtained should be sufficient to accomplish all of the objectives for which the experiment was designed.

The experiment was initially operated on day 2 for 13 minutes. The first planned raster scan was cancelled; however, the second raster scan was accomplished as planned on day 5 for 12 minutes. Data operations began on day 6 and ran very close to the planned schedule through day 10. During this time, approximately 37 hours and 28 minutes of data were obtained.

The most significant achievement of the observing program was the detection of an intense extreme ultraviolet source on day 8. The source is located in Coma Berenices at the hour angle of 13 hours 13 minutes and 0.506 radian above the celestial equator. Positive detections have been made in the following wavelength bands: 170 to 620 angstroms, 114 to 160 angstroms, and 55 to 150 angstroms. This is the first known detection of a cosmic source of extreme ultraviolet radiation. The object appeared very strongly in the parylene and beryllium filters.

6.1.5 Helium Glow Experiment

The purpose of the helium glow experiment (MA-088) was to extend the investigation of the interstellar medium by observing two resonance lines of the helium component of the medium in the vicinity of the solar system.

The helium glow detector was activated for checkout on day 2. Data operations began on day 6 and ran very close to the planned schedule through day 10. During this time, approximately 29 hours and 36 minutes of data were obtained. There was some concern over possible excessive background noise on detector 3. Otherwise, the instrument performed well throughout the mission.

Data obtained on day 8 included supplementary data on the daytime and nighttime earth helium glow and a wide-angle raster scan. Good spectroscopic absorption cell data were obtained using the helium glow detector, although the extreme ultraviolet telescope (MA-083) was inadvertently left switched off with the consequence that the morphology of this complex region was not mapped.

The experiment is expected to provide significant new information concerning the local distribution of both neutral and ionized helium of terrestrial, solar, and interstellar origin. Specifically, a number of roll scans of the sky were obtained and these will allow preparation of a coarse all-sky map of the intensity and spectral characteristics of the local helium resonance radiation.

6.1.6 Doppler Tracking Experiment

The doppler tracking experiment (MA-089) was performed to map earth gravity field anomalies of magnitude $10^{-5}g$ and larger. The technique used was to separate the Apollo spacecraft and the docking module by a distance of approximately 300 kilometers and to measure and record velocity variations between the two satellites while they were under the influence of gravitational anomalies.

The experiment was operated in the warmup mode to achieve oscillator stabilization starting on day 6. After nearly 50 hours of warmup, the transmitter was inadvertently turned off for about 10 minutes but the subsequent data acquisition was not affected. At receiver turn-on, no tape motion was observed on either of the two (redundant) tape recorders and no telemetry was received. This condition was corrected by recycling the control switch. The operation of the system was normal except that the tape on one of the recorders adhered to the heads, preventing tape movement. (This malfunction is discussed in sec. 14.2.4.)

The quality of the data on the recorder which operated appeared to be very good and 108 frames of data were collected. The data printed out from the tape corresponds closely to the signatures found in premission simulations. The data have not been evaluated in detail as of publication of this report.

6.1.7 Geodynamics Experiment

The geodynamics experiment (MA-128) was performed to demonstrate the feasibility of detecting and recovering high-frequency components of the geopotential by use of a synchronous relay satellite tracking a low altitude spacecraft. Signals originating at the Spaceflight Tracking and Data Network (STDN) Madrid tracking station were relayed to the Apollo spacecraft via Applications Technology Satellite 6 (ATS-6). The prime area for collecting experiment data was the Indian Ocean depression, centered at approximately 3 degrees north latitude and 75 degrees east longitude.

The experiment data collection phase was satisfactory. Originally, 28 experiment revolutions were planned with the spacecraft passing through the center of the Indian Ocean depression on ten of these. The experiment relay doppler data and all supporting data were collected for all of the 28 planned experiment revolutions. Additional unscheduled data were acquired during parts of four revolutions in which the ground track passed through the center of the gravity anomaly, twelve revolutions in which the ground track passed near the center, and sixty-three revolutions in which the ground track passed through the outer periphery. The data sampling rate was less than desired from revolution 38 through revolution 74 because of ground network equipment difficulties.

Preliminary analysis of the data shows a possible correlation between the spacecraft traversing the center of the Indian Ocean depression and a sudden increase in the amplitude of range rate residuals. In addition, the possible detectability of other gravity anomalies, such as those in the Himalaya Mountains, were noted to correlate with a sharp increase in the amplitude of range rate residuals. These results must be verified, however.

6.1.8 Earth Observations and Photography Experiment

The earth observations and photography experiment (MA-136) was a continuation of operations to gather visual observational data from earth orbit pertaining to the fields of geology, hydrology, oceanography, meteorology and desert studies.

6.1.8.1 Mapping.- Of the 11 mapping passes that were scheduled, one had to be cancelled. In general, the photography was good to excellent. Table 6-I lists the mapping passes and indicates the passes on which data were acquired.

6.1.8.2 Visual observation sites.- About 20 percent of the 100 planned observations of 60 sites could not be performed because of cloud cover. (Cloud cover was predicted for about 30- to 40-percent of the observations.) Table 6-II lists the visual observation sites and shows the sites for which data were acquired.

Coordination with ground truth parties went well; pertinent information obtained from both aircraft flights and ships at sea were relayed to the crew in real time so that specific sites surveyed on the ground would be observed from orbit.

Television transmissions were made over several tracking stations and images of the daylight portion of revolution 124 were recorded on the video tape recorder. This imagery turned out well and will be valuable in both scientific investigations as well as for educational and public information purposes.

6.1.9 Artificial Solar Eclipse Experiment

The artificial solar eclipse experiment (MA-148) was conducted to study the feasibility of performing solar corona observations from space using simple instrumentation. The procedure was to perform photography simultaneously from both spacecraft as the Apollo occulted the sun as seen from the Soyuz docking hatch window. The Soyuz photography was to obtain the solar corona data and the Apollo photography was to provide data on separation distance and umbra location on the Soyuz as a function of time.

The planned operation was to maneuver the docked spacecraft so that the aft end of the Apollo was toward the sun. The spacecraft were to be undocked 75 seconds past orbital sunrise. The Apollo was then to coast for 15 seconds, thrust away from the Soyuz along the X axis for 3 seconds, coast for 12 seconds, thrust for 4 additional seconds, and coast for 206 seconds at a separation rate of approximately 1 meter per second. The Apollo crew reported that the maneuver was performed nominally.

TABLE 6-I.- MAPPING PASSES

Pass	Accomplished	Not accomplished	Remarks
M1 Gulf Stream		X	Cancelled because of flight plan problems.
M2 New Zealand	X		
M3 Southern California	X		
M4 Himalayas	X		
M5 Arabian Desert	X		
M6 Australia	X		
M7 African drought area	X		
M8 Falkland Current	X		Attitude not nominal.
M9 Sahara Desert	X		
M10 Northern California	X		
M11 New England	X		Out of focus.

TABLE 6-II.- VISUAL OBSERVATION SITES

Site	Accomplished	Not accomplished	Remarks
1 New Zealand	X		The Alpine Fault was cloud covered, but visual observations of ocean waters northeast of New Zealand were made.
2A Southern California	X		Photographs of the Baja Peninsula were obtained, but there were no crew comments.
2B Baja California	X		
2C California Current	X		
2D Great Salt Lake	X		
3A Cloud features	X		
3B Tropical storms	X		
3C Hawaii		X	Cloudy.
4A Snow peaks	X		Visual observation comments. No photography.
4B Puget Sound	X		
4C Superior iron	X		
4D Sudbury nickel		X	Cloudy.
5A Gulf of Mexico	X		Cancelled.
5B Gulf Stream		X	
5C Labrador Current	X		Visual observation comments. No photography.
5D Central American structures	X		
5E Florida red tide		X	Cloudy.
5F New England red tide	X		Boothbay cloudy; no red tide observed.
5G Chesapeake Bay	X		
6A Oil slicks		X	No oil slicks observed.

TABLE 6-II.- VISUAL OBSERVATION SITES - Continued

Site	Accomplished	Not accomplished	Remarks
6B London	X		Cloudy.
7A Humboldt Current	X		
7B Nazca Plain	X		
7C Internal waves	X		
7D Peruvian Desert	X		
7E Orinoco River Delta	X		
7F Galapagos Islands	X		
7G Caribbean Sea	X		
8A Falkland Current		X	Cloudy.
8B Andes	X		Cloudy.
8C Dune field	X		
8D Parana River	X		
8E Circular structures	X		
9A Afar Triangle		X	Cloudy.
9B Arabian Peninsula	X		
9C Guinea Current		X	Cloudy.
9D Desert colors	X		
9E Oweinat Mountain	X		
9F Nile Delta	X		
9G Levantine Rift	X		
9H Niger River Delta		X	Cloudy.
9I Algerian deserts	X		
9J Tripoli	X		
9K Strait of Gibraltar	X		
9L Alps		X	Cloudy.
9M Danube Delta	X		
9N Anatolian Fault	X		
9O Volcanics	X		
9P Bioluminescence		X	None observed.

TABLE 6-II.- VISUAL OBSERVATION SITES - Concluded

Site	Accomplished	Not accomplished	Remarks
10A Great Dike		X	Photographs out of focus.
10B Somali Current		X	Photographs out of focus.
10C Arabian Sea		X	Photographs out of focus.
10D Himalayas		X	Photographs out of focus.
10E Takla Makan Desert		X	Photographs out of focus. Visual observation comments.
11A Playas	X		
11B Coral Sea	X		
11C Simpson Desert	X		
11D Tasmanian Sea		X	Cloudy.
12A Icebergs		X	None observed.

As the spacecraft separated, a camera in each spacecraft obtained data. The Soyuz photography was performed with a camera mounted in the orbital module docking hatch window and viewing along the X-axis toward the Apollo. The Apollo photography was performed by the window-mounted 16-mm data acquisition camera viewing along the X-axis toward the Soyuz. The data acquisition camera operation commenced just prior to undocking and continued through the separation. The edge of the Apollo shadow began crossing the Soyuz docking ring approximately 120 seconds after undocking and began crossing the light baffle on the Soyuz docking hatch window approximately 170 seconds after undocking.

6.1.10 Crystal Activation Experiment

Crystals of sodium iodide (thallium-activated) and germanium are detector materials for appropriate instrumentation for the new field of gamma-ray astronomy. However, a limitation in their use is that they are susceptible to radioactive activation by particle bombardment (mostly protons and neutrons) in the space environment. This activation yields background noise which can obscure the desired gamma-ray signal. The crystal activation experiment (MA-151) was conducted to study the effects of particle radiation on instrument noise levels in these detector materials.

The experiment package was received within 80 minutes of landing and all items were in good condition. About 4 hours of early measurements were made on the recovery ship. The delivery of the materials to the appropriate laboratories was as scheduled, and preliminary measurements were completed.

The high-energy proton activity was at least a factor of three below that of Apollo 17. However, iodine-124 and iodine-126 have been identified in the sodium iodide spectra and gallium-67 in the germanium spectra. It was determined from the scandium analysis that the thermal neutron flux was 0.2 neutron/sq cm. sec.

6.2 LIFE SCIENCES

6.2.1 Microbial Exchange Experiment

The purpose of the microbial exchange experiment (AR-002) was to evaluate components of the infectious disease process in space flight by measuring alterations in three factors: (1) the composition of the microbial populations inhabiting the crewmembers and spacecraft, (2) the ability of each crewmember's defense mechanism to resist infection, and (3) alterations in the ability of certain microorganisms to originate infections. The impetus for performing the experiment arose from the variety

of flight and ground-based microbial studies in the USA and USSR which have indicated that the conditions of spaceflight alter man and microorganisms in such a way that the normal fine balance between them may be adversely affected. The monitoring of two separate crews, which differ microbiologically and immunologically, provided an opportunity to study inflight cross-contamination patterns. Because crewmembers came from widely different geographical and ecological areas, it was possible to identify specific, naturally occurring, marker microorganisms for detailed analysis.

Microbiological samples were to be collected from the five prime crewmembers, the five backup crewmembers, and 15 areas on the inner surface of each spacecraft at specific times before, during, and after the flight. Blood and saliva samples were to be collected from each of the 10 crewmembers at specific times before flight and from the five flight crewmembers after the flight. All inflight samples, and all samples collected in the Soviet Union were to be processed in the presence of American and Soviet specialists and subsequently divided and distributed to both sides for complete and detailed analysis. Samples collected in the USA were to be processed in the presence of Soviet specialists at the option of the Soviet Union.

The purpose of this experiment could be completely realized only by satisfying each of the following objectives. The extent to which each objective was accomplished is evaluated.

a. Identify the normal microbial autoflora and immunocompetence level of each crewmember.- The collection of five sets of preflight saliva and microbial samples and of two sets of serum samples per crewmember provided adequate establishment of baselines for the Apollo crewmembers. However, data pertaining to the total microbial load cannot be retrieved from microbiological specimens collected in the Soviet Union. All serum samples, with the exception of that from the backup Soyuz Commander taken 15 days prior to launch, were received in a condition which permitted the required analysis. All of the astronaut saliva samples, with the exception of the Command Module Pilot's, were collected as planned. (An alternate collection method was employed with the Command Module Pilot which provided adequate, but not optimum, saliva.) All cosmonaut saliva samples were collected differently than agreed upon and, hence, the validity of the samples was uncertain. This objective was completely satisfied for the six astronauts and partially satisfied for the four cosmonauts.

b. Identify qualitative and quantitative changes in crew and spacecraft microbial populations and evaluate demonstrable intercrew microbial exchange occurring in flight.- The most important marker microorganisms were properly monitored in the USA and USSR laboratories, allowing for a reasonably thorough evaluation of intercrew microbial exchange. Likewise,

shifts in many of the major components of the autoflora can be evaluated. Variations in the total autoflora can be calculated only for the Apollo crew because the cosmonaut specimens were not always processed in the agreed upon manner.

c. Critically evaluate selected microorganisms in a manner which will detect postflight changes in their ability to demonstrate pathogenicity, infectivity, or toxicity in man.- Several of the microorganisms which are most useful for these evaluations have repeatedly been isolated from Apollo-Soyuz crewmembers. The prognosis for satisfying this objective is very good. Thirty-day postflight samples of cosmonauts were not collected by USSR specialists, making analysis of return-to-normal unlikely.

d. Evaluate immunological parameters of blood and saliva to detect postflight changes in the ability of crewmembers to resist infection.- The required astronaut specimens have been obtained and analyses are proceeding as planned. Data from cosmonaut specimens will be less complete because of incomplete blood collection and the different saliva collection technique referred to earlier.

In summary, the majority of activities were performed as planned and a large percentage of the cooperatively derived data will be preserved. At present, all samples have been collected and divided among the two laboratories, with the exceptions noted above. Analyses of specimens are currently progressing according to the joint laboratory procedures previously approved by both countries.

6.2.2 Light Flash Experiment

The light-flash phenomenon consists of tiny star-like, brief flashes of white light noted by astronauts on Apollo 11 and subsequent missions including Skylab. The need for dark adaptation and the dependence of the perception of light flashes on latitude explains why the flashes were not seen by the Gemini astronauts and all the Apollo astronauts. The purpose of the light flash experiment (MA-106) conducted on the Apollo-Soyuz mission was to characterize the cosmic particles that cause this phenomenon using solid-state detectors and to quantify sensory perception relative to latitude, shielding, and dark adaptation.

Flux measurements were made during two revolutions: Revolution 110 was devoted to measurement of heavy charged particles using a silicon telescope-spectrometer. Particles of stopping power greater than 10 keV/ μ in water would be detected and described by a digital word comprised of the trajectory and energy loss of the particle as it traversed 4 gm/cm² of copper. Events were recorded during revolution 110 without visual

perception being attempted by the crew. On revolution 111, the Commander and Command Module Pilot, wearing light-tight masks, noted perceived light flashes while the Docking Module Pilot operated the experiment control switches and recorders. During this orbit, silver chloride crystals doped with cadmium were used in addition to the silicon detector telescope-spectrometer for particle identification.

The first light flash was reported when the spacecraft was approaching the Gulf of Alaska near the end of revolution 110. Between this point and Florida, 20 events were reported. Only 4 events were noted during the 15 minute transit from Florida to Brazil and only 5 events were noted while passing through the South Atlantic anomaly region. The greatest activity was noted during the southernmost portion of the orbit. Between 45 degrees south, over the South Atlantic Ocean, and 30 degrees south, over the Indian Ocean, 41 events were noted. During the remainder of the orbit from 30 degrees south to the northernmost point of revolution 111 over the Aleutian Islands, only 9 events were noted. At each event, a pushbutton signal from the observer was recorded on the digital tape recorder and a voice description was recorded on the voice tape recorder.

Preliminary data from the silver chloride crystals show that most of the heavy particles were registered over the northern latitudes, as expected, but very few particles were detected through the southernmost portion of the orbit where the light flashes were most frequent. The silicon detector recorded 36 events during the unmanned orbit and 132 events on the manned orbit. A total of 79 visual events was reported for each of the two crewmen.

The results of this experiment indicate that the abundance of ions that cause the light flash phenomenon is minimal between latitudes 30 degrees north and 30 degrees south. The frequency of light flash events between latitudes of 30 degrees and 50 degrees is 25 times that noted in the equatorial latitudes. There were no reports of increased light flashes through the South Atlantic anomaly, as might be expected from the results of the Skylab third visit (ref. 1) where increased activity through the South Atlantic anomaly was noted. However, at the 223-kilometer altitude of Apollo-Soyuz, the proton flux is much less than at Skylab altitudes, and the shielding of the Apollo spacecraft is generally greater than that of Skylab.

Through the South Atlantic anomaly, the number of protons with a stopping power of $15 \text{ keV}/\mu$ which would intersect a crewman's eye is about two per minute for each eye with the shielding of the Apollo spacecraft. The range of these particles in the retina is about 100μ . Based on the assumption that stopping protons of this energy will give visual perception, and the fact that two observers reported events at one per minute through the South Atlantic anomaly, the investigators conclude that the efficiency is about 10 percent. If the threshold is $10 \text{ keV}/\mu$, then the

efficiency is reduced to 5 percent. The efficiency should be a function of stopping power. Careful ground-based experiments with stopping protons are being conducted to corroborate these observations. The efficiency for seeing high-atomic-weight/high-energy particles is about 50 percent for the dark adapted eye.

6.2.3 Biostack Experiment

The Biostack experiment (MA-107) was conducted to investigate the effects of high-energy particles of cosmic radiation on selected biological materials. Two canisters containing the biological materials and cosmic ion track detectors were stowed in the command module. The specific effects to be studied include evidence of chromosome damage, changes in cell division, growth and development, and the appearance of mutations.

The flight units and ground control units were disassembled 5 days after recovery and all components were found to be in good condition. The visual track detectors were developed without difficulty, showing analyzable tracks. Preliminary data on the fluence of cosmic high-energy/high-atomic-number particles were obtained. Studies of the non-hit biological material of the flight unit showed that the viability was not remarkably different from that of the ground controls.

6.2.4 Zone-Forming Fungi Experiment

The zone-forming fungi experiment (MA-147) was conducted to study effects of the space environment on the zone-forming microorganism *Streptomyces (Actinomyces) levoris*. The purpose was similar to that of the Biostack experiment (MA-107) except that growing rather than dormant organisms were studied. An additional important objective was to investigate the influence of the space flight on the circadian periodicity of the microorganism by exchanging during flight one set of cultures launched from the USA with one set launched from the USSR.

From the time of final culture selection on July 13, 1975, both control and flight specimens were scheduled to be photographed for monitoring of growth at approximately 12-hour intervals until postflight termination of the experiment. The controls were photographed as scheduled from July 13 until August 11, 1975. Although the flight specimens were scheduled for photography 18 times, only 15 series of photographs were obtained in the Apollo spacecraft because of various photographic difficulties. Precise determination of specific growth characteristics from each series of photographs was not possible because some of the photographs were out of focus or lighting conditions were poor. Photographic data from the Soviet activities of this phase of the experiment are not available at this time.

The exchange of one set of cultures (one device with two cultures) occurred on schedule. One set of specimens launched from the USA and one set launched from the USSR were recovered from the Apollo spacecraft and returned to JSC. Visual assessment and comparison of the cultures showed that the cultures initiated in the USA (one flight unit and two ground control units) exhibited 9 and 11 rings, while the cultures in the device received in the inflight exchange from the Soyuz contained 6 and 7 rings at the time of recovery. On the basis of photographic evidence, alteration in the growth periodicity appeared to have occurred during the mission.

Analysis of the dosimeters for hard particle (high-energy radiation) tracking will be performed by the Soviets. The possible genetic alteration of the bacteriophage contained in the dry gelatin film which was situated beneath the culture dish within the hardware device will also be studied in the Soviet laboratories.

Densitometry studies of the photographic film for evaluating the growth patterns of the organisms will be performed by the Soviets and are currently in progress.

Following termination of the experiment at JSC, both the flight and control cultures were forwarded to the Soviet principal investigator. Secondary culturing of the specimens for comparison with controls should currently be in progress in the USSR.

Specimens for transmission and scanning electron microscopy examination, which will be performed by the United States, will not be obtained until the dosimetry studies are completed by the Soviets. These data will be used as a guide for locating specific sectors of cultures which may have been affected by radiation during the mission.

The only difficulty with the hardware was loss of a retainer ring from the device lid which held a Lexan dosimeter in place. This occurred in a flight specimen during the mission and since the dosimeter was removed from the original position, the use of this dosimeter for radiation particle tracking on that particular sample is precluded.

6.2.5 Killifish Hatching and Orientation Experiment

The killifish hatching and orientation experiment (MA-161) was conducted to evaluate the gravity dependence of vestibular structure and functional development during embryogenesis.

Two plastic film packages were carried on the Apollo spacecraft - one containing fry and the other containing a graded series of embryonated eggs. The fry package was photographed and visually observed during flight for indications of vestibular adaptation.

Relatively little hatching of eggs occurred during the flight. By day 9, only ten of the 336-hour eggs had hatched but four of these were visually observed to exhibit looping disorientation behavior. Immediately after opening of the package at recovery, hatching proceeded very rapidly and had reached 60, 77, and 98 percent in the 128-, 216- and 336-hour eggs, respectively. All eggs were in exceptionally good condition. Only two of the 450 flight eggs had died at time of recovery. Hatching and health were noticeably better than any of the control treatments.

Sampling for subsequent histological analysis was begun approximately 6 hours after landing on both fry and hatchlings. Sampling of hatchlings continued for 6 days after landing, and additional samples were taken at 15 and 30 days. Further samples will be taken at 6 months. Preliminary postflight examination revealed no major differences between control and flight treatments.

The following observations are based on written flight notes and preliminary observation of the films. Partial habituation to zero-g had occurred by flight day 6 and there was some evidence that lowered sensitivity to disturbance, as indicated by reduced looping, had occurred by day 9. Nutritional factors may have had some influence. There was considerable variation in individual reaction to the null gravity environment.

6.2.6 Cellular Immune Response Experiment

The cellular immune response of the three prime crewmembers (MA-031) was studied before and after the 9 days of flight. The findings from this study are to be correlated in the future with lymphocytic changes noted during the Skylab spaceflight. Briefly, at the conclusion of the Skylab second and third visits, the functional capacity of lymphocytes was depressed along with a suppression in the T lymphocyte numbers. No such changes were noted following a 14-day bedrest study. In the present study of shorter flight, functional suppression was again noted but no quantitative changes in lymphocytes occurred.

6.2.7 Polymorphonuclear Leukocyte Response Experiment

A variety of different physiological, hematological, and serological measurements have been performed throughout the manned space flight program. There has been no previous formal effort to investigate the possible effects of space flight and weightlessness on polymorphonuclear leukocyte function. Since these phagocytic cells are important in the defense of the body against disease, and since situations in which the function of these leukocytes is abnormal are associated with increased susceptibility to infection, a variety of experiments was designed to look at changes in the polymorphonuclear leukocytes in relationship to the events of the Apollo Soyuz mission. The goal of this set of experiments (MA-032) was to identify any measurable alterations in polymorphonuclear leukocytes which might be of significance in planning future, more prolonged space missions.

The results of this set of experiments are still being analyzed, and correlated with a large body of normal data in which the same laboratory methods are also utilized. The experiments went as planned, and all blood samples were obtained during baseline, postflight, and later convalescent periods. The unplanned inhalation of propellant gases, and subsequent hospitalization of the crew with the administration of adrenal corticosteroid therapy, gave an opportunity for the evaluation of these factors in relation to leukocyte function.

Overall, this experiment was successful in documenting that no consistent, potentially serious abnormalities in polymorphonuclear leukocyte function were produced (detectable by the methods employed) in the crew members who participated in the Apollo Soyuz Test Project. A broader experience, including studies of a similar nature on future space flight missions, will be required before any definite conclusions can be drawn.

6.3 TECHNOLOGY

6.3.1 Multipurpose Furnace

The multipurpose furnace (MA-010) in the docking module was used to process seven sets of samples, each having a separate experiment number. The seven experiments are discussed individually.

6.3.1.1 MA-041, surface tension induced convection.- The MA-041 experiment objective was to evaluate surface tension effects due to concentration gradients. The experiment consisted of melting three bimetallic material samples (lead/lead with 0.05 atomic percent gold) in wetting (iron) and non-wetting (graphite) capsules, allowing them to interdiffuse, and then solidify.

The MA-041 sample processing was performed on mission days 2 and 3. The processing was not entirely nominal because the samples were held at the "soak" temperature longer than planned and the cooldown timeline was also extended. Visual inspection of the samples indicates that they are in excellent condition with no breaks evident in the ampoule walls. Surface ridges which were observed in earlier Skylab experiment samples are not present in the MA-041 samples. The cause and significance of this will be further evaluated. Sample sectioning and polishing will be followed by metallurgical evaluation. The impact of the extended heating will be determined during these detailed analyses.

6.3.1.2 MA-044, monotectic and syntectic alloys.- Aluminum-antimony compounds have promise as a high efficiency solar cell material; however, the large difference in specific gravities of the two elements has hampered growth of quality single crystals on earth. Two samples of aluminum-antimony were processed to determine the homogeneity of mixing during solidification of this syntectic compound. As a companion experiment, three samples of lead-zinc alloy were processed to determine the effects of near-zero gravity on the degree of immiscibility of this monotectic system.

The three MA-044 cartridges containing the samples were successfully processed in the multipurpose furnace on mission days 8, 9, and 10. When the cartridges were opened for postflight analysis, one of the three aluminum-antimony ampoules was found to have leaked, rendering it less valuable because of contamination. The two remaining samples were in good condition and provide adequate sample material for all planned analysis effort. The three lead-zinc samples were in good condition. After completion of topographical studies, the detailed metallurgical investigation of the microstructure of all samples will begin.

6.3.1.3 MA-060, interface marking in crystals.- Gravity-induced thermo-hydrodynamic perturbations in the melt have been identified as a primary cause for defects that limit chemical and crystalline perfection of electronic devices (especially semiconductor devices) resulting in performance below their theoretical levels. A new process, growth interface demarcation, used with differential etching and spreading resistance measurements make quantitative studies of the detailed growth and segregation behavior of electronic materials possible. In MA-060, single crystals of germanium (gallium doped) were subjected to partial melting and regrowth in the MA-010 furnace system. Throughout the growth process, "interface demarcation" was achieved by Peltier heating associated with the transmission of electric current pulses across the crystal-melt interface. Interface demarcation will be revealed by differential etching. Absolute time reference can be obtained from the demarcation lines, and growth interface morphology and growth changes will be obtained from the shape of the demarcation line. These data will be combined with the spreading resistance measurements to analyze the numerous factors involved in crystal growth.

The three MA-060 flight cartridges were successfully processed on mission days 7 and 8. Upon postflight examination, one of the three quartz ampoule covers was found to be cracked, but there was no apparent adverse effect on the sample material. The Peltier-effect striations are extremely well pronounced in the one sample opened to date.

6.3.1.4 MA-070, zero-g processing of magnets.- High coercive strength permanent magnets are being investigated for advanced technology applications. At present, the major limitation to the use of high-coercive-strength-cobalt/rare-earth permanent magnets is the method of fabrication (sintering of powders). The processing of these magnetic materials in the low-gravity environment should, in one operation, limit the possibility of oxidation and increase the density and magnetic properties of the product.

The experiment involved solidification of an immiscible mixture of a magnetic phase in a non-magnetic matrix without separation due to density differences or stirring caused by convection that are characteristic of earth-based processing. Three separate ampoules, each with an individual experimental objective, were contained in each of the three cartridges.

The three MA-070 flight cartridges were successfully processed on mission days 5 and 6. After the cartridges were returned for analysis, two of the ampoules were found to be cracked with a small loss of sample material. The cracking probably occurred during processing due to expansion of the manganese-bismuth material. All other ampoules were intact and the lost material is insignificant to the planned analyses.

6.3.1.5 MA-085, crystal growth from the vapor phase.- Single crystal growth by chemical vapor transport in a temperature gradient is macroscopically and microscopically affected by gravity-induced convection currents. In addition, changes in surface morphology and defect structure have also been observed in ground-based experimentation.

Germanium-selenium and germanium-tellurium alloys were selected for this experiment since this system seems to be ideal for study of the effects of convection current. A second area of investigation that was addressed is the question of whether and to what extent surface defects on grown and pretreated crystals propagate during subsequent growth by vapor deposition. Investigations of this phenomenon under low-gravity conditions will determine the possible existence of gravity-independent fluctuations in the gas phase in the vicinity of the crystal surface. Thirdly, the experiment studied the possible existence and significance of thermochemically driven convection currents as compared to gravity-induced convection.

The three MA-085 cartridges were processed in the MA-010 furnace system on mission days 6 and 7. No problems with sample materials were apparent upon postflight examination.

6.3.1.6 MA-131, sodium chloride-lithium fluoride eutectic.- Solidification of a eutectic liquid of sodium chloride and lithium fluoride results in lithium fluoride fibers being formed in a matrix of sodium chloride. This material is one of great interest in the area of fiber optics. However, defects in these materials grown on earth cause these solid-state devices to be inefficient. The defects include banded structure, fiber discontinuity, and faults due, in part, to the presence of vibration, and to gravity-induced convection currents in the melt during solidification. This material was processed in near-zero gravity to determine if the defects can be eliminated.

The three MA-131 experiment cartridges were processed on mission day 6. Postflight examination showed that the samples were in good condition.

6.3.1.7 MA-150, USSR multiple material melting.- Convective stirring during solidification and segregation in the melt due to gravity contribute to inhomogeneities, voids, and structural imperfections in materials when processed on earth. This experiment processed three different material systems in each of three cartridges. A sample of aluminum with tungsten spheres and a sample of powdered aluminum were processed in isothermal regions, and a sample of germanium with 0.5-percent silicon by weight was processed in the gradient region.

The three MA-150 experiment cartridges were launched and returned in the Soyuz spacecraft. The sample processing in the MA-010 furnace system was performed on mission days 3 and 4. The furnace system provided a nominal temperature profile and no anomalies are expected in the returned samples.

6.3.2 Electrophoresis Technology Experiment

Electrophoresis is an important tool in biological and medical research. Most of the development in the field has been in an area described as zone electrophoresis in stabilized media. This technique separates a single narrow zone of sample mixture in an electrolyte medium into many zones containing a single component of the mixture and electrolyte between them. Since the densities of the separated zones generally differ from that of the intervening medium, such systems are gravitationally unstable and stabilization is required.

Two major causes of disturbances during one-g processing are the density of the particles or solute being separated and the thermal convection generated by the joule heating of the column during electrophoresis. Although a variety of techniques have evolved to overcome these problems on earth, none have been completely successful and the elimination of gravity-induced sedimentation and thermal convection can be accomplished best in space.

The Apollo-Soyuz mission provided an opportunity to test new techniques in a low-gravity field with a potential improvement in the degree of separation. The electrophoresis technology experiment (MA-011) was a further development of technology begun on experiments performed on the Apollo 14 and 16 missions and the Skylab third visit. Specifically, the MA-011 experiment investigated the following areas: sample insertion techniques, verification that electro-osmosis can be minimized in a static fluid electrophoresis device, evaluation of two methods of electrophoresis, and verification that the viability of biological samples prepared for electrophoresis can be maintained during a space mission.

Six samples were provided to test free fluid zonal cell electrophoresis using fixed blood cells and live biological samples. In addition, two samples of fixed blood cells were provided to test the isotachopheresis process.

The processing unit, seven of the eight sample columns, and the cryogenic freezer were returned. One sample column, kidney cells, developed a leak during processing and no usable data were acquired. The cause of the leak is discussed in section 14.2.5. Usable data can be obtained from the returned samples and film coverage of seven of the eight sample runs. Sectioning and examination of the returned frozen columns is currently in progress.

The returned photographic results reviewed to date indicate that planar band separations were accomplished in at least one of the two red blood cell model columns. A significant number of photographs were out of focus and these data are currently being subjected to special photographic processing for visual enhancement.

6.3.3 Electrophoresis-German Experiment

The electrophoresis-German experiment (MA-014) was designed to investigate the area of free-flow electrophoresis in contrast to the stabilized media method tested in experiment MA-011. In free-flow electrophoresis, the sample flows continuously through an electric field perpendicular to the flow.

At present the efficiency of ground-based free-flow electrophoresis is limited by the resolution of the electrophoresis apparatus and the low output of separated cells. Higher output could be achieved by using higher flow-rate, wider separation gap, and higher electric field densities. This, however, increases the temperature with resultant increase in convection currents and sedimentation. In zero-gravity, these conditions would not prevail.

The MA-014 experiment equipment analyzed the sample flow by opto-electronic means and the results were recorded in duplicate in digital form on the MA-014 tape recorder which was returned to earth. One of the returned tapes has been delivered to the Max Planck Institute in Germany for analysis of the data. The second tape is currently in bonded storage at JSC. To date, data analysis indicates excellent separation for two of the samples processed (rat spleen cells and rat lymph node cells).

6.3.4 Crystal Growth Experiment

The crystal growth experiment (MA-028) was conducted to determine if crystals can be grown by diffusion through water in near-zero-gravity conditions.

The experiment was performed according to plan. Photographs of the reactors taken during the flight were of good quality. Because of the index of refraction of the crystals being close to that of water, a darker background would have enhanced the contrast of the photographs. A total of 40 photographs was taken to document experiment progress.

Crystals formed in all six reactors. Therefore, the experiment was very successful. As expected, the largest crystals obtained are calcium tartrate. Both calcium tartrate reactors produced numerous well-formed crystals up to 5 millimeters in length and a few up to 10 millimeters in length. The two calcium carbonate reactors each produced a large number of well-formed clear crystals up to 0.5 millimeter on an edge. The two lead sulfide reactors were less successful but did produce crystals up to 0.1 millimeter in size. The crystals grown in all six reactors are at least as large as those obtained by gel growth methods on the surface of the earth in the same length of time. All six reactors contained crystals suitable for detailed postflight analysis.

No finely divided precipitate was observed in the calcium tartrate or calcium carbonate reactors. This indicates that in these four reactors, mixing of the reactant solutions and crystal growth were completed by the time of spacecraft entry.

During the flight, the cabin temperature varied more than was desirable for solution growth. The effects of the temperature variation on crystal growth will be considered during the data analysis.

7.0 INFLIGHT DEMONSTRATIONS

Four inflight demonstrations were performed: capillary wicking, liquid spreading, chemical foams, and physics demonstrations.

7.1 CAPILLARY WICKING

The capillary wicking demonstration was conducted to obtain data on capillary wicking rates of various materials in near-zero gravity. Four wicking assemblies were used for the demonstration, each containing three metal wicks and one durrell cloth wick. Oil and water solutions were introduced with syringes. The data were recorded with the 16-mm data acquisition camera and the voice tape recorder.

Initial examination of the film and crew comments indicates that the wicking rate was greater than expected. The reason for this is presently under investigation.

7.2 LIQUID SPREADING

The liquid spreading demonstration was performed to qualitatively measure the rate and manner of liquid spreading on solid surfaces in near-zero gravity. Drops of dyed silicone oil and water were to be deployed inside three boxes with syringes and the liquid spreading was to be recorded with the 16-mm data acquisition camera.

Only one of the three boxes was photographed. However, the data are sufficient to essentially satisfy the objectives. The spread of oil drops on the wall of the box will be measured and the behavior of liquid on the bottom of the box will be studied.

7.3 CHEMICAL FOAMS

The chemical foams demonstration was performed to demonstrate the rate of chemical reactions and the stability of liquid foam mixtures in near-zero gravity. To demonstrate the rate of chemical reactions in foams, tubes containing water, thymal blue indicator, and ethyl alcohol were to be prepared and shaken to form a foam. The color change would indicate the time required for the chemical reaction to be completed. To demonstrate the stability, chemical solutions were to be prepared and shaken vigorously to effect thorough mixing. The demonstration was to be photographed using the 16-mm data acquisition camera.

Out-of-focus photography limited analysis of the foam stability, but the color change was detectable. The timing of the color change was significantly different than that obtained with earth gravity and on a KC-135 aircraft zero-g flight.

7.4 PHYSICS DEMONSTRATIONS

Simple demonstrations of basic physical principles for classroom instruction were performed and photographed as planned in the Soyuz spacecraft. The demonstrations were recorded with a USSR motion picture camera.

8.0 JOINT FLIGHT ACTIVITIES

Following a nominal rendezvous, docking with the Soyuz spacecraft was successfully achieved. After placing the Apollo spacecraft systems into a post-docked configuration, the Apollo Commander and the Docking Module Pilot prepared for the first transfer into the Soyuz orbital module. Upon entering the docking module, the crew smelled a pungent odor likened to that of acetate. Oxygen masks were donned until an assessment of possible sources of the odor indicated that it was safe to proceed. The atmosphere mixing system subsequently dispersed the odor.

Hatches 3 and 4 were opened on schedule and the first international handshake in space was televised to much of the world (fig. 8-1). All planned operations for the first joint activity period in the Soyuz spacecraft were completed. The activities included the exchange of each nation's flags, exchange of the United Nations flag, connection and checkout of the cable communication system between the two spacecraft, setup of Apollo television and data acquisition camera equipment, signing of joint flight certificates (fig. 8-2), eating (fig. 8-3), exchange of zone-forming fungi and microbial exchange experiment devices, and insertion of experiment cartridges into the multipurpose furnace for subsequent heatup. In addition to the planned activities, messages of congratulations were received by all crewmembers from the heads of state of both countries.

At the end of the first day's joint activities in the Soyuz spacecraft, the Apollo Commander and Docking Module Pilot began their return to the command module. Following the depressurization of tunnel 2, the cosmonauts reported that their integrity check of the orbital module/docking module hatches 3 and 4 had failed and one of the two hatches appeared to be leaking into the tunnel at a rate of 0.01 newton/cm²/min. The tunnel was repressurized by the Apollo crew, hatches 3 and 4 were reopened, and their respective seals checked. A second integrity check subsequently failed. A decision was made to have the Apollo crew continue docking module depressurization procedures en route to the command module in the belief that the apparent hatch leakage was actually thermal instability in the tunnel following its rapid depressurization. Further checks on the tunnel by the Soyuz crew during the ensuing sleep period convinced both sides that there was no problem and the Soyuz integrity check go/no-go tolerances were biased to reflect the thermal condition.

The second day's joint activities were begun on time with the Command Module Pilot transferring into the Soyuz spacecraft and the Soyuz Commander transferring into the command module. Milestones accomplished in Soyuz included an orbital module and descent vehicle television tour conducted by the Flight Engineer (fig. 8-4), the Command Module Pilot's signing of

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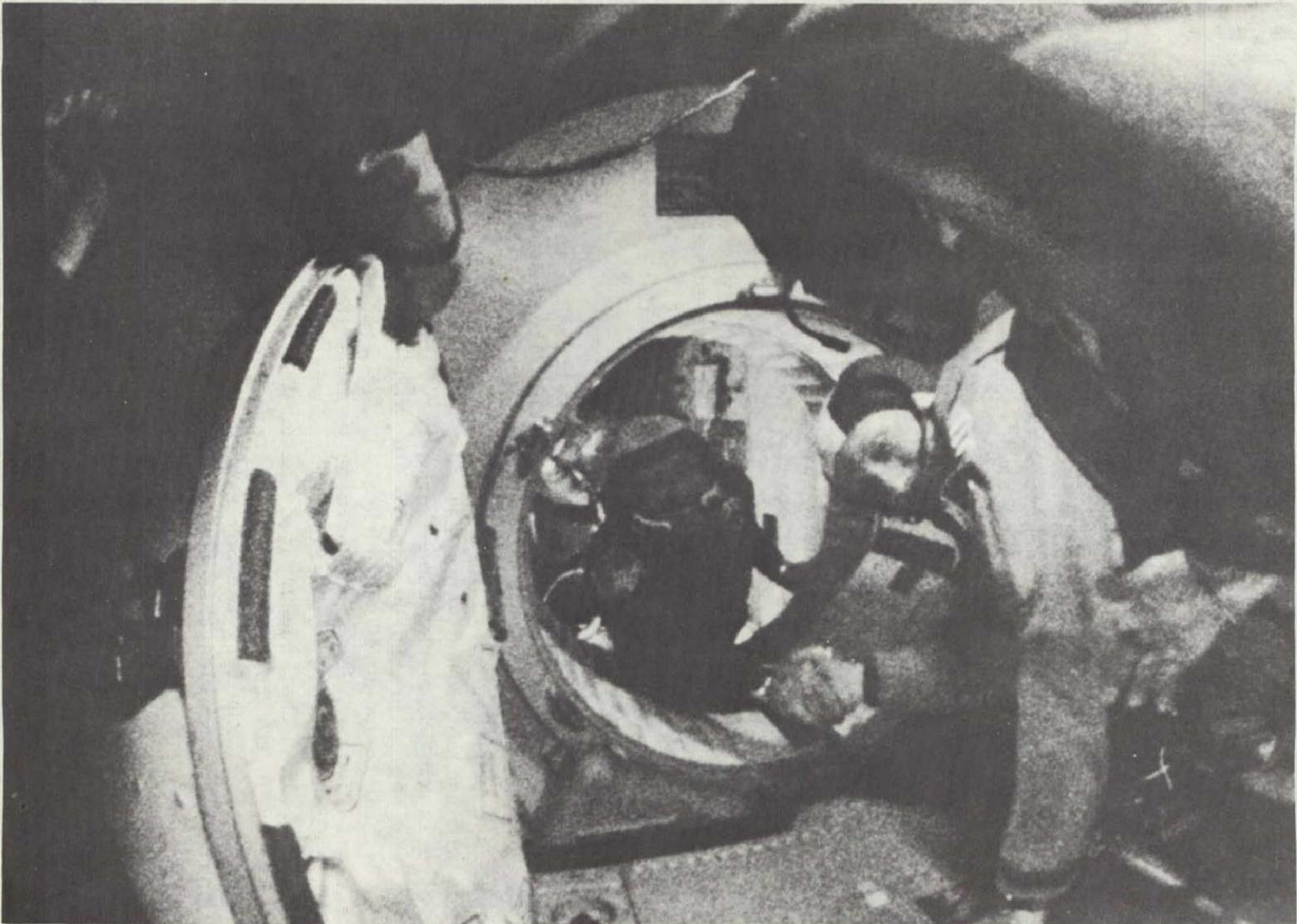


Figure 8-1.- Apollo and Soyuz Commanders greeting each other during initial transfer.

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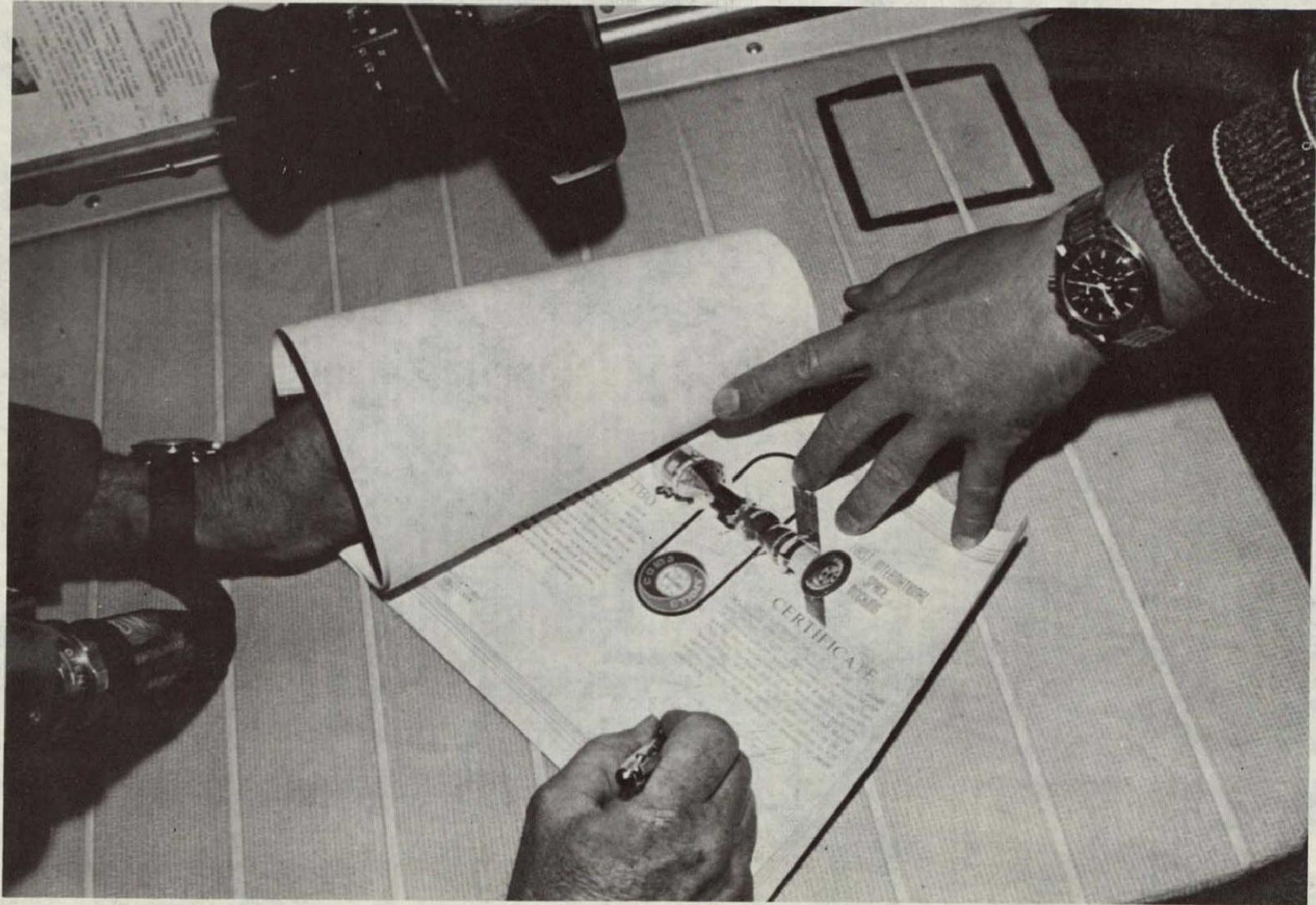


Figure 8-2.- Docking Module Pilot signing joint flight certificate.

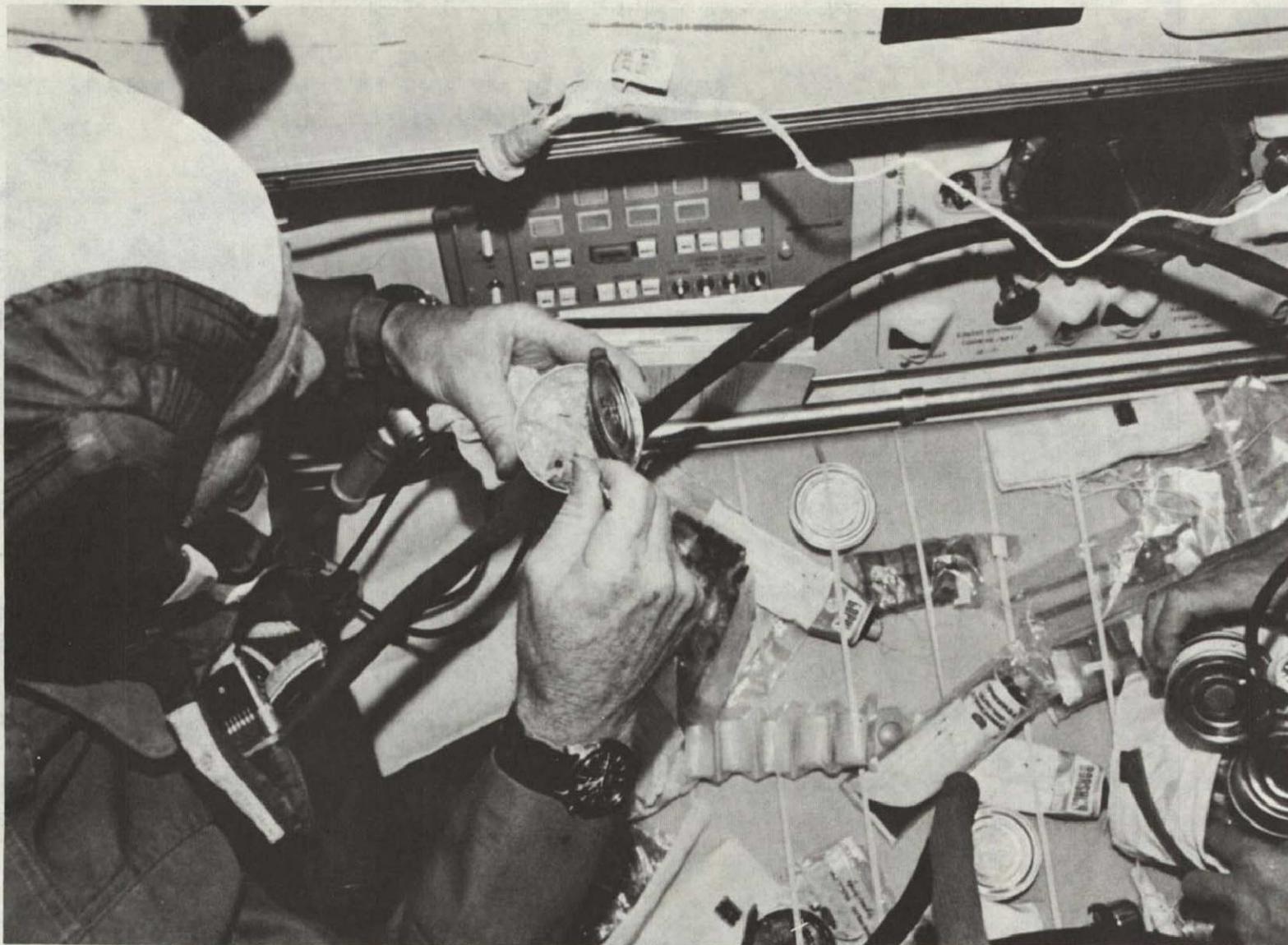


Figure 8-3.- Apollo Commander eating in the Soyuz spacecraft.



Figure 8-4.- Soyuz Flight Engineer .

the joint flight certificates, the joining of a commemorative plaque (fig 8-5), an exercise period for the Command Module Pilot using Soyuz equipment, a visual tour of the USSR conducted by the Flight Engineer, science demonstrations performed by the Command Module Pilot, a joint meal - with comments on Russian space food by the Command Module Pilot, and television and photographic requirements associated with the activities. Milestones accomplished in Apollo during this time included a television tour of the command module conducted by the Apollo Commander, signing of the joint flight certificates, the joining of a commemorative plaque, television camera support of the tour of the USSR conducted in the Soyuz spacecraft, an exercise period for the Soyuz Commander using Apollo equipment, a joint meal with commentary from the Soyuz Commander on American space food, orbital science, and television and photographic requirements associated with the activities.

Following the third crew transfer - with the Soyuz Commander (fig. 8-6) and Apollo Commander in the Soyuz spacecraft and the Docking Module Pilot, Command Module Pilot and Flight Engineer in the Apollo spacecraft - the milestones accomplished in Soyuz included a joint press conference, microbial exchange experiment sampling, presentation of tree seeds as a symbolic gift to the Soviet people, and the joining of a commemorative medallion. Simultaneously, in Apollo, a joint press conference, microbial exchange experiment sampling, presentation of tree seeds as a symbolic gift to the American people, the joining of a commemorative medallion and a television tour of Florida were accomplished along with earth observations. The activities in both spacecraft were televised and photographed as planned. The fourth transfer returned all crewmembers to their own spacecraft on schedule.

The complex network of communications between the two vehicles and their respective ground stations led to numerous small communications problems during the duration of the joint phase. In most instances, switches in one vehicle or the other were out of configuration for the particular mode desired and the problem was solved by reconfiguration. (The Apollo speaker boxes in each vehicle caused a feedback squeal when operated simultaneously with the crew headsets.) Planned television procedures went very well with good quality coverage of the activities.

The general quality of the joint photography was good to excellent. Several types of photographs were taken: 16-mm motion pictures, 35-mm stills and 70-mm stills. All the interior photography of joint crew activities met or exceeded expectations including those shot in the docking module where lighting was poor. The 35-mm flash photographs are particularly good. Out-the-window photography (16-mm and 70-mm) of the Soyuz during rendezvous and docking, initial undocking, test docking, and final undocking are good with the exception of one 16-mm magazine taken of the

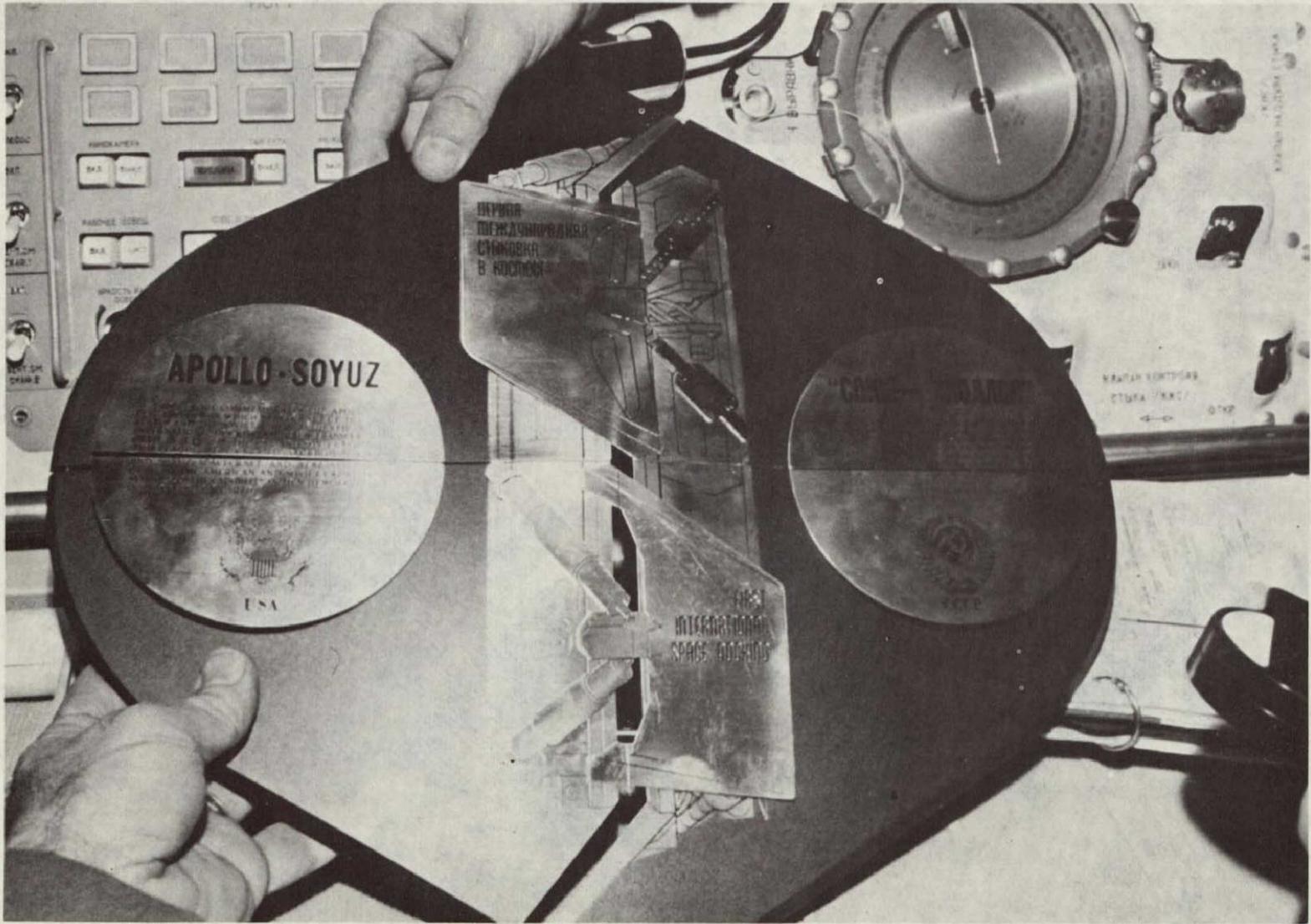


Figure 8-5.- Joining of commemorative plaque in the Soyuz spacecraft.



Figure 8-6.- Soyuz Commander.

9.0 BIOMEDICAL EVALUATION

This section summarizes the medical findings of the Apollo Soyuz mission based on the preliminary analysis of the biomedical data. The three crewmen accumulated 652.4 man-hours of space flight experience during the mission. All inflight medical objectives were successfully completed. All physiological parameters during the preflight and inflight medical evaluation periods remained within the expected ranges.

Exposure of the crew to nitrogen tetroxide vapors during the landing phase of the mission resulted in cancellation of the majority of post-flight medical protocols, and hospitalization of the crew at Tripler Army Medical Center in Honolulu, Hawaii.

The biomedical experiments that were conducted in conjunction with the mission are discussed in section 6.2. The results of limb volume measurements that were made to obtain additional data on body fluid shifts during space flight are included in this section. The results of other biomedical evaluations are contained in reference 2.

9.1 BIOMEDICAL INSTRUMENTATION AND PHYSIOLOGICAL DATA

All physiological measurements remained within expected limits.. Electrocardiographic and respiratory rate data were obtained through the bioinstrumentation data system during launch, on mission days 2, 7, and 8 in conjunction with the exercise periods, and during the entry phase of the mission. Table 9-1 summarizes the physiological data obtained from the biomedical instrumentation.

On mission day 2, interference of the Docking Module Pilot's exerciser harness with his bioinstrumentation electrodes resulted in poor quality data not suitable for analysis (sec. 14.1.12). On mission day 6, the Apollo Commander's biomedical data were not received in the Mission Control Center because of ground support technical difficulties. Although 5 minutes of instrumented periods of rest, exercise, and postexercise data were formally requested before flight, the lack of knowledge in real time of the exercise start and end times made it impossible to correlate the biomedical data with the actual activity periods.

No medically significant arrhythmias were detected during the mission. Isolated premature heart beats were observed in all three crewmembers. The fact that the frequency and character of these prematurities remained consistent with previously obtained data on these same crewmembers during ground-based studies indicates that they were not related to the space flight.

TABLE 9-I.- PHYSIOLOGICAL DATA SUMMARY

9-2

Mission phase	Heart rate, beats/min						Respiration rate, breaths/min		
	Commander		Docking Module Pilot		Command Module Pilot		Commander	Docking Module Pilot	Command Module Pilot
	Average	Peak	Average	Peak	Average	Peak			
Launch	105	130	95	123	82	117	14	14	9
^a Orbital flight									
Mission day 2	74	90	(b)	(b)	66	91	10	(b)	21
^c Mission day 6	--	--	--	--	--	--	--	--	--
Mission day 7	70	17	--	--	94	137	22	--	26
Mission day 8	--	--	74	140	--	--	--	16	--
Entry	96	150	67	91	76	124	24	16	13

^aValues obtained during exercise.

^bPoor data.

^cNo biomedical data received.

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9.2 INFLIGHT MEDICAL OBSERVATIONS

9.2.1 Adaptation to Weightlessness

All three crewmembers experienced the now classical fullness-of-the-head sensation immediately after earth orbital insertion; however, this symptom was mild and did not interfere with the crew's performance. The crew members commented that they did not experience sensations of nasal stuffiness or sinus congestion. Head movements and moving around the spacecraft did not intensify the feeling of fullness-of-the-head, and did not provoke symptoms of motion sickness. There were no instances of nausea, vomiting, disorientation, or loss of appetite.

9.2.2 Body Fluid Shifts

Major inflight headward fluid shifts have been observed on Apollo and Skylab crews. The most important data are pre- and postflight leg volume determinations calculated from multiple circumferential measurements. Inflight leg volume measurements were performed previously only on the Skylab third visit crewmembers, and leg volume decrements of a full liter (10 to 15 percent of preflight values) occurred as early as the third mission day. For this mission, inflight leg volume measurements were conducted on the crewmembers (1) to substantiate further this important finding, (2) to obtain earlier inflight volume determinations, and (3) to document the time course of headward fluid shifts.

A simple jig template with accompanying flexible tape measure similar to the limb volume measuring kit flown on the Skylab third visit was provided. The single difference from the Skylab kit design was reduction in the number of circumferential position measurements from 25 to 12, chiefly because of crew time constraints. These 12 template positions were optimized by preflight crew trials in order to preserve the 1- to 2-percent accuracy of the method.

The earliest inflight determination was obtained on the Command Module Pilot about 6 hours after Apollo lift-off (fig. 9-1). This value fell only 0.26 liter below the value obtained 1 day prior to launch, but was still a statistically significant volume reduction. By 32 hours after Apollo lift-off, all three crewmen had evidenced substantially greater inflight decrements (5 to 10 percent). All four of these values were obtained earlier than the previous earliest measurements (obtained during the Skylab third visit on mission day 3). Subsequent determinations revealed a definite downward trend. Most variations were parallel in all three crewmen despite the fact that no single crewmember performed all measurements. All inflight volumes dropped below the lower 95-percent confidence limit established by preflight volumes determined over a 45-day period.

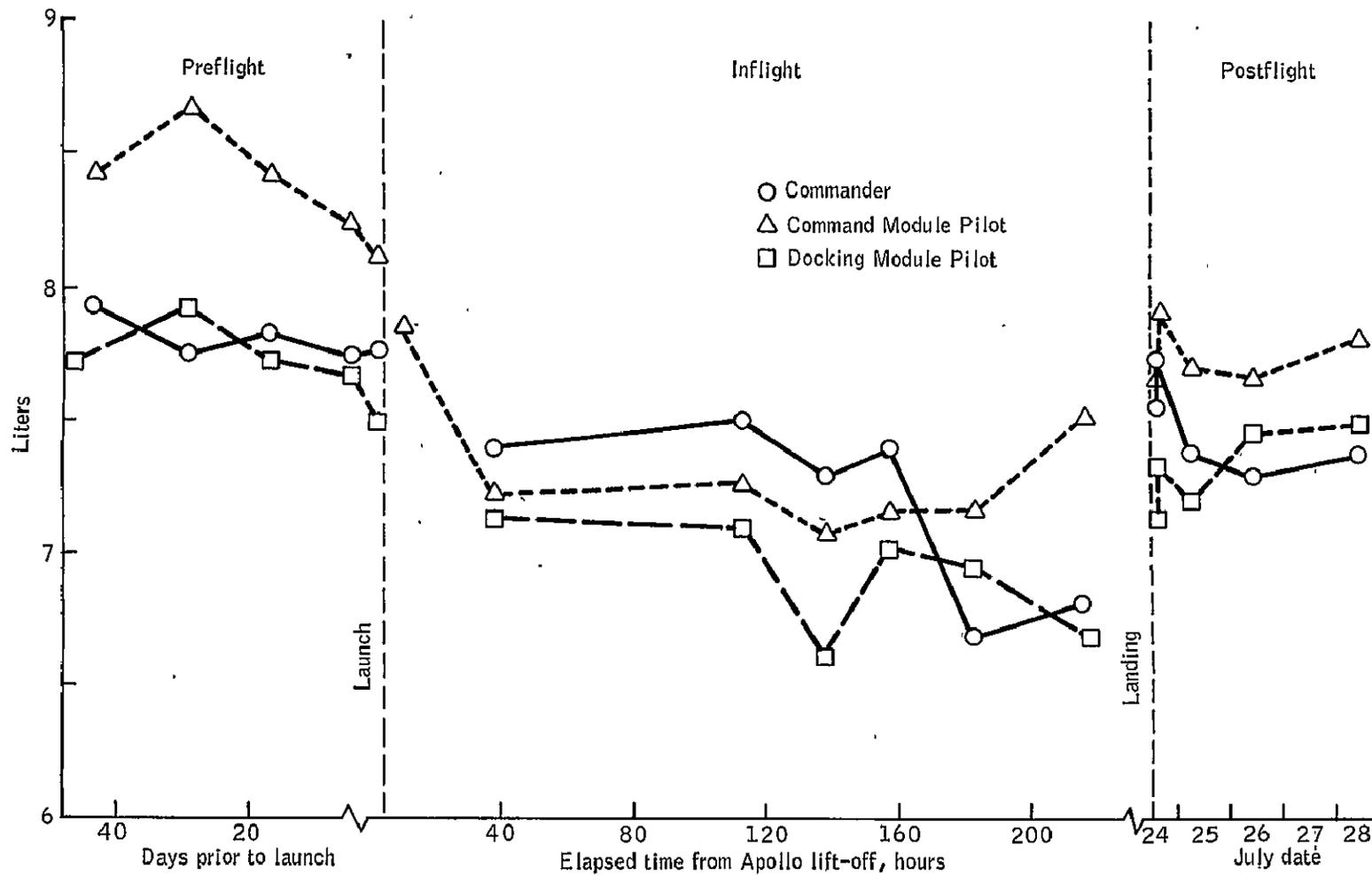


Figure 9-1.- Left leg-volume.

The earliest postflight determinations were obtained between 1 1/2 and 2 hours after landing, earlier than any such data have been acquired on United States space crews. Even so, leg volumes on all three crewmen had already increased well above the last inflight values. Volume measurements obtained a second time on recovery day, some 2 to 5 hours after landing evidenced yet greater volumes, reflecting the reversing effects of early readaption to earth gravity. Volumes on the day following recovery, July 25, were obtained prior to arising that morning and demonstrate clearly the established diurnal pattern of minimal volume at the end of the horizontal sleep period. Last postflight values, on July 28, reveal distinct increasing trends toward preflight values.

9.2.3 Crew Transfers .

No significant problems with the inflight docking module compression and decompression profiles were encountered during the mission. There was no evidence of adverse effects to crew health from the acetone-like odor and mild irritation of the eyes experienced by the crew on opening the docking module after docking with the Soyuz (sec. 4.1.8).

9.2.4 Medications

The medications taken by each crewman are as follows.

Medication	Apollo Commander	Docking Module Pilot	Command Module Pilot
Actifed (decongestant)	2	0	0
Lomotil (antiperistalsis)	7	2	2
Scopolamine-dextroamphetamine (anti-motion-sickness)	0	0	2
Aspirin	*2	0	0

*Number not definitely established.

On mission day 3, the Apollo Commander took three Lomotil tablets prophylactically in an attempt to decrease the frequency of inflight bowel movements. He took another two tablets on mission day 4 because of a loose bowel movement, and again took two Lomotil tablets prophylactically on mission day 8 prior to docking module jettison. The Docking Module Pilot and Command Module Pilot took two Lomotil tablets each prophylactically

on mission day 3. The Command Module Pilot took one scopolamine-dextro-amphetamine tablet prophylactically immediately after orbital insertion, and repeated the same dose approximately 5 hours later.

On entry day, approximately 1 hour after the sleep period, the Apollo Commander took two Actifed tablets prophylactically in order to prevent possible ear blockage during the entry phase. Since no significant medical problems requiring specific treatment occurred in flight, the medications used by each crewmember were minimal when compared to most previous space flights. No medications for sleep were taken at any time.

9.2.5 Sleep

Changes to the flight plan occasionally impacted planned crew sleep periods. In general, however, an adequate amount of restful sleep was obtained by all crewmembers. The estimate of sleep duration made by ground personnel was in general agreement with the subjective evaluations of the crew. On mission days 2 and 3, the crew averaged 3 to 4 hours of sleep. For the remaining days, the average amount of sleep was between 6 and 7 hours daily.

9.2.6 Radiation

One personal radiation dosimeter and one passive dosimeter each were assigned to the Apollo Commander, Command Module Pilot, and Docking Module Pilot. During the mission, the passive dosimeters assigned to the Apollo Commander and Docking Module Pilot were worn in the left leg pocket of the inflight coveralls and the one assigned to the Command Module Pilot was worn in the left thigh pocket. All flight dosimeters were recovered and returned for evaluation. The dosimeters indicated that the radiation dose absorbed by each crewman was 10 to 15 millirads per day. This value is among the lowest reported for any Apollo mission and approaches the minimum response sensitivity of the passive radiation dosimeters. The total space radiation exposure of the crewmen is insignificant from a medical standpoint. Details are given in the following paragraphs.

9.2.6.1 Passive dosimeters.- The flight passive dosimeters were disassembled and the component detectors were forwarded to the cognizant analysts. The component detectors consisted of thermoluminescent dosimeters, nuclear emulsions, neutron resonance foils, and Lexan track detectors. No evidence of contamination or component damage was observed during disassembly of the dosimeters.

The results from the thermoluminescent dosimeters are as follows.

Assignment	Mission dose, millirads
Apollo Commander	110 ±11
Command Module Pilot	108 ±11
Docking Module Pilot	100 ±10
Control	Less than 1

A preliminary analysis of the nuclear emulsions based on photodensitometry measurements and ratios of those measurements to proton doses determined from the previous Apollo earth-orbital missions agreed well with the thermoluminescent dosimeter data. The results are:

Crewman	Proton dose (estimated), millirads
Apollo Commander	102
Command Module Pilot	99
Docking Module Pilot	90

Analysis of the neutron resonance foils and Lexan track detectors will continue to provide a more accurate measurement of the crew dose from the trapped radiation environment.

9.2.6.2 Personal radiation dosimeters.- The mission doses are reported in the following table. The tolerances shown represent the 1 register count (0.01 rad) uncertainty inherent in digital readouts.

Crewman	Mission dose, rad
Apollo Commander	0.15 ±0.01
Command Module Pilot	5.58
Docking Module Pilot	0.12 ±0.01

A malfunction of the Command Module Pilot's dosimeter was observed during the flight and two dents in the aluminum housing, not present during pre-installation acceptance testing, were observed when the dosimeter was returned, indicating that the instrument had been dropped and/or struck. Failure analysis is not planned.

9.3 FOOD AND WATER

9.3.1 Food

Flight menus were designed to meet comparable individual earth-gravity energy requirements, specified nutrient levels, and crew-selected preferred foods. Energy requirements calculated for each crewman were 2815, 2760, and 2554 for the Apollo Commander, Command Module Pilot, and Docking Module Pilot, respectively. Based on crew menu acceptance, evaluations, and compatibility tests, an average daily caloric intake of 2820 kilocalories was provided for the Apollo Commander and Command Module Pilot, and 3165 kilocalories was provided for the Docking Module Pilot. Estimates of inflight food consumption based on daily reports indicate that averages of 2900, 3000, and 2867 kilocalories per day were consumed by the Apollo Commander, Command Module Pilot and Docking Module Pilot, respectively.

In order to meet the specified daily nutrient levels, some of the beverages were fortified with either calcium lactate or potassium gluconate. Calcium fortified beverages were limited to two per man per day, while only one potassium fortified beverage was required for each 4-day menu cycle. The crew selected a 4-day menu cycle as used previously on Apollo missions rather than a 6-day cycle which was used during Skylab. The average daily nutrient intakes for the proposed menus and estimated inflight food consumption for each crewman are shown in table 9-II.

In addition to the scheduled meals, a pantry containing beverages and snack foods was supplied. These foods could be used to substitute or supplement the normal meal items.

New foods for this mission included dehydrated compressed pea bars and spinach bars; irradiated breakfast rolls; thermostabilized/irradiated turkey, corned beef and charcoal broiled steak; thermostabilized cranberry sauce, tuna and salmon in cans which required a can opener; commercial cookies and graham crackers; dehydrated beef patty, pears, and potato patty; and intermediate moisture almonds, cheese slices and dried beef jerky.

In general, the crew was satisfied with the quality and quantity of flight food provided. No gastrointestinal problems were encountered during the mission. Inflight appetites were reported to be the same as during the preflight period. The Command Module Pilot reported changes in the taste of foods during flight, and indicated that salty foods tasted best to him. As on previous Apollo missions, the crew reported gas in the hot water supply which interfered with complete rehydration of the food. Throughout the mission, high priority activities and work schedules frequently precluded adequate time for meal preparation and food consumption.

TABLE 9-II.- AVERAGE DAILY NUTRITIONAL INTAKE

Crewman	Energy, kilocalories	Protein, grams	Calcium, milligrams	Phosphorus, milligrams	Sodium, milligrams	Potassium, milligrams	Magnesium, milligrams
Proposed Intake							
Commander	2820	99.7	1076	1832	4983	2942	313
Command Module Pilot	2820	98.1	1458	1996	4724	2984	288
Docking Module Pilot	3165	112.2	1375	2113	6402	3745	355
^a Estimated Actual							
Commander	2900	98.0	1295	1830	4970	2983	299
Command Module Pilot	3000	101.9	1661	2071	5318	2975	290
Docking Module Pilot	2867	107.7	1422	1964	6079	3748	322

^aAverage of 7 nominal days; incomplete days have been omitted.

9.3.2 Water

Postflight crew comments indicated that the potable water was of good quality. No out-of-specification conditions were noted in the microbiological and chemical analyses conducted.

Preflight chlorination was accomplished at 19 hours before launch and the level of chlorine measured 2 hours later was sufficient for microbiological control.

Inflight chlorinations were accomplished approximately on schedule and no inflight problems were experienced. As in previous flights, some gas was present in the water - particularly the hot water.

Postflight analyses indicated a lack of residual chlorine in the potable water. This remains unexplained since the records indicate that the last inflight chlorination was accomplished 17 hours prior to landing. Chemical analysis of postflight samples showed all levels within specification limits. Microbiological results were positive in that *Flavo* bacterium species at levels of 10^5 organisms/milliliter were detected.

9.4 PREFLIGHT PHYSICAL EXAMINATIONS

Comprehensive medical examinations were conducted 30, 15 and 5 days before launch. Additional abbreviated physical examinations were conducted daily starting 3 days prior to launch. No significant deviations from the normal were observed.

9.5 CREW EXPOSURE TO NITROGEN TETROXIDE FUMES AND POSTFLIGHT PHYSICAL EXAMINATIONS

During the entry phase of the mission, all three crewmembers were inadvertently exposed to toxic fumes of nitrogen tetroxide. Initial medical examinations failed to reveal significant abnormalities. Because of the potentially serious exposure to the nitrogen tetroxide, all scheduled biomedical procedures and all official social functions were cancelled. After showers and the evening meal on recovery day, the crewmen were transferred to the primary recovery ship sick bay for sleep and observations. On the following day, signs of chemical pneumonitis were detected on the chest roentgenograms of all three crewmen. The crew were forthwith transferred to Tripler Army Medical Center for further medical care, where they remained until July 30, 1975. After discharge from the hospital, the crew

remained in Hawaii until August 7, 1975, for convalescence and further medical observation. They were subsequently pronounced fit to return to regular duty. Final medical examinations conducted 30 days after recovery failed to reveal any residual effects of the exposure to nitrogen tetroxide.

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Apollo and Soyuz flight crews

Astronaut Donald K. Slayton, Docking Module Pilot, Astronaut Thomas P. Stafford, Apollo Commander, Astronaut Vance D. Brand, Command Module Pilot, Cosmonaut Alexei A. Leonov, Soyuz Commander, and Cosmonaut Valeri N. Kubasov, Flight Engineer.

slightly to one side, he could observe a faint vertical line for roll reference. Closure was made using this reference and a soft dock was obtained. The retraction sequence was then initiated and a hard dock was obtained. The entire timeline from then on, including the docking module extraction and the evasive maneuver, was normal.

10.1.3 Rendezvous and Docking

All rendezvous maneuvers were nominal and the crew was confident that things would go well because of the many simulations that had been conducted. The two midcourse maneuvers after transfer phase initiation were probably the smallest yet encountered. The first was 0.2 meter per second; the second was 0.4 meter per second. The line-of-sight rates were completely nulled, and the Commander used the control mode of autopilot on and the spacecraft under automatic control for the 1.85-kilometer and 0.93-kilometer braking gates to save fuel. Even at the 0.46-kilometer gate, there was no relative movement of the Soyuz. A slight relative movement was subsequently detected and was immediately corrected for.

After a period of stationkeeping, a flyaround over the top of the Soyuz was initiated. As the Apollo spacecraft flew above the Soyuz, the reticle pattern completely disappeared when looking down at the earth. As planned, stationkeeping was continued above the Soyuz and looking down, holding inertial attitude, with the Soyuz slowly moving up relative to the horizon. When the Soyuz approached the horizon, the reticle pattern reappeared and the Commander was able to hold a more precise position.

The Soyuz went to its programmed inertial attitude in pitch and then completed the roll maneuver. At that time, Apollo was ready to close for docking. The Commander initiated the final closing maneuver with very precise alignment. At the point of contact he estimated that he had a closing velocity somewhat in excess of 0.1 meter per second. The docking was very soft. Simultaneous indications were obtained of both contact and capture. The Commander went to the control mode of autopilot on and the spacecraft under manual control, and the reticle was aligned precisely with the Soyuz docking target standoff cross. After hard docking was completed, the Commander observed that the reticle pattern (the center of the crosshairs) was aligned with the center of the bolt that fastened the standoff cross; thus, the alignment was very good. The crew had earlier been concerned about the accuracy of the fixtures that had been used to align the docking targets between the two spacecraft.

10.1.4 Joint Activities

The period of joint activities with the Soviet crew commenced with the first docking and was completed on mission day 5 when the Apollo and Soyuz spacecraft separated. The entire phase proceeded in a nominal manner. Fortunately, it was not necessary to use many of the backup procedures that were available in the event of slippage in this part of the timeline.

The transfers between spacecraft are well documented by television, motion picture, and still camera film. Activities consisted of initial greetings, talks with heads of state, meals, presentations, signing of documents, exchanges of seeds and medallions, joint experiments, exercise periods, out-the-window tours, descriptions of the two spacecraft, and other activities. Additional details are given in section 8. At the end of the first transfer the crews lagged behind the planned timeline because of unscheduled activities, but this was not a serious problem. On the second day of joint activities, the crew made an extra effort to stay on the planned timeline to insure that all phases of the joint activities would proceed as planned.

In the way of preparation, two things were invaluable for the joint activities. First, the Soviet and American crews were well acquainted. They knew what to expect from each other and worked together in a relaxed manner. Second, ground-based training in mockups served the intended purpose of preparing both crews for the joint activities. The flight plan, including the television plan, worked well.

The ultraviolet absorption experiment was performed on mission day 5 and the Apollo trajectory was nominal. Data could not be obtained during the 150-meter ultraviolet absorption sequence, so it was necessary to change to the Soyuz aft reflector for the 500-meter sequence. This required modification of the Soyuz maneuvering during the 500-meter run. Data collection for the 500-meter and the 1000-meter phases of the experiment went well. After separation, the Soyuz went below and ahead of Apollo. The crews were able to chat for a few hours until spacecraft separation exceeded line-of-sight VHF range. From the standpoint of the Apollo crew, the joint phase of the flight, a most important 3 days, went well.

10.1.5 Solo Orbital Activities

Solo orbital activities were those which were performed between Apollo/Soyuz separation and the beginning of entry. The solo activities proceeded in a nominal manner and the experiment operations were accomplished essentially as planned. The only major exception was the X-ray telescope. A sensor problem required the crew to perform some additional switching.

The visual observations experiment was performed from a very low orbit - especially when compared to Skylab. As a result, there was little time to acquire, discuss, and photograph planned sites; but it was easier to see objects on the ground. The following ideas are recommendations in the visual observations area:

a. Spacecraft attitude should be such that the astronaut is viewing the earth forward and beneath him in a heads-up rather than a heads-down attitude.

b. The Shuttle visual observations program should be thought of in terms of a Block 1 and Block 2 system. Block 1 would be simply an observer with equipment to assist his viewing. The Block 2 system would employ a turret or observation bubble and the astronaut would be used, primarily, as a pointer of equipment and, secondarily, as a sensor for collecting data.

c. Cameras such as the 70-mm reflex should have an automatic exposure meter to automatically adjust the f-stop. In addition, the camera should have a conventional manual f-stop adjustment capability for special applications such as cloud photography. There should be a continuing attempt to develop films that have improved color resolution for out-the-window photography.

d. The use of television cameras for earth observations probably offers great growth potential. Television camera improvements are needed for better spatial and color resolution.

e. A color measurement device is needed to replace the color wheel. The color wheel was not very effective simply because the hues of the color samples on the wheel changed with lighting conditions.

10.1.6 Entry

Entry activities down to 15 240 meters were essentially normal. This includes the deorbit firing, command module/service module separation, activities at 0.05g, and descent into the lower atmosphere. At around 15 240 meters, the crew noticed an increase in the cabin noise level, and at 9140 meters, the automatic portion of the earth landing system was not armed. The forward heat shield, drogue parachutes, and main parachutes were deployed manually by means of pushbuttons. An irritating gas invaded the cabin at about the time of drogue parachute deployment and was in the cabin until after landing (sec. 14.1.2).

10.1.7 Landing and Recovery

Landing was slightly harder than expected, and the command module immediately turned to the stable II position. The crew donned oxygen masks after landing, and the command module was uprighted to the stable I position. The side hatch was opened following flotation collar attachment and the gas was cleared from the cabin. Later, the hatch was closed again so that the command module could be lifted aboard the aircraft carrier. The crew received no communications from the ground from approximately 15 240 meters to landing. They had the impression that their transmissions were not being received by the recovery forces.

10.2 FLIGHT PLANNING

The flight plan was written assuming a minimum of distraction and that there would be few hardware failures. Fortunately, this was the case, and it was possible to carry out the flight plan without too much real-time change. It would have been nice had the flight plan included time for personal hygiene. Although the crew always received less sleep than was allotted in the flight plan, flight planning was considered adequate regarding sleep.

10.3 HOUSEKEEPING

The docking module added much needed volume for this mission and, in addition to its other uses, served as a general exercise area, storeroom, and experiment room. Living and operating in the command module was slightly more difficult than expected - mainly because many activities which had been planned to be conducted in parallel had to be accomplished in series. In future designs, an attempt should be made to simplify housekeeping as much as possible to allow more time for experiments and important flight operations. For example, future communications cables should be lightweight, easy to plug in, and should require no switching when plugged and unplugged.

10.4 MEALS

The food system was adequate but required much of the crew's time. Future systems ideally should be designed around a concept such as TV trays and a microwave oven to save time. The crew followed a low residue

diet beginning 3 days before the flight and considers that appropriate for future short missions. The water dispensing panel was a bottleneck in the food preparation process, since only one man could use this panel at a time.

10.5 TRAINING

10.5.1 General

The amount and quality of crew training was sufficient and the training system worked well. The fidelity of the command module simulator was very good except in the area of optical systems. Perhaps the optical systems did not represent real life very well simply because the command module simulator was becoming very old by the time it was used for the Apollo Soyuz Test Project. Technicians maintained the command module simulator in good working condition - and did very well in this regard. The instructor crews did a good job and showed once again that the concept of having instructors on a console outside the cockpit worked well. Mockups and trainers were quite adequate and were a real necessity for this mission. Mockups were representative of the flight hardware.

Joint training with the Russians was generally adequate. If such a project were to be undertaken again, a similar joint training plan should be pursued. Training trips to the USSR, although the training was not as concentrated as that at home, were quite necessary. It simply was good to go there to see Russian facilities and personnel, to get a glimpse of the flight hardware, and to become better acquainted with the cosmonauts. Additionally, it was good to touch base with the cosmonauts every few months during the training phase so that operational questions and any misunderstandings could be resolved. A comfortable working relationship was developed through these visits to the USSR and the cosmonaut visits to the United States. This relationship paid off during the flight.

10.5.2 Russian Language

About one-third of the training hours for the Apollo crew in the 16 months before the flight were spent in language training. Most crewmembers received 600 to 1000 hours of language training conducted by NASA. Additional training had been obtained earlier by the prime crewmembers about 2 years before the flight. Experimentation and iteration were required to determine the best organization and means for giving the NASA instruction. Finally, four instructors were located full-time in the astronaut area and they concentrated their efforts in preparing the astronauts for the mission. The texts were those used in the basic Russian

course at the Defense Language Institute, Monterey, California. Much instruction was given on a one-to-one basis. In addition, supplementary work was done in the area of technical vocabulary with emphasis on vocabulary needed for joint activities with the Soviets. Occasional trips to the USSR were helpful for practice in everyday speaking. Should there be a similar mission in the future, the crew recommends an intensive language course starting at least 2 years before launch. Having in-house instructors is a recommended approach. Defense Language Institute textbooks are recommended. The Apollo crew had sufficient language instruction to perform the mission but needed every hour of Russian training that was received - and would not have wanted less.

11.0 MISSION SUPPORT PERFORMANCE

11.1 FLIGHT CONTROL

Flight control provided satisfactory operational support. The pre-flight plan for coordination of joint flight activities between the Houston and Moscow control centers was followed closely during the mission with excellent results, and the primary objective of control center interaction was successfully achieved. Continuous manning of both control centers began on July 14, 1975, approximately 24 hours prior to the Soyuz launch. Visiting specialist teams were exchanged to provide support to the host country regarding spacecraft and operations. The primary means of exchanging voice information between control centers was through joint flight directors who were in continuous communication. Each joint flight director negotiated flight plan and procedures changes, and provided status information to the flight director at the other control center. Since most of the problems that were encountered are discussed elsewhere in this report, only those problems that are unique to flight control or have operational considerations not previously mentioned are present in this section.

Because of the actual Soyuz orbital inclination, the Apollo launch azimuth was changed from 0.789 to 0.783 radian to adhere to the nominal launch time of 19:50:00 G.m.t.

Docking module checkout required more time than planned, and the crew was 1-1/2 hours behind at lunchtime on the second day. To recover lost time, several activities were deleted. These included the first visual observations pass, the extreme ultraviolet survey (MA-083) raster scan, and the visual observations portion of the second earth observations experiment period. All other planned activities for the second day were accomplished.

A switching device in the Soyuz television system failed. The cosmonauts performed a corrective procedure which recovered the use of two interior color television cameras.

Numerous operating instruction changes were made for the experiments located in service module bay 1. Most of the changes were due to the failure in the soft X-ray experiment (MA-048) hardware. Several alternate/contingency pads were utilized.

The spacecraft attitude for visual observations passes was modified at crew request. The premission attitude was heads down, posigrade, local horizontal. The new attitude pitched the docking module 524 milliradians toward the earth.

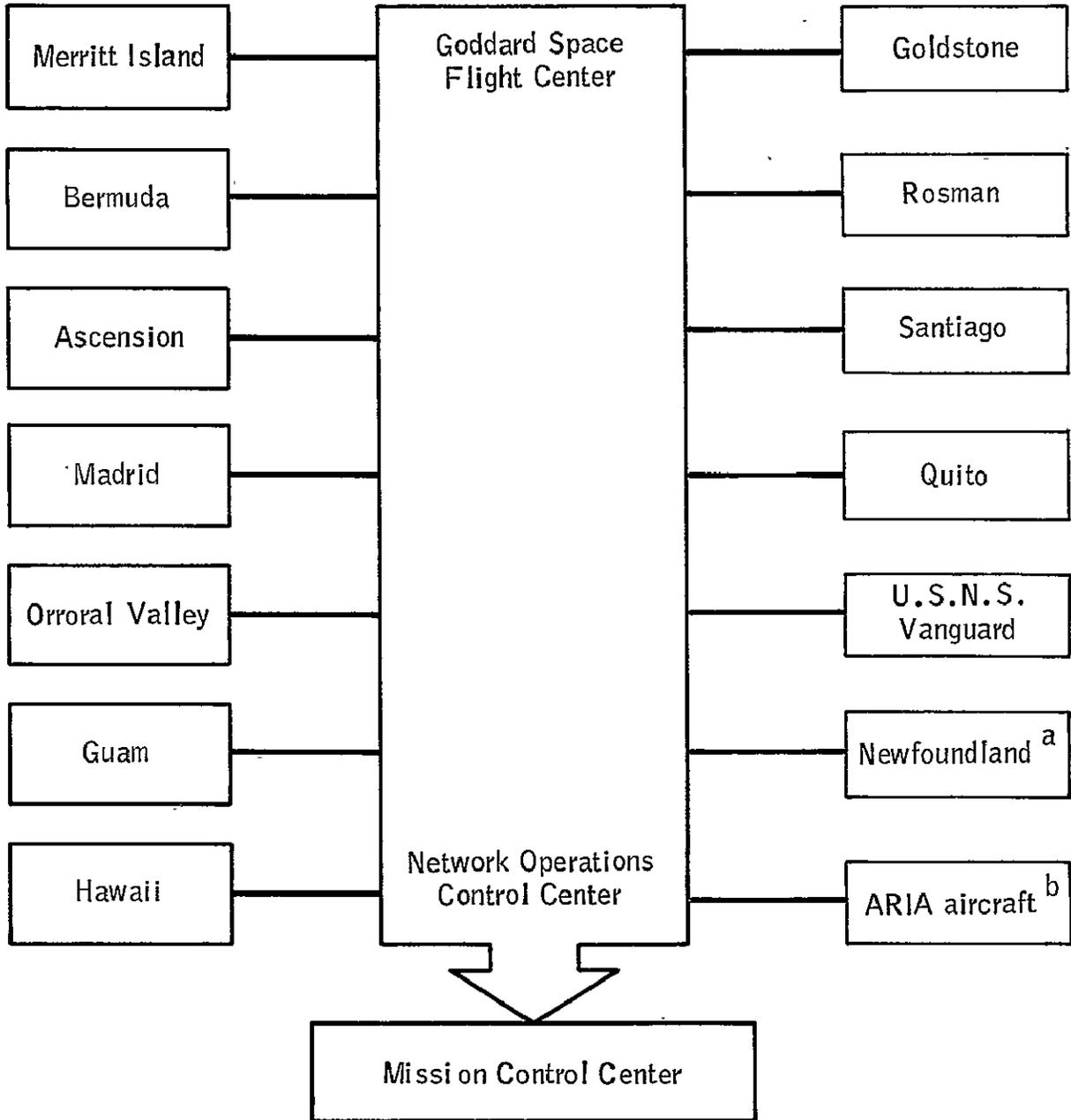
11.2 NETWORK

The Spaceflight Tracking and Data Network consisted of the S-band stations shown in figure 11-1 and the C-band stations shown in figure 11-2. The S-band network was similar to that used for Skylab except that the Canary Island; Honeysuckle, Australia; and Tananarive stations were not used, and stations at Rosman, North Carolina; Santiago, Chile; Quito, Ecuador; and Orroral Valley, Australia, were used for the first time in support of a manned mission. The C-band network was considerably expanded over that used on the Skylab and the later Apollo missions.

The most significant additions to the Spaceflight Tracking and Data Network were (1) the use of Applications Technology Satellite 6 (ATS-6) for relay of S-band communications between the command and service module and the satellite ground station near Madrid, Spain, and (2) the provision of television, voice, teletype, and facsimile facilities between the United States and Soviet control centers. The ATS-6 extended communications with the command and service module as shown in figure 11-3. The Spaceflight Tracking and Data Network coverage alone would have provided communications with the command and service module 17 percent of the time, while the combined Spaceflight Tracking and Data Network and ATS-6 provided communications 63 percent of the time. ATS-6 supported as expected and no significant problems were experienced. The USA/USSR mission control center interface included 14 voice circuits, 3 teletype circuits, and television services via commercial facilities. Temporary circuit outages were experienced as expected but there were no significant problems.

The Spaceflight Tracking and Data Network went on full mission status on June 30, 1975, and went off mission status on July 24, 1975. The network hardware, software, personnel and procedures performed very well throughout the mission with only minor problems. No significant loss of data or mission support failures were experienced.

An orbital navigation via synchronous satellite relay data investigation was made during the mission to obtain direct and relay (ATS-6/CSM/ATS-6) synchronous satellite navigation tracking data to support the Space Shuttle tracking and data relay satellite system design. A total of 28 revolutions of relay tracking data were planned. During the mission, data were obtained for these revolutions plus additional data for 67 partial revolutions.



^a VHF - air to ground/UHF - command only

^b VHF/Unified S-band - air to ground only

Figure 11-1.- S-band ground support network.

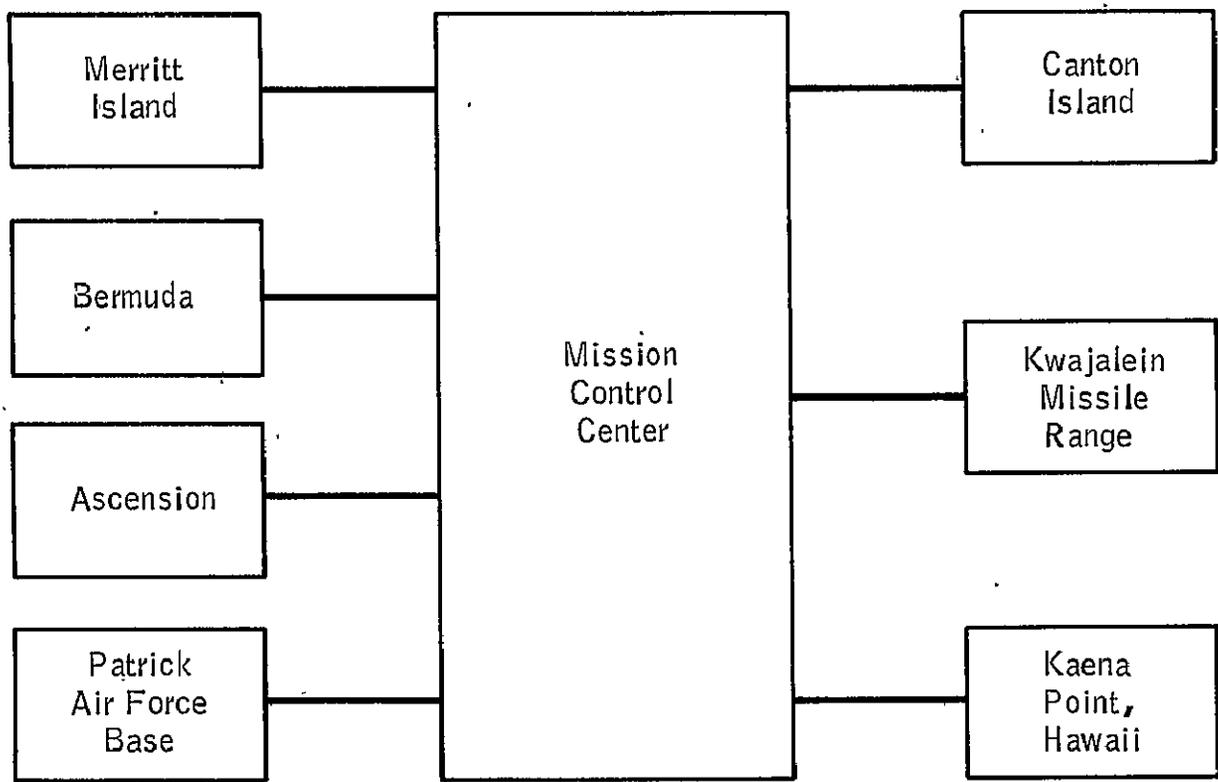


Figure 11-2.- C-band radar network.

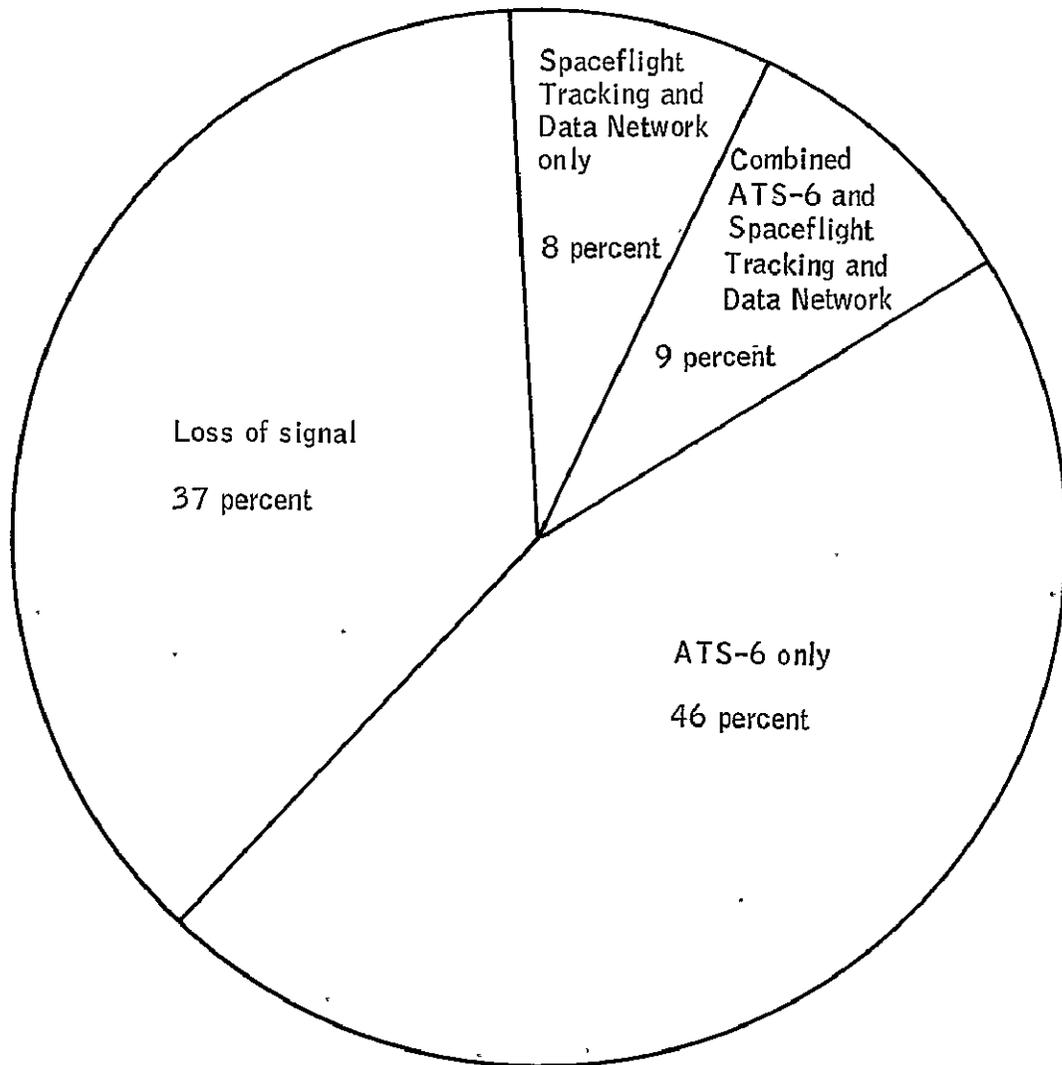


Figure 11-3.- Communications with command and service module for 24-hour period.

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11.3 RECOVERY OPERATIONS

The Department of Defense provided recovery support as requested by the National Aeronautics and Space Administration. The recovery force support is outlined in table 11-I.

11.3.1 Prelaunch Through Orbital Insertion

The primary recovery ship - the *U.S.S. New Orleans* - departed from San Diego, California, on July 5, 1975, and arrived at Pearl Harbor, Hawaii, on July 12. Recovery personnel trained while en route to Hawaii. The ship departed for mission support on July 13. Twenty-four hours prior to scheduled launch, the recovery forces reported under the command of the Department of Defense Manager for Manned Space Flight Support except for the primary recovery ship which reported on July 5. After orbital insertion on July 15, the recovery forces were placed on alert or released, as appropriate.

11.3.2 Orbital Operations

The primary recovery support from orbital insertion through the end of the mission consisted of the primary recovery ship, which provided on-scene support for a daily go/no-go target point, and HC-130 aircraft at Hickam AFB, Hawaii. In addition, air rescue units located at various bases around the world were prepared to provide support should a contingency occur.

11.3.3 End-of-Mission Support

Recovery support for the end-of-mission area off Hawaii was provided by the *U.S.S. New Orleans*. Air support consisted of five SH-3A helicopters from the primary recovery ship and two HC-130 rescue aircraft staged from Hickam AFB, Hawaii. Three of the helicopters carried U.S. Navy swimmers. The first, designated "Recovery," also carried the flight surgeon and was used for command module retrieval operations. The second helicopter, designated "Swim," served as backup to "Recovery" and aided in the retrieval of the forward heat shield. The third helicopter, designated "ELS" (earth landing system), aided in the retrieval of the three main parachutes. The fourth helicopter, designated "Photo," served as a photographic platform for motion picture photography and live television. The fifth helicopter, designated "Relay," served as a communications relay aircraft. The HC-130 aircraft, designated "Hawaii Rescue 1" and "Hawaii Rescue 2," were positioned to track the command module after it exited

TABLE 11-I.- RECOVERY FORCE SUPPORT

Ship/aircraft	Number	Ship name/aircraft staging base	Responsibility
Ships			
LPH	1 ^a	<i>U.S.S. New Orleans</i>	Primary recovery ship
ARS	1 ^a	<i>U.S.S. Preserver</i>	Launch site recovery ship
Operational Support Aircraft			
HH-53	3 ^a	Patrick AFB, Florida	Launch site area. Provided short access time for early launch aborts.
HC-130	2 ^a	Pease AFB, New Hampshire	Supported for launch aborts.
HH-3	1 ^b	Pease AFB, New Hampshire	Provided minimum crew retrieval time in West Atlantic launch abort area.
HH-3	1 ^b	Gander International, Newfoundland	Provided minimum crew retrieval time in West and Mid-Atlantic areas for launch aborts.
HH-53	1 ^b	Woodbridge RAF, England	Provided crew retrieval time in East and Mid-Atlantic areas for launch aborts.
HC-130	1 ^a 1 ^b	Woodbridge RAF, England	Provided launch abort and early earth orbital landing support (active), and contingency de-orbit support (on alert) in East and Mid-Atlantic areas.
HC-130	2 ^a 1 ^b	Hickam AFB, Hawaii	Provided daily go/no-go and contingency landing support in Mid-Pacific area (on alert), and end-of-mission landing area support (active).
HC-130	1 ^b	McClellan AFB, California	Provided contingency support in East Pacific area.
HC-130	1 ^b	Eglin AFB, Florida	Provided contingency support in the West Atlantic area.
HC-130	1 ^b	Kadena Air Base, Japan	Provided contingency support in the East Pacific area.
WC-130	1 ^a	Hickam AFB, Hawaii	Provided weather reconnaissance support in end-of-mission area.

^aActive.
^bOn alert.

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TABLE 11-I.- RECOVERY FORCE SUPPORT - Concluded

Ship/aircraft	Number	Ship name/aircraft staging base	Responsibility
Logistics Support Aircraft			
C-5	1 ^a	Travis AFB, California	Transported mobile laboratory, experiment equipment, and passengers from Ellington AFB to North Island NAS on July 1, 1975.
H-53	2 ^a	Hickam AFB, Hawaii	Transported experiment equipment, isotopes, and passengers from Hickam AFB to the primary recovery ship on recovery minus 2 days.
H-53	2 ^a	Hickam AFB, Hawaii	Transported passengers from Hickam AFB to the primary recovery ship on recovery minus 1 day.
SH-3	2 ^a	<i>U.S.S. New Orleans</i>	Transported flight and recovery film, and passengers from primary recovery ship to Hickam AFB.
SH-3	2 ^a	<i>U.S.S. New Orleans</i>	Transported experiment MA-151 and passengers from primary recovery ship to Hickam AFB.
ARIA	1 ^a	Hickam AFB, Hawaii	Transported film from Hickam AFB to Ellington AFB and to Patrick AFB.
C-5	1 ^a	Travis AFB, California	Transported mobile laboratory, experiment equipment and passengers from Hickam AFB to Ellington AFB.

^aActive.

from S-band blackout, as well as to provide pararescue capability and to optimize the capability for locating the command module had it landed up-range or downrange of the target point. Figure 11-4 shows the relative positions of the recovery ship, its aircraft, and the HC-130 aircraft prior to landing. The figure also shows the target point, the landing point indicated by the command module computer, and the landing point estimated by the recovery ship.

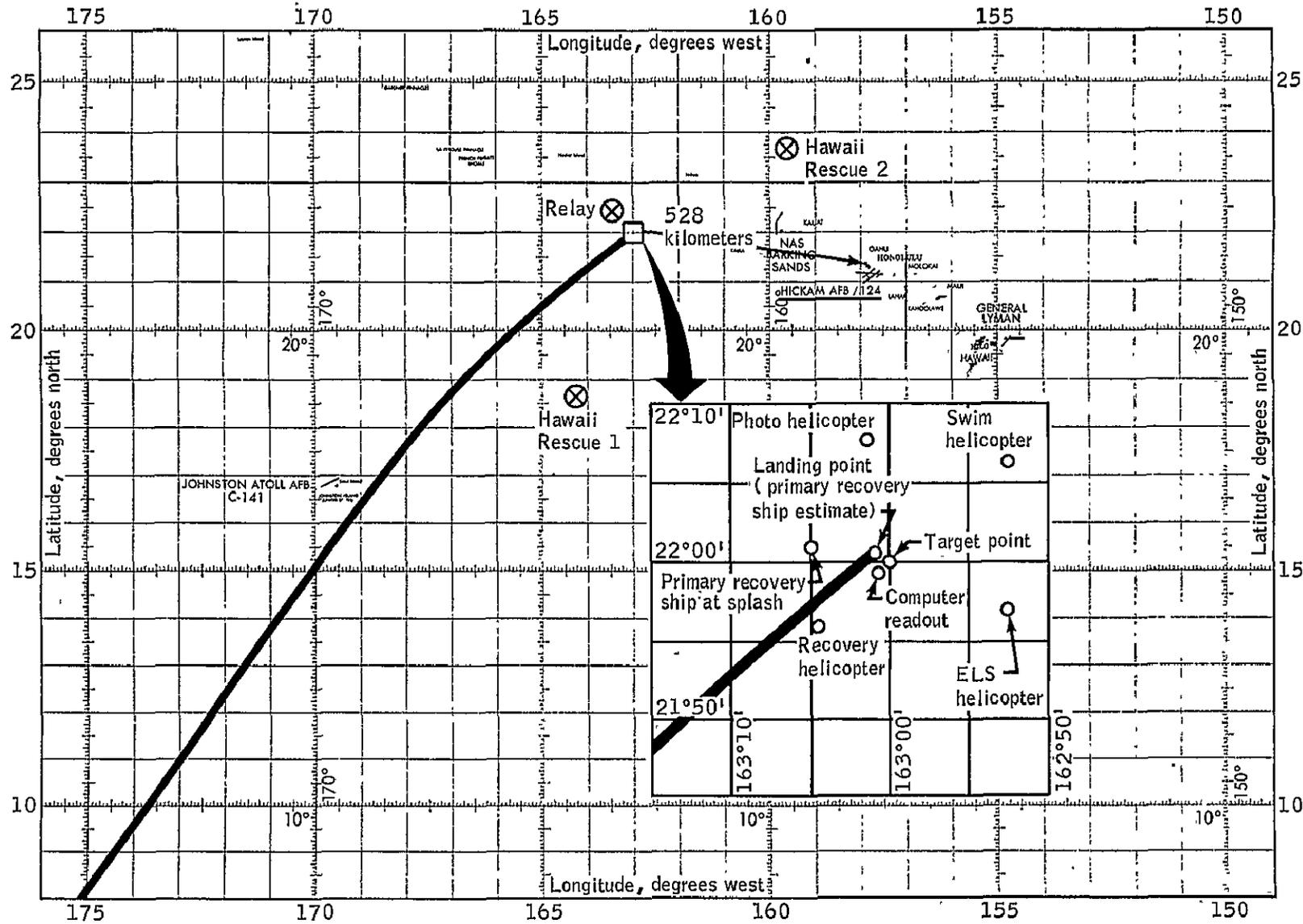
11.3.4 Command Module Location and Retrieval

Weather was good in the end-of-mission area on recovery day. At landing, the lowest cloud coverage was 10 percent at 450 meters and the total cloud coverage was 50 percent. Winds were 4.6 meters per second from 1.22 radians referenced to true north. The sea state was 0.3-meter waves at 1-second intervals on top of 0.9-meter swells at 4-second intervals, and the air and water temperature was 299° K.

The command module landed at 21:18:24 G.m.t. on July 24, 1975. Based upon navigation satellite fixes, the *U.S.S. New Orleans* position at the time of landing was calculated as latitude 22 degrees 0 minutes 54 seconds north and longitude 163 degrees 5 minutes 24 seconds west. Using the calculated ship position, visual bearings, and radar to determine range, the landing point coordinates of the command module were determined to be latitude 22 degrees 0 minutes 36 seconds north and longitude 163 degrees 0 minutes 54 seconds west. As the spacecraft descended on the main parachutes, the computer readout was latitude 21 degrees 59 minutes 24 seconds north and longitude 163 degrees 0 minutes 36 seconds west.

The command module went to the stable II attitude upon landing and was righted in 4 minutes and 24 seconds by the uprighting bags. The swimmers were deployed to the command module and the flotation collar was installed and inflated. Because of the presence of nitrogen tetroxide in the cabin (sec. 14.1.2), the flight crew opened the hatch for a period while the command module was still on the water to allow fresh air into the cabin. The flight crew remained inside the command module while it was hoisted aboard ship. The crew egressed from the command module, participated in a welcome-aboard ceremony, and entered the mobile medical laboratory. The flight crew remained aboard the *U.S.S. New Orleans* until it arrived in port at Pearl Harbor, Hawaii, at 18:30 G.m.t. on July 25. About an hour later, the crewmen were transported by automobile to Tripler Army Hospital in Honolulu, Hawaii, where they received additional treatment for exposure to the toxic gas. The times of these and subsequent events are given in table 11-II.

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Figure 11-4.- Recovery force deployment at end of mission.

TABLE 11-II.- RECOVERY EVENT TIMELINE

Event	Time, G.m.t.	Time relative to landing, day:hr:min
	<u>July 24, 1975</u> (recovery day)	
VHF voice contact (receive only)	21:10	-0:00:08
Radar contact by primary recovery ship	21:11	-0:00:07
Visual contact by "Photo" helicopter	21:14	-0:00:04
Command module landing (21:18:24.4 G.m.t.)	21:18	0:00:00
Command module in stable I attitude (21:22:48 G.m.t.)	21:23	0:00:05
Flotation collar inflated	21:33	0:00:15
Flight crew and command module aboard primary recovery ship	21:59	0:00:41
^a Hatch opened	22:07	0:00:49
Flight crew in mobile laboratory	22:29	0:01:11
	<u>July 25, 1975</u> (recovery plus 1 day)	
Time-critical experiment removal completed/reaction control system depressurization started	3:42	0:06:24
Reaction control system depressuri- zation completed	8:30	0:11:12
Primary recovery ship arrived Pearl Harbor	18:30	0:21:12
Flight crew departed ship	19:39	0:22:21
Flight crew arrived Tripler Army Hospital	19:55	0:22:37
Mobile laboratory offloaded from ship	22:00	1:00:42
Command module offloaded from ship	22:30	1:01:12
Command module arrived deactiva- tion site at Hickam AFB	23:45	1:01:27
	<u>July 26, 1975</u> (recovery plus 2 days)	
Experiments offloaded from primary recovery ship	1:00	1:03:42
Mobile laboratory and experiments loaded on C-5 aircraft	2:45	1:05:27
C-5 departed Hickam AFB	5:00	1:07:42
C-5 arrived Ellington AFB	13:22	1:16:04
Mobile laboratory in place at JSC	20:50	1:23:32
Experiments delivered to JSC	21:00	1:23:42

^aFlight crew opened hatch while the command module was still on the water.

TABLE 11-II.- RECOVERY EVENT TIMELINE - Concluded

Event	Time, G.m.t.	Time relative to landing, day:hr:min.
Command module deactivation completed	July 29, 1975 (recovery plus 5 days) 22:00	5:00:42
Flight crew transferred from Tripler Army Hospital to Kaneohe Marine Corps Air Station	July 30, 1975 (recovery plus 6 days) 1:00	5:03:42
Flight crew departed Kaneohe Marine Corps Air Station	August 7, 1975 (recovery plus 14 days) 20:00	13:22:42
Flight crew departed Honolulu	21:45	14:00:27
Flight crew arrived Washington, D.C.	August 8, 1975 (recovery plus 15 days) 20:50	14:23:22
Flight crew departed Washington, D.C.	August 10, 1975 (recovery plus 16 days) 14:30	15:17:02
Flight crew arrived Ellington AFB	19:00	15:21:32
^b Command module departed Pearl Harbor	August 13, 1975 (recovery plus 20 days) 17:57	19:20:39
Command module arrived San Diego, California	August 20, 1975 (recovery plus 27 days) 17:30	26:20:06
Command module arrived prime contractor's facility, Downey, California	August 21, 1975 (recovery plus 28 days) 20:00	27:22:36

^bVia U.S.S. Okinawa.

The command module was offloaded at Pearl Harbor and transported to the deactivation site at Hickam Air Force Base, Hawaii. After deactivation, the command module was returned to the continental United States on the *U.S.S. Okinawa*.

12.0 ASSESSMENT OF MISSION OBJECTIVES AND
TECHNICAL INVESTIGATIONS

The Apollo Soyuz Test Project objective was to develop and test systems for rendezvous and docking of future manned spacecraft and stations that would be suitable for use as a standard international system. This objective was further detailed and defined by the primary and supplementary objectives listed in table 12-I. In addition to the joint USA/USSR objectives, several unilateral experiments and technical investigations were performed as reflected in table 12-I.

TABLE 12-I.- SUMMARY OF MISSION OBJECTIVES AND TECHNICAL INVESTIGATIONS

Objectives/Investigations	Achievement status
Primary Objectives:	
1. Spacecraft Rendezvous	Yes
2. Spacecraft Docking and Undocking	Yes
3. Intervehicular Crew-Transfer	Yes
4. Interaction of Control Centers	Yes
5. Interaction of Spacecraft Crews	Yes
Supplementary Objectives:	
1. Docked Spacecraft Attitude Control	Yes
2. Radio and Cable Communications	Yes
3. Test Docking and Undocking	Yes
4. Onboard Documentary Photography	Yes
5. Television Transmission	Yes
6. Joint Experiments (see Experiments)	Yes
Experiments:	
AR-002 Microbial Exchange (Joint)	Yes
MA-007 Stratospheric Aerosol Measurement	Yes
MA-010 Multipurpose Furnace (Joint ^a)	Yes
MA-011 Electrophoresis Technology	^b Partial
MA-014 Electrophoresis - German	Yes
MA-028 Crystal Growth	Yes
MA-048 Soft X-Ray	^c Partial
MA-059 Ultraviolet Absorption (Joint)	Yes
MA-083 Extreme Ultraviolet Survey	Yes
MA-088 Helium Glow	Yes
MA-089 Doppler Tracking	Yes
MA-106 Light Flash	Yes
MA-107 Biostack	Yes
MA-128 Geodynamics	Yes
MA-136 Earth Observations and Photography	Yes
MA-147 Zone-Forming Fungi (Joint)	Yes
MA-148 Artificial Solar Eclipse (Joint)	Yes
MA-161 Killifish Hatching and Orientation	Yes

TABLE 12-I.- SUMMARY OF MISSION OBJECTIVES AND TECHNICAL INVESTIGATIONS
(Concluded)

Objectives/Investigations	Achievement status
<p>Detailed Objectives:</p> <p>Lower Limb Volume Measurements</p> <p>Crew Height Measurements</p> <p>Orbital Navigation via Synchronous Satellite Relay Data</p> <p>Inflight Demonstrations:</p> <p>Capillary Wicking in Zero Gravity</p> <p>Liquid Spreading in Zero Gravity</p> <p>Chemical Foams in Zero Gravity</p> <p>Physics Demonstrations</p>	<p>Yes</p> <p>^d Partial</p> <p>Yes</p> <p>Yes</p> <p>Yes</p> <p>Yes</p> <p>Yes</p> <p>Yes</p>

^aOne of seven sets of samples was a joint experiment.

^bOne of eight samples leaked and was not completed.

^cDegraded instrument operation significantly reduced data collection.

^dAll scheduled measurements were not obtained.

13.0 LAUNCH PHASE SUMMARY

13.1 WEATHER CONDITIONS

High cirrus clouds, and scattered middle and low clouds were present over the launch area at the time of launch but did not threaten the launch. At midmorning, a line of cumulonimbus clouds had been present about 40 kilometers offshore and had moved westward, but it dissipated by the time it reached a distance of 20 kilometers from the launch pad. This was the closest approach of any clouds that could have threatened the launch. Visibility was excellent at the time of launch. The surface wind was from the southeast and ranged from 5 to 8 meters per second in velocity. Surface pressure was 10.2 newtons/sq cm and surface temperature was 303° K. The maximum wind velocity observed in the troposphere was 15 meters per second blowing from 0.524 radian at an altitude of 12 kilometers.

13.2 ELECTRICAL FIELD MEASUREMENTS

A large network of ground-based electrical field mills is operated at the Kennedy Space Center to identify clouds which might be an electrical hazard to space vehicles prior to and during launch (fig. 13-1). Information from the field mills is displayed at the Kennedy Space Center weather station. Based on the data received, the Kennedy Space Center weather office provides advice to the Launch Director on impending weather conditions. Because of this capability and the fact that the Apollo Soyuz launch was scheduled to occur during a period in which thunderstorm activity would be high, the Launch Mission Rules regarding launching during severe weather conditions were proposed to be changed on June 18, 1975, to permit the Launch Director to base his decision to launch on the quantitative data from the field mill measurements as well as information obtained by other observations. Subsequent to this, an investigation was made on using instrumented aircraft to obtain electric field measurements in the projected path of the space vehicle. As a result of tests comparing the data obtained from the airborne and ground-based instruments, the Launch Mission Rules were modified on July 14, 1975, to include the inputs from both sources. Figure 13-2 and 13-3 are pages from the Launch Mission Rules containing the revised rules and a graphical representation of the acceptable risk for launch using the airborne and ground instrumentation.

The aircraft flown on launch day, the altitudes at which they were flown, and the flight patterns are shown in figure 13-4. Field mill readings were communicated from the aircraft to the Kennedy Space Center weather office from T minus 35 minutes to T minus 5 minutes. All measurements were very near 0 kilovolts per meter.

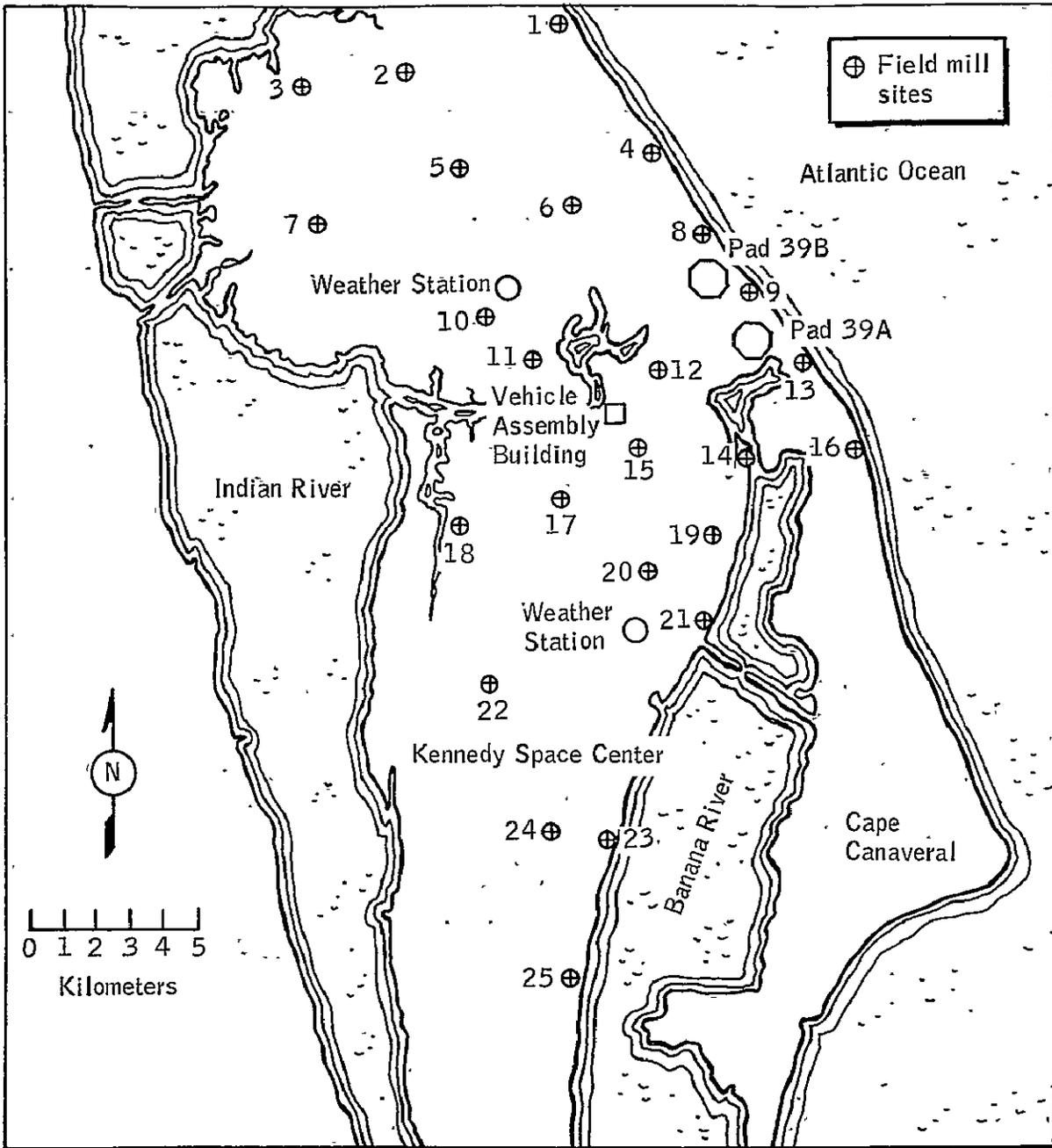


Figure 13-1.- Field mill network at Kennedy Space Center.

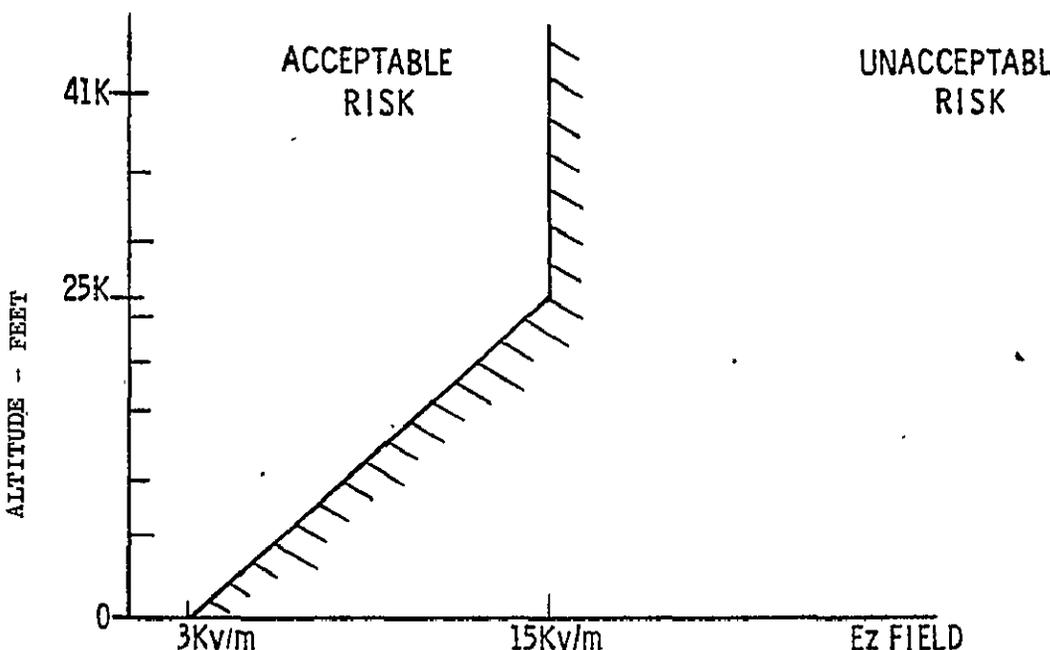
JOHN F. KENNEDY SPACE CENTER, NASA - LAUNCH MISSION RULES

REV	ITEM	DESCRIPTION			
		<u>WEATHER RESTRICTIONS (CONTINUED):</u>			
3	1-406	<p><u>SEVERE WEATHER CONDITIONS:</u></p> <p>1. THE SPACE VEHICLE WILL NOT BE LAUNCHED IF THE NOMINAL FLIGHT PATH WILL CARRY THE VEHICLE:</p> <ul style="list-style-type: none"> A. THROUGH A CUMULONIMBUS (THUNDERSTORM) CLOUD. B. WITHIN 5 STATUTE MILES OF A CUMULONIMBUS (THUNDERSTORM) CLOUD OR WITHIN 3 STATUTE MILES OF AN ASSOCIATED ANVIL. THIS RULE MAY BE RELAXED AT THE DISCRETION OF THE LAUNCH DIRECTOR IF THE ELECTRIC FIELD AT THE LAUNCH PAD IS LESS THAN 1 KILOVOLT PER METER. C. THROUGH COLD-FRONT OR SQUALL-LINE CLOUDS WHICH EXTEND ABOVE 10,000 FEET. D. THROUGH MIDDLE CLOUD LAYERS 6,000 FEET OR GREATER IN DEPTH WHERE THE FREEZE LEVEL IS IN THE CLOUDS. E. THROUGH CUMULUS CLOUDS WITH THE FREEZE LEVEL IN THE CLOUDS. F. RULES C, D, AND E ABOVE MAY BE RELAXED AT THE DISCRETION OF THE LAUNCH DIRECTOR WHEN ELECTRIC FIELD MEASUREMENTS IN THE LAUNCH PAD AREA ARE STABLE AND MEASURE LESS THAN 1 KILOVOLT PER METER. G. RULES C, D, AND E ABOVE MAY BE FURTHER RELAXED PROVIDED THAT AIRBORNE AND GROUND ELECTRIC FIELD MEASUREMENTS MEET THE CRITERIA DEFINED IN FIGURE 1. <p>2. FOR FLIGHT CREW EGRESS RULES RELATIVE TO SEVERE WEATHER CONDITIONS, SEE "FLIGHT CREW SAFETY RULES" SUBSECTION, ITEMS 1-609 (9) AND 1-610 (6).</p>			
	1-407	<p><u>PAD OR LAUNCH ABORT RECOVERY WEATHER CONDITIONS:</u></p> <p>THE LAUNCH OPERATIONS MANAGER WILL OBTAIN A GO/NO-GO DECISION FROM THE FLIGHT DIRECTOR BASED UPON THE PAD OR LAUNCH ABORT RECOVERY WEATHER CONDITIONS.</p>			
MISSION	SECTION	STAGE SUBSECTION	SYSTEM	CHANGE NO	PAGE
ASTP	SPACE VEHICLE	WEATHER RESTRICTIONS		JULY 14, 1975	1-18

NSC FORM 4-47A (REV 1 68)

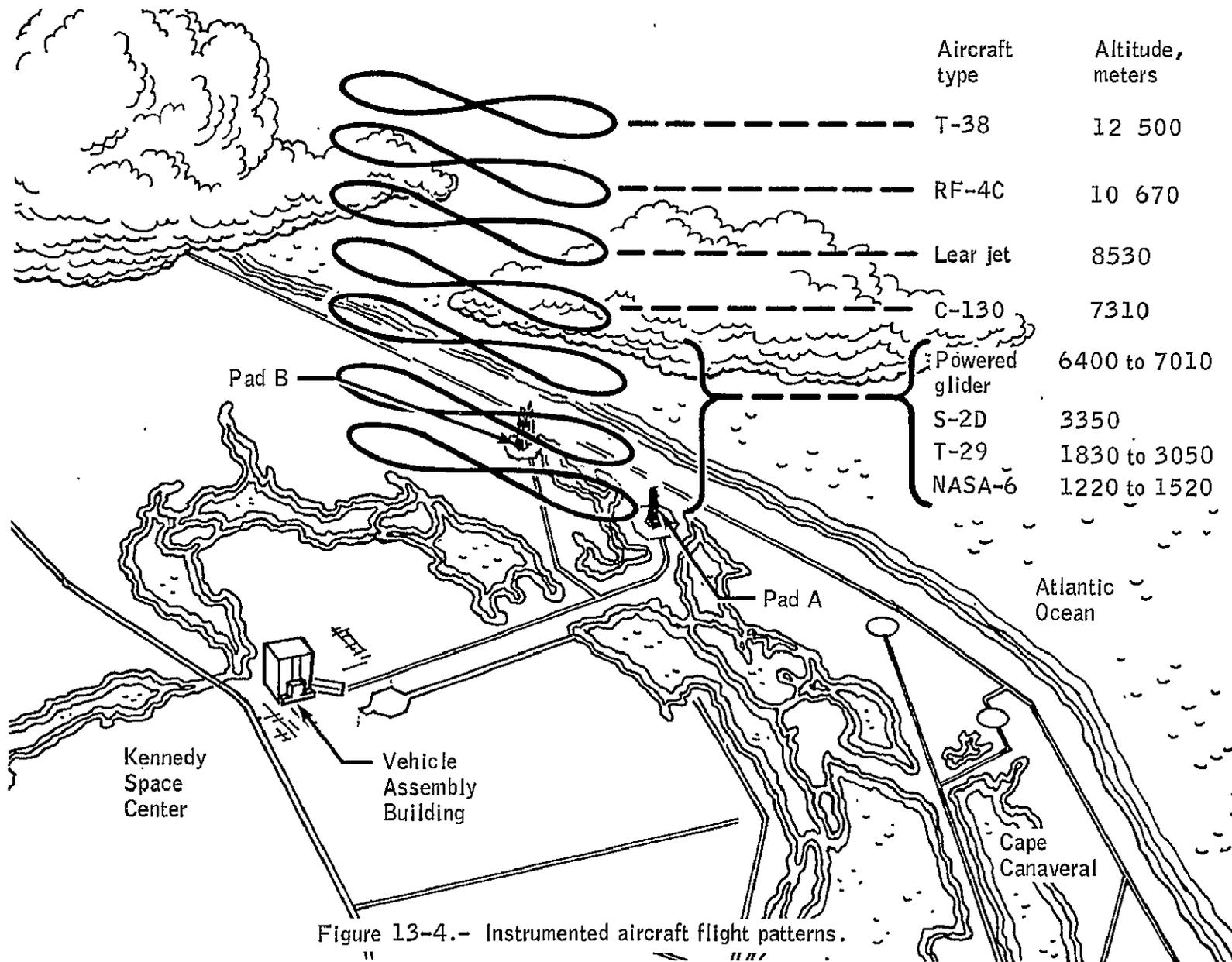
Figure 13-2.- Launch mission rules for launching during severe weather.

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REV	ITEM	DESCRIPTION			
3		<p style="text-align: center;">ASTP LAUNCH GUIDE LINES WITH AIRBORNE AND GROUND INSTRUMENTATION</p>  <ul style="list-style-type: none"> ● VERTICAL FIELD MEASUREMENTS ALONG FLIGHT PATH IN A 3 MILE AREA ARE WITHIN THE ABOVE ENVELOPE ● AND NO RAPID FLUCTUATION OF ABOUT 3 Kv/m AT ABOUT 1 MIN INTERVALS WITHIN THE 3 MI AREA MEASURED BY THE GROUND MILLS. 			
MISSION ASTP	SECTION SPACE VEHICLE	STAGE SUBSECTION WEATHER RESTRICTIONS	SYSTEM FIGURE 1	DATE JULY 14, 1975	PAGE 1-18A

NASC FORM 4-47A (REV. 1-68)

Figure 13-3.- Launch guidelines with airborne and ground instrumentation.



13.3 LAUNCH VEHICLE PERFORMANCE

The Apollo space vehicle for the Apollo Soyuz mission was launched from Pad B of Launch Complex 39, Kennedy Space Center, Florida. The launch was normal and all launch vehicle objectives were attained, as discussed in reference 4. Range zero occurred at 19:50:00 G.m.t., July 15, 1975, and the times given in the following paragraphs are referenced to range zero.

The S-IB stage provided normal thrust with inboard engine cutoff at 136.6 seconds and outboard engine cutoff 3.6 seconds later. Stage separation occurred at 141.5 seconds. The longitudinal accelerometers, liquid oxygen inlet pressure, and combustion pressure data showed no evidence of POGO-type resonance instability.

All S-IVB propulsion system start conditions, control functions and maneuvers were normal. The S-IVB stage was cut off by guidance command at 585.7 seconds. The firing duration was 1.9 seconds longer than predicted. The vehicle was inserted into earth orbit at 595.7 seconds.

Most S-IVB vibration levels were similar to those observed on Skylab Saturn IB vehicles. However, the thrust structure vibration amplitude was higher in the 19-hertz frequency range than on previous flights. The amplitude built up gradually during S-IVB operation, reaching a peak of $\pm 0.5g$ 52 seconds after engine ignition. The profile was similar to that observed on the Skylab second visit but the magnitude was greater. Consequently, the level was higher than had been expected from the preflight POGO analysis.

During orbital coast, the S-IVB fuel tank ullage pressure recovery was below the predicted minimum value following the third programmed vent. Two extra tank vents (compared to previous flights) were added because of the docking of the command and service module to extract the docking module. The different vent schedule may have caused errors in prediction, or the heat input to the tank may have been less than expected.

The S-IVB/instrument unit deorbit was initiated at 5 hours 15 minutes and 45 seconds during revolution 4. About 33 minutes later, the stage fragmented when it was at an altitude of approximately 72 kilometers. The estimated impact footprint extended from latitude 10 degrees 54 minutes north, longitude 175 degrees 36 minutes east to a point located at latitude 18 degrees 30 minutes north, longitude 178 degrees 30 minutes west.

14.0 ANOMALY SUMMARY14.1 COMMAND AND SERVICE MODULE, DOCKING MODULE, AND
GOVERNMENT FURNISHED EQUIPMENT

14.1.1 Docking Probe Pyrotechnic Connector Interference

During probe removal operations, the crew was unable to unlatch the capture latches using the capture latch retraction tool. The crew observed that one of the pyrotechnic electrical connectors was obstructing the tool entrance hole in the center of the probe as shown on the close-out photograph, figure 14-1. A workaround procedure was used to remove the protective cover and the interfering connector. The cover was then reinstalled and the connector taped to the cover. The interfering connector (P3) was used for the expended pyrotechnic initiator for probe retraction.

The connectors were keyed and color coded. The wiring was cut to fit on a standard mockup board at the spacecraft contractor's facility. The connectors were oriented and the harness spot-tied during ordnance installation and harness fit check at the Kennedy Space Center Field Ordnance Test Laboratory. A pyrotechnic cover modification for command and service modules 111, 117, 118, and 119 deleted a tool insertion guide tube; however, the KSC test and checkout procedure was not modified. The procedure specified that the connector backshells be oriented in accordance with the docking system probe specification. The specification stated that the connectors should be oriented and spot-tied in accordance with figure 14-2. No sequence was given for installation of pyrotechnic connectors. Further, there was no caution note, either in the test and checkout procedure or the specification, providing clearance for the capture latch retraction tool.

As shown in figure 14-1, the adjacent pyrotechnic connector (P6) was mated and safety-wired prior to installation of P3. This sequence of installation caused the wire from P3 to ride over instead of under connector P6 as shown in figure 14-2. As a result, the wire to P3 was so short that that proper connector backshell orientation could be achieved only by pulling on the wire, contrary to good wiring practice. The connector installation team decided to reclock the P3 connector backshell to the position shown in figure 14-1 to prevent overstressing the wire. The reclocked connector backshell obstructed the capture latch release tool access hole.

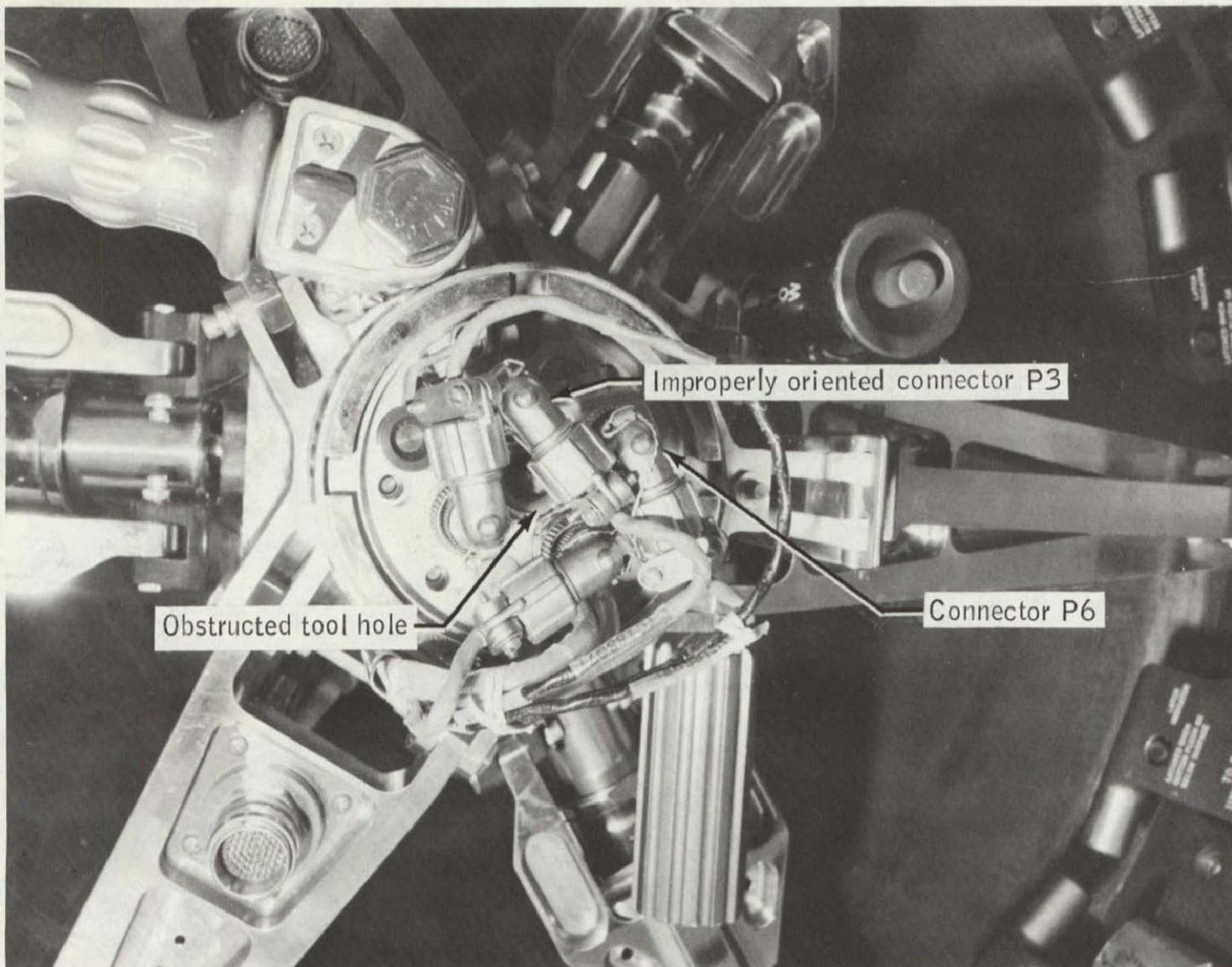


Figure 14-1.- Closeout photograph of improperly installed docking probe pyrotechnic connector.

No corrective action is required since the Apollo docking probe will not be used for future missions. However, for similar systems installation, proper sequence of installation and any caution notes to avoid interference will be defined in the installation procedure.

This anomaly is closed.

14.1.2 Toxic Gas Entered Cabin During Earth Landing Sequence

Toxic gas entered the command module cabin during repressurization for 30 seconds from manual deployment of the drogue parachutes at 5650 meters (18 550 ft) to disabling of the reaction control system at 2925 meters (9600 ft).

Nominal operation of the earth landing sequence for this mission was for the crew to arm the sequencer pyrotechnic buses at 15 240 meters (50 000 ft) altitude. Then the crew was to arm the automatic earth landing system (ELS) at 9145 meters (30 000 ft) by switching the two ELS switches to LOGIC and AUTO. As shown in figure 14-3, arming the earth landing system applies sense power to 7315-meter (24 000-ft) baroswitches so that, upon closure of the baroswitches, the sense power latches the earth landing system activate relay. This applies power to the reaction control system disable relay and the 0.4-, 2.0-, and 14.0-second timers for the forward heat shield, the drogue parachutes and the main parachutes, respectively. Timeout of the 14-second timer applies power to the 3050-meter (10 000 ft) baroswitches. Closure of these baroswitches releases the drogue parachutes and deploys the main parachutes. If the redundant (A and B) automatic earth landing systems fail, manual switches must be used to perform the automatic functions.

As the landing was actually accomplished, the two switches which arm the automatic earth landing system were not positioned in time to initiate the automatic sequence at the proper altitude (fig. 14-4). In addition, the RCS CMD switch, which backs up the automatic disabling of the reaction control system thruster firing commands, was not placed in the OFF position. As a result, the reaction control system command inhibit function did not occur and a chain of events took place that caused nitrogen tetroxide vapors to be drawn into the command module cabin.

When the crew realized that the drogue parachutes had not been deployed automatically, the forward heat shield and drogue parachutes were deployed manually. The spacecraft motions that were induced by the drogue parachute deployment caused the reaction control system thrusters to respond vigorously. The thruster activity was terminated when the crew armed the automatic earth landing system 30 seconds after drogue parachute deployment, disabling the reaction control system (fig. 14-4).

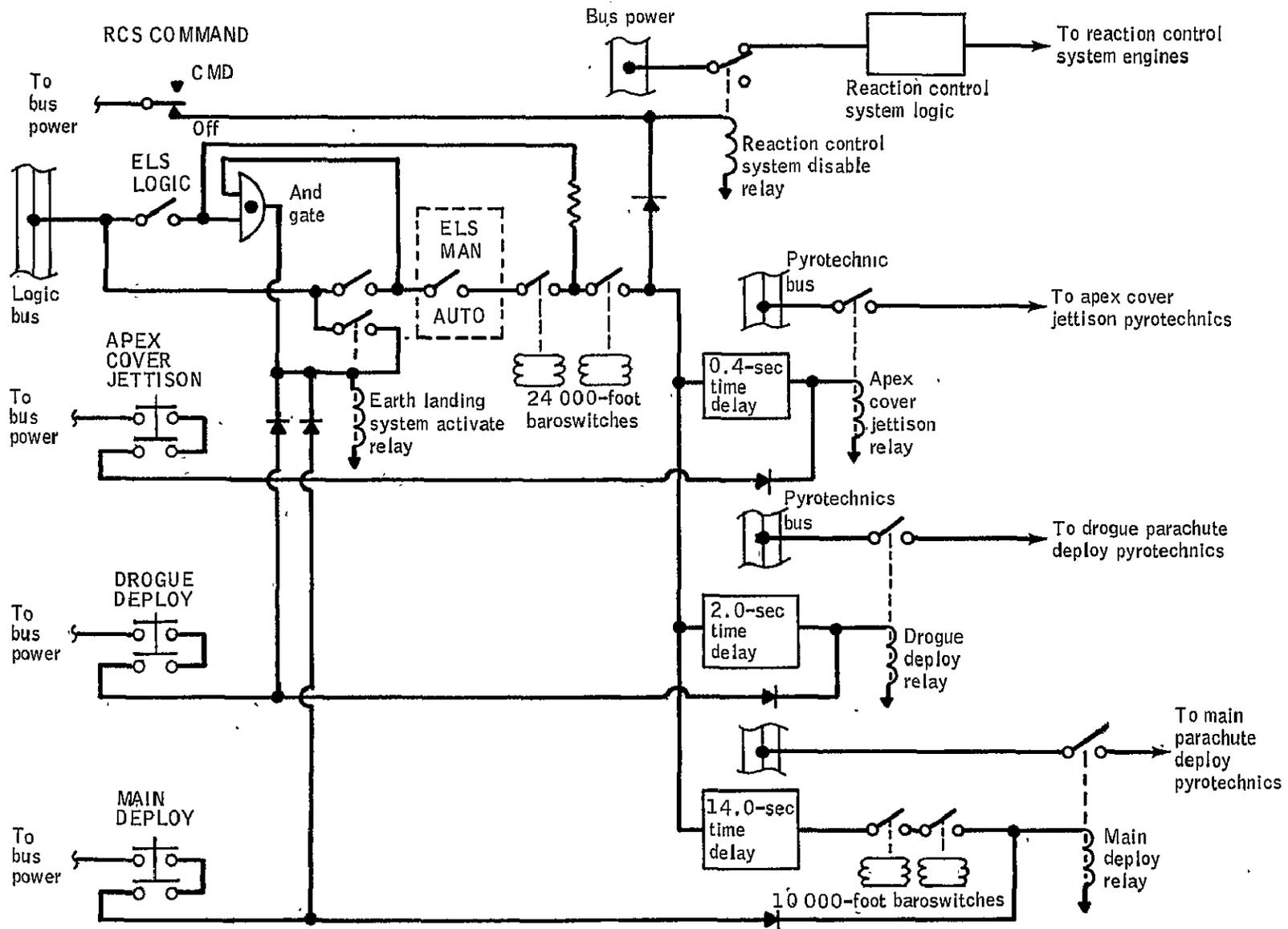


Figure 14-3.- Simplified manual and automatic circuits of earth landing system (system A).

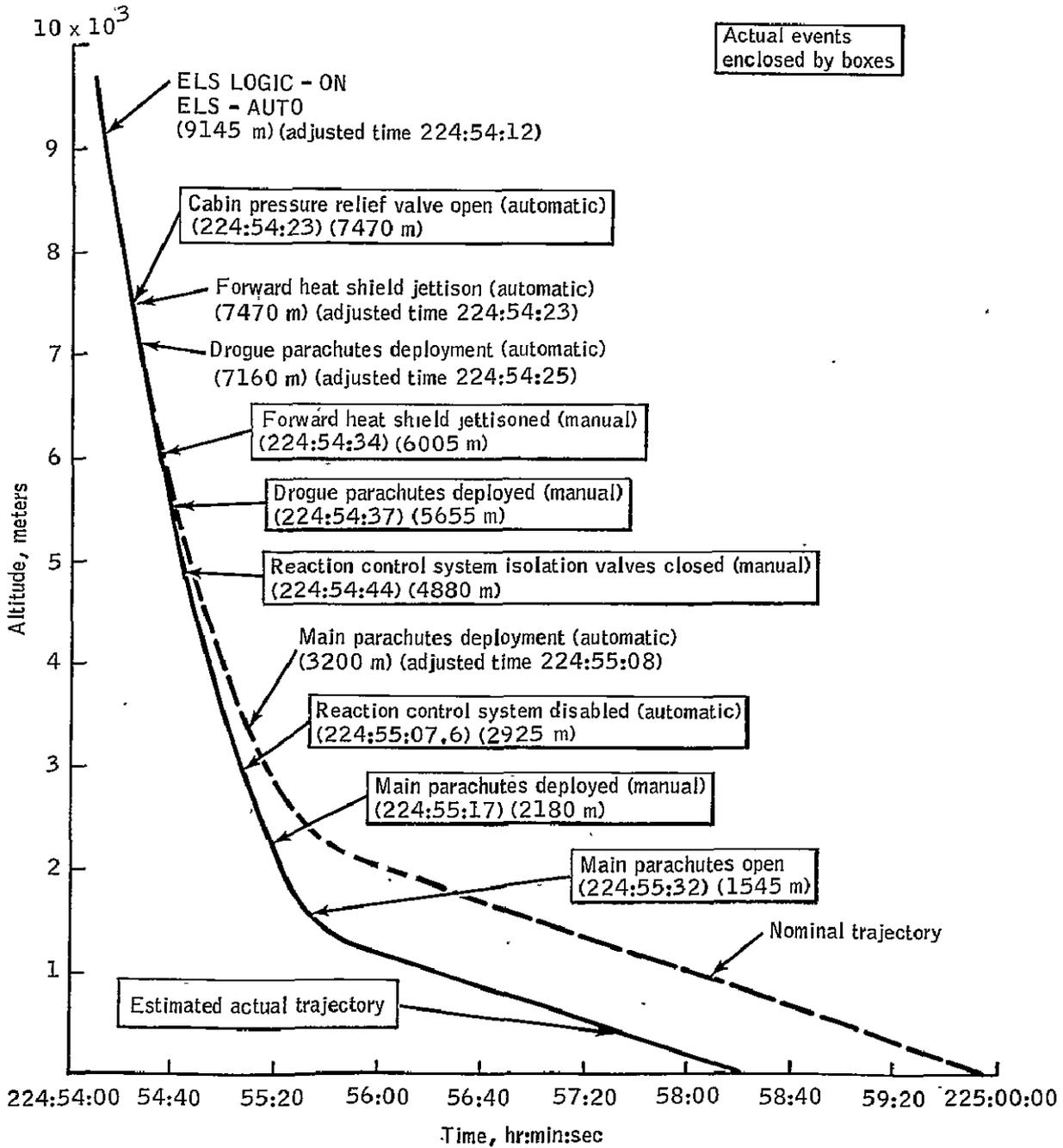


Figure 14-4.- Comparison of actual and nominal descent sequences.

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The cabin pressure relief valve opens automatically prior to drogue parachute deployment to repressurize the command module cabin. The valve pulls outside air into the cabin through the steam duct vent, located about 0.6 meter (2 ft) away from a pair of reaction control system roll thrusters (fig. 14-5). During the first 7 seconds of the 30-second period of thruster activity, a mixture of air and combustion products of monomethyl hydrazine (fuel) and nitrogen tetroxide (oxidizer) were pulled into the command module cabin. Then, the reaction control system isolation valves were closed when the CM RCS PRPLNT switches were positioned to OFF. As a consequence, a mixture of air and oxidizer vapors was sucked into the cabin during the remaining 23 seconds of thruster activity before the reaction control system was disabled.

As shown in figure 14-6, propellants are trapped between the isolation valves and the thruster solenoid valves when the isolation valves are closed. With the solenoid valves open, the oxidizer boils at the altitudes through which the command module was descending during the 23-second period. The positive roll thruster near the steam vent fired in the direction of the vent and was open for 19 of the 23 seconds. During this time, 0.5 kilogram of oxidizer was expelled through this thruster. The negative roll thrusters did not fire during this period. Each pitch and yaw thruster fired for about 11 of the 23 seconds. Of the 4.1 kilograms of oxidizer trapped between the isolation valves and thrusters, 3.4 kilograms boiled off.

Total crew exposure to the oxidizer vapors was for 4 minutes and 40 seconds from closure of the reaction control system isolation valves until the crew donned oxygen masks after landing. (The masks were not accessible during the entry and landing phases.) Fifty percent of the suit loop flow is through the lithium hydroxide canisters. The remainder flows through a hole in the middle of each canister. All the suit loop flow passes through the glycol heat exchanger where the toxic gases not removed by the lithium hydroxide canisters would be converted to nitric acid after reacting with water.

The peak cabin concentration of toxic gases immediately after the reaction control system was disabled was estimated to be about 700 parts per million of nitrogen dioxide at 1 atmosphere (fig. 14-7). Nitrogen tetroxide reacts with air to produce, principally, nitrogen dioxide. Because of its prevalence, stability, and toxicity, nitrogen dioxide is the primary concern for crew exposure. The average crew exposure was estimated to be about 250 parts per million from closure of the reaction control system isolation valves to donning of the oxygen masks. The medical findings support the evidence that the crew was exposed to a high level of oxidizer products and that there were no other toxic compounds such as monomethyl hydrazine (ref. 2).

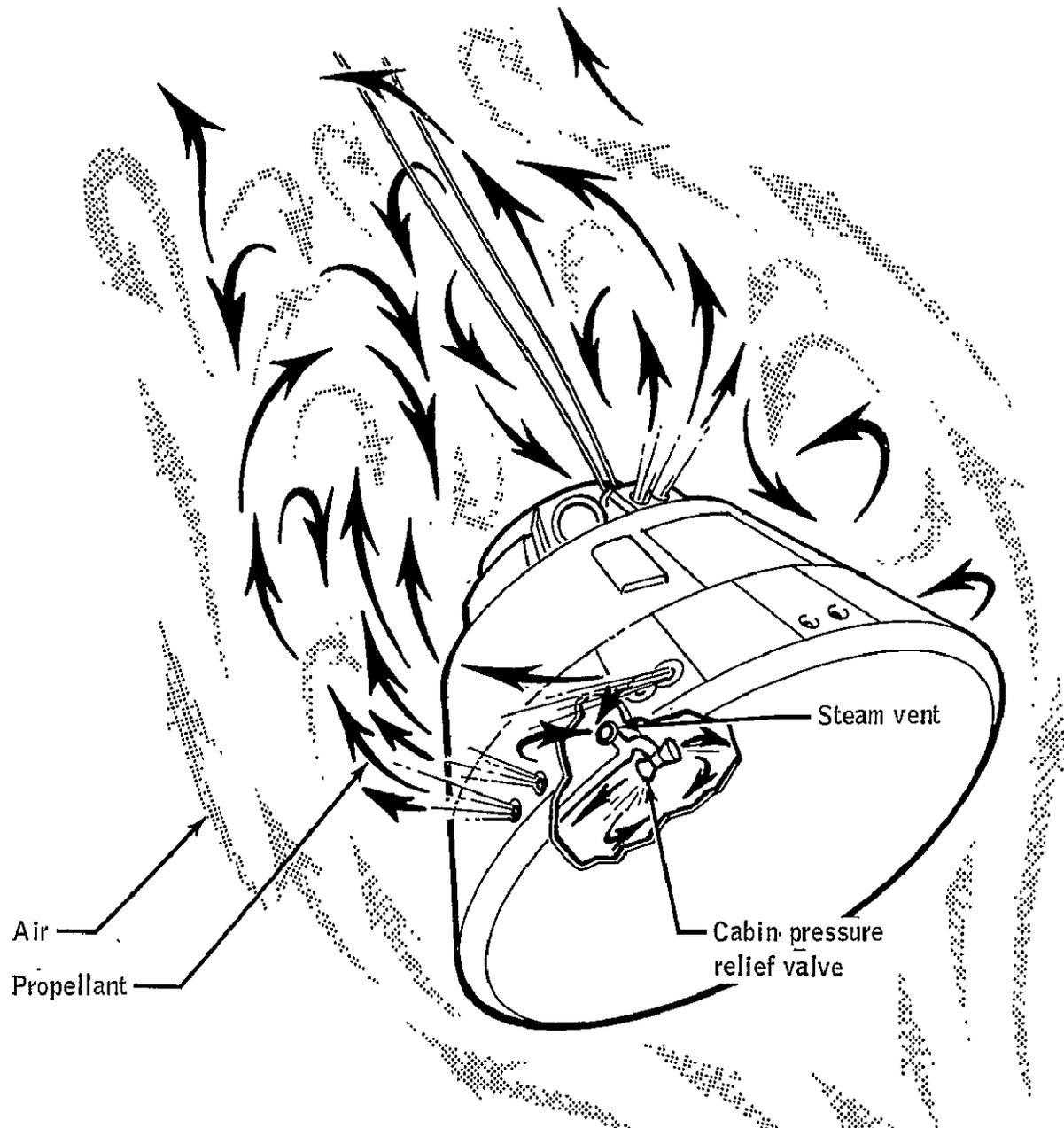


Figure I4-5.- Flow field mixing around command module at time of incident.

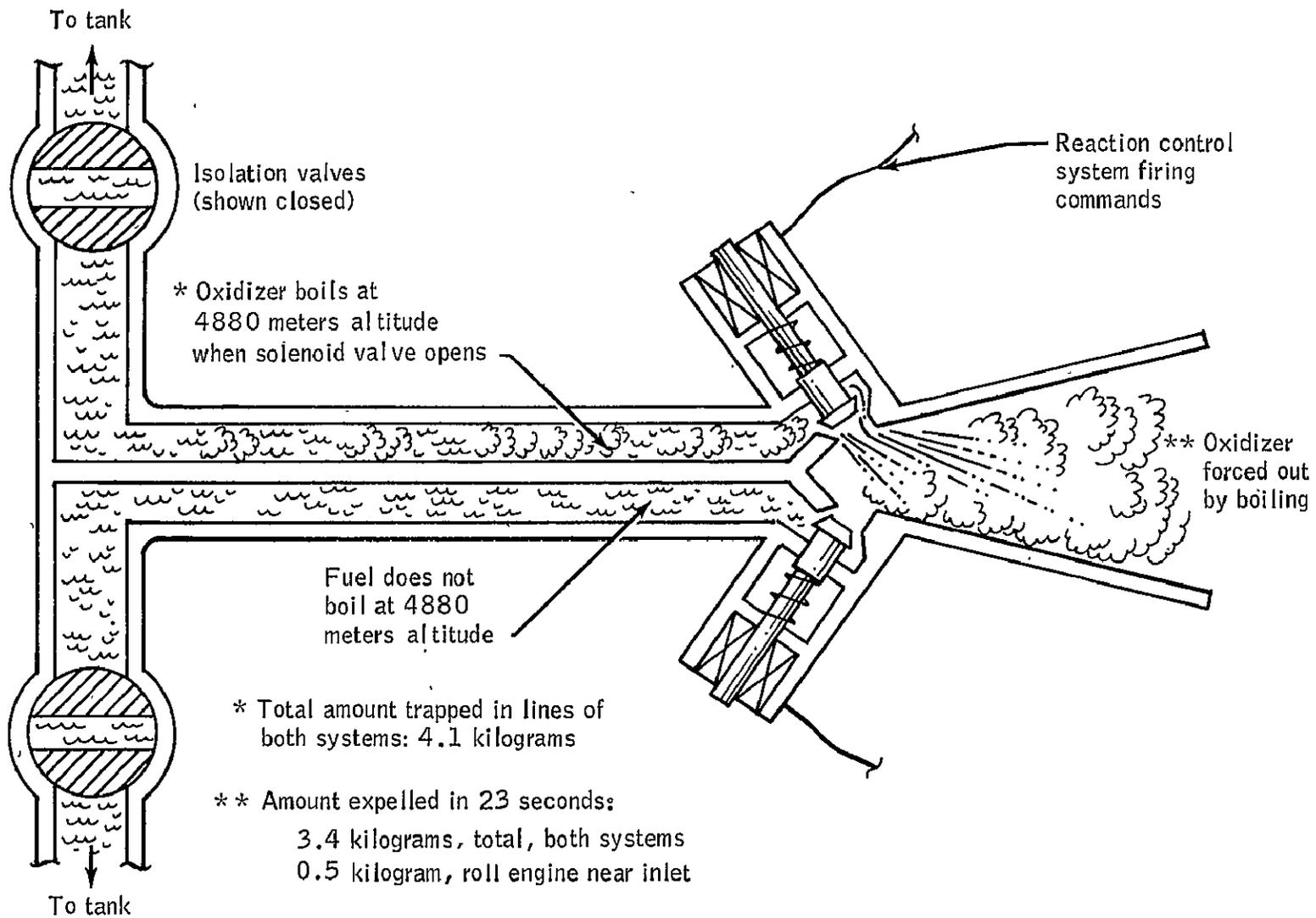


Figure 14-6.- Oxidizer boiloff.

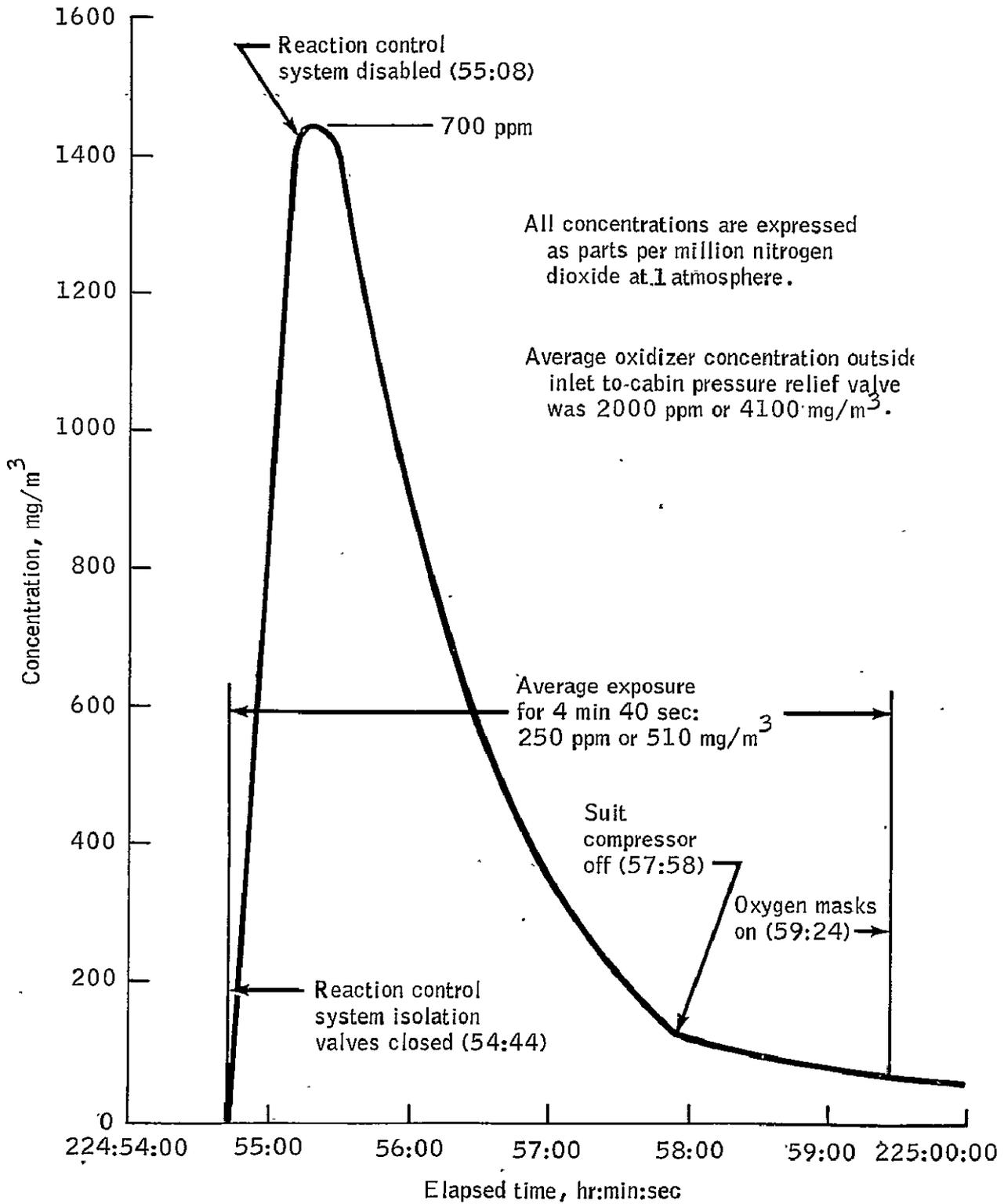


Figure 14-7.- Oxidizer concentration in cabin

The crew reported trouble communicating due to noise starting at about 15 240 meters (50 000 ft). The onboard tapes revealed an intermittent warbling tone received on the S-band for about 8 to 10 seconds at 27 430 meters (90 000 ft). From this time until landing there was no apparent problem with intercommunications and crew conversation recorded by the onboard recorder was clear with the reaction control system thrusters firing in the background. As on previous missions, high cabin vent noise was present between 7315 meters (24 000 ft) and 1525 meters (5000 ft).

Three crucial manual functions had to be completed within a few seconds prior to reaching 7315 meters before the automatic sequencer could accomplish the tasks for which it was designed. Time-critical manual switching had been required on early Apollo flights to protect against single-point failures. Redesign for the Apollo 15 spacecraft and subsequent spacecraft allowed all three functions to be performed prior to entry. However, the pre-Apollo 15 procedure was used on this mission. The crew was about 15 seconds late in arming the pyrotechnic buses, arming them just before reaching 9145 meters during the time-critical earth landing sequence. Two crucial functions missed by the crew inhibited the time-critical automatic earth landing sequencer and left the reaction control command enabled. This resulted in toxic gas entering the cabin.

The following corrective actions are being taken for future designs.

- a. Crew procedures for time-critical operations are being reviewed by cognizant subsystem managers to insure proper use of manual capability.
- b. Manual overrides and inhibits to crew-safety time-critical automatic functions are being evaluated.

A more thorough discussion of this anomaly is presented in reference 5.

This anomaly is closed.

14.1.3 Color Information Lost From Video Tape Recorder Dump

Color information was absent from the onboard video tape recorder dump of inflight exercise activity obtained with television camera 4002 at 130 hours 22 minutes.

Figure 14-8 shows the color wheel housing and color wheel drive train of the television camera. The color wheel motor is a synchronous-drive type that drives the color wheel in phase with the color wheel pulse generator. The color wheel is divided into six equal sectors of

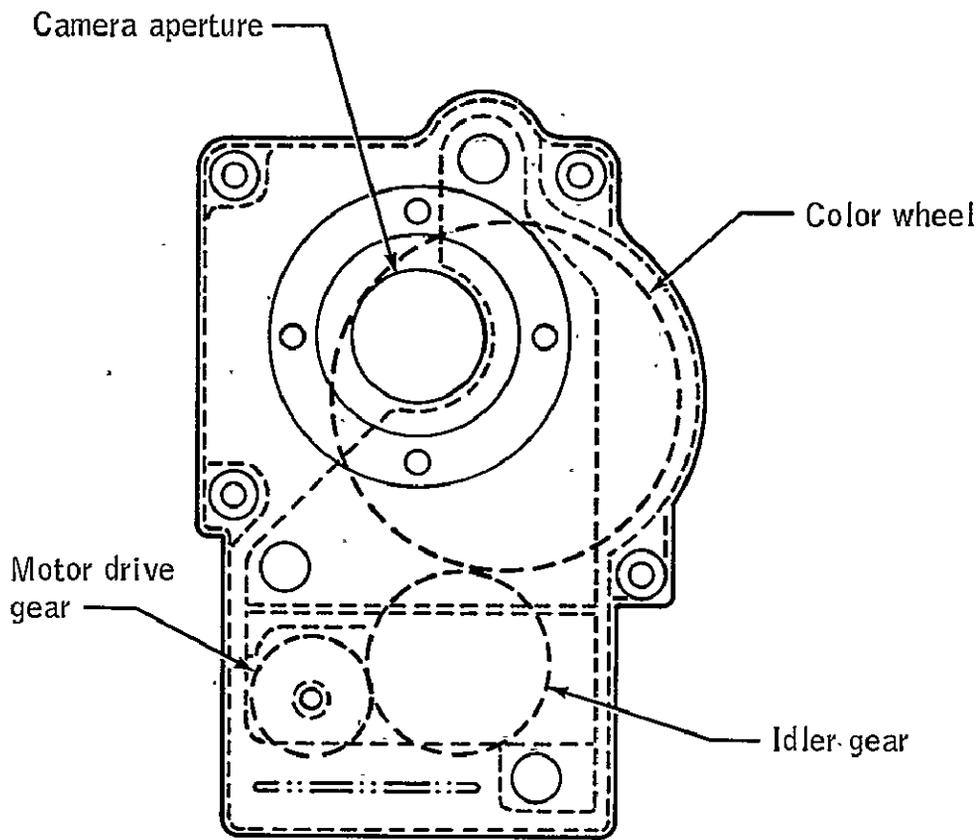


Figure 14-8.- Color wheel housing and drive train.

alternating red, blue, and green. The color pulse-coded information, when received on the ground, is used to generate a color burst and color carrier for commercial-type color compatibility by the ground scan converter.

When the postflight failure analysis and inspection was performed, the color wheel operated normally, but a piece of cable bundle sleeve lacing was found adrift in the color wheel housing. Introducing a frayed piece of lacing of the same size into the wheel gears resulted in an immobile color wheel.

In zero gravity, the frayed lacing probably found its way into the gears, rendering the wheel immobile. This resulted in the absence of color synchronization pulses, producing a black-and-white picture.

There are at least three combinations of corrective action to prevent this type of failure: (1) modularize the color wheel assembly for plug-in inflight replacement, (2) improve sleeving to prevent fraying, if sleeving must be used, and (3) use a motor with greater torque capacity.

This anomaly is closed.

14.1.4 Television Camera Exhibited Spurious Flashes in the Video and Color Breakup

Aperiodic flashes and loss of color synchronization were noticed while using television camera 9.

The image tube (fig. 14-9) is an electron bombardment silicon (EBS) type. The photocathode consists of a semitransparent deposition on the inner surface of a fiberoptic face. Light photons striking the photocathode produce electrons by photo-emission. These are accelerated from the photocathode to the image tube target by the large (9000 volts) potential difference between the photocathode and target. The photocathode is maintained at minus 9000 volts with respect to ground to obtain this potential difference.

Postflight testing at a sustained lowered pressure produced corona which was observed through the glass faceplate going from the lip of the photocathode fiberoptic face across the inner surface of the faceplate to the front support. Figure 14-10 shows an enlarged section of the region where the corona occurred and includes a photograph of the corona taken in a darkened room, viewing from the front of the image tube.

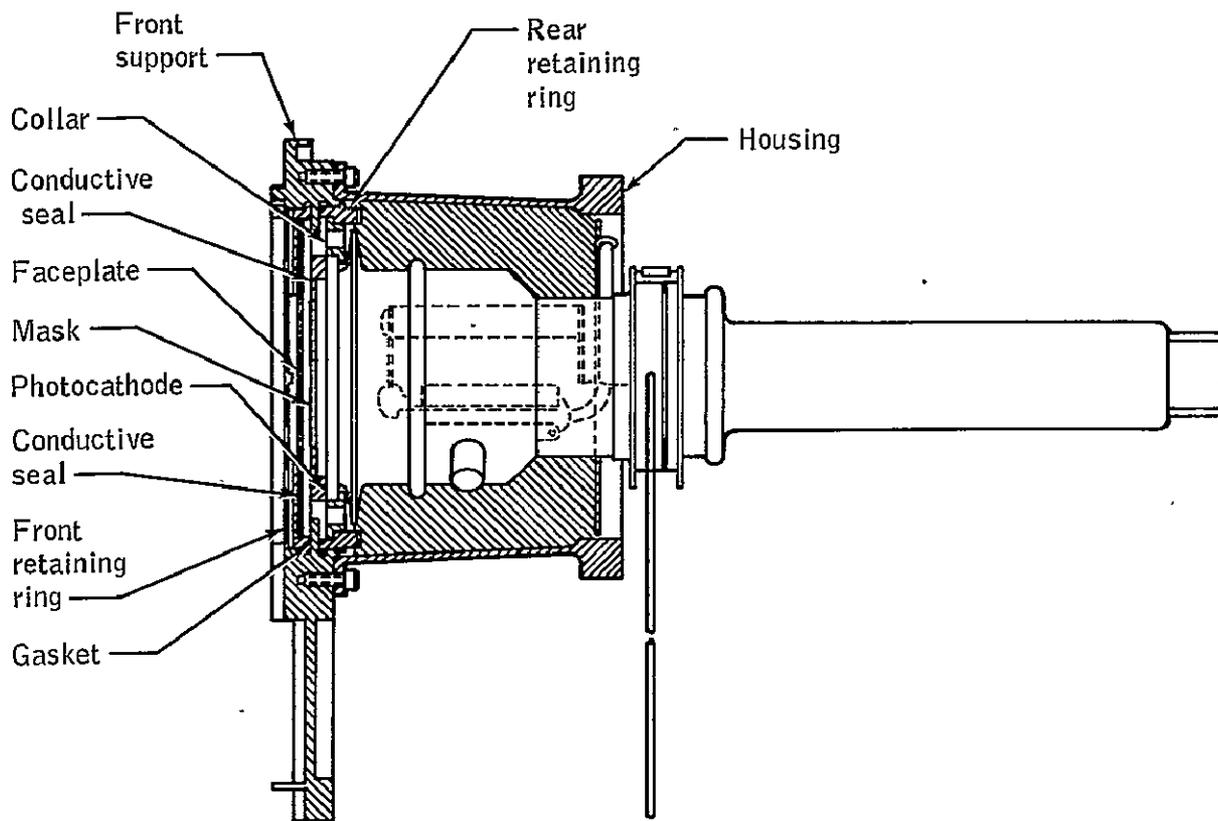


Figure 14-9.- Image tube.

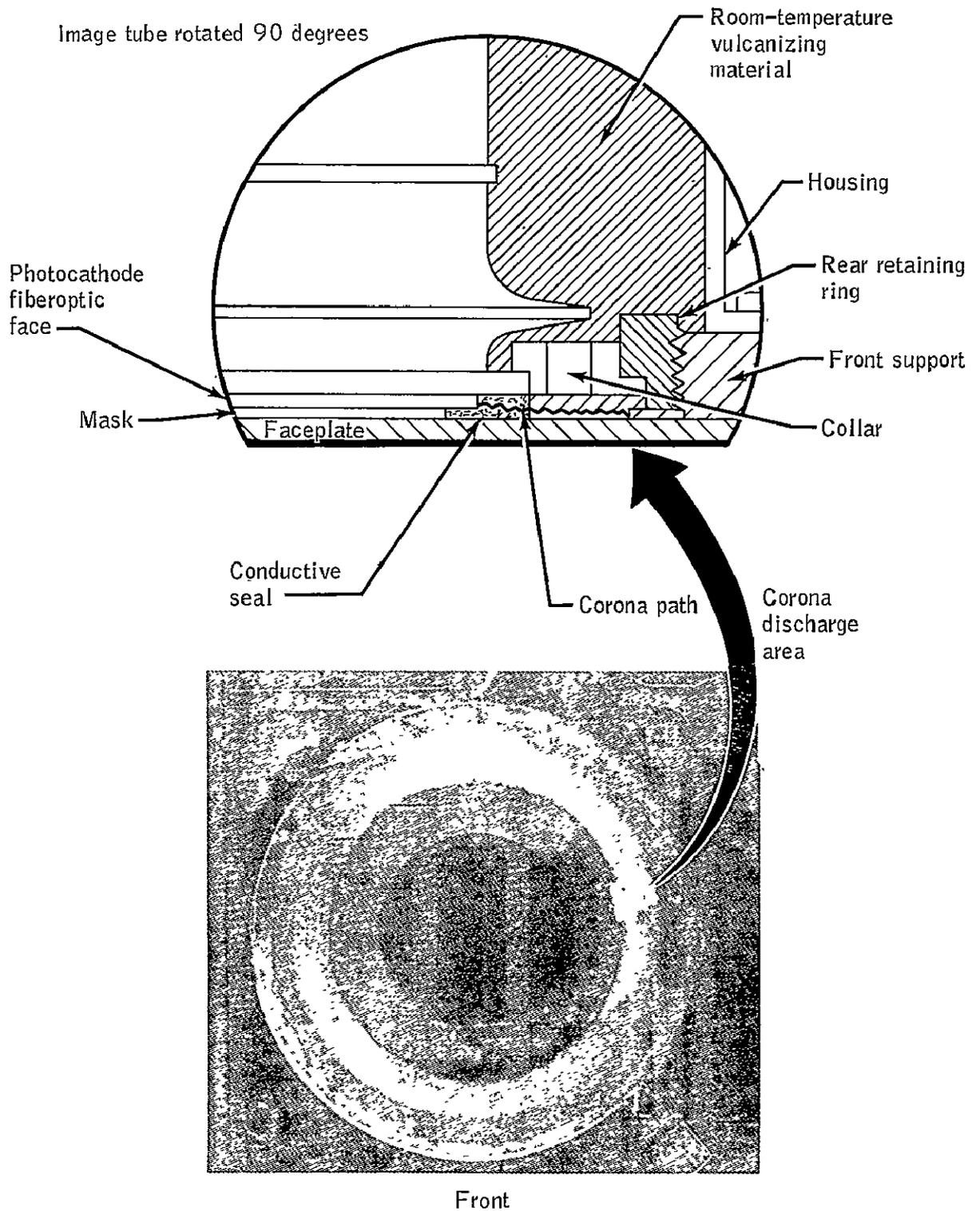


Figure 14-10.- Corona discharge observed during postflight testing.

The corona was not produced at normal sea level pressure, so there must have been air bubbles in the room-temperature-vulcanizing (RTV) material fill between the conductive seal and front support. The corona commenced when these bubbles lost pressure after a low pressure dwell time. In addition to the corona at the 3 o'clock position of the EBS tube, there was random arcing reported at the 6 and 11 o'clock positions.

A darkened region behind the faceplate was carbonization from previous arcing. This arcing produced fast-rise light pulses visible to the photocathode at the periphery of the tube. This, in turn, upset the automatic light level control circuitry in the camera system.

Bubbles in the potting will be detected by flight article acceptance testing at the sustained low pressures, including vacuum testing if the camera is to be used outside the vehicle pressure vessel.

This anomaly is closed.

14.1.5 Cryogenic Oxygen Tank 1 Quantity Measurement Failed

The oxygen quantity measurement output decreased to zero volts at about 8 hours 40 minutes. The decrease was exponential with a time constant of approximately 0.02 second.

The oxygen tank quantity gage senses the average dielectric constant of oxygen in the cylindrical annular volume between two concentric aluminum tubes. The dielectric constant is proportional to density which, in turn, is proportional to the quantity of oxygen in the tank. The gage is approximately 2 feet long and is mounted in the center of the tank.

The gage capacitance is connected in series with a reference capacitor to form a capacitive voltage divider, as shown in figure 14-11, and is adjusted to apply zero volts input to the amplifier when the tank is empty. As the tank is filled, the gage capacity increases, applying a voltage to the amplifier input. This voltage is amplified and rectified to provide an output signal voltage which increases to 5 volts dc when the tank is full.

The reactive voltage developed across the probe capacitance will change as rapidly as capacitance changes. The rectifier filter on the output of the signal conditioner introduces a time constant of about 0.022 second in the instrument response.

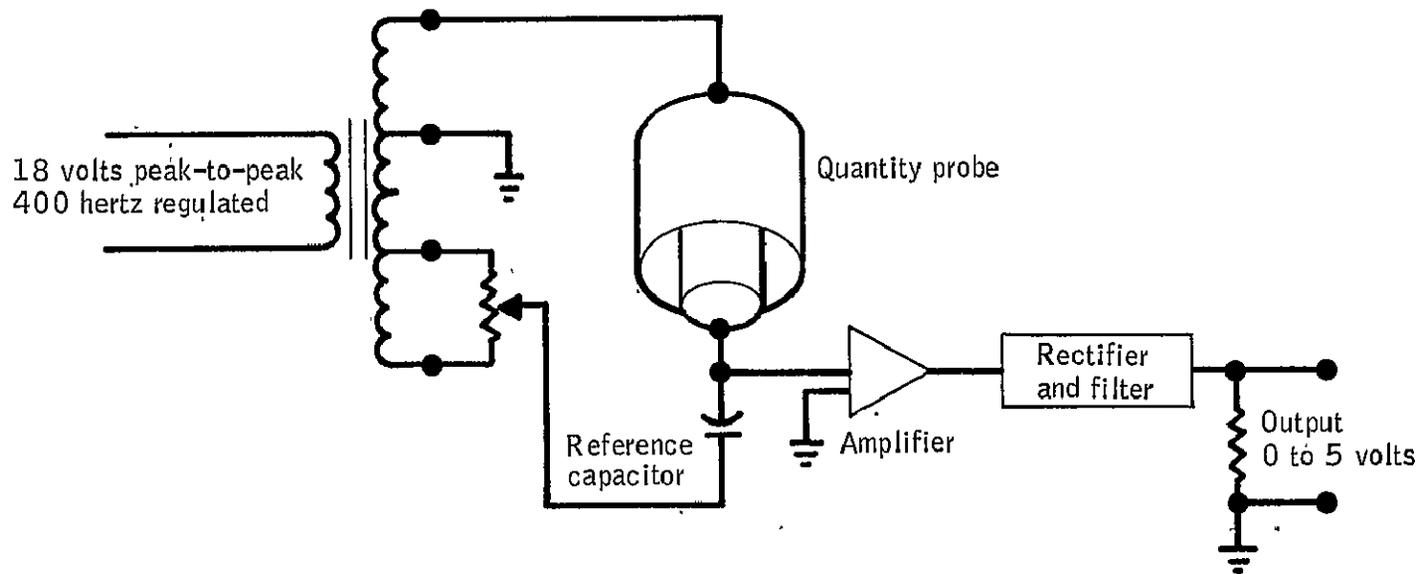


Figure 14-11.- Cryogenic oxygen tank quantity measurement schematic.

Since the time constant of the measurement output decrease was about 0.02 second, and the time constant of the signal conditioner output was 0.022 second, the failure must have been either in the transducer or in the 400-hertz power supplied to the transducer. Any one of many possible failures in the signal conditioner, including open and shorted leads and failed semiconductors, could cause the indications seen. Insufficient data exist to further isolate the failure.

No corrective action can be taken because the exact cause of failure cannot be determined.

This anomaly is closed.

14.1.6 Service Module Reaction Control System Quad A Helium Tank Temperature Failed

The service module reaction control system quad A helium tank temperature measurement decreased from a nominal value of 297.6° K to 255.4° K in 1 second at about 8 hours 44 minutes. This was observed both on telemetry and on the cabin meter. The measurement remained at 255.4° K through service module separation. The quad A primary fuel tank temperature measurement was used as a backup measurement because its sensor was located close to the quad A helium tank. The redline limits of this measurement were available to assure that the helium tank temperature did not exceed its safe upper and lower limits.

The temperature transducer (fig. 14-12) is a platinum resistance thermometer that operates in a resistance bridge. The transducer is connected to the signal conditioner by a shielded three-wire cable. Two of the wires are connected to the resistance thermometer, and the third wire is for temperature compensation and common-mode return for both sides of the bridge. The signal conditioner is powered by a current limiter box that provides 28 volts dc, and the power input is fused with a 1/4-ampere fuse. The signal conditioner filters this power and then performs a dc-to-dc conversion to produce plus 12.5 volts dc and minus 11.5 volts dc to excite the bridge and differential amplifier, respectively.

The differential amplifier output is transmitted from the service module, across the command and service module umbilical to the command module instrumentation junction box. The junction box then feeds the 0- to 5-volt signal to the pulse code modulation equipment for telemetry, and to the helium temperature meter on panel 2.

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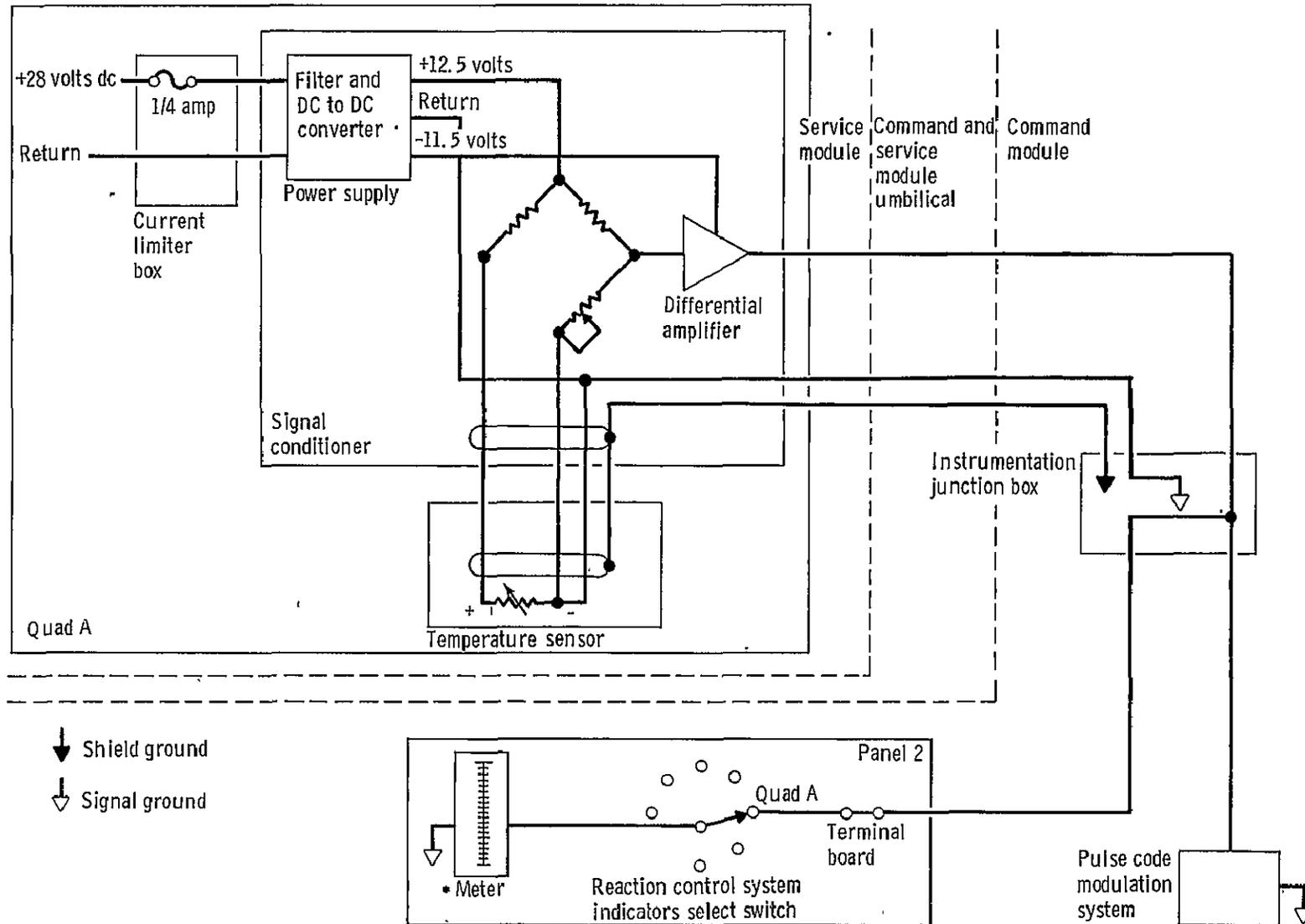


Figure 14-12. - Service module reaction control system quad A helium tank temperature measurement schematic.

Numerous failure modes caused by opens and shorts in the components and wiring are possible. These include:

- a. An open sensor positive lead.
- b. A short or open in the power supply or current limiter box.
- c. An open or short in the differential amplifier in the signal conditioner.
- d. An open or short in the amplifier output leading to the instrumentation junction box.
- e. A short in the junction box output leading to telemetry or to the select switch in panel 2. (The meter operated properly for the other select switch positions.)

Since most of the circuitry was located in the service module, which is not recoverable, failure isolation testing was not possible.

Since failure isolation could not be performed, no corrective action can be taken.

This anomaly is closed.

14.1.7 Cryogenic Oxygen Tank 2 Pressure Measurement Was Intermittent

The measurement dropped to 22.1 newtons/sq cm (50 millivolts), at 53 hours 40 minutes. One minute later, the pressure reading recovered to 608.6 newtons/sq cm, but after about 7 minutes it dropped again to 22.1 newtons/sq cm. The measurement returned to the normal reading at 54 hours 4 minutes.

The transducer (fig. 14-13) consists of a pickup comprised of a silicon strain gage bridge mounted on a damped-edge diaphragm, and of an integral signal conditioner. The diaphragm separates a vacuum reference chamber from the pressure chamber. The signal conditioner output is a 0- to 5-volt dc analog signal which is linearly proportional to pressure in the range of 34.5 to 724.0 newtons/sq cm. The transducer is powered by the spacecraft 28 volts dc.

Analysis of the transducer circuit (fig. 14-14) indicates that the likely cause of the failure was an intermittent open at a joint in the pressure gage bridge excitation circuit. This conclusion is supported

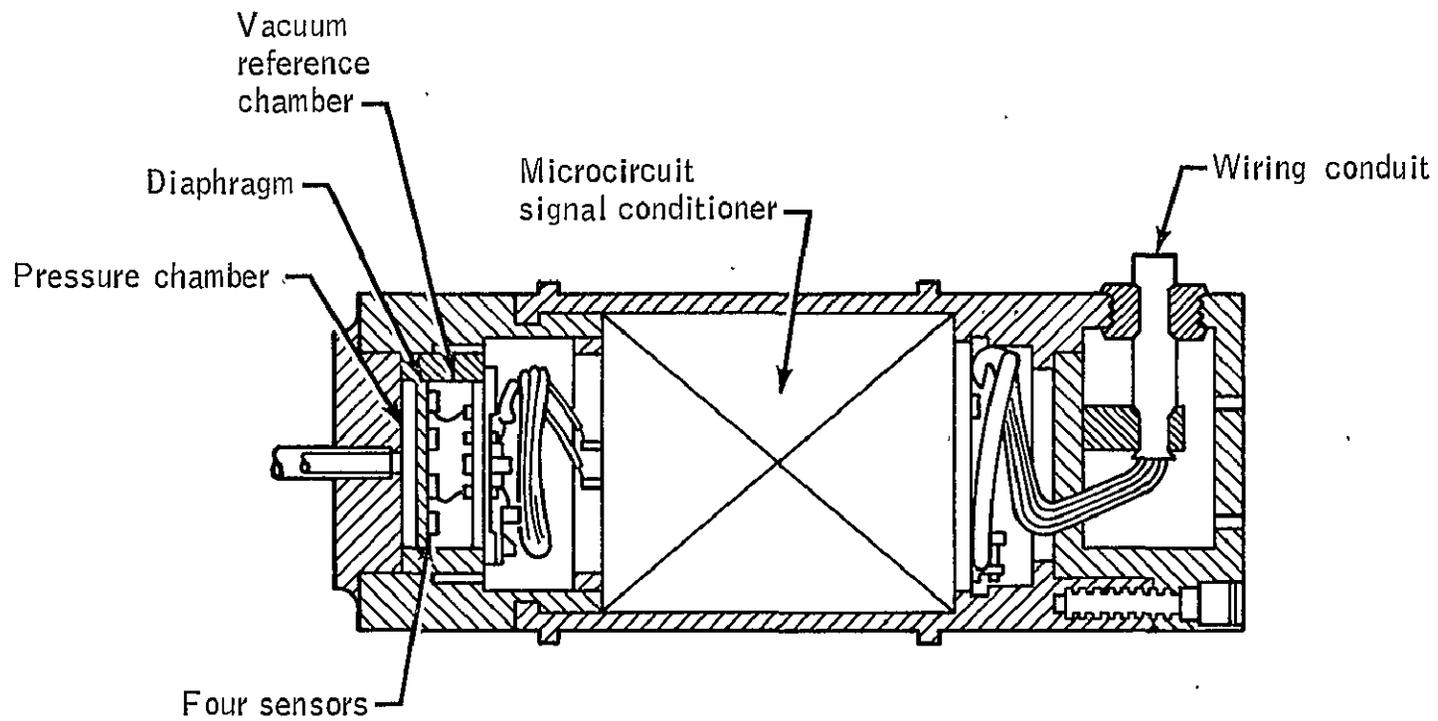


Figure 14-13.- Oxygen pressure transducer.

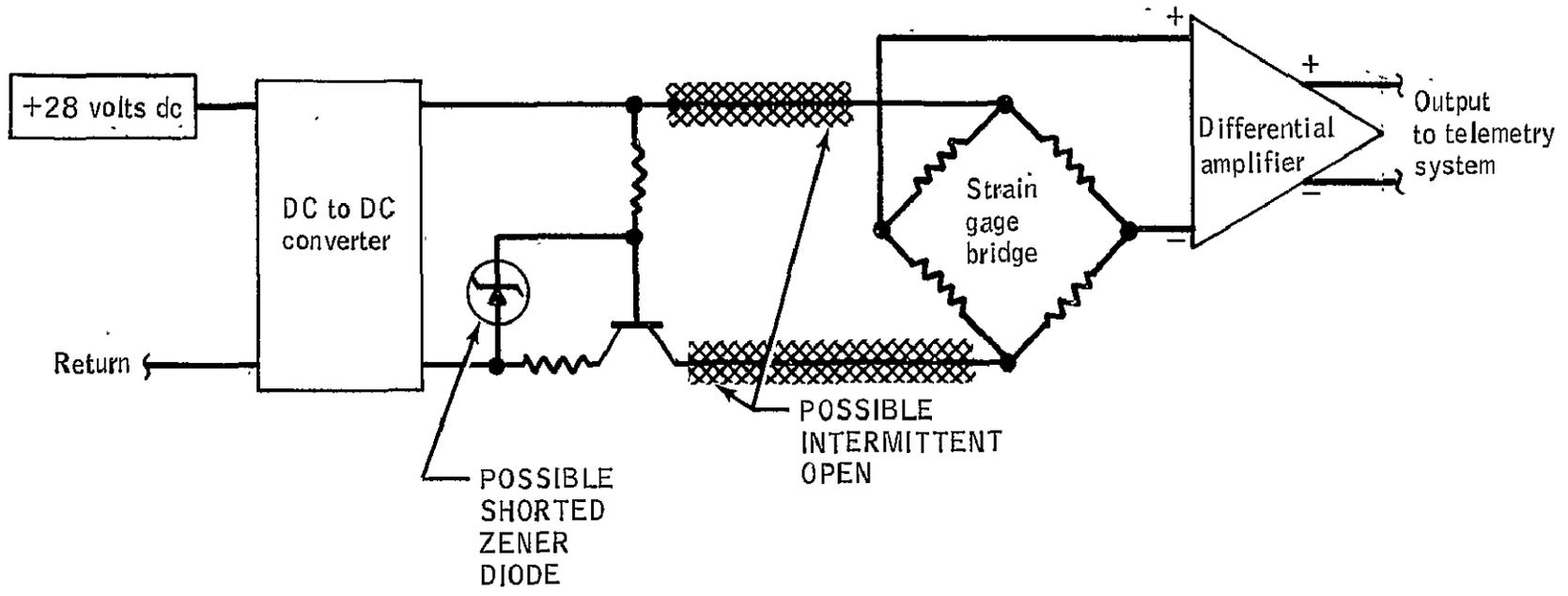


Figure 14-14.- Oxygen tank pressure transducer schematic.

by the fact that the "cordwood" type construction used in this unit characteristically can stress joints and conductors to failure. Contamination in semiconductors and shorts can be ruled out except for a possible, but not likely, short in one of the four zener diodes in the signal conditioner.

This transducer is not planned for use on future missions. Therefore, no corrective action is required.

This anomaly is closed.

14.1.8 Primary Evaporator Inlet Temperature Measurement Failed

The primary evaporator inlet temperature measurement failed off-scale high at 106 hours 52 minutes and subsequently intermittently spiked down from the off-scale-high reading.

The transducer (fig. 14-15) is a platinum resistance thermometer that operates in a resistance bridge. The transducer is connected to the signal conditioner by a twisted shielded four-wire cable. Two of the four wires are connected to the resistance thermometer and the remaining two are shorted together to cancel errors introduced by wiring resistance.

The bridge output is amplified by a differential amplifier and supplied to the instrumentation system. The negative side of the amplifier output is connected to the signal ground of the pulse code modulation assembly. The positive output is connected to the multiplexer of the pulse code modulation assembly.

The intermittent spiking indicates that the cause of the failure is an intermittent short or open in the transducer (fig. 14-15). A similar failure occurred in an identical transducer used to measure service module quad B engine temperature on the first Skylab command and service module.

One possible failure mechanism is that the resistance thermometer circuit may have become open. If this occurred, the resistance bridge would have been unbalanced in a direction to cause the off-scale-high indication.

A second possible failure mechanism is a short to ground in any one of three of the four wires in the twisted shielded quad cable. Such a short would tie the differential amplifier negative output to the negative input through the ground path and drive the amplifier to maximum output, giving the off-scale-high indication. The transducer cable shield is connected to ground at the signal conditioner. The shield connection

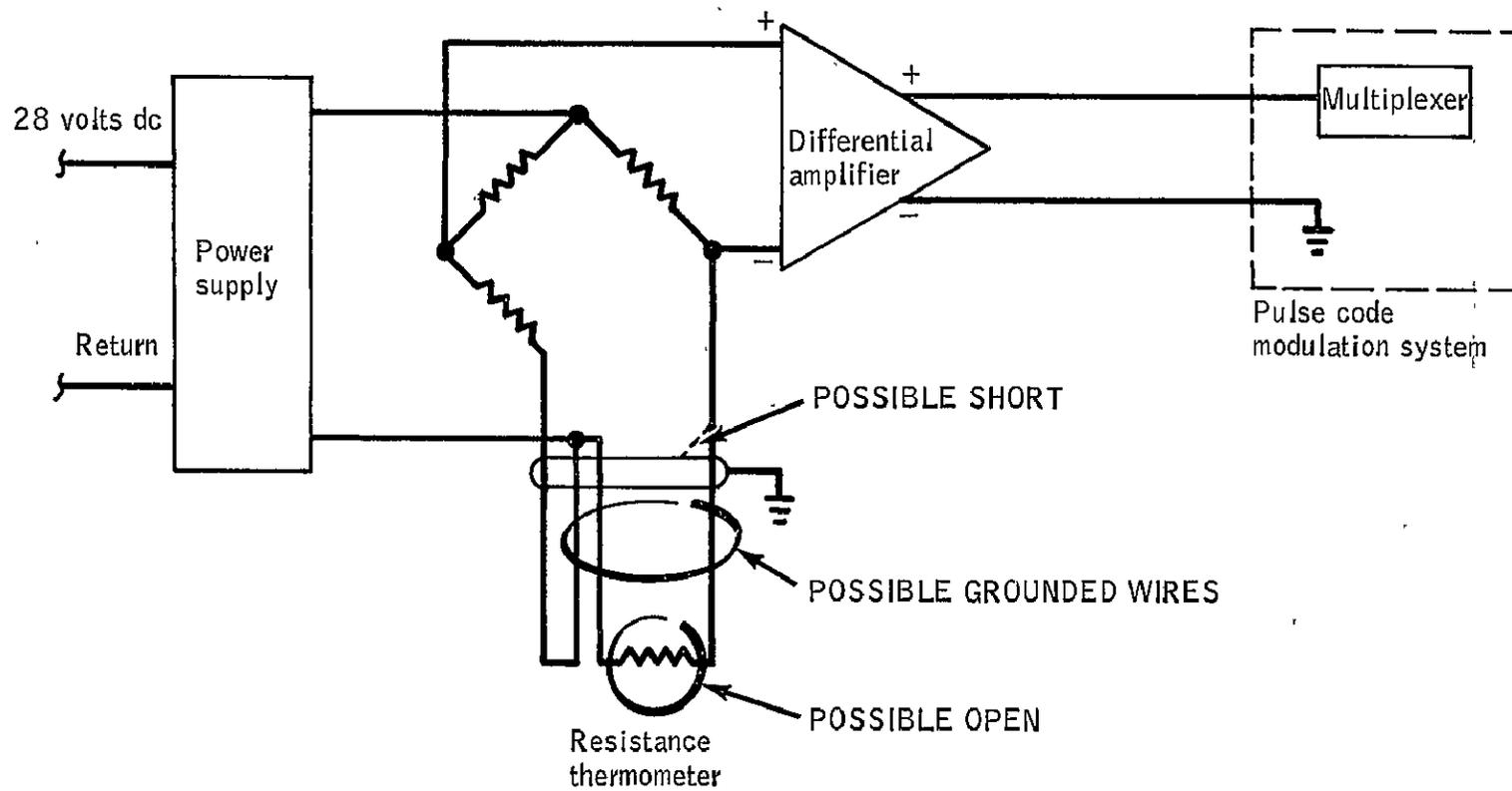


Figure 14-15.- Primary evaporator inlet temperature measurement schematic.

is made by crimping two ferrules around the cable, the first under the shield and the second over the shield. Shorts have occurred previously because the inner ferrule was too small for the wire bundle and it cut through the wire insulation during assembly (fig. 14-16). During subsequent operations, the wire conductor shorted to the inner ferrule. When this problem was first identified, the assembly drawing was changed to allow the use of a ferrule having a larger inner diameter. In addition, a post-assembly shield-to-conductor resistance test was added. Even if the failed transducer was assembled using the small ferrule and was subjected to the resistance test, the wire insulation could have been cut through with the conductor positioned in the ferrule in such a manner that a short did not exist when the resistance was measured.

A third possible failure mechanism is that some failure occurred in the differential amplifier or the portion of the resistance bridge contained in the signal conditioner.

This transducer is not planned for use on any future mission, so no corrective action is required.

This anomaly is closed.

14.1.9 Caution and Warning Tone Inaudible

The caution and warning tone was reported as inaudible on several occasions during the mission.

The caution and warning tone is generated by an oscillator which is enabled by the occurrence of an alert. This tone is fed through isolation resistors to the command and service module audio stations (panels 6, 9 and 10) and the crew communication umbilicals to the headsets.

Postflight testing of the tone amplitude via all 27 possible combinations of audio station panels, crew communication umbilicals and headsets showed all to be normal. The caution and warning tone level was the same as it was prior to the mission setting of 350 millivolts, peak-to-peak. The inaudibility of the tone was probably the result of differences in background noise level between ground testing and the actual flight conditions.

Future designs should consider the use of a variable caution and warning tone that will allow an increased output as mission conditions require.

This anomaly is closed.

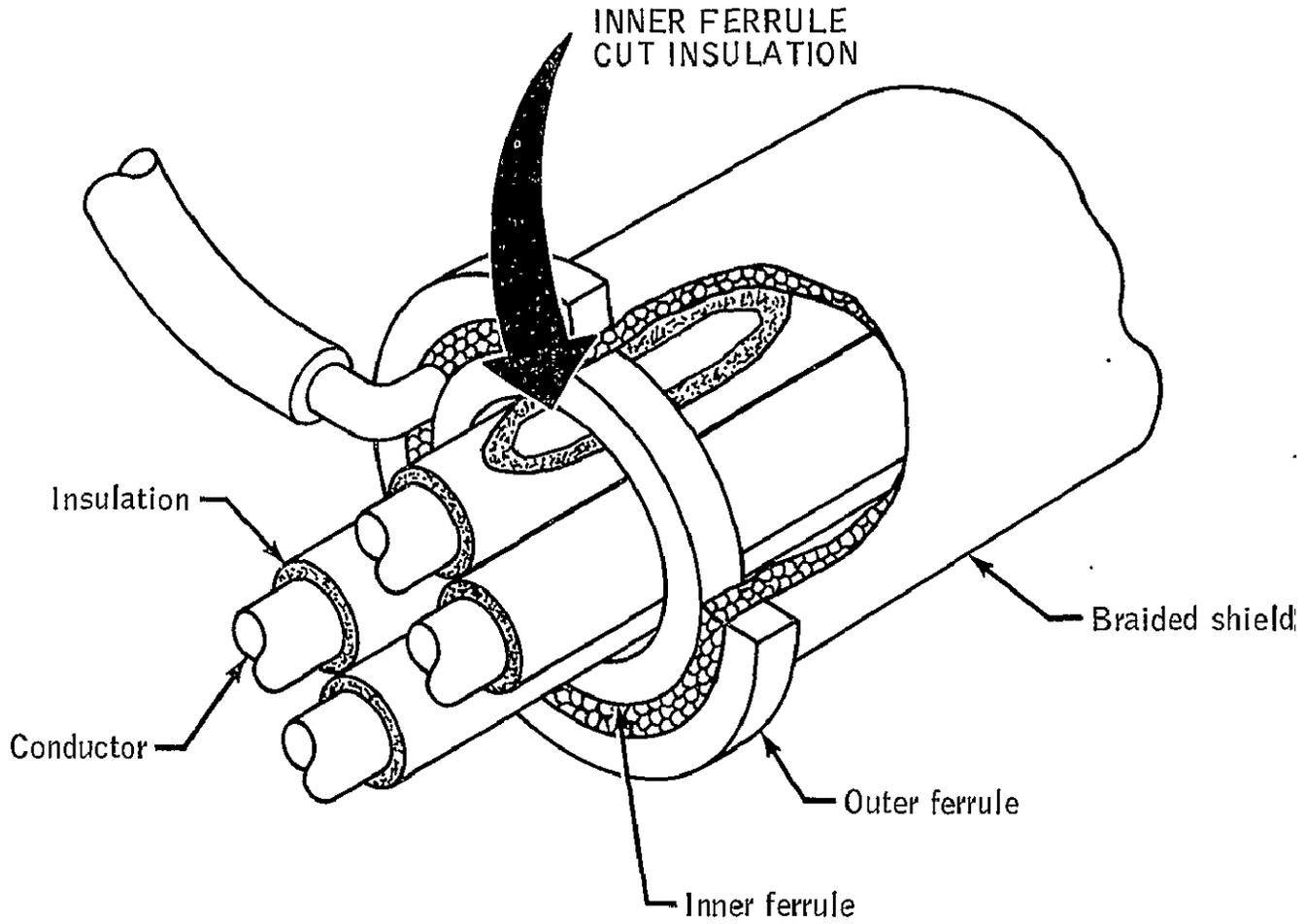


Figure 14-16.- Short in wire bundle from conductor to inner ferrule to braided shield.

14.1.10 Inertial Subsystem Warnings and Coupling Data Unit Fail Indications

The inertial subsystem warning light was observed to illuminate numerous times commencing at 42 hours 32 minutes. Each illumination was accompanied by a program alarm of an electronic coupling data unit failure. The analysis of system data during the flight showed that the alarms were false and the inertial subsystem performance was normal. The coupling data unit fail discrete was inhibited from generating an inertial subsystem warning and master alarm, but was available on display and keyboard readout and on telemetry. Twenty-seven coupling data unit fail indications occurred during the mission.

The coupling data unit assembly contains failure detection circuits which monitor the performance of the assembly. There are individual failure detect circuits for the inertial subsystem and the optical subsystem coupling data unit sections. Figure 14-17 is a functional representation of the pertinent circuits. A failure discrete is sent to the computer if a failure occurs in either subsystem coupling data unit section. The failure discrete is sent to the computer if one or more of the following conditions exist:

- a. Coarse error detect - a disagreement of approximately 0.52 radians between a read counter and the 1X gimbal resolver.
- b. Fine error detect - A disagreement of more than 12.2 milliradians between a read counter and the 16X gimbal resolver.
- c. Cosine ($\theta - \psi$) - Voltage less than 2 Vrms. This voltage is normally 4 Vrms when the read counter (θ) agrees with the resolver (ψ) voltage.
- d. Read counter limit cycle - The read counter changes the direction in which it is counting at a rate greater than 160 times a second.
- e. 14-Vdc power supply - The 14-Vdc power supply in the coupling data unit decreases to 8 volts or less.

During the mission, an analysis was performed in an effort to correlate mission activities and spacecraft operations to the indicated anomaly. Power bus fluctuations, thruster firings, guidance and navigation operations, and experiment activities were examined. No pattern of events or characteristics of operations could be found that could cause or influence the fail indications.

An evaluation of the inertial subsystem was performed in an effort to determine the possible failure source. The units isolated by this evaluation are the electronic coupling data unit, the A-harness, and the computer.

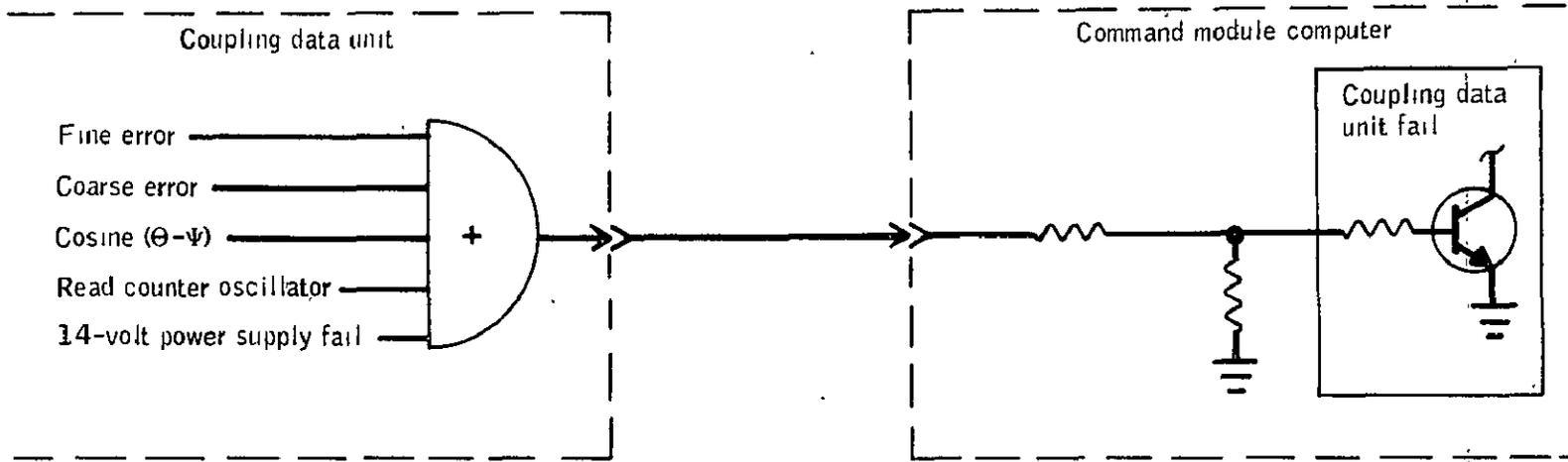


Figure 14-17.- Coupling data unit fail indication.

The electronic coupling data unit mode module fail detect circuit and its fail inputs were evaluated for circuit anomalies that could have caused a fail indication. Circuit analysis indicated six part-type failures (shorts or opens) that could have caused the anomaly. Failures of circuits monitored by the fail detect circuit are not suspect because these failures would have resulted in abnormal inertial subsystem operations, and the guidance and navigation system operations were normal.

The interface circuitry within the coupling data unit between the operational circuits and the fail detection circuitry was tested after the mission for possible high sensitivity to noise. None of the interfaces appeared to be sensitive. The coupling data unit was then subjected to thermal and mechanical shock without inducing the problem. The interface welds in the mode module of the coupling data unit that could have caused the problem were X-rayed and found to be good. The module was then disassembled and the welds were physically checked and found good. The components that could have caused the problem were then tested (including particle impingement noise testing of the semiconductors) and no problems were found. The A-harness and its connectors (P9 and P10) that transmit the coupling data unit fail circuit signal to the computer were also evaluated. Shorts within the harness, as shown in figure 14-18, could have caused a failure indication. However, postflight thermal and mechanical shock testing did not cause the problem to recur. Therefore, the intermittent nature of the anomaly would make this source unlikely.

The computer modules that monitor the coupling data unit fail outputs are the input/output module and interface module. Past failure analyses of these types of modules for particle contamination or lifted bonds, part degradation, or poor workmanship have revealed no problems related to coupling data unit fail indications. The computer modules were X-rayed and disassembled and the components were checked but, again, no problems were found.

The cause of the failure was not determined so no corrective action can be taken.

This anomaly is closed.

14.1.11 Intermittent Attitude Display on Flight Director Attitude Indicator 1 When in Orbital Rate

The Docking Module Pilot reported prior to docking module jettison that the attitude ball on flight director attitude indicator 1 did not always flip to orbital attitude when the "orbital rate 1" switch was set from "inertial" to "orbital rate." The attitude ball did not always indicate the correct orbital attitude when it did flip.

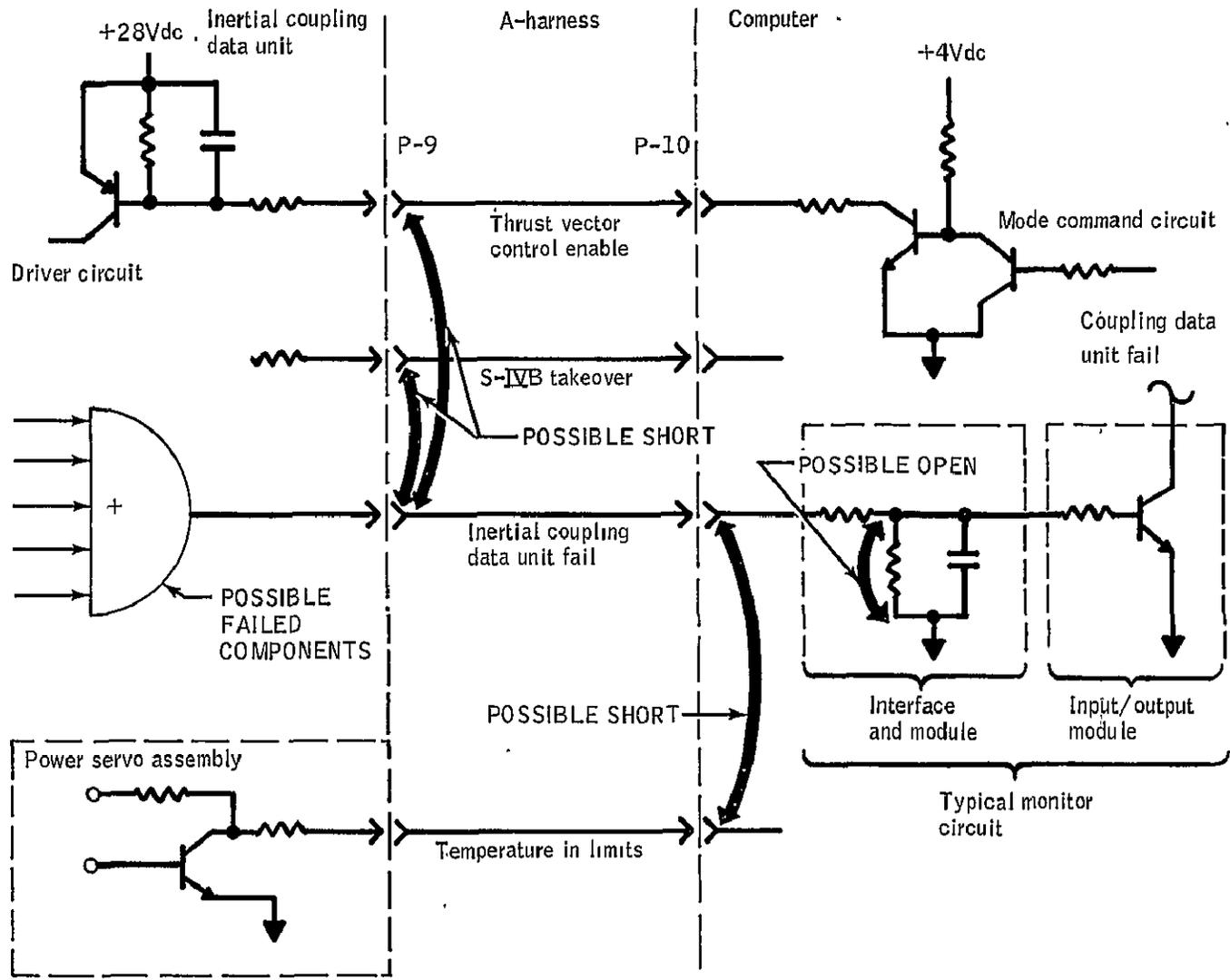


Figure 14-18.- Inertial coupling data unit fail - interconnect diagram.

The inertial attitude angle sources are the inertial measurement unit and the gyro display coupler. Either attitude source may be switched to feed either or both flight director attitude indicators. The yaw and roll attitude signals are transmitted to the flight director attitude indicators directly (fig. 14-19). Each pitch attitude signal is routed through an orbital rate switch. When an orbital rate switch is in the inertial position, the pitch signal is transmitted unchanged to the flight director attitude indicator. When the switch is in the orbital rate position, the pitch signal is transmitted to an orbital rate drive resolver. The resolver converts the inertial pitch attitude signal to an orbital pitch attitude signal. The resolver output is then transmitted back to the switch and from there to the flight director attitude indicator, where the attitude ball displays the local vertical of the spacecraft.

Since both flight director attitude indicators operated in the inertial mode, the attitude sources and flight director attitude indicators themselves were operational. Since flight director attitude indicator 2 operated in orbital rate and flight director attitude indicator 1 did not and both resolvers are driven by the same stepper motor, there are only four possible failure mechanisms, all in the orbital rate drive. They are an intermittent short or open in resolver assembly 1 or orbital rate switch 1 (fig. 14-19).

An orbital rate drive will not be used on the Space Shuttle. Therefore, a teardown of the orbital rate drive electronics assembly for failure isolation is not required.

This anomaly is closed.

14.1.12 Bioinstrumentation System Data Erratic

The Commander's biomedical data were erratic during an exercise period in the docking module at about 157 hours. Earlier, the Docking Module Pilot's data had been erratic, and the Pilot used the Commander's biomedical belt successfully after some initial intermittent data while exercising.

The bioinstrumentation system (fig. 14-20) is attached to the crewman's constant wear garment during exercise sequences, and provides a physical status check of the crewman's heart and lung action. The electrocardiograph circuitry receives heart signals via the sternal harness which has three electrodes placed on the crewman's skin. The impedance pneumograph circuitry receives respiration rate signals via the axillary harness that has two electrodes placed on the crewman's skin. Electrodes are held to the skin with doughnut shaped stomaseal tape, as shown in figure 14-21, that has adhesive applied on both sides.

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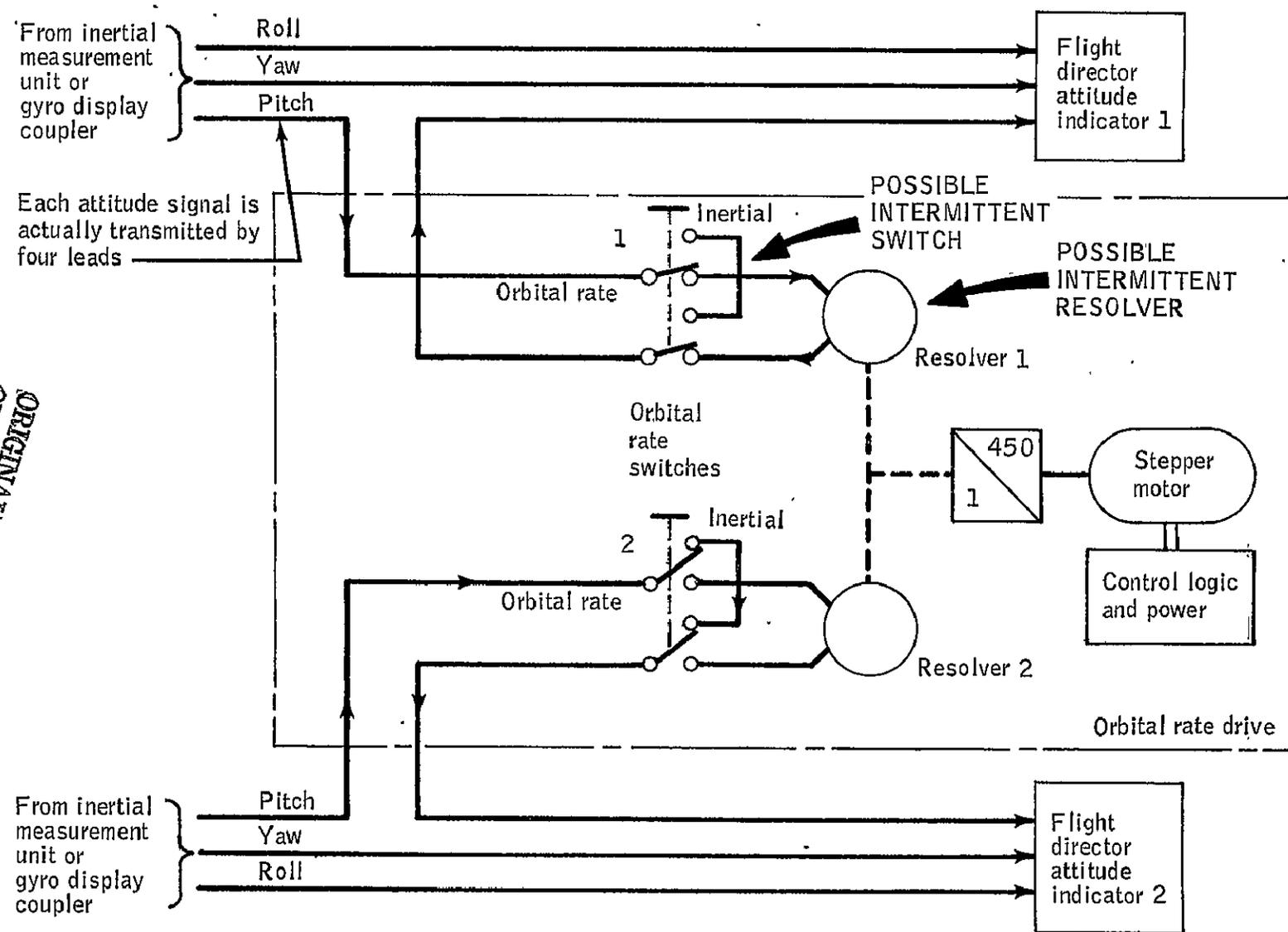


Figure 14-19.- Orbital rate display/flight director attitude indicator interface.

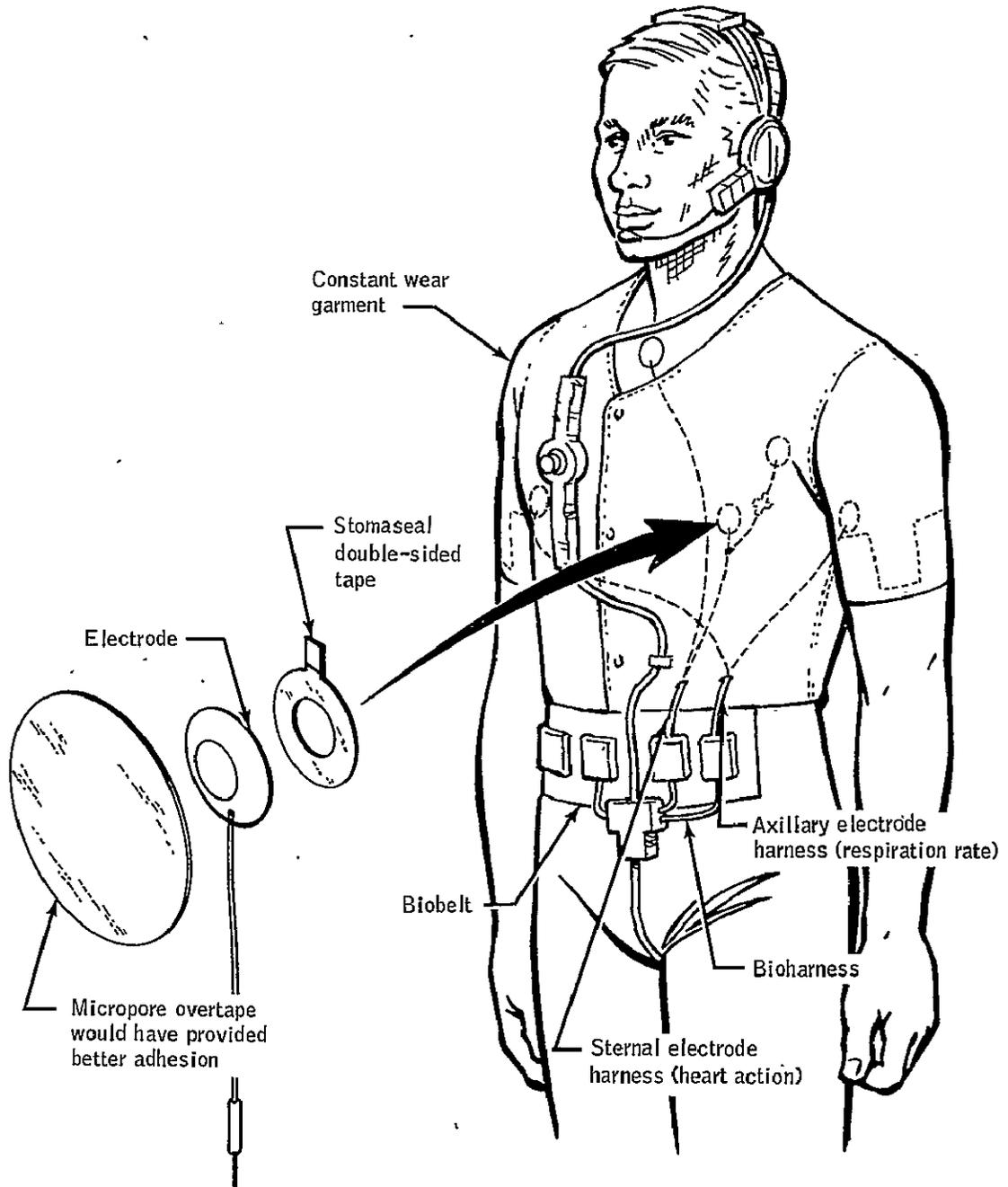


Figure 14-20,- Electrode installation.

During exercise sequences, each crewman used a restraining strap between the docking module hand rails to tie himself down. The strap slid across the biomedical belt and the lower portion of the electrode harness as he moved back and forth, putting considerable strain on the biomedical belt and electrode harness. This strain caused the electrodes to pull away from the crewman's skin.

The problem was recreated during postflight testing by lifting electrode sensors from the test subject's skin. When the ground electrode of the sternal harness was lifted, all the data channels became noisy. All the bioinstrumentation hardware was functional.

When a crewman wears bioinstrumentation with the pressure garment assembly, the electrodes are not easily accessible, and a micropore over-tape is used over the regular stomaseal tape for greater adhesion. This micropore overtape was not used when a crewman was dressed in a constant wear garment. When a crewman expects to exercise with a restraint strap, he should apply micropore overseals on the electrodes for greater adhesion to the skin.

This anomaly is closed.

14.2 EXPERIMENTS

14.2.1 Soft X-Ray Experiment Problems

The soft X-ray experiment instrument exhibited two distinct and unrelated anomalies during the mission. The first was that a high voltage discharge occurred each time the instrument was operated, usually after an initial period of correct operation that lasted from 1 to 20 minutes. This problem caused the loss of approximately 80 percent of the expected data from the experiment. The second was that one of the two calibration sources stuck in the field of view. This problem did not cause a serious degradation of the data.

The X-ray instrument consisted of a multiwire proportional counter, a high voltage power supply, and pulse detection and analyzing circuits. The multiwire proportional counter consisted of a primary detector surrounded by a 2.54-centimeter-thick background detector on five of the six sides. Each detector contained an array of anode and cathode wires. The primary detector anode wires were tied together and were connected to the high-voltage supply through a high resistance. These anode wires, with their surrounding cathode wires and the detector front face, constituted the primary detector. The background counter anode wires were tied together and were connected through a separate high resistance to the same

high-voltage supply as the primary detector. These wires, along with their surrounding cathode wires and the sides and back of the detector case, constituted the background detector. One background detector anode wire went around the entire periphery of the detector. All cathode wires were connected to the case and, hence, were at ground potential. Two radioactive sources were provided that could be rotated into the detector field of view by solenoid actuators. These sources were used for calibration of the instrument. The counter is shown in figure 14-21.

A single primary and background section of the counter is shown in figure 14-22. X-rays in the energy region of interest can enter only through the detector face entrance foil. In this energy range, an X-ray photon interacts with the counter gas predominantly through the photoelectric effect, ejecting a photoelectron from a gas molecule. The ejected electron loses its energy by ionizing other gas molecules. The electrons liberated are accelerated toward the anode wire which is at a high positive potential. If the counter is operated at a voltage in the proportional region, these electrons gain enough energy to ionize other electrons, resulting in an avalanche. The resultant avalanche of electrons is collected at the anode wire and causes an electrical pulse having an amplitude proportional to the energy of the incident X-rays photon. This pulse is fed through a capacitor to the input of the primary detector amplifier for further amplification and shaping. In this case, the X-ray is removed from the picture by the initial interaction. An energetic charged particle entering the detector (fig. 14-22), however, will traverse both the primary and background sections of the detector, ionizing gas molecules along its entire path. Ionization of the gas in the primary section will produce a pulse on the primary detector anode wire and in the background counter anode wire. These pulses are then fed to their respective amplifiers and put into a coincidence unit whose output blocks analysis of the pulse.

Some X-rays can, of course, be mistaken for particles. For example, photoelectrons created near the back of the primary counter can enter the background section, thereby creating pulses in both sections. This effect can be "calibrated out," however, when calibrations are performed on the sensors. Charged particles which have enough energy to penetrate the entrance foil but not enough energy to traverse the primary counter and enter the background section can also be mistaken for X-rays. Knowledge of the anticipated particle spectrum allows correction for this effect.

High-voltage discharge.- High-voltage breakdown in proportional counters is a common problem. The gas gain region of the detector amplifies any discharge or breakdown due to sharp points, metal chips, or inadequate spacing of elements. The soft X-ray instrument exhibited an intermittent breakdown problem. It operated correctly for periods ranging

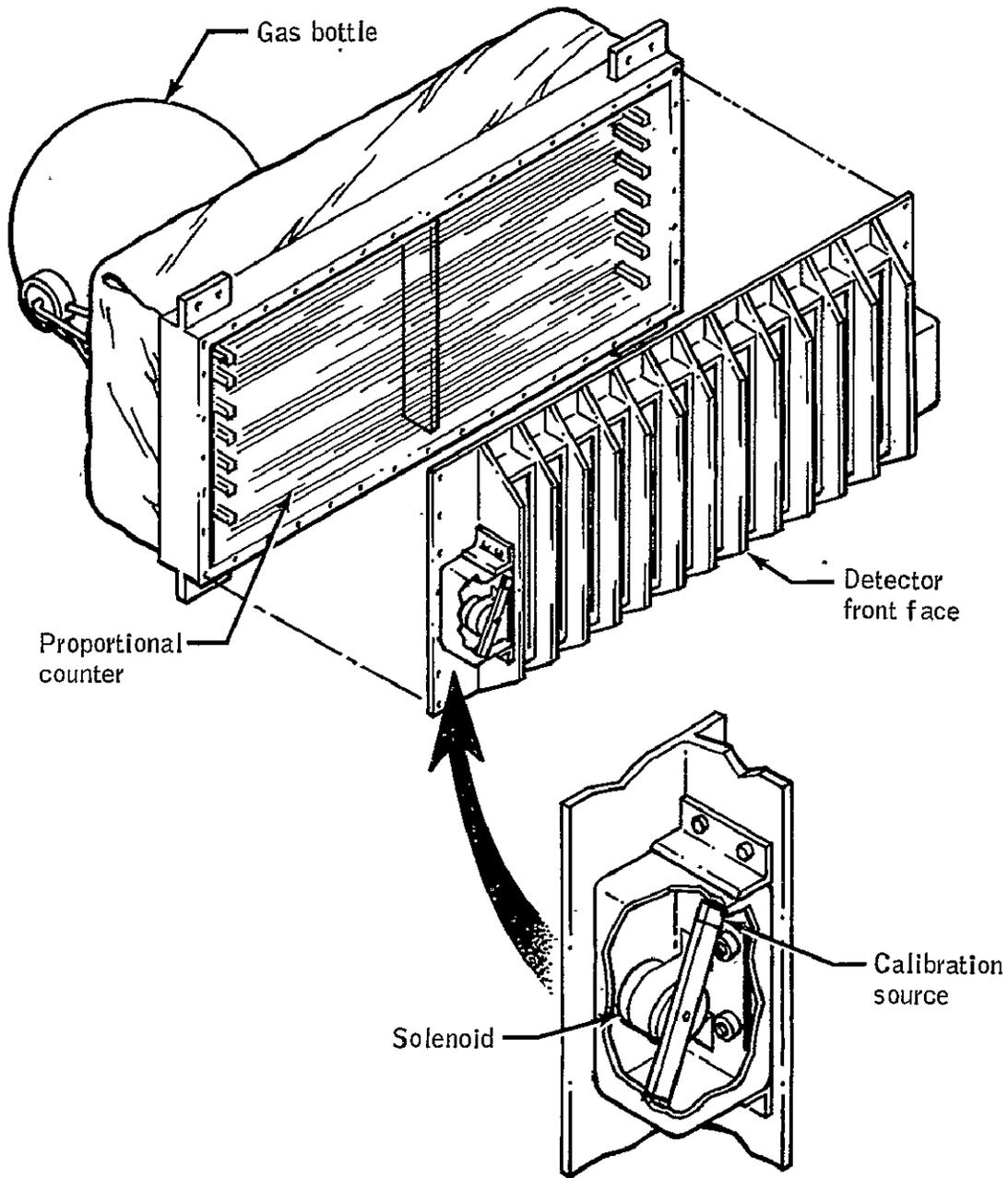


Figure 14-21.- Detector assembly.

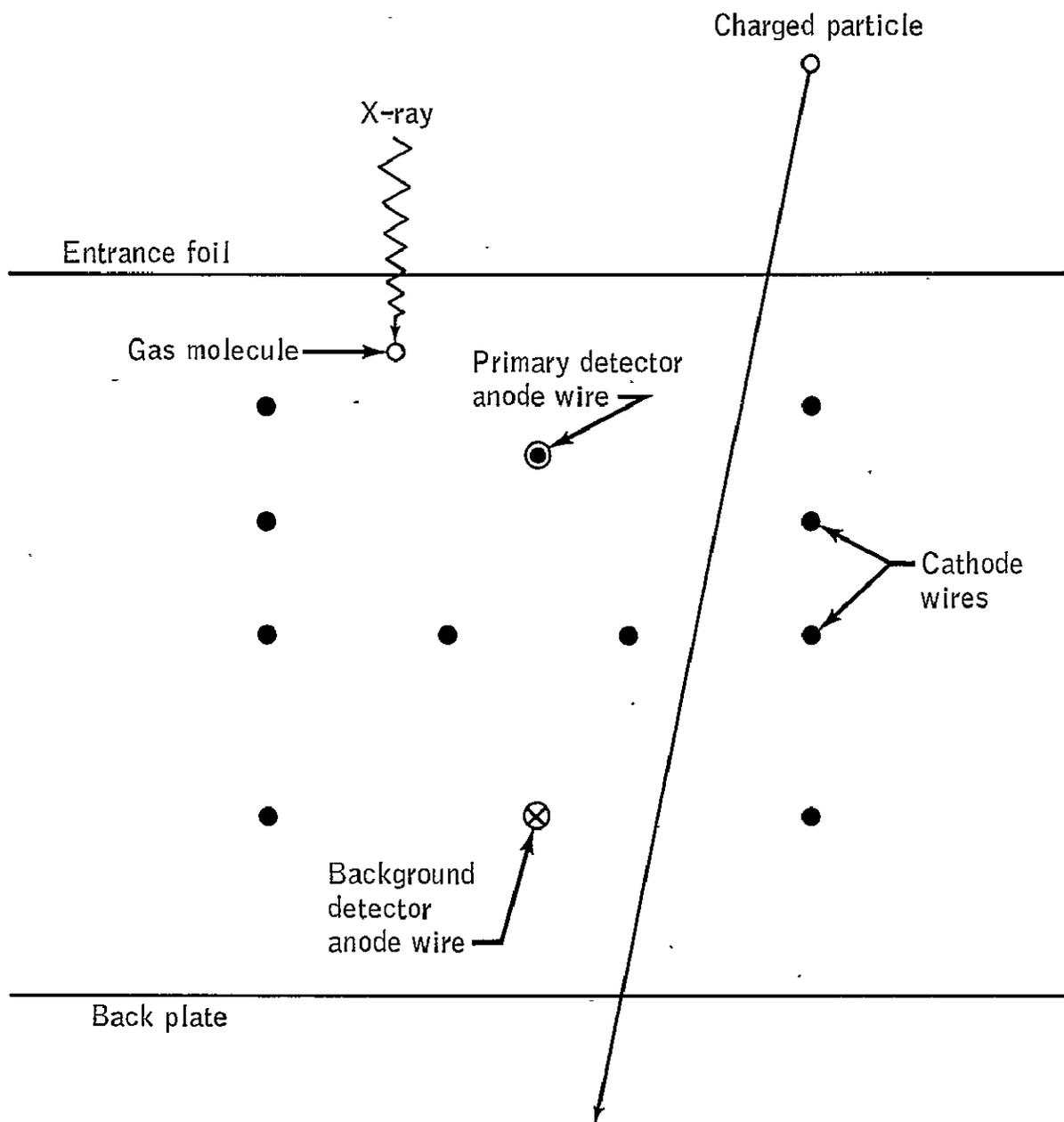


Figure 14-22.- Detector operation.

from 1 to 20 minutes before going into discharge. A brief operating history for the flight is given. The count rates went full-scale when the experiment was operated during the raster scan for the extreme ultraviolet survey experiment (MA-083). In anticipation of a possible instrument problem, a gas purge was performed. After purging, the instrument was operated on the sixth day of flight and 25 minutes of good data were obtained before the problem reappeared. On day 7, a test was performed in which the instrument was operated with the high voltage on for 2 minutes and then off for 2 minutes. The instrument operated properly for approximately 3 minutes out of the 5 minutes during which the high voltage was on. The experiment operation procedure for day 8 was revised to incorporate the 2-minute-on mode of operation and at least 30 more minutes of good data were obtained.

On days 8 and 9, an engineering test was performed in an attempt to determine the cause of the instrument malfunction. At the end of the data-take on day 8, the instrument was left in a "contingency power down" mode. This mode evacuated the gas volume of the detector. The detector was allowed to pump down to a hard vacuum overnight and, on day 9, the high voltage was turned on with no gas in the detector. No counts were observed over a period of 30 minutes or more, indicating that the detector malfunction was located in the gas gain volume of the detector. High-voltage potting, power supplies, the other high-voltage components external to the gas volume of the detector were eliminated as possible causes of the malfunction.

The high voltage was turned off, the detector gas volume was refilled to a nominal pressure of 11 newtons/sq cm, and the high voltage was activated again. The detector malfunction reappeared in approximately 2 minutes, thus confirming that the breakdown was occurring in the gas volume of the detector.

A metal chip or other point discharge in the detector gas volume is the most probable cause of the problem. A chip could have shaken loose during launch, and the chip's floating in zero-g could explain the intermittent behavior. Metal chips were a constant problem during window changeouts due to the binding of the screws in the steel locking inserts. Precautions were taken to prevent them from falling into the detector, and a close inspection of the detector was made; however, a chip may not have been detected.

Calibration source.— One calibration source stuck in the field of view of the detector. The problem first occurred during the initial turn-on on July 21. The calibration source retracted after several momentary applications of power by the crew using the PURGE/CAL switch. However, the source was seen to be back in the field of view several orbits later, apparently because of crew operation.

A calibration source problem occurred during fabrication and in qualification testing. It was thought to be due, at that time, to inadequate space between the rotary solenoid pivot axis and cover. This problem was supposedly corrected and no such problems were found in flight acceptance testing.

The experiment is not presently planned for subsequent missions, therefore, no corrective action is required.

This anomaly is closed.

14.2.2 Ultraviolet Absorption Experiment Nitrogen Lamp Intensity Monitor Output Was Erratic

The ultraviolet absorption experiment (MA-059) nitrogen lamp intensity monitor output was erratic at about 105 1/2 hours during the final data scan. The closed-door flight calibration and the lamp current indicated that the lamp was operating properly. Oscillations first occurred when the ultraviolet absorption experiment housing temperature reached 305.5° K. The temperature rose as high as 312.5° K and never returned below 305.5° K.

As shown in figure 14-23, the reflected signal is amplified by a feedback amplifier (U2). Increased gain with increasing temperature is typical of feedback amplifiers. Increased gain in a feedback amplifier will cause oscillation and is the probable cause of the anomaly.

Since this experiment will not be flown on future missions, no corrective action will be taken.

This anomaly is closed.

14.2.3 Doppler Tracking Experiment Data Recorder Reel Immobile

When the experiment dual redundant tape recorders were activated at 191 hours and 45 minutes, neither tape recorder reel transport moved. Switch toggling caused tape recorder 1 to start but tape recorder 2 did not start.

Figure 14-24 shows the arrangement of the recorder components with the tape threaded through the transport mechanism traveling across the erase, record, and playback heads. It is important to note that the tape is always resting against the heads, even when it is not being transported.

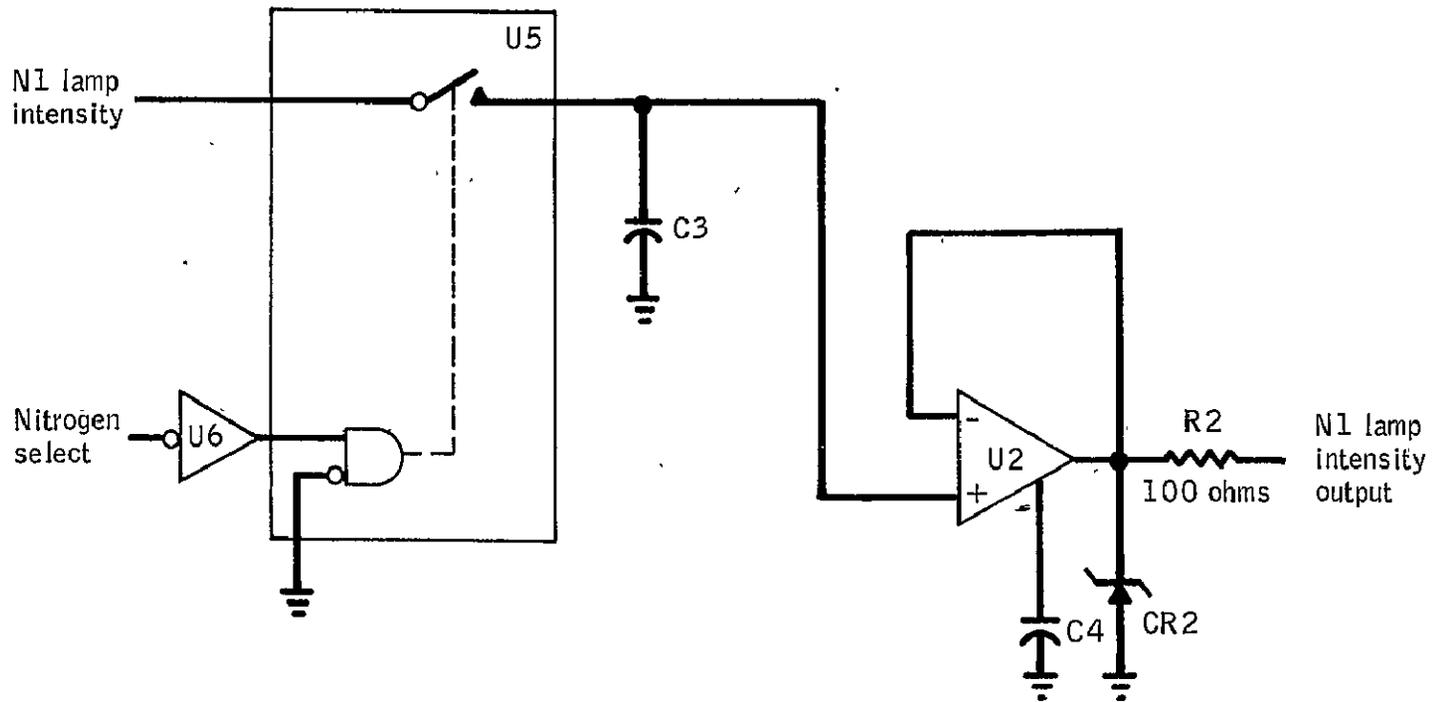


Figure 14-23.- Nitrogen lamp intensity monitor circuit schematic.

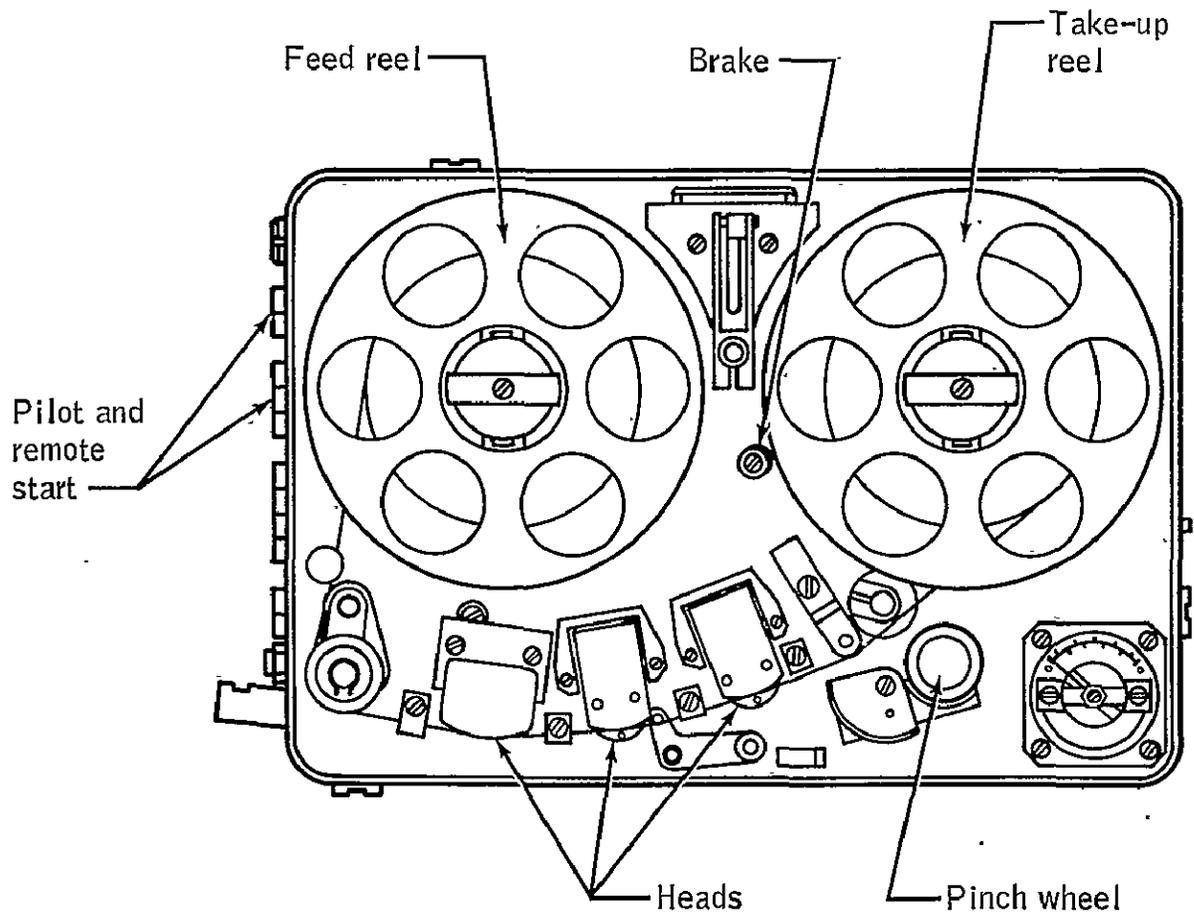


Figure 14-24.- Doppler tape recorder.

During postflight inspection, the recorder 2 tape was found stuck to all three heads, thus rendering the transport ineffective. A chemical analysis of the tape surface material adjacent to the heads was performed. The sample was so small, however, that no positive identification of the material was obtained. At the same time, the pair of backup flight tape recorders was energized for comparison testing. The same problem occurred in that one of the redundant recorders was immobile because of a stuck tape. Both tape recorder systems operated normally once the tape was freed from the heads.

The Apollo Soyuz mission prelaunch countdown sequence entailed setting these recorders at the ready 5 days before lift-off. Adding about 6 days to the operate point in the mission makes 11 days that the tape was resting against the heads at the same contact points. The backup system went through the same time profile.

The solvent used to clean the recorder heads was a mixture of xylene and trichloroethane. All tape manufacturers consulted state that aromatic hydrocarbons, such as xylene, should not be used for recorders that contain any plastic parts and also that such solvents should not be allowed to contact the tape. Also, since xylene has a relatively high boiling point (417.6° K), complete vaporization of the residuals from cleaning takes a relatively long time. Adsorbed residuals tend to leach out, carrying semidissolved tape emulsion to the point of contact of tape and heads. The emulsion would behave like a paste.

The xylene constituent of the head cleaner was not completely vaporized (baked out) of the vicinity of the tape recorder heads when the tape was placed against them for 11 days without movement. The result was adherence of the tape to the heads.

If a tape recorder which permits continual contact between tape and head must be used, then it should be cycled to pull through some tape periodically to prevent adhesion. Secondly, isopropyl alcohol should be used as a solvent instead of aromatic solvents. (The boiling point of isopropyl alcohol is 355.5° K, and thus is more readily vaporized.)

This anomaly is closed.

14.2.4 Electrophoresis Technology Experiment Unit Leaked

The Commander reported at 176 hours 34 minutes that the Docking Module Pilot had stopped operating the electrophoresis technology experiment because of a leak. A blob of electrolyte had leaked out around the column assembly left-hand housing and there were gas bubbles in the left and right fluid lines.

The electrophoresis unit electrolyte circulation system provided a constant flow of electrolyte around electrodes in a column assembly as shown in figure 14-25. (The experiment used eight column assemblies, six of which used the circulator system.) The electrolyte flow carried away oxygen and hydrogen products of electrolysis from the electrodes. A phase separator was connected to the anode side to remove liberated oxygen and another phase separator was connected to the cathode side to remove hydrogen. A glass plug and fluid-tight membrane separated buffer solution and sample materials from the electrolyte.

To prepare the experiment for operation, a column assembly was electrically connected to the electrophoresis unit as shown in figure 14-26. The fluid disconnects were then secured at the electrode housings and a frozen sample slide was inserted in a recess of the cathode housing. Each disconnect assembly had two needles (fig. 14-27) that pierced a plastic septum when inserted into the housing orifices. One needle provided gas-free electrolyte that flowed around the electrode where the gas products of electrolysis were entrained. The gas laden electrolyte was drawn out through the other needle.

Postflight testing has shown that none of the components of the fluid system and the column assembly leak. However, if the disconnect assembly is lifted 1.5 millimeters from a fully seated condition, or rocked in place, electrolyte does seep out. In a 3.45 newton/sq cm cabin atmosphere, the pressure differential would be greater, tending to allow more leakage.

Since the circulation system contained about 100 cubic centimeters of electrolyte, and about 2 cubic centimeters were found in the system during postflight testing, it appears that most of the liquid leaked out in flight. (The Commander noted that about a cupful of liquid leaked out.) After most of the electrolyte leaked out, the transparent left and right feed lines filled with cabin air, accounting for the gas bubbles observed.

The disconnect assembly needles tore openings in the thin septum when inserted, rather than clean holes, providing a poor seal. That, combined with a disconnect assembly that was not completely seated would allow leakage of electrolyte.

If an experiment like this is performed in the future, a locking clasp should be provided to secure the disconnect against the septum. In addition, a thicker septum should be used to provide a better seal between the needle and septum.

This anomaly is closed.

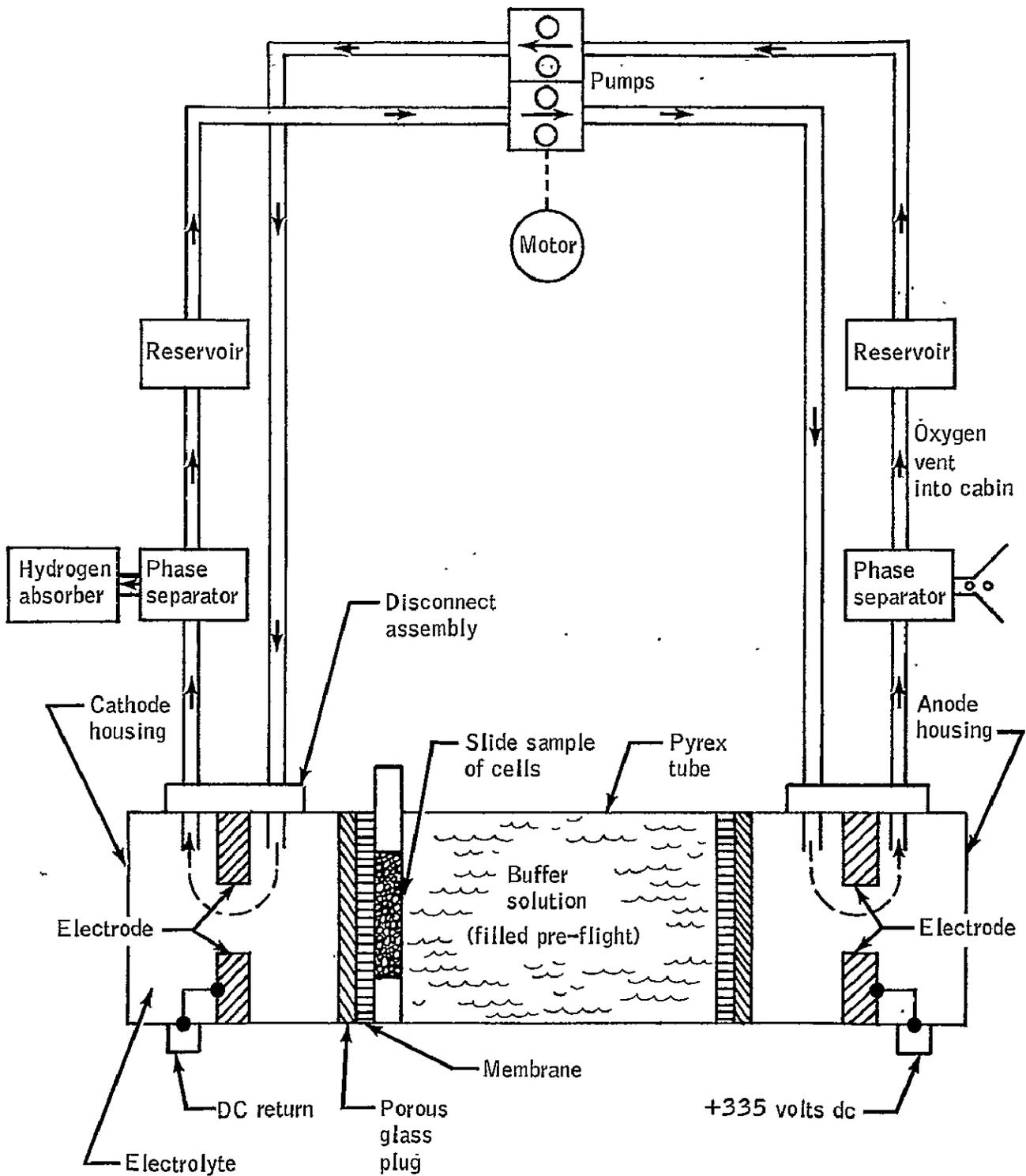


Figure 14-25.- Electrophoresis technology unit fluid system functional schematic.

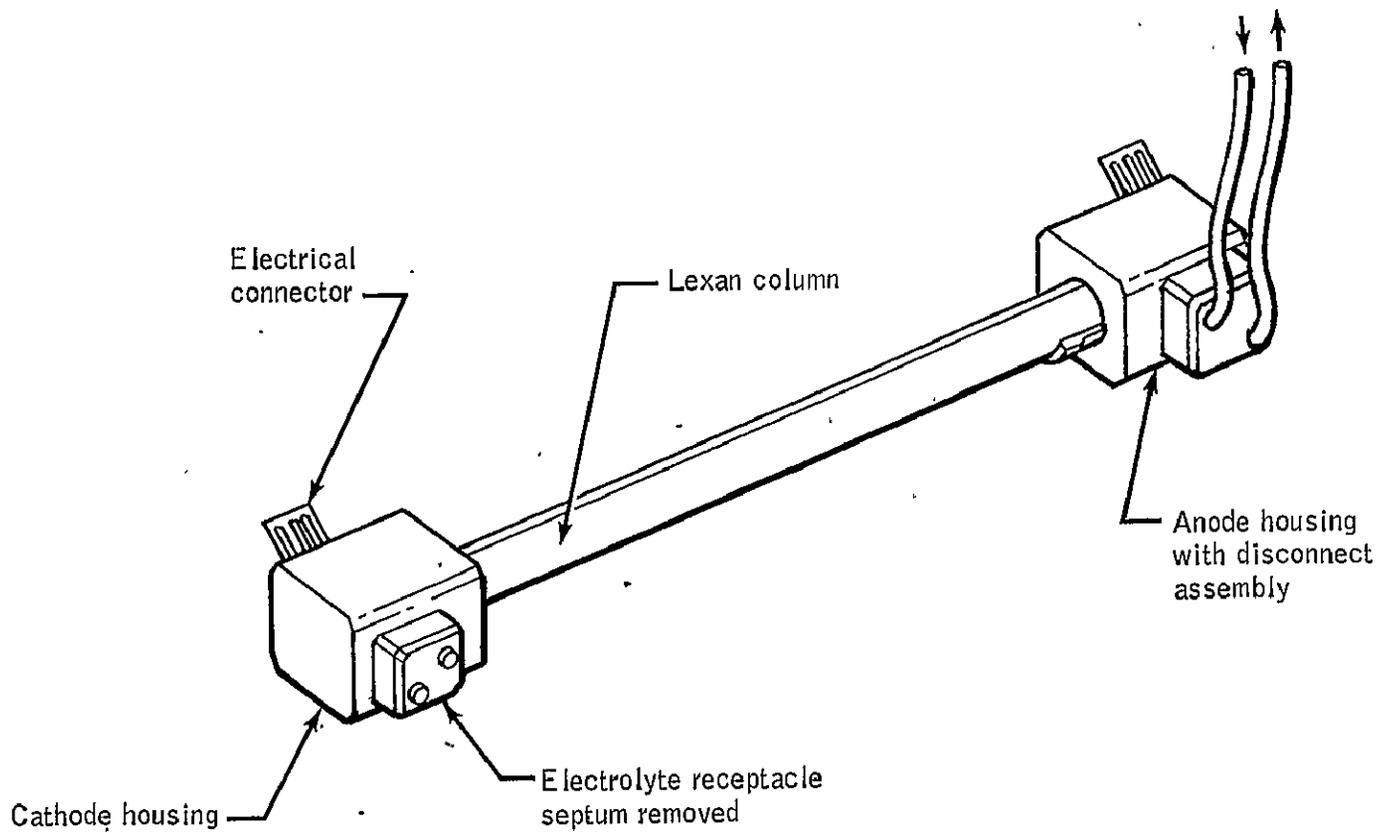


Figure 14-26.- Electrophoresis column assembly.

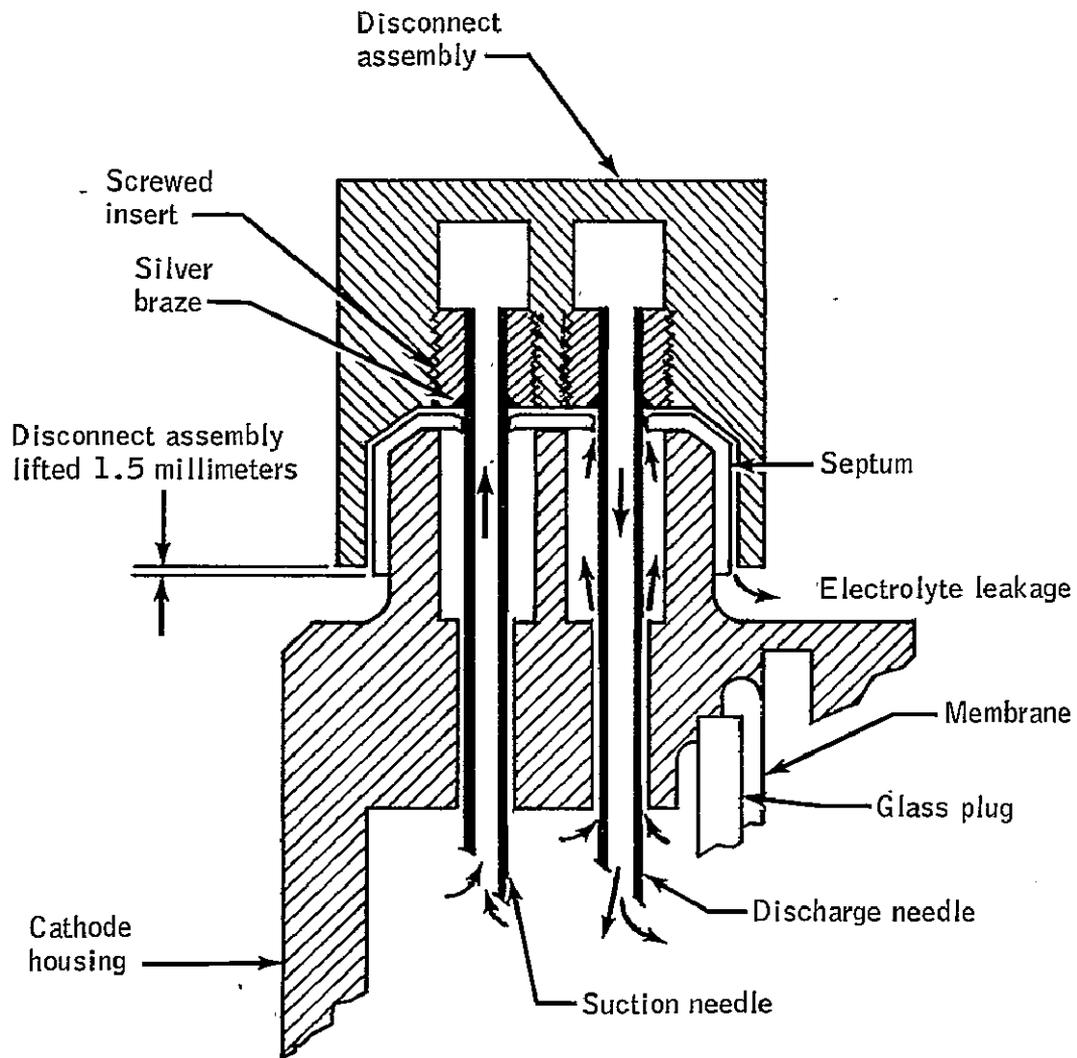


Figure 14.27.- Liquid disconnect interface on the cathode housing.

15.0 CONCLUSIONS

1. The Apollo Soyuz Test Project demonstrated that two nations with similar space technology capabilities can cooperate to successfully carry out complex joint manned space flight operations even though languages and cultures are different.

2. The satisfactory performance of the systems developed to permit rendezvous and docking of the Apollo and Soyuz spacecraft and transfer of crews between spacecraft demonstrated the feasibility of achieving an international docking capability.

3. Reliance on checklists for critical manual functions may result in dangerous situations. Automation should be relied upon, where practical, for all time-critical functions.

4. Expanded Spaceflight Tracking and Data Network coverage provided by the use of a communication relay satellite in geosynchronous orbit is a significant improvement in the method of air/ground communications for earth orbiting spacecraft.

5. The design and the techniques employed for control of spacecraft atmospheres having differing pressures and compositions during docked operations were satisfactorily demonstrated.

6. Electrical field measurements by airborne and ground instrumentation at the launch site was a satisfactory technique for determining an acceptable risk for lightning involvement with the launch for marginal weather conditions.

7. The doppler tracking experiment and the geodynamics experiment demonstrated that satellite-to-satellite tracking is an effective technique in determining small variations in the earth gravitational field.

8. Discovery of an extreme ultraviolet source demonstrated this new field of astronomy and the continued value of space scientific research.

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November 1975.

APPENDIX A - SYSTEMS AND EQUIPMENT DESCRIPTION

This section contains descriptions of the systems and equipment flown for the first time, and descriptions of changes to previously flown hardware.

The space vehicle configurations are shown in figure A-1. The Apollo and Soyuz spacecraft were similar in most respects to those flown previously.

Different Apollo and Soyuz atmosphere total pressures prevented the spacecraft from docking directly with each other. (The normal Apollo operating atmosphere was 100 percent oxygen at 3.44 newtons/sq cm and the Soyuz atmosphere was a mixture of approximately 40 percent oxygen and 60 percent nitrogen by volume at 10.1 newtons/sq cm.) Therefore, a docking module was built to serve as an airlock that isolated the spacecraft atmospheres from each other and allowed the crews to transfer between the spacecraft. In addition, the docking module served as a structural base for the docking mechanism that interfaced with the Soyuz docking mechanism. In order to facilitate transfer operations, the Soyuz was modified so that total pressure could be decreased from 10.1 newtons/sq cm to about 6.9 newtons/sq cm prior to the joint activities phase of the mission while maintaining an oxygen partial pressure of about 2.3 newtons/sq cm. Figure A-2 shows the spacecraft coordinate systems and figure A-3 shows the hatch and tunnel designations of the docked spacecraft.

A.1 DOCKING MODULE

A.1.1 Structure

The docking module was a cylindrical pressure vessel with an internal diameter of 1.42 meters and an overall length of 3.15 meters between the docking interfaces. The wall of the docking module was fabricated from aluminum-alloy rolled plate formed into two semicylindrical sections and welded together to form a hollow cylinder. A truncated conical assembly welded to one end of the cylinder contained a tunnel and provisions for installation of a hatch and an Apollo-type docking drogue. A ring welded to the other end provided for attachment of a bulkhead, a hatch, and the docking system that was compatible with the Soyuz docking system. The bulkhead and docking system were bolted to the ring. The two docking module hatches were machined aluminum structures fitted with existing command module tunnel hatch mechanisms. Each hatch had a pressure differential gage and a pressure equalization valve. Both hatches opened into the docking module. When the hatches were closed, the docking module was a self-contained pressure vessel.

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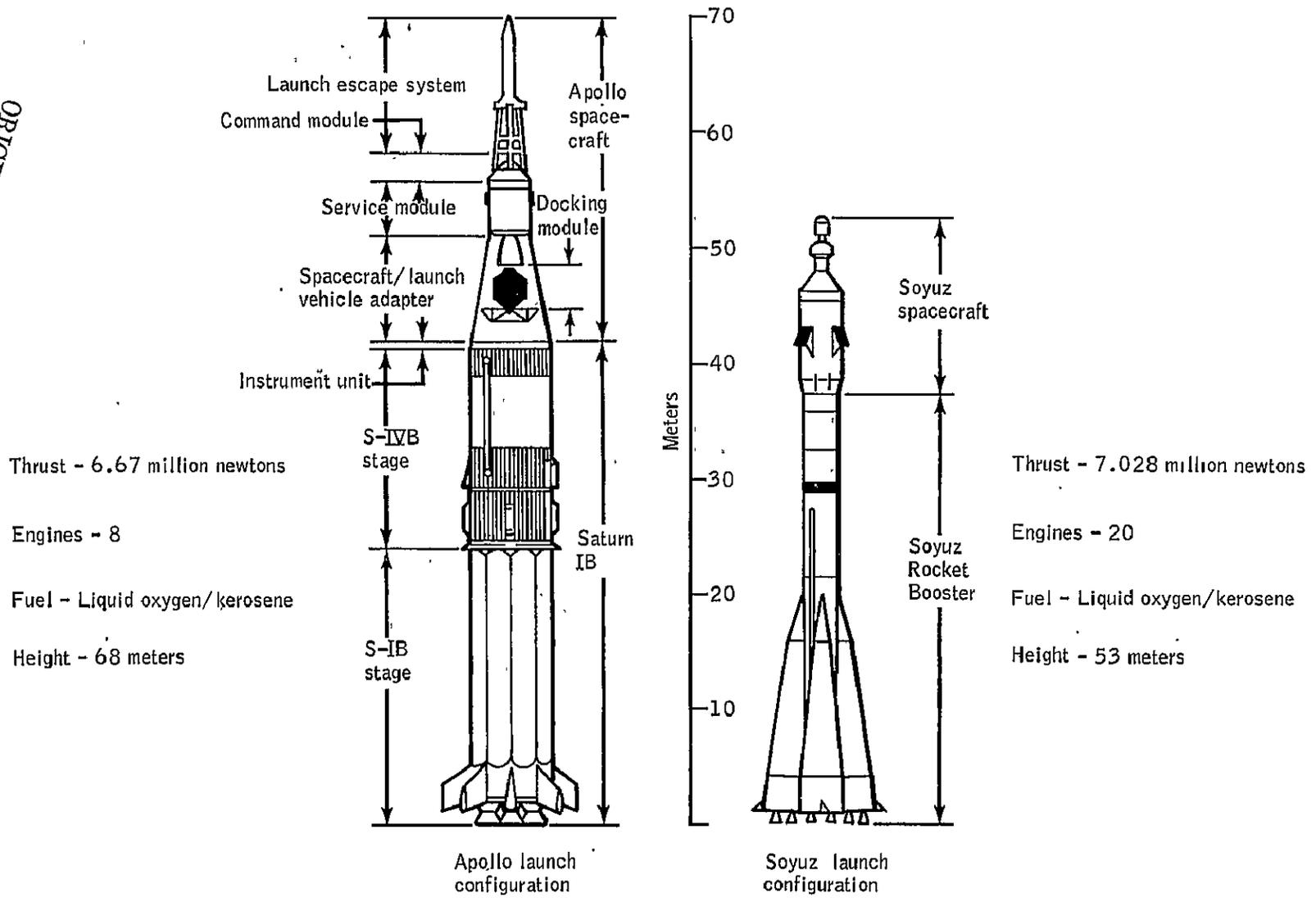


Figure A-1.- Apollo and Soyuz space vehicles.

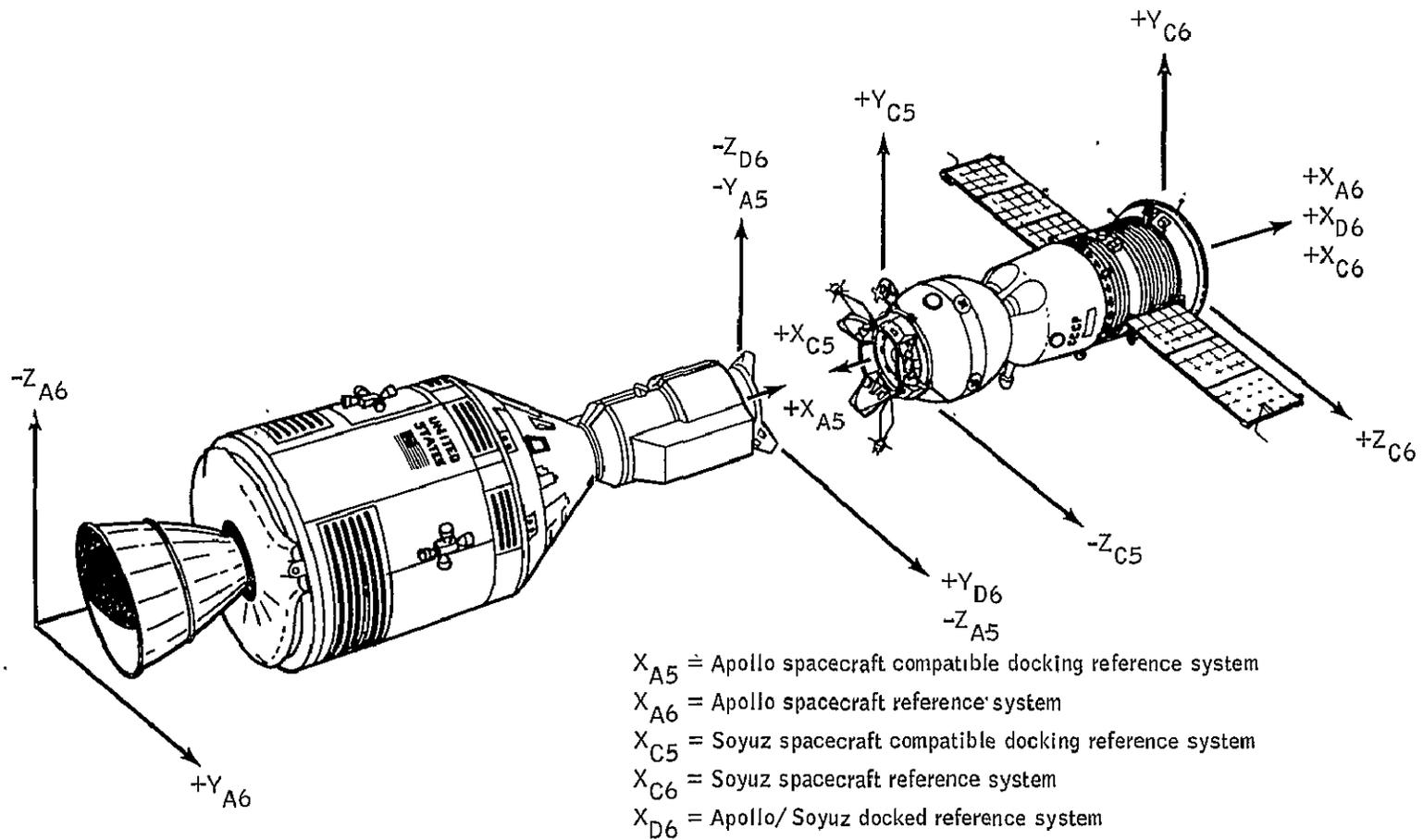


Figure A-2.- Apollo/Soyuz coordinate systems.

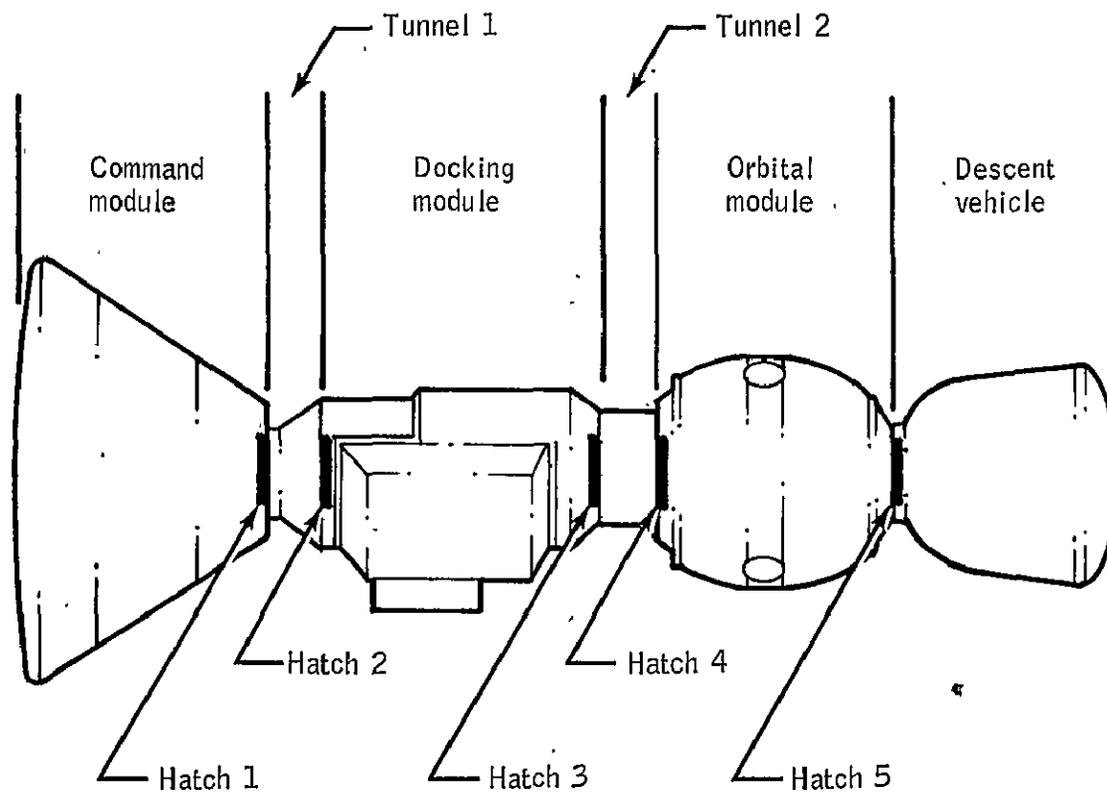


Figure A-3.- Command module/docking module/Soyuz hatch and tunnel designations.

The main external attachments were four gas storage tanks, three VHF antennas, a doppler transmitter antenna for the doppler tracking experiment (MA-089), an ultraviolet spectrometer for the ultraviolet absorption experiment (MA-059), three adapter mountings, a vent housing, and a docking target that allowed the Soyuz crew to observe the docking approach through the Soyuz periscope. The tanks were mounted in pairs. The two tanks closer to the Apollo docking end contained oxygen, while the other two tanks contained nitrogen. The external and internal arrangements of the docking module are shown in figures A-4 and A-5.

A.1.2 Thermal Control System

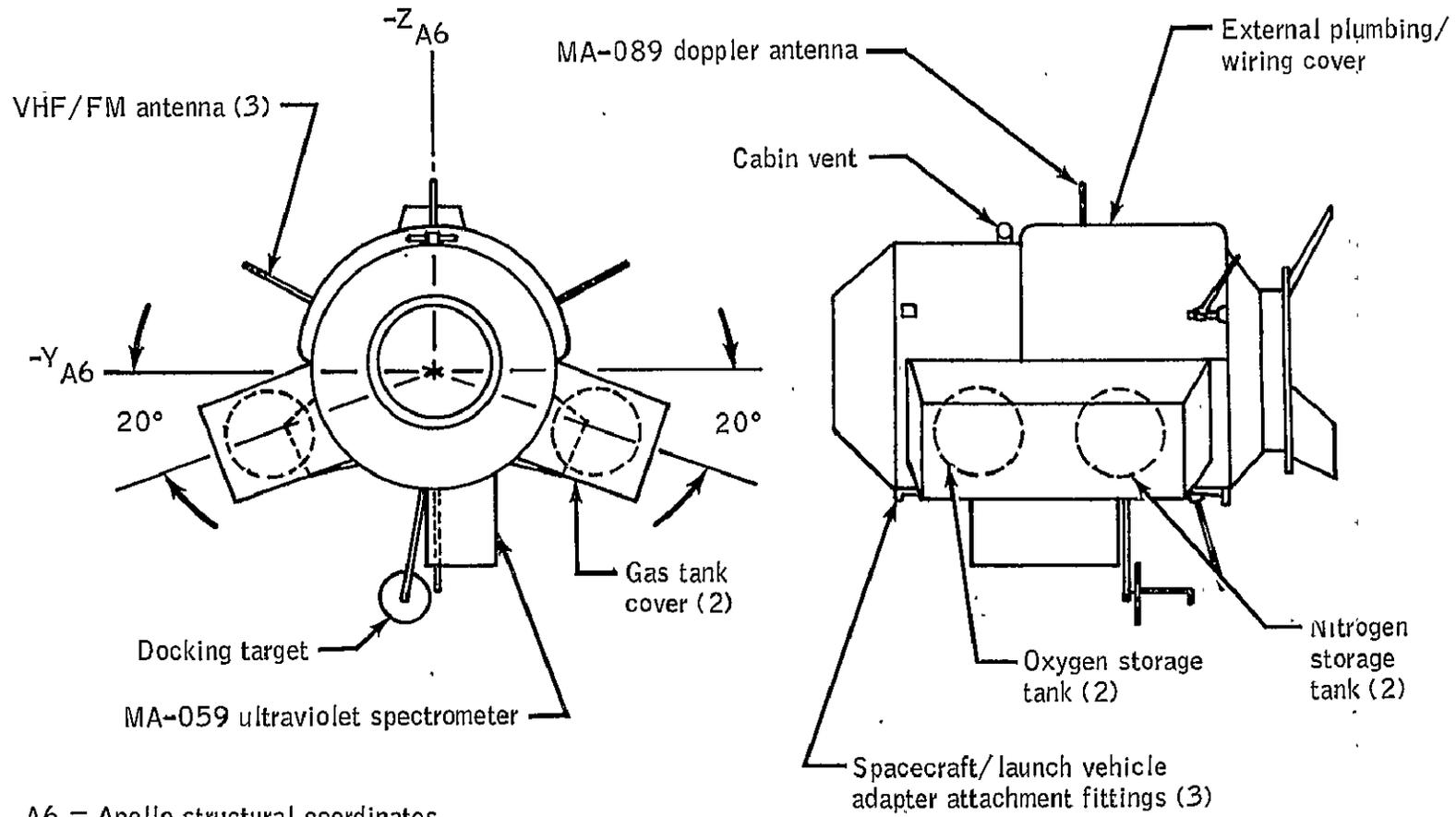
The thermal control system utilized external insulation and thermal coatings, internal equipment, crew metabolism, and electrical heaters to maintain the temperatures within acceptable limits. The overall design was cold-biased with three 45-watt heaters to be used as required.

Most of the external surfaces were insulated with 30 layers of aluminized Kapton covered with inconel foil and nickel foil to protect the Kapton from service module reaction control system plume impingement. The equipment area (between the gas storage tanks) was protected by fiberglass covered by inconel foil. In addition, the internal wall of tunnel 2 (fig. A-3) was insulated by two layers of beta felt enclosed in two layers of beta cloth.

A.1.3 Electrical Power Distribution System

All electrical power for the docking module was supplied by the command and service module. Four umbilical connectors, which were mated following docking of the command and service module with the docking module, conveyed power to the docking module; provided for control of the docking system; and carried experiment, telecommunications, and data signals. (See sec. A.1.5).

Three umbilical connectors provided for interfacing the docking module with the Soyuz. One of these carried command and service module 28-Vdc power and signals for television and voice communications in the Soyuz; the other two brought Soyuz power and signals for television and communications to a junction box in the docking module. The Russian crewmen plugged their equipment into the junction box using a drag-through cable long enough to enable them to enter and work in the command module when desired. The connector which carried command and service module power into the Soyuz was backed up by a drag-through cable for contingency operations.



A6 = Apollo structural coordinates

Figure A-4.- Docking module exterior.

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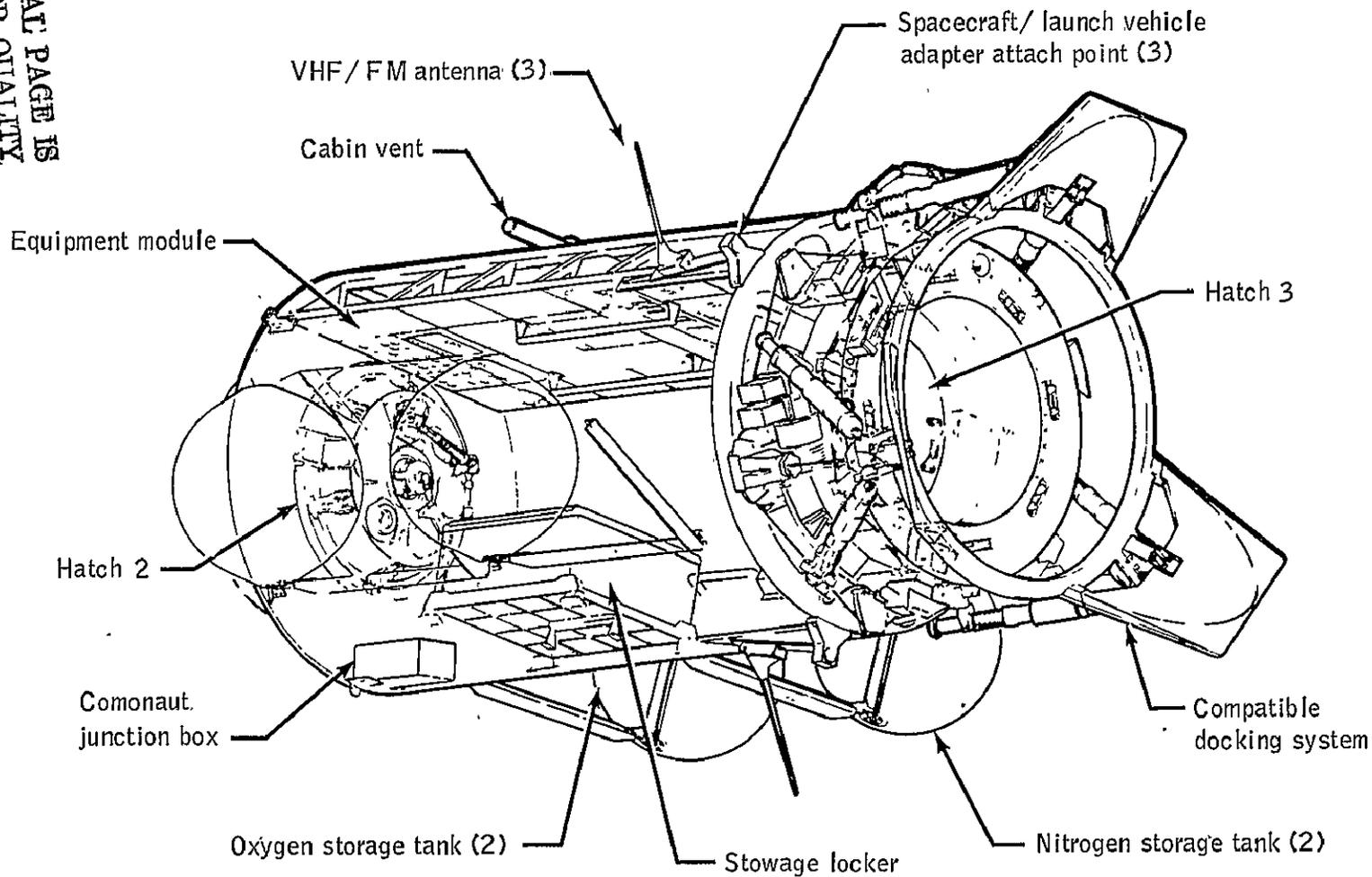


Figure A-5.- Docking module inboard profile.

A.1.4 Environmental Control System

The general capabilities of the docking module environmental control system were as follows.

- a. Pressurization gas (nitrogen) and manual controls for pressurization of the docking module for integrity checks, transfers, and leakage makeup for the docking module and Soyuz combined volume.
- b. Enrichment gas (oxygen) and manual controls for enrichment of the docking module and leakage makeup for the docking module and Soyuz combined volume.
- c. Pressure vent valves for overpressurization and planned cabin pressure dump operations.
- d. Emergency 100-percent oxygen breathing facilities for two men.
- e. A ventilation system for atmosphere circulation.
- f. Emergency pressurization (oxygen) of the docking module in the event of an unexpected pressure loss.
- g. Provisions for pressurizing and venting tunnel 2 (fig. A-3) for pressure integrity checks and hatch integrity checks.
- h. Provisions for equalization of the docking module pressure with that of the Apollo and Soyuz spacecraft (or the tunnels between the docking module hatches and the spacecraft hatches).
- i. Provisions for measurement of atmospheric conditions and out of tolerance conditions.

Docking module cabin pressure was controlled by the Apollo pressure regulation system when the hatches between the docking module and the command module were open, and by the Soyuz pressure regulation system when the hatches between the docking module and Soyuz were open. There was no automatic pressure regulation of docking module cabin pressure when both of its hatches were closed. A simplified schematic of the environmental control system is presented in figure A-6.

The launch composition of the docking module atmosphere was 60 percent oxygen and 40 percent nitrogen.

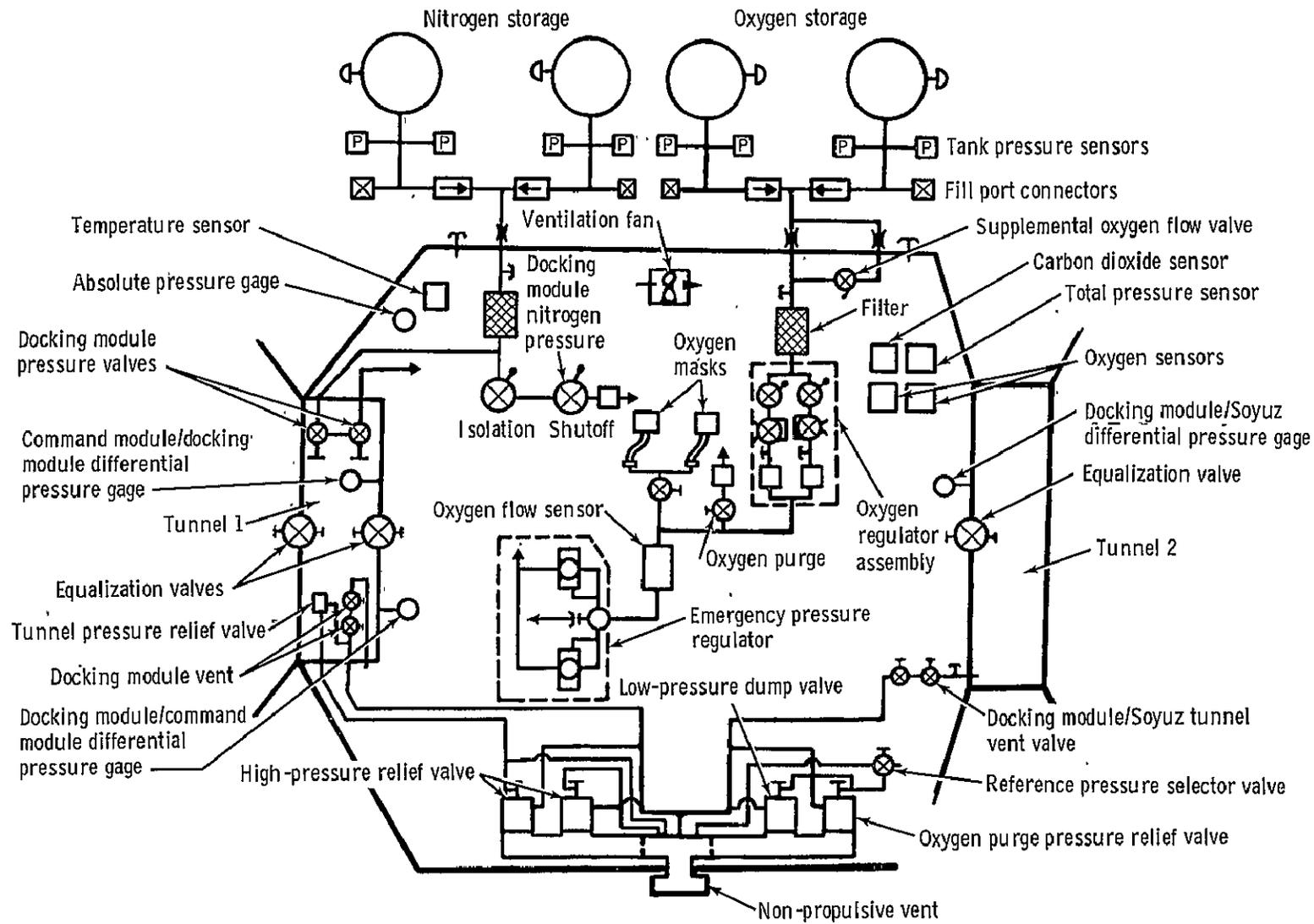


Figure A-6. - Simplified schematic of the docking module environmental control-life support system.

A.1.5 Telecommunications System

The docking module telecommunications system consisted of:

- a. Redundant VHF/FM simplex transceivers operating at the Soyuz spacecraft frequency of 121.75 megahertz.
- b. A three-element antenna array (fig. A-4).
- c. A Skylab-type speaker-box for voice communications with the command and service module and Soyuz.
- d. Provisions to allow use of a television camera in the docking module, with video signals transmitted to the ground by the command and service module S-band transmitter.
- e. Sensors to provide data on docking module systems status for display in the docking module and command module and for transmission via the command and service module telemetry system.

The umbilical connectors between the command module and docking module and between the command module and Soyuz provided hardline audio communications in conjunction with the command and service module audio centers and three Skylab-type speaker boxes. Besides the speaker box mounted in the docking module, a speaker box was mounted in the command module, and another was launched in the docking module and transferred to the Soyuz spacecraft during the first crew transfer. The crewman communications umbilicals were connected to the Soyuz or docking module hardline receptacles in place of the speaker boxes for headset communications. Switching operations at the command module audio center provided intercommunication between all three modules and/or transmission/reception capability via the VHF/FM, VHF/AM, or S-band equipment. The VHF/FM and VHF/AM transceivers provided redundant communication links with the Soyuz spacecraft during docked operations. A simplified schematic of the system is given in figure A-7.

A.1.6 Displays and Controls

With the exception of the speaker box and the cosmonaut junction box, all display and control panels in the docking module were located in the equipment rack. Display and control notations and instructions were lettered in both English and Russian on all panels except the docking system test panel.

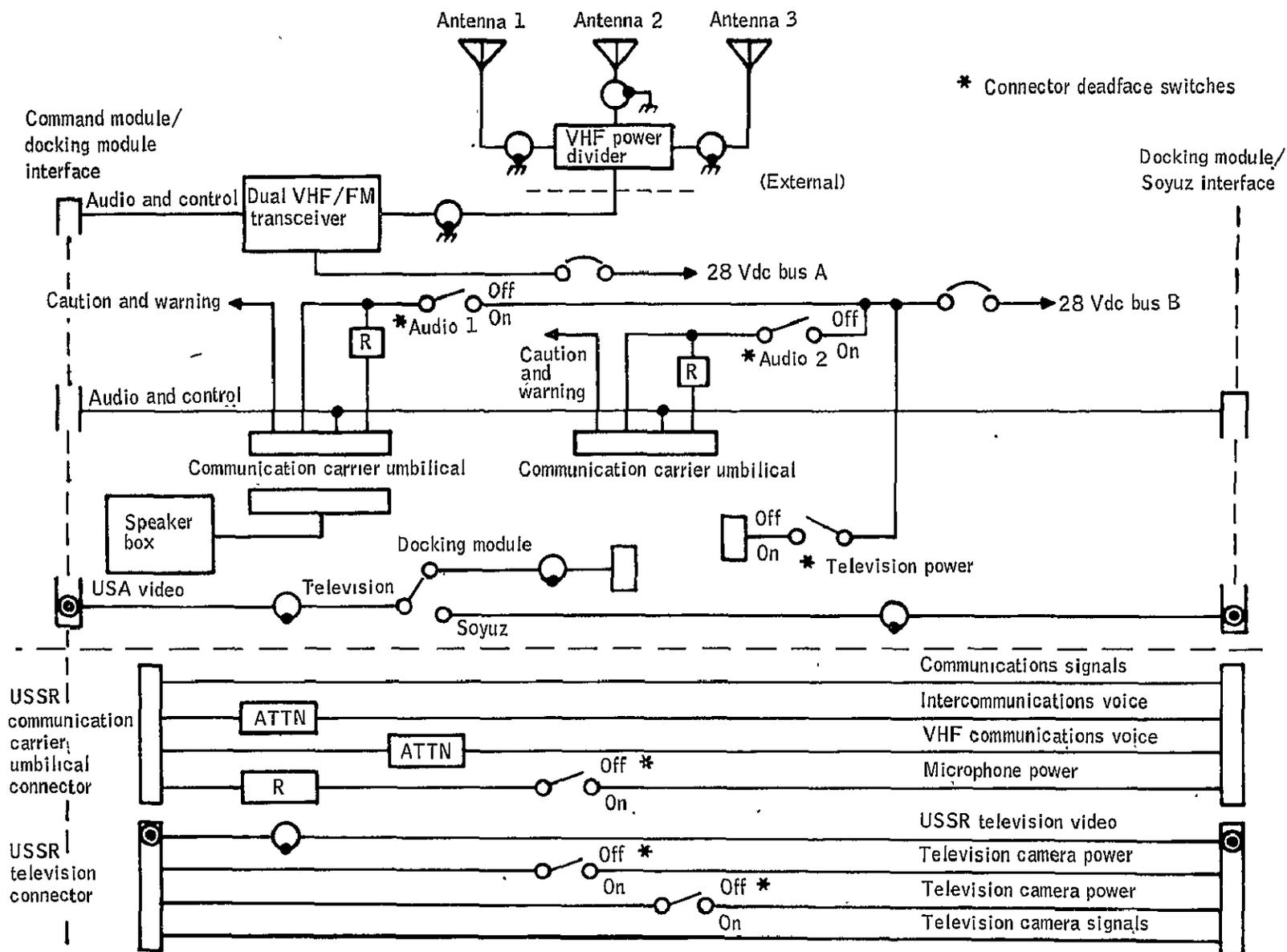


Figure A-7.- Simplified schematic of the docking module telecommunications system.

The docking module caution and warning system consisted of a master alarm and three warning lights, namely, oxygen flow high, oxygen partial pressure A, and oxygen partial pressure B. The master alarm was activated if one of the three monitored parameters went out of tolerance range. The alarm system was similar to that in the command and service module, a red light accompanied by an audible tone. The audible tone was confined to the docking module speaker box and the communications carrier umbilicals.

A.1.7 Docking System

The docking system was designed to function as either an active or passive mechanism for docking and undocking with an identical system or the compatible USSR docking system (fig. A-8). During docking operations, one system was active and the other was passive. The active system required no assistance from the passive system.

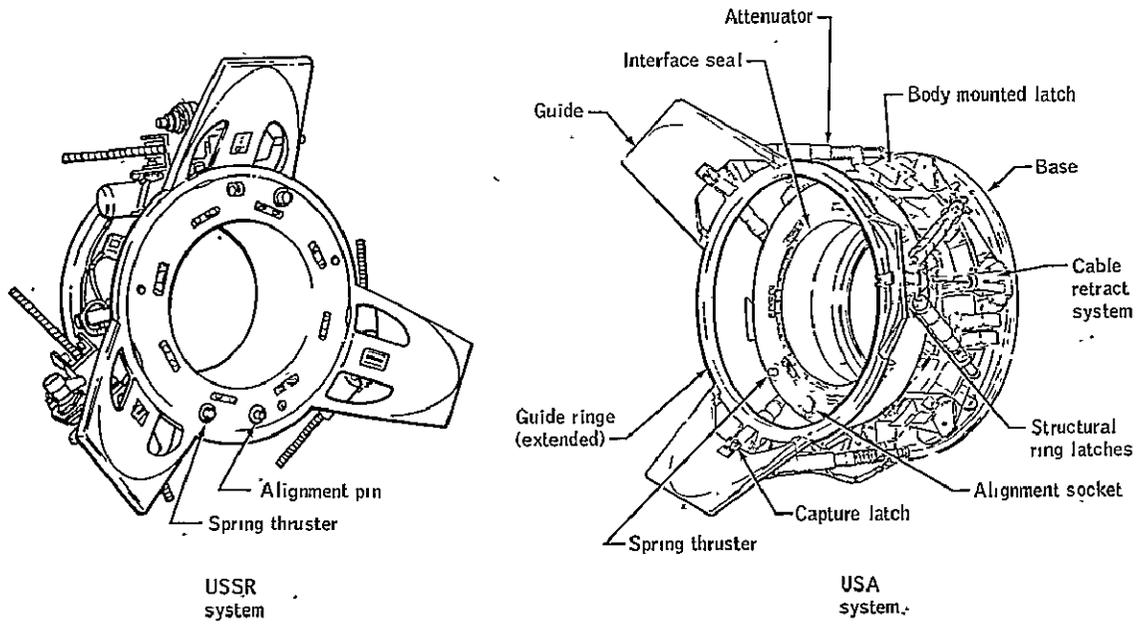
Each system had three guides spaced equally apart around the extendible guide ring (fig. A-8). The guide ring of the active system was extended and the guide ring of the passive system was retracted. Three capture latch sets on the active system engaged with three body-mounted latches on the passive system, but the capture latches of the passive system did not engage with the body-mounted latches of the active system. Impact energy was dissipated on the USA system when it was active by six hydraulic attenuators.

Following capture latch engagement, the guide ring was retracted by a drum and cable system. Retraction was followed by crew-initiated engagement of eight structural latches on the active system with a similar set of eight latches on the passive system. Each latch consisted of an active hook and a passive hook. The eight active hooks of the active system engaged with the eight passive hooks of the passive system. In the passive mode, the active hooks of the passive system were rotated in a manner which precluded their engagement with the passive latches of the active system.

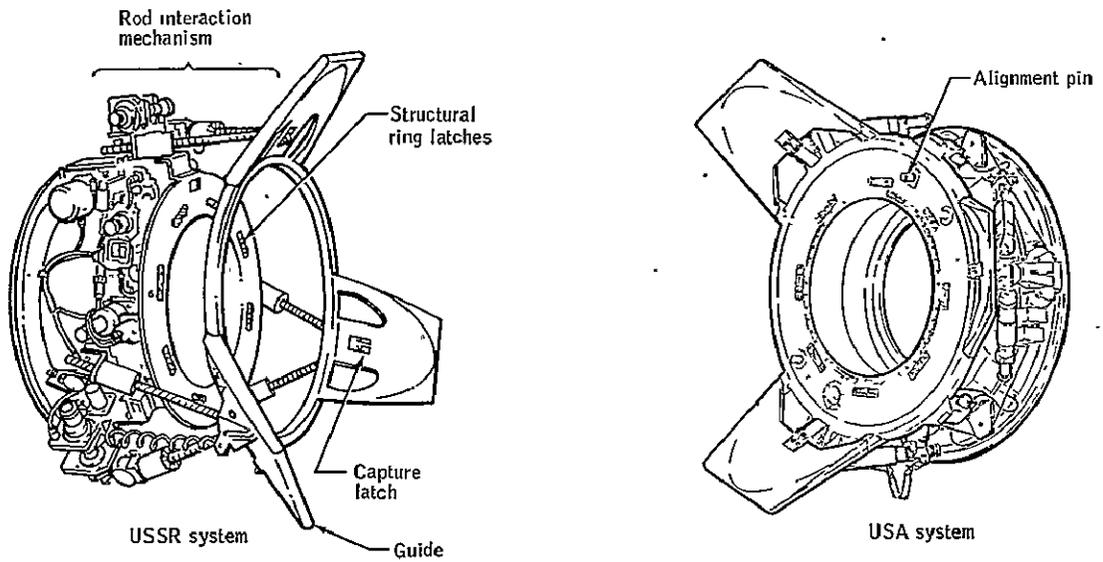
The two docking systems were initially aligned to the correct orientation for capture and structural latch mating by the interaction of the guides. Final alignment of the structural rings was performed by engagement of a pin and socket on each structural ring.

Two spring thrusters were mounted on each structural ring interface surface to provide initial separation velocity at undocking.

The passive system had the capability to effect an emergency separation from the active system.



(a) USA system active



(b) USSR system active.

Figure A-8.- USA and USSR docking systems.

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A.2 SOYUZ SPACECRAFT

A.2.1 Structure and General Description

The Soyuz spacecraft consisted of an orbital module, a descent vehicle, and an instrument-assembly module (fig. A-9).

A.2.1.1 Orbital module.— The orbital module was the crew station for non-dynamic orbital operations. The forward end of the module incorporated the docking mechanism as well as a tunnel for crew transfer into the docking module (fig. A-10). The tunnel was entered by manually opening a hatch that was attached to the inboard end of the docking assembly body. The module also contained a side hatch for crew entry prior to launch and an aft tunnel for crew transfer to the descent vehicle. Two windows were provided - one forward of the side hatch for earthward viewing, and the other on the opposite side of the module for outward viewing. Interior details of the orbital module are shown in figure A-11.

A.2.1.2 Descent vehicle.— The descent vehicle (fig. A-12) was occupied during insertion into orbit, dynamic orbital operations, and for entry and landing. The interior was functionally divided into two areas - a work area and an instrument area. The work area contained the pilots' couches, controls and displays, and equipment bays. The instrument area contained life support and vehicle attitude control systems.

A.2.1.3 Instrument-assembly module.— The instrument-assembly module (fig. A-13) consisted of intermediate, instrument, and assembly sections. The intermediate section interfaced with the descent vehicle and contained fuel tanks for the approach and orientation engine system (par. A.2.2.6) as well as the majority of the approach and orientation engines. In addition, the smaller of two thermal control system radiators was mounted on the exterior of the intermediate section. The sealed instrument section housed thermal control system heat exchangers and other equipment. The assembly section contained the main propulsion system (par. A.2.2.7), additional approach and orientation engines, and storage batteries. The larger of the two thermal control system radiators was mounted on this section. The aft end of the assembly section had a base ring that mated the spacecraft to the launch vehicle.

Two sets of wing-like solar battery panels were mounted on opposite sides of the instrument-assembly module. Figure A-9 illustrates the panels and other external equipment.

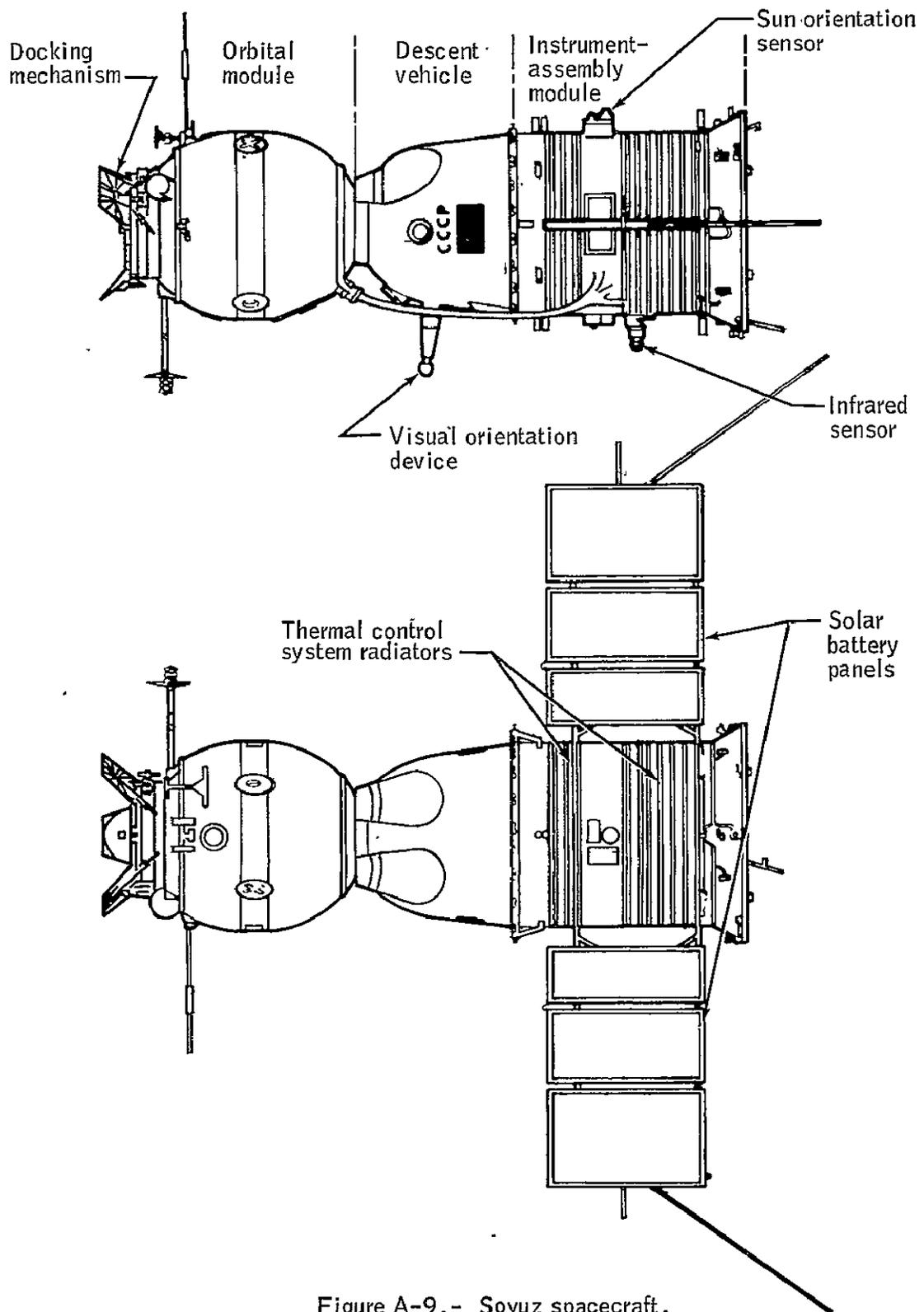


Figure A-9.- Soyuz spacecraft.

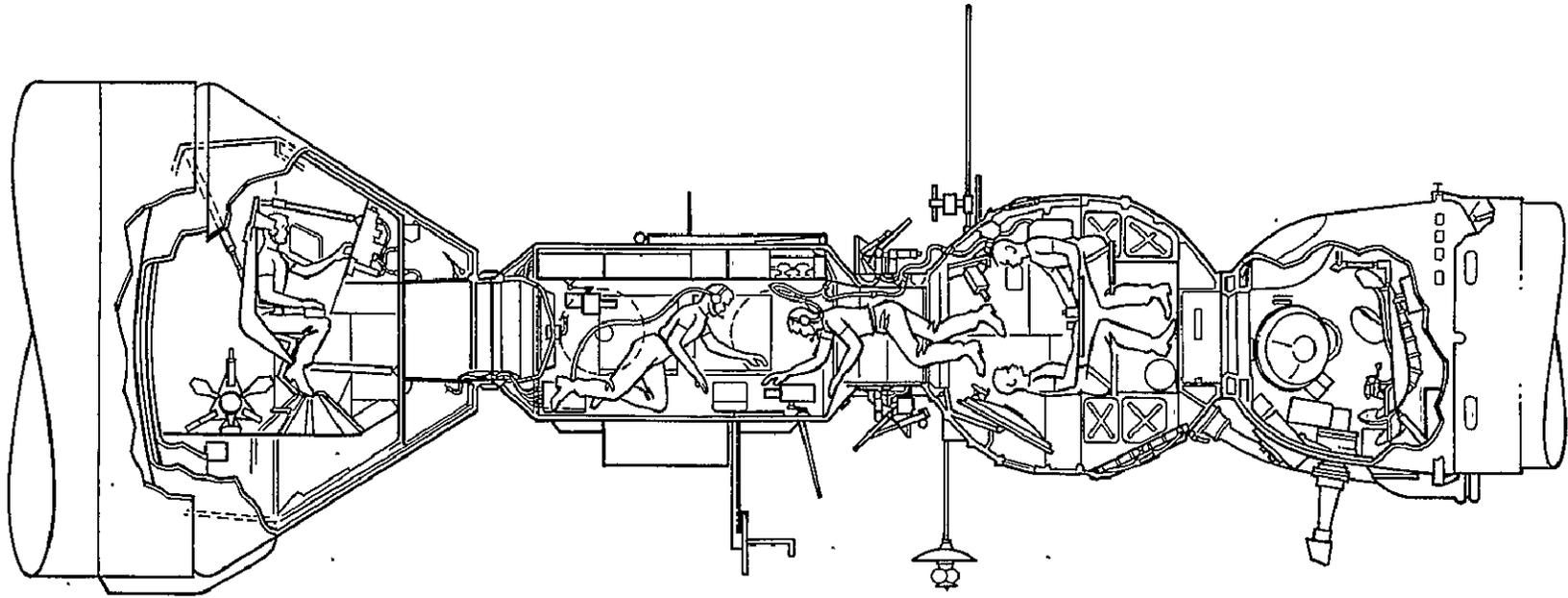


Figure A-10. - Crew transfer.

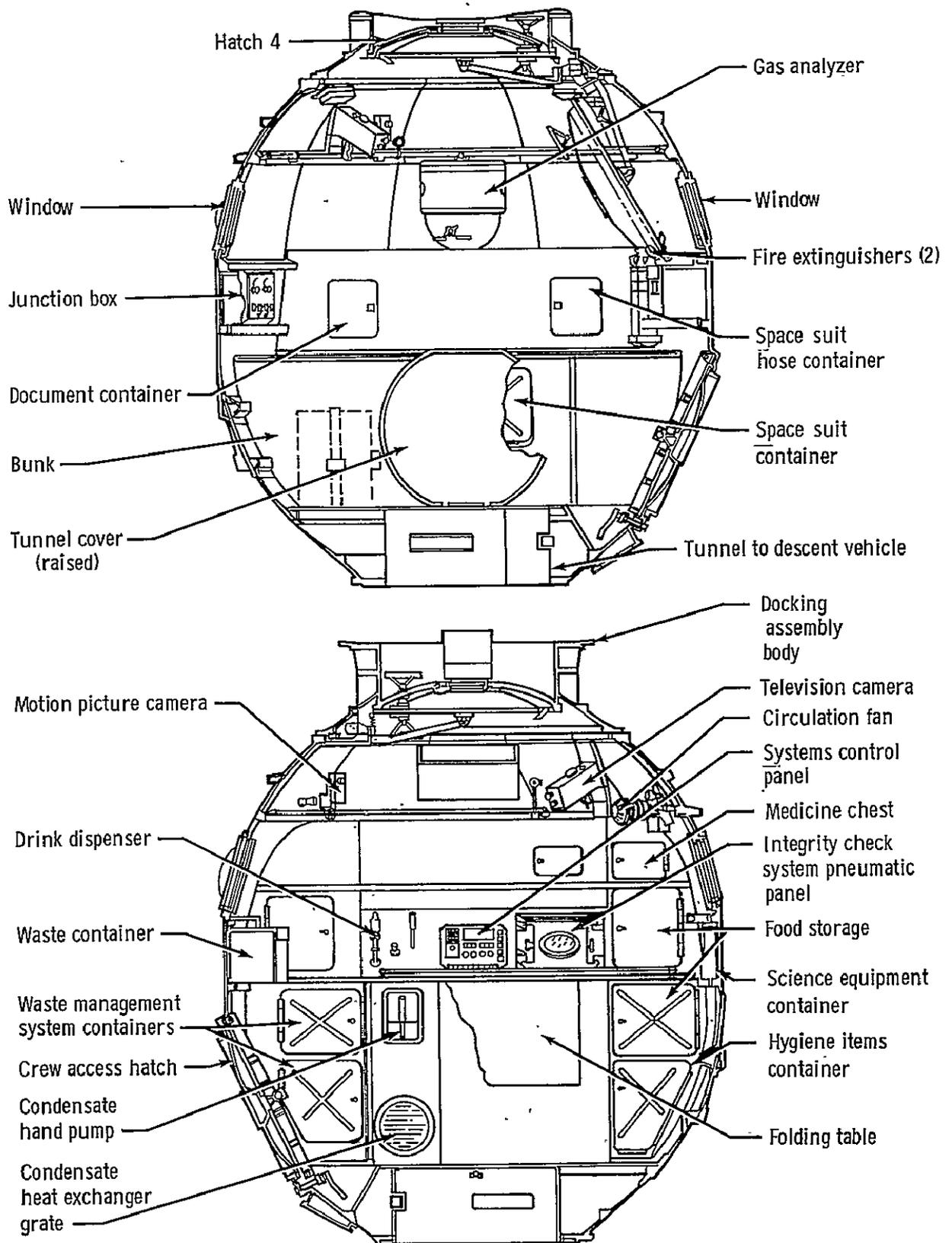


Figure A-11. - Orbital module interior.

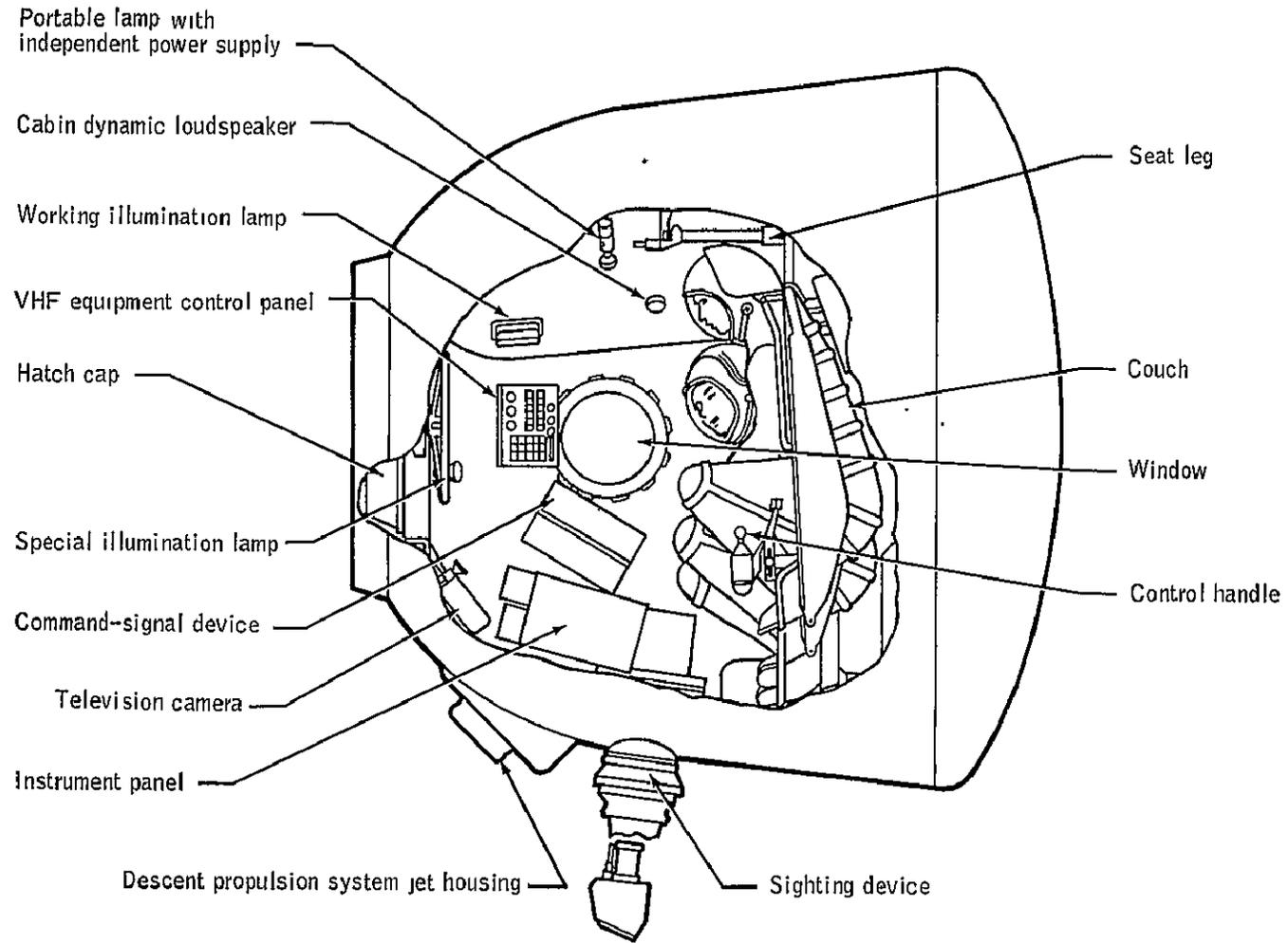


Figure A-12.- Descent vehicle.

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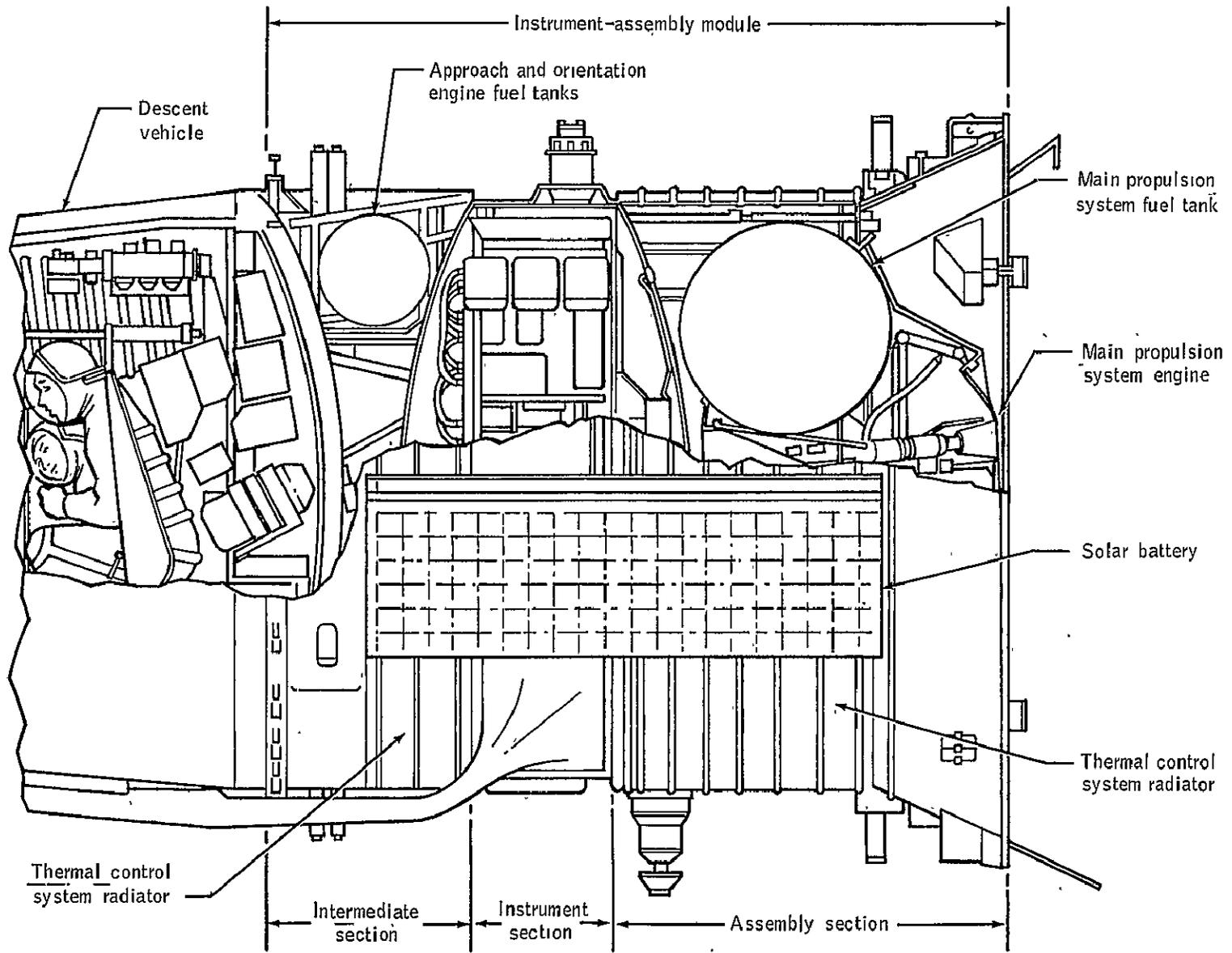


Figure A-13.- Instrument-assembly module.

A.2.2 Soyuz Systems Description

A.2.2.1 Docking system.- The Soyuz docking system was similar in design to the one on the docking module (sec. A.1.7). Compatibility between the Soyuz system and the docking module system was achieved by standardizing elements which connected or interacted. The major differences between the two systems are illustrated in figure A-8.

A.2.2.2 Electrical power system.- The Soyuz electrical power system provided d-c power for the spacecraft. The main system consisted of solar battery panels (fig. A-9) and storage batteries which were used in a buffer mode. The onboard voltage was 23 to 34 volts. A backup storage battery was available for short-time operation of the spacecraft in case of main system failure.

A.2.2.3 Thermal control system.- The Soyuz thermal control system employed both active and passive thermal control methods. The active portion of the system, shown schematically in figure A-14, consisted of two loops. One loop maintained the required temperature and humidity of the descent vehicle and orbital module. The other loop controlled the temperature of the instrument-assembly module. Both loops were thermally interconnected by a liquid-to-liquid heat exchanger. Passive thermal control was obtained by using thermal control coatings, multilayer insulation, and thermal bridges to reduce unregulated heat exchange between the spacecraft and space to a minimum.

Conduction and radiation were the primary methods of heat exchange within the unpressurized compartments. Forced air convection was the primary method of heat exchange within the pressurized modules. In most cases, equipment cooling was obtained by transfer of heat from the equipment to the atmosphere, and subsequently to the liquid of the heat exchange units through forced air convection. In specific cases, equipment was cooled and heated by passing heat-carrier liquid through channels on the structure. The air temperature was automatically controlled within $\pm 3^\circ$ of selected temperature settings of 288° , 293° , and 298° K. The liquid temperature was controlled within $\pm 2^\circ$ of selected temperature settings of 276° , 278° , 280° , and 282° K.

A.2.2.4 Atmospheric composition control system.- The atmospheric composition control system, schematically shown in figure A-15, performed the functions of (1) analysis and regeneration of the orbital module and descent vehicle oxygen-nitrogen atmospheres; (2) pressure equalization and relief; (3) verification of pressure integrity; (4) module repressurization; (5) supply of breathing gas to the pressure suits; and (6) supply of pressurant gas to the parachute expansion bladder.

Gas analysis: Gas analyzers in the orbital module and descent vehicle measured oxygen, carbon dioxide, and absolute humidity levels.

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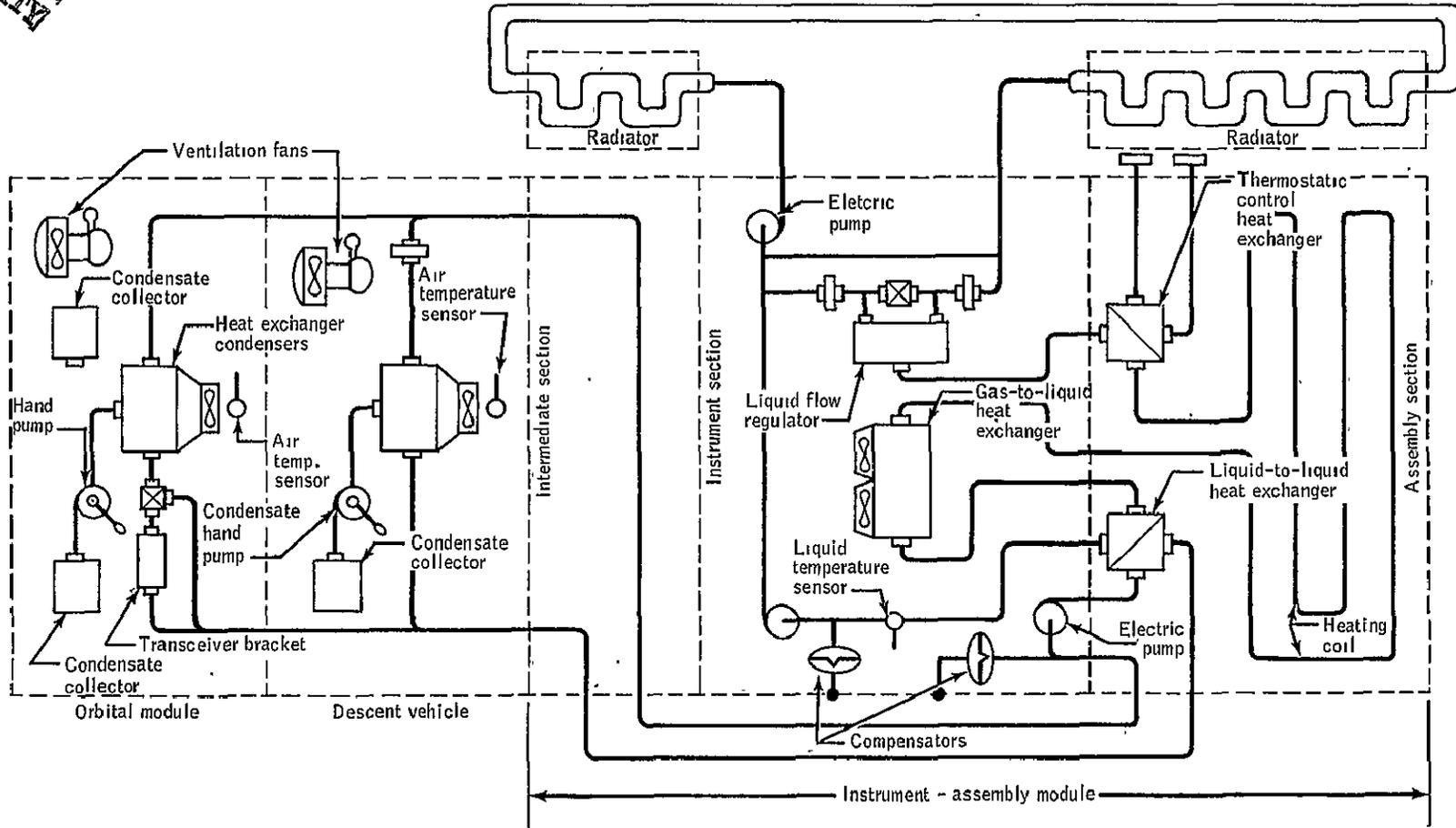


Figure A-14.- Thermal control system schematic.

Atmospheric regeneration: The atmosphere regeneration system consisted of a regeneration assembly and a carbon dioxide absorber in both of the habitable modules. The system was designed to provide the capability of maintaining the oxygen partial pressure between 2.0 and 2.8 newtons/sq cm at a total pressure of 6.5 to 7.3 newtons/sq cm, and between 2.1 and 3.6 newtons/sq cm at a total pressure of 10.1 to 11.9 newtons/sq cm total pressure (but not more than 40 percent oxygen by volume). The regeneration assemblies contained potassium superoxide, which absorbs carbon dioxide while releasing oxygen, and an activated charcoal filter for removal of other contaminants. The carbon dioxide absorbers contained lithium hydroxide granules. The cabin atmosphere was circulated through the regeneration assemblies and carbon dioxide absorbers by means of fans.

Pressure equalization and relief: The pressure equalization and relief equipment consisted of three valve assemblies: the tunnel 2 pressure vent valve, the orbital module pressure vent valve, and the descent vehicle/orbital module pressure equalization valve. The tunnel 2 pressure vent valve, located on the docking assembly body, was provided to vent the tunnel leading to the docking module, if required, prior to undocking. However, the tunnel venting was normally performed by docking module valves. The orbital module pressure vent valve, located in the orbital module cabin, was used to vent the module when checking the hatch 5 (descent vehicle hatch) seal before separating the descent vehicle from the orbital module. The descent vehicle/orbital module pressure equalization valve, located on hatch 5, was used to equalize the pressure in the descent vehicle and orbital module prior to opening hatch 5.

Pressure integrity check system: The pressure integrity check system provided the capability to monitor the pressure integrity of the modules, tunnel 2, the docking interface, and the hatches. The system also provided the capability of venting the orbital module pressure before docking, pressurizing the interface volume between docking seals, and equalizing orbital module/tunnel 2 pressure. The system consisted of a pressure-and-vacuum gage, four electropneumatic valves, two pressure sensors, plumbing, and a module pressure integrity check unit (fig. A-15). The valves and gage were mounted on a control panel in the orbital module. The pressure integrity check unit, also located in the orbital module, measured the rate of pressure loss in the modules and, if excessive, provided an emergency signal.

Modules pressurization system: The basic component of the modules pressurization system was an external spherical steel tank containing 4.5 kilograms of air (fig. A-15). The system was used for repressurizing the orbital module and descent vehicle after undocking, and could also have been used for emergency pressurization in case of an accidental pressure loss.

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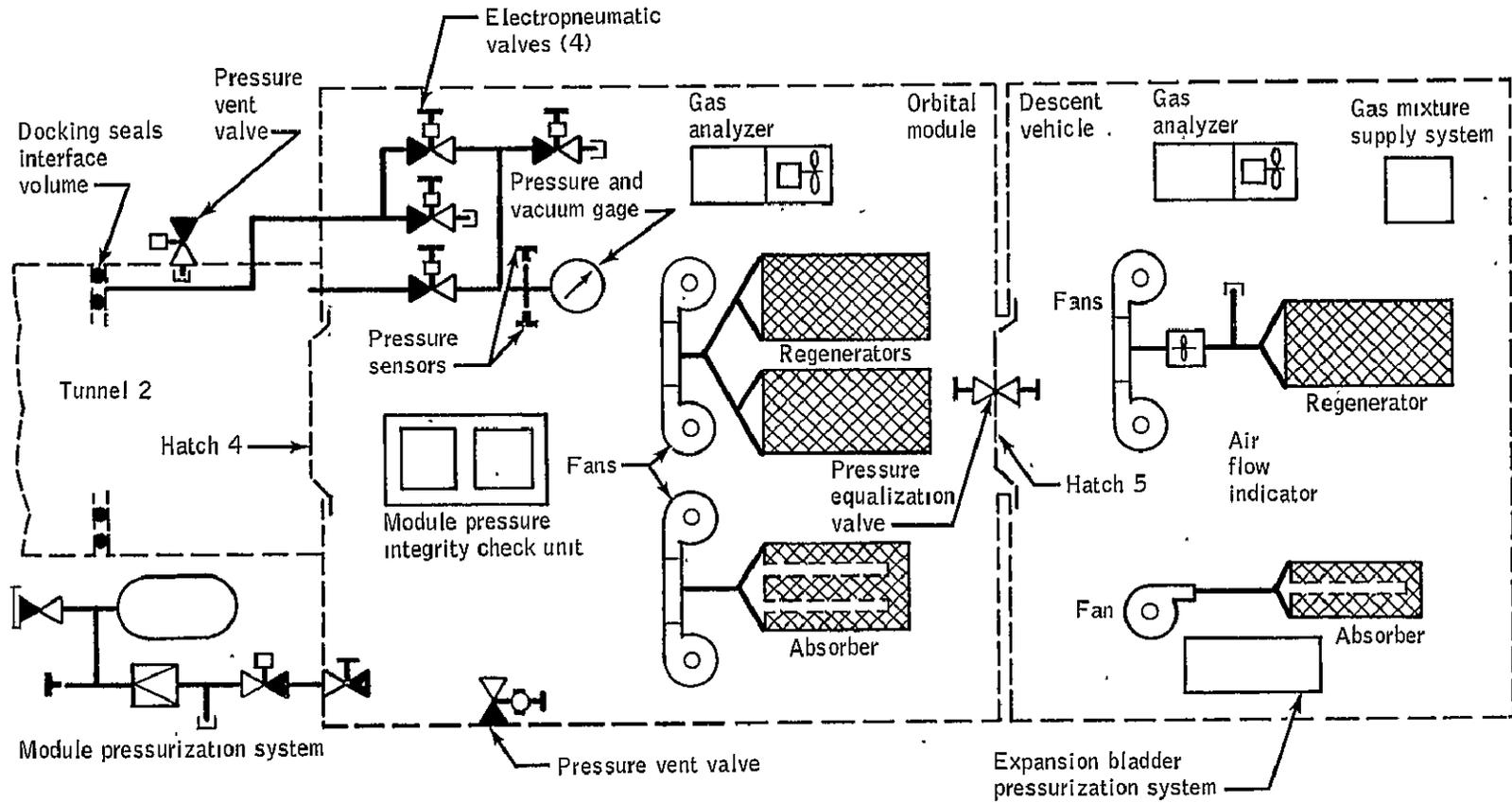


Figure A-15.- Atmospheric composition control system schematic.

Gas mixture supply system: The gas mixture supply system, located in the descent vehicle, supplied breathing gas to the pressure suits. The composition of the gas was approximately 40 percent oxygen and 60 percent nitrogen. In addition to the gas supply, the system included a contingency cabin pressure relief unit, descent and landing pressure equalization valves, vents, and fans.

Expansion bladder pressurization system: This system was used to pressurize the expansion bladder of the parachute during the landing sequence. During descent, an explosive valve was actuated in response to an automatic sequencer command to permit air flow from a tank to the parachute expansion bladder.

A.2.2.5 Food, water and waste management systems.-

Food: The Soyuz food system consisted of 15 daily food rations packaged in cans, tubes, and plastic containers; eating utensils; and an electric heater for heating tubes. The nutritional value of a crewman's daily ration was about 3000 kilocalories and the ration was consumed in four meals. Four crewmen could be accommodated for a meal around a folding table.

Water: The water supply system was located in the orbital module and included a 0.03-cubic-meter water storage tank, a hand pump, a dispenser assembly, and a safety unit. The hand pump was used to pressurize an air chamber in the tank which was separated from the water by a diaphragm. The dispenser assembly consisted of a flexible hose with a tapered tip for connection of individual mouthpieces, and a valve assembly for dispensing water. The safety unit consisted of a valve assembly that maintained sufficient pressure for water expulsion while preventing overpressurization of the tank.

Waste management: The waste management system was located in the orbital module and provided for collection, isolation, and storage of body waste. A collection receptacle was used to collect both liquid and solid waste. Urine was transported from the collection receptacle to a liquid waste receptacle by air flow from the action of two centrifugal fans. The air passed through the liquid waste receptacle to a purifying filter, and the purified air was then vented to the cabin. An insert was placed in the collection receptacle for containment of feces. After use, the insert was closed and removed, placed in a container, and the container sealed.

A backup collection receptacle was provided for urine collection in the event that the primary system fans failed. In addition, separate rubber containers were furnished for urine collection in the descent vehicle.

A.2.2.6 Approach and orientation engine system.- The approach and orientation engine system (fig. A-16) consisted of fourteen 10-kilogram-thrust approach and orientation engines; eight 1-kilogram-thrust orientation engines; fuel tanks and lines; and engine actuating devices. The engines damped angular disturbances during separation of the spacecraft from the booster, controlled the attitude of the spacecraft, and provided thrust for translational movement during orbital flight. The descent vehicle was equipped with six additional engines along with the associated fuel supply system, fuel distribution system, and engine actuating devices for attitude control after separation of the spacecraft modules.

A.2.2.7 Main propulsion system.- The main propulsion system (fig. A-16) consisted of a single-chamber 417-kilogram-thrust main engine, a two-chamber 411-kilogram-thrust backup engine, and the associated fuel supply and engine actuation elements. The main propulsion system was used for orbit correction and braking for spacecraft deorbit. The spacecraft was stabilized by the approach and orientation engines during main engine firing. Control nozzles (fig. A-16) were provided for stabilization during backup engine operation.

A.2.2.8 Orientation and motion control system.- The orientation and motion control system furnished attitude and translational control commands to the propulsion systems. The system had both automatic and manual control loops. Devices for sensing spacecraft orientation are shown in figure A-17. An infrared sensor which detected the earth horizon was located on the underside of the instrument-assembly module. A sighting device was used by the crew for visual observation when approaching and docking, and for local vertical indication. An ionic boost velocity vector sensor was located on the forward end of the orbital module. Two ionic braking velocity vector sensors were located on the aft end of the instrument-assembly module. Finally, a sun orientation sensor was located on the upper side of the instrument-assembly module. Attitude and translation controllers were mounted on the couches in the descent vehicle, as illustrated in figure A-12. The right controller was used for attitude control and the left controller was used for translational control. The general manual control concept is shown in figure A-18.

A.2.2.9 Communications and ranging systems.-

Radio communications: The radio communications systems included a command-signal radio link, a telemetry system, and a radio telephone and telegraph system. The equipment consisted of a VHF/FM transmitter operating on a frequency of 121.75 MHz, a VHF/FM receiver operating on a frequency of 121.75 MHz, and VHF/AM transceiver/transponder operating at frequencies of 259.7 and 296.8 MHz with a range tone transfer assembly. The command-signal radio link provided for reception by the spacecraft of commands and digital data, transmission of signals verifying reception, and trajectory measurement. The telemetry system provided for

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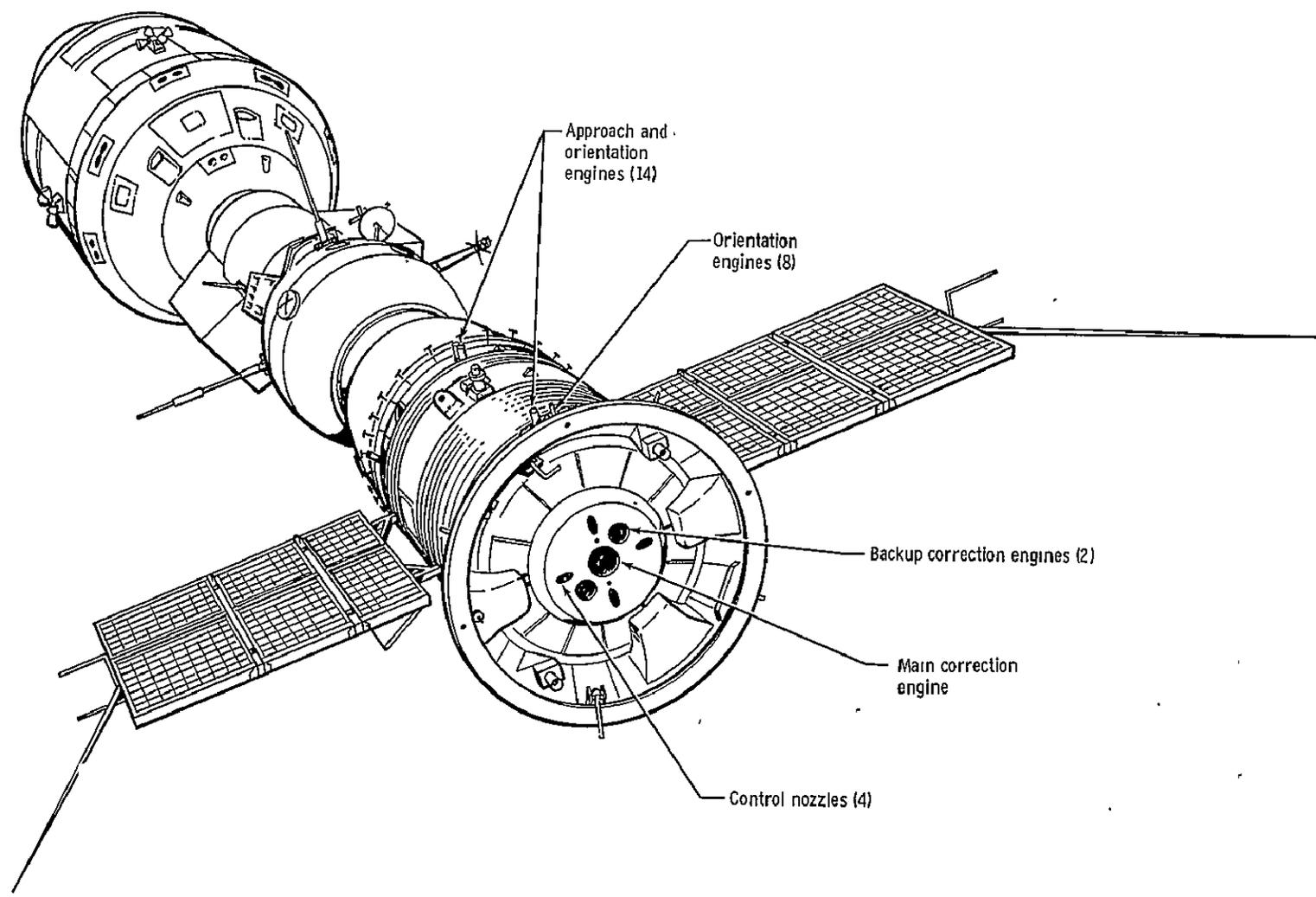


Figure A-16. - Propulsion system engines.

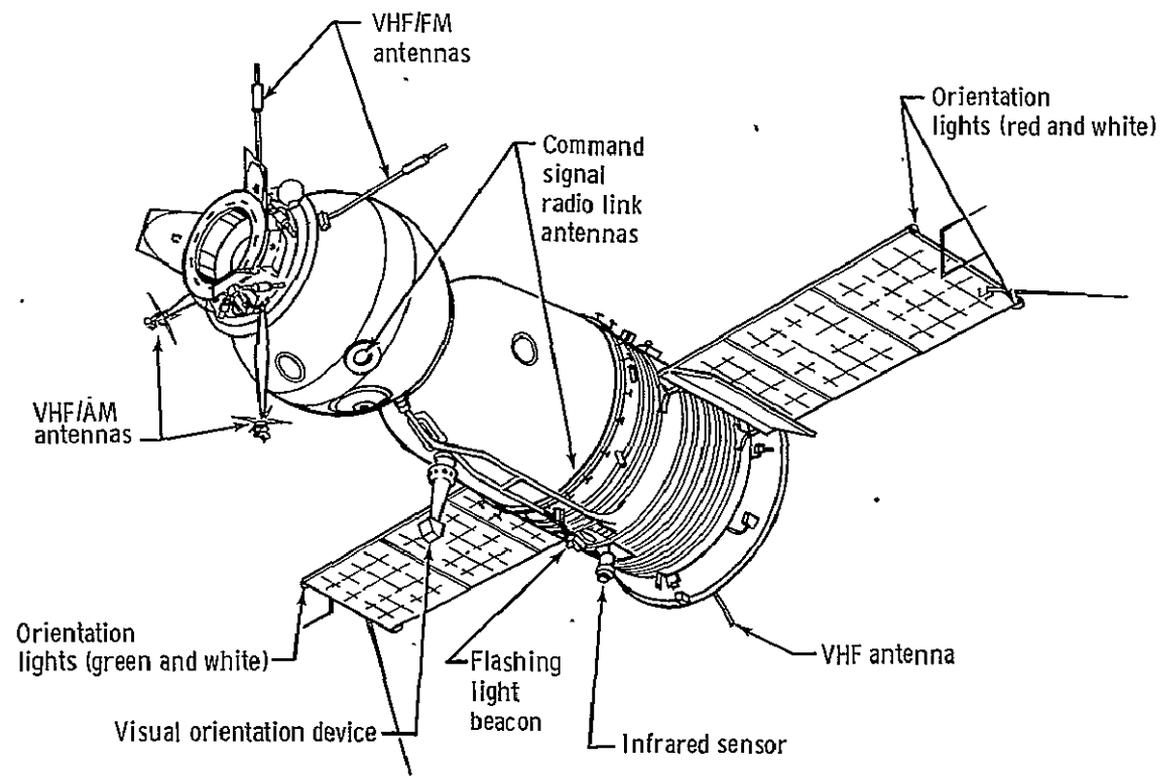
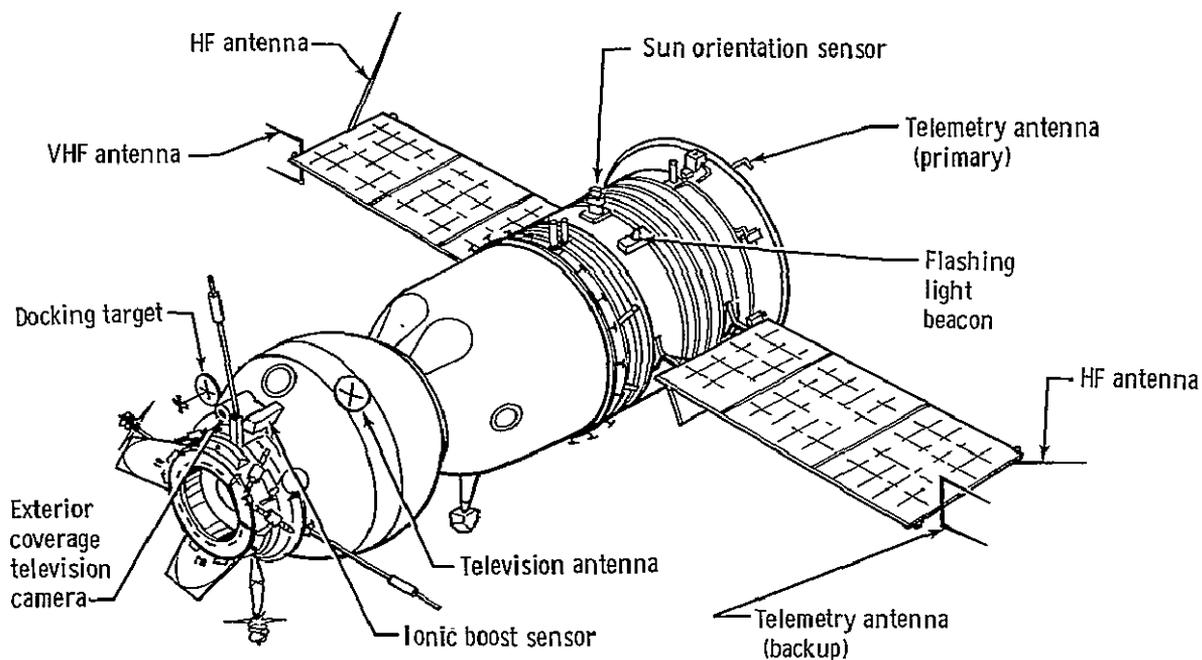


Figure A-17. - Orientation sensing devices, antennas, docking aids and external lights.

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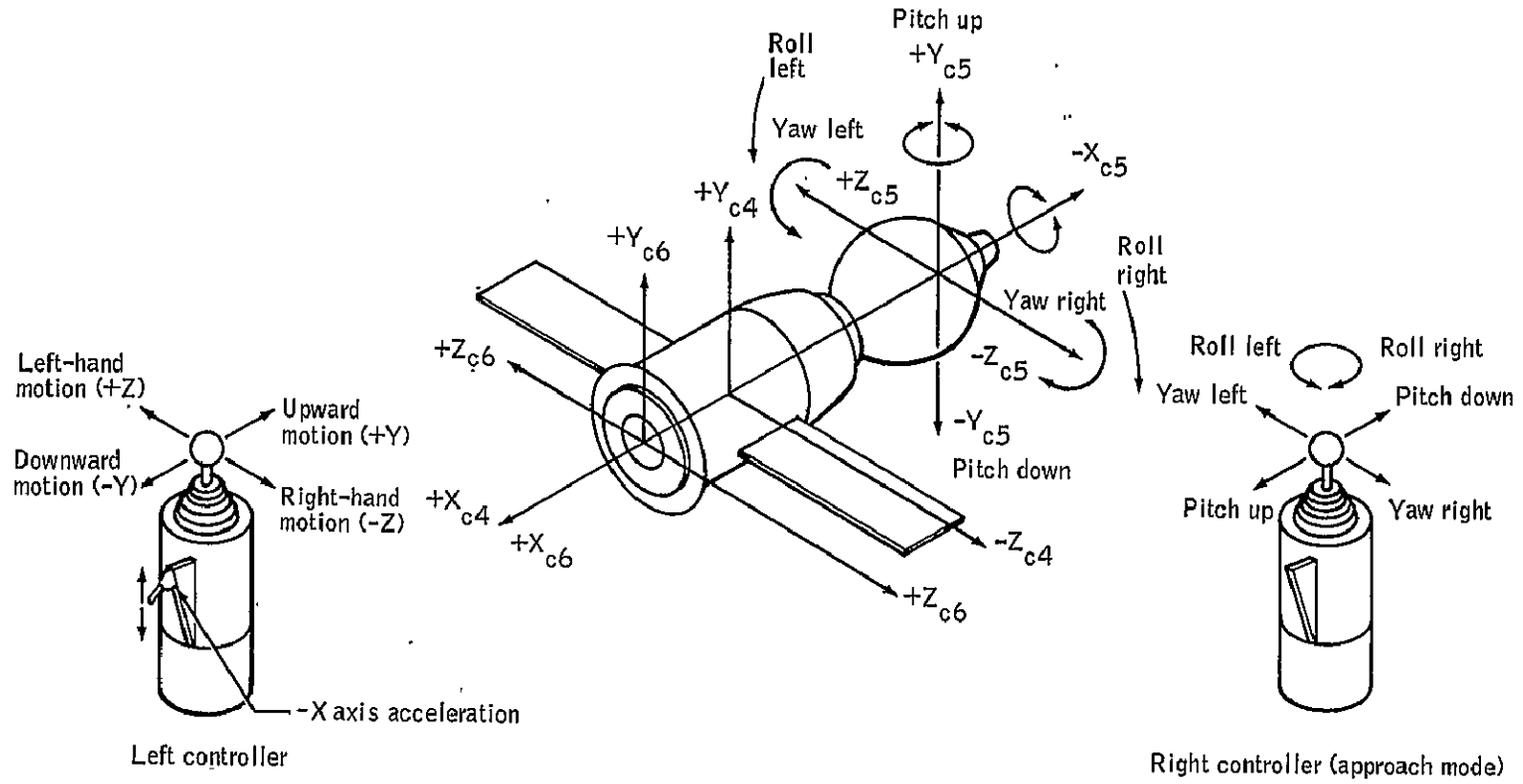


Figure A-18.- Onboard systems manual control.

transmission of spacecraft systems status data and medical data. The radio telephone and telegraph communication system provided for two-way radio communication between the crew and the Soviet ground stations, transmission of operational short-wave telemetry signals from the spacecraft, transmission of direction finding signals for descent vehicle location during parachute descent and landing, and intercommunication between the Soyuz crewmembers. Additional capabilities for joint operations were two-way radio communication between spacecraft crews, Soyuz crew communication with the United States Spaceflight Tracking and Data Network, and intercommunication with the Apollo crew after docking of the spacecraft. The various modes of radio communication are shown in figure A-19.

Ranging: The VHF transponder installed in the Soyuz spacecraft provided the capability for transmission of ranging tones when corresponding tones were received from the Apollo spacecraft. The distance between the two spacecraft was derived from the ranging signals.

Television: The Soyuz television system was designed to permit monitoring of docking, to allow observation of the crewmembers, and to provide exterior views from the vantage point of the Soyuz. During joint operations, the television system performed the additional function of televising crew activities in the docking module and in the Apollo command module. The television pictures could be displayed to the crew by means of an onboard monitoring system as well as transmitted to the ground via the radio link.

A.2.2.10 Docking aids and exterior lights.- An erectable docking target of the type used in the Apollo program was mounted on the Soyuz docking assembly as shown in figure A-17. The target was sighted through the crewman optical alignment sight mounted adjacent to a window of the Apollo command module. A backup fixed target was also provided for use in the event that the primary target failed to erect.

The Soyuz was equipped with two flashing light beacons (fig. A-17) for visual acquisition during the final phase of rendezvous. The spacecraft was also equipped with four orientation lights located on the outboard corners of the solar panels. The forward light on the left (+Z) side was red, and the forward light on the right (-Z) side was green. Both aft lights were white.

A.2.2.11 Displays and controls.- The Soyuz contained the displays and controls described in the following paragraphs for onboard-monitoring and operation of the spacecraft systems.

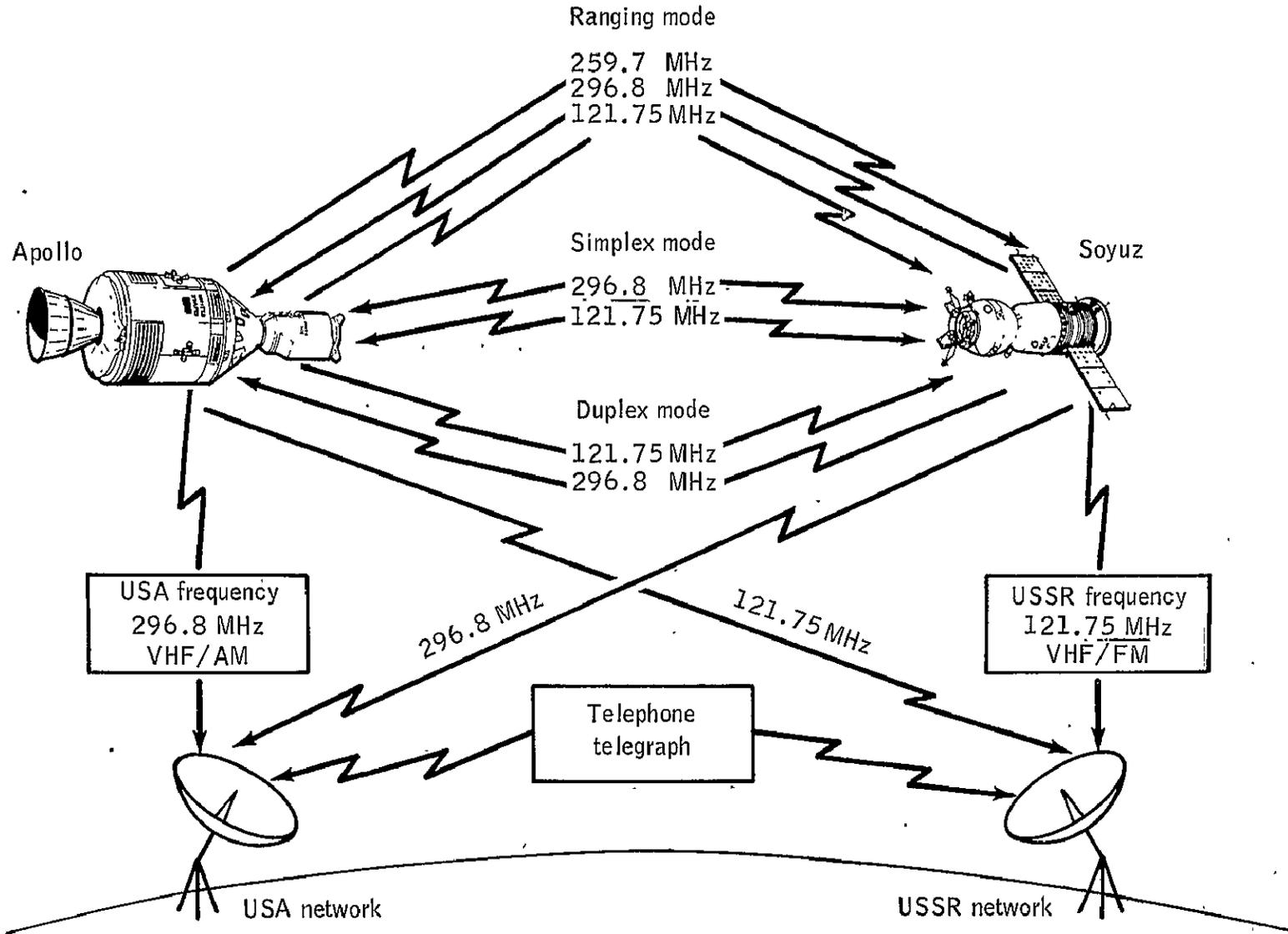


Figure A-19.- Radio communication modes.

Descent vehicle: Figure A-12 shows the location of the descent vehicle displays and controls. The instrument panel contained the following:

- a. A space navigation display simulated relative spacecraft-earth motion and provided other data for navigation.
- b. An electroluminescent indicator displayed system status, and provided caution/warning and alarm indications for out-of-tolerance conditions.
- c. A combined electronic display allowed the crew to monitor, via television, docking operations and areas of the spacecraft that could not be directly observed, and also displayed systems data in analog form.
- d. A program monitoring indicator displayed the status of automatic sequences.
- e. A digital display unit allowed the crew to enter and monitor control data, displayed propellant quantity and velocity changes due to main engine operation, and indicated backup engine operation.
- f. A voltage/current indicator showed the status of the electrical power supply system.
- g. A pressure/temperature indicator showed the pressure and temperature in the orbital module, descent vehicle, and instrument module, and also the temperature of the thermal control system heat-carrier liquid.
- h. A clock unit showed current and elapsed time, and incorporated a timer.
- i. Pressure gages indicated the pressure in the modules pressurization system and the gas mixture supply system.
- j. Additional controls on the instrument panel allowed the crew to initiate control commands, adjust radio volume, and control the brightness of the descent vehicle interior lights.

Two command/signal devices were provided, one on each side of the cabin. These devices contained selection keys, electroluminescent lights and on-off keys for spacecraft systems control. Systems control could be effected using either command/signal device, or the two devices could be used in combination.

The radio communication systems control panel was designed for manual control of VHF communications and exterior lights during rendezvous. The panel was mounted on the right-hand side of the cabin as shown in figure A-12.

Orbital module: The locations of the orbital module displays and controls are shown in figure A-11. The orbital module systems control panel contained circuit breakers, camera and lighting controls, communications controls, and atmospheric composition controls and displays. The integrity check system pneumatic panel incorporated a pressure-and-vacuum gage and electropneumatic valves that were used to perform the pressurization and pressure integrity check functions described in section A.2.2.4. A junction box in the orbital module was used to connect radio and television equipment transferred into the Soyuz spacecraft.

A.3 APOLLO SPACECRAFT, CREW EQUIPMENT, AND STOWAGE CONFIGURATIONS

A.3.1 Spacecraft

A.3.1.1 General.- CSM-111 was the Apollo spacecraft for the Apollo Soyuz mission. The spacecraft was initially manufactured for use on a lunar-type mission, but was modified to meet the specific requirements of the Apollo Soyuz mission. The major differences among the four basic types of Apollo spacecraft are given in table A-I. As indicated by the table, the spacecraft for the Skylab and Apollo Soyuz missions were similar in most respects. The major differences are listed in figure A-20.

A.3.1.2 Color television equipment.- Four color television cameras were carried; two were launched in the command module and two in the docking module. The cameras could be connected to any one of five television input stations (two in the command module, two in the docking module, and one in the Soyuz spacecraft). Video signals from the docking module or Soyuz were transmitted by cable to the command module and by S-band from the command module to the Spaceflight Tracking and Data Network. A switch in the docking module provided source selection between the docking module and the Soyuz spacecraft.

A video tape recorder in the command module provided the capability of recording 30 minutes of video signals from any one of the five input stations. The recorder had two audio inputs - intercommunication and downlink. The intercommunication audio input provided for interleaving audio with video signals while the recorder was in the "record" mode. The downlink audio input provided for interleaving audio in the same manner for real-time television transmission via the ATS-6 communications link (sec. A.4). Real-time video transmission was also available via the unified S-band communications link; however, in this mode, audio was transmitted over a separate carrier.

TABLE A-I.- SIGNIFICANT APOLLO SPACECRAFT CONFIGURATION DIFFERENCES

System	Area of difference	Mission type			
		Apollo Soyuz	Skylab	^a Lunar H	^b Lunar J
Command Module					
Electrical power system	Lunar module umbilicals			X	X
	Skylab tunnel umbilical		X		
	Docking module umbilicals	X			
	Drag-through umbilical	X	X		
Experiments	Stowable	X	X	X	X
	Coldplate mounted	X			
Television	Cameras and monitors	X	X	X	X
	Video tape recorder	X			
Communications	Speaker box	X	X		
	ATS-6 equipment	X			
Environmental control system	Extravehicular activity capability		X		X
Displays and controls	Experiments	X	X		X
	Docking module	X			
	Compatible docking system	X			
	ATS-6 communications	X			
Stowage	Apollo Block II			X	X
	Skylab		X		
	Modified Skylab	X			
Service Module					
Experiments	Scientific instrument module	X			X
	Lunar sounder				X
	Doppler tracking receiver	X			
	Remotely controlled doors	X			X
	Extravehicular retrieval capability				X
Service propulsion system	Propellant utilization gaging system (flight)			X	X
	Propellant utilization gaging system (ground)	X	X		
	Four propellant/two pressurant tanks			X	X
	Two propellant/one pressurant tank	X	X		
Environmental control system	Heaters deactivated	(c)	X		
	Coldplates for experiment cooling and ATS-6 equipment	X			

^aLunar landing mission.

^bLunar landing mission with expanded scientific data return capability.

^cECS radiator heater motor switches placed in open position prior to launch.

^dFor long-duration RCS firings.

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TABLE A-I.- SIGNIFICANT APOLLO SPACECRAFT CONFIGURATION DIFFERENCES - Concluded

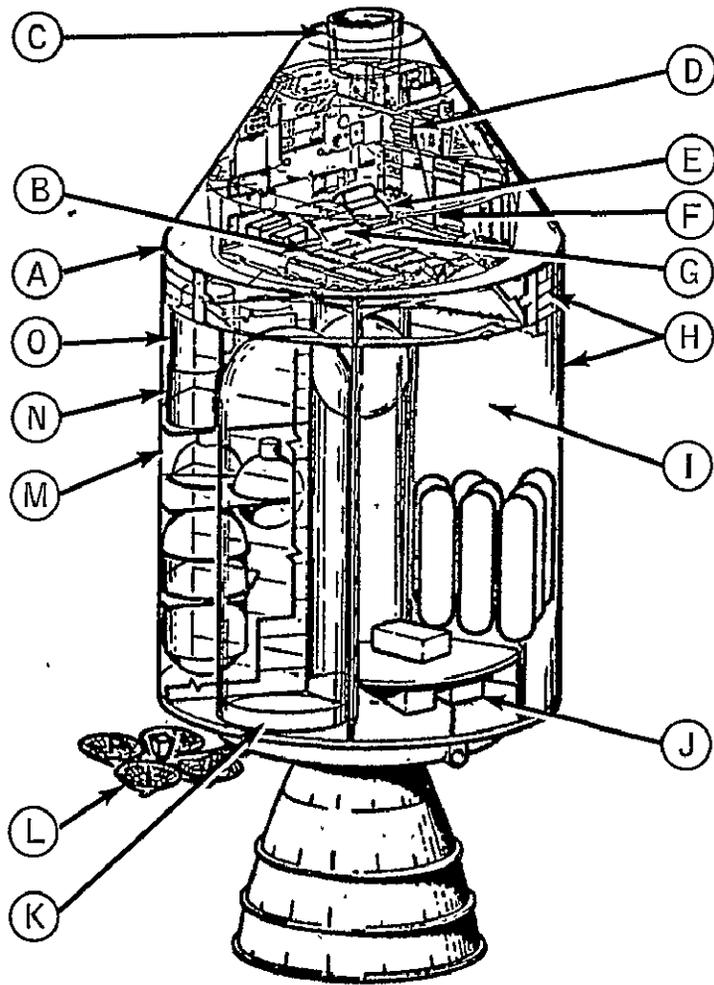
System	Area of difference	Mission type			
		Apollo Soyuz	Skylab	^a Lunar H	^b Lunar J
Reaction control system	Propellant storage module	X	X		
	RCS quad heaters	X	X		
Thermal protection	Increased cork insulation	X	X		
	^d Additional cork insulation	X			
Communications	Rendezvous radar transponder			X	X
	ATS-6 power amplifier system	X			
	High-gain antenna	X		X	X
Electrical power system	Two fuel cells		X		
	Three fuel cells	X		X	X
	Descent batteries		X		
	Extra water tank		X		
	Increased cryogenic storage capacity			X	X
	Return enhancement battery			X	X
Spacecraft Adapter					
Panels	Jettisonable Deployable	X	X	X	X
Structural support	Lunar module Docking module	X		X	X

^aLunar landing mission.

^bLunar landing mission with expanded scientific data return capability.

^cECS radiator heater motor switches placed in open position prior to launch.

^dFor long-duration RCS firings.



- A Provisions for experiments added.
- B Stowage provisions modified.
- C Docking module umbilicals replaced Skylab tunnel umbilical.
- D Displays and controls added for compatible docking system, docking module, experiments, and ATS-6 communications.
- E Modified unified S-band equipment and premodulation processor added for ATS-6 communications.
- F Extravehicular activity station deleted.
- G Video tape recorder added.
- H Receiver and antenna added for doppler tracking experiment.
- I Descent battery pack deleted.
- J Experiments and remotely controlled covers added (service module bay 1).
- K Power amplifiers added for ATS-6 communications.
- L High-gain antenna added for ATS-6 communications.
- M Water storage tank deleted.
- N Insulation added adjacent to RCS thrusters.
- O Three fuel cells installed instead of two.

Figure A-20.- Major spacecraft changes from Skylab configuration.

In preflight tests, cabin floodlights caused hot spots on the television screen, diminishing clarity and scene detail. Light polarization was determined to be the best method of eliminating the problem. The offending floodlights were covered with a polarized Lexan material with a transmissibility of 32 percent. Cross-polarization of the floodlight filters was accomplished by installing a 32-percent-transmissibility adjustable polarizing filter on the television camera lens. By rotating the television camera polarizing filter, the floodlight hot spots were eliminated while allowing the desired scene to be displayed.

A.3.2 Crew Equipment

A.3.2.1 Space suits.— The space suits for this mission were similar to the Command Module Pilots' suits used in the Apollo 15, 16 and 17 missions except that the provisions for extravehicular activity were deleted. In addition, a Tygon tube insert was incorporated in the crotch area between the pressure sealing and restraint entry zippers to prevent recurrence of a low pressure leakage problem encountered with the Command Module Pilot's suit during the countdown demonstration test. Figure A-21 shows the suit configuration.

A.3.2.2 Photographic and ancillary equipment.— The photographic equipment included four camera systems: a 16-mm data acquisition camera, a 35-mm camera, a 70-mm data camera, and a 70-mm reflex camera. The first three systems were similar to those used on the Skylab missions.

The 16-mm system (fig. A-22) included a 5-mm lens, a 10-mm lens, a 25-mm lens, a 75-mm lens, a 140-foot film magazine, a cine actuator, a right angle mirror, a power cable, and brackets. The 35-mm system included a 35-mm f/1.4 lens, a 35-mm film cassette, a cable release, a 300-mm f/4.5 lens, an eyepiece, a 300-mm lens bracket, and an automatic flash assembly. The 300-mm lens, eyepiece, and bracket (fig. A-23) were used in conjunction with a target on the top guide vane of the docking module docking mechanism to verify that the Soyuz docking system, or the docking module system, had extended the required distance to insure a safe dock. The 300-mm lens was a specially modified version of the 300-mm lens utilized on Skylab and provided a vernier scale with 2.54-centimeter graduations from 4.6 to 5.5 meters. The 70-mm data camera system included a 60-mm lens, an 80-mm lens with a 21-mm extension tube, a 100-mm lens, and eight film magazines of approximately 150 exposures each (fig. A-24). The 70-mm reflex camera system included a prism viewfinder, a 50-mm lens, a 250-mm lens, and 15 film magazines of 70 exposures each, along with infrared filters. These items are shown in figure A-25. Other items of crew equipment added for this mission are briefly described in the following paragraphs.

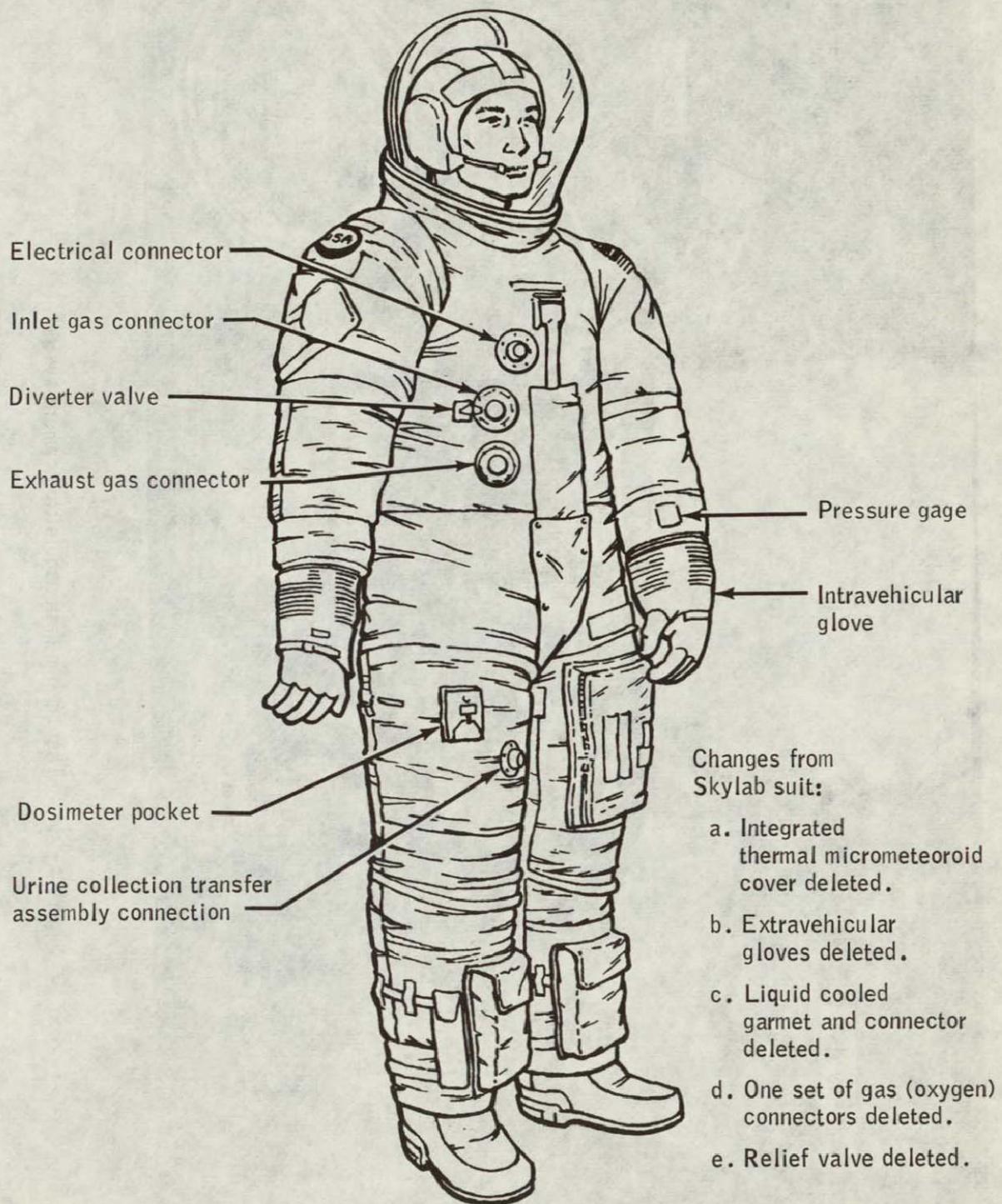


Figure A-21.- Space suit configuration.



Figure A-22.- The 16-mm data acquisition camera system.

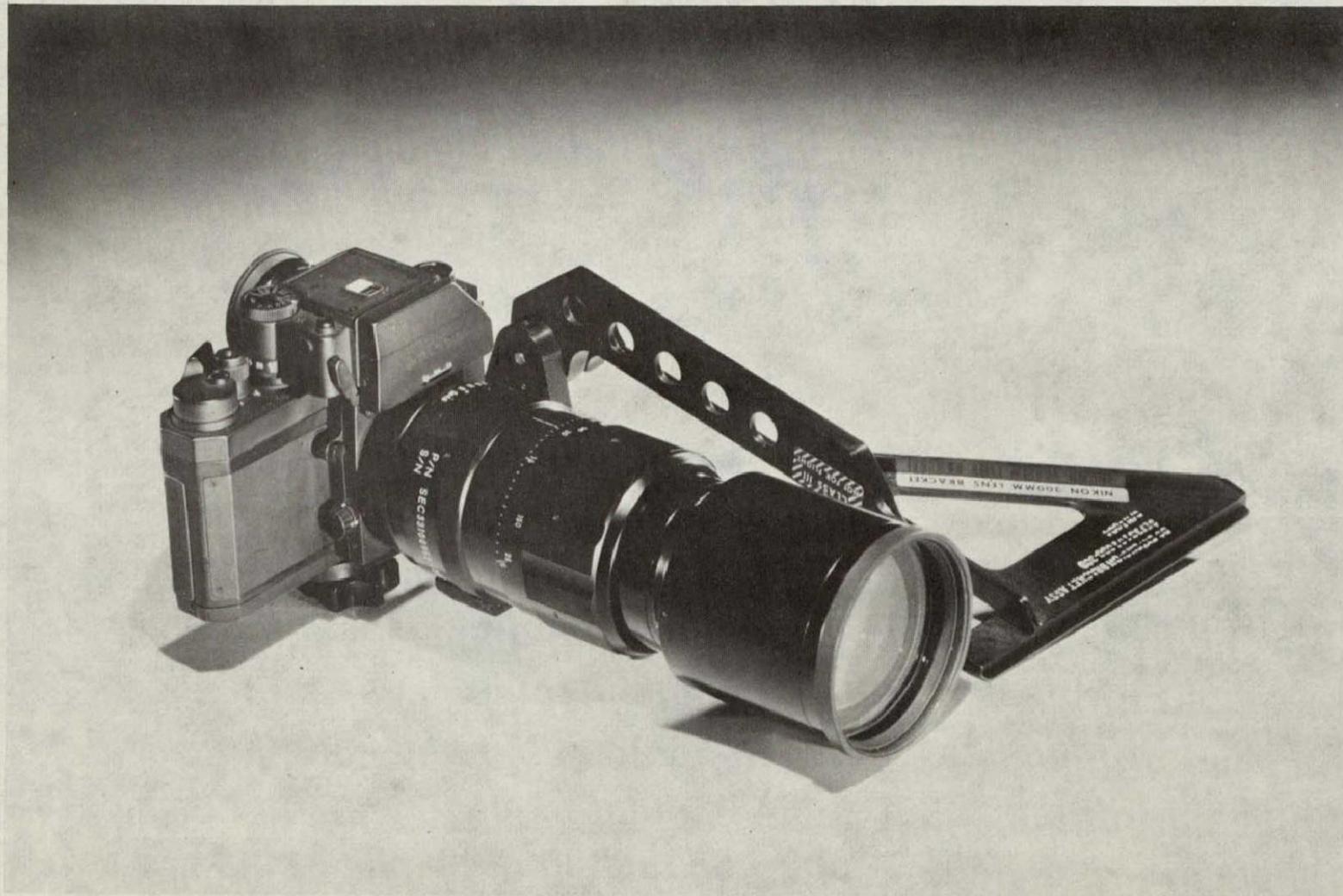


Figure A-23.- The 35-mm camera configured for docking system extension verification.

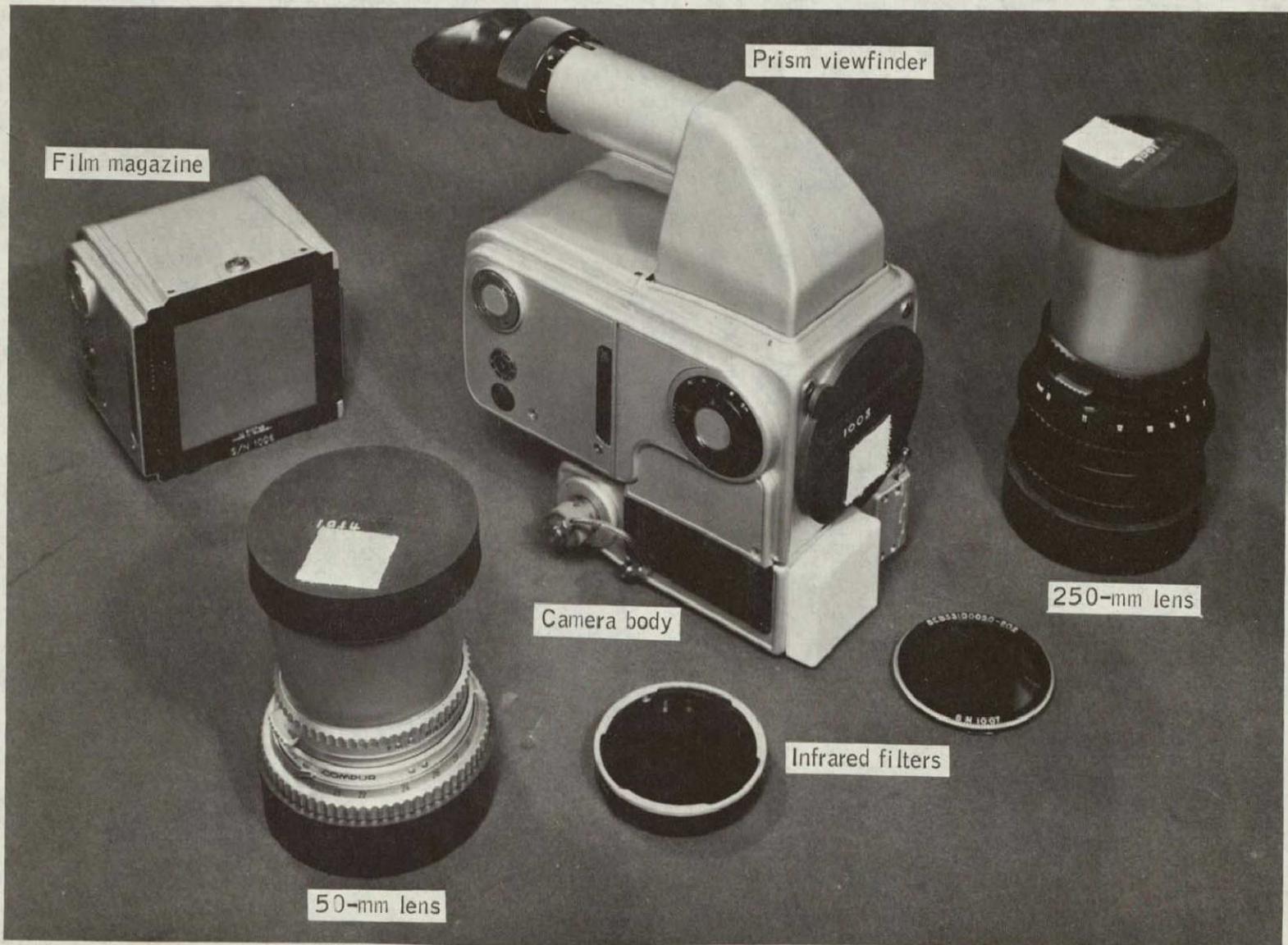


Figure A-25.- The 70-mm reflex camera system.

General-purpose optical viewing device: The general purpose optical viewing device was a 20- to 45-power zoom spotting telescope with an additional 20-power wide-angle eyepiece. The device was handheld and had a cushion on the front to protect the windows of the spacecraft.

Binoculars: Twenty-power binoculars were added to support the earth observation and photography experiment (MA-136). The binoculars were modified from a commercially available type and incorporated a reticle pattern to enable the crew to estimate object sizes on the ground.

Variable intervalometer: The variable intervalometer was a modified version of the Skylab unit. This unit provided picture intervals of 2.5, 6.25, 10, and 180 seconds. The interval selected depended on the lens focal length used. The intervalometer is shown in figure A-24.

Cine actuator: The cine actuator (fig. A-22) was a remote-control device for the 16-mm data acquisition camera that allowed continuous unattended operation of the camera in the 24-frames-per-second mode. It actuated the camera by means of a toggle switch for on-off control. The actuator had a bracket for mounting to the accessory shoe on the camera and Velcro was used for other mounting as required.

Twenty-one-millimeter extension tube: The 21-mm extension tube was a metal coupling ring between the 70-mm data camera body and lens that permitted the lens to be focused at a distance of 4 to 4 1/4 meters. The extension tube is shown in figure A-24.

Data acquisition camera/television camera adapter bracket: The data acquisition/television camera adapter bracket was used to mount the 16-mm data acquisition camera to the United States television camera, allowing the data acquisition and television camera to view the same scene from approximately the same location with the lines of sight of both cameras parallel.

Color wheel: The color wheel was part of the earth observation and photography experiment (MA-136). The wheel was 20.3 centimeters in diameter, and contained 108 different colors consisting of different shades of red, blue, and yellow. This enabled the crew to compare colors observed on the ground to the colors on the color wheel.

Photographic auxiliary light: The photographic auxiliary light was a portable fluorescent light fixture that supplied additional lighting in the docking module and command module for 16-mm data acquisition photography (fig. A-22). The light was capable of providing two levels of light output and operated from a 14-volt dc self-contained battery that could be changed in flight.

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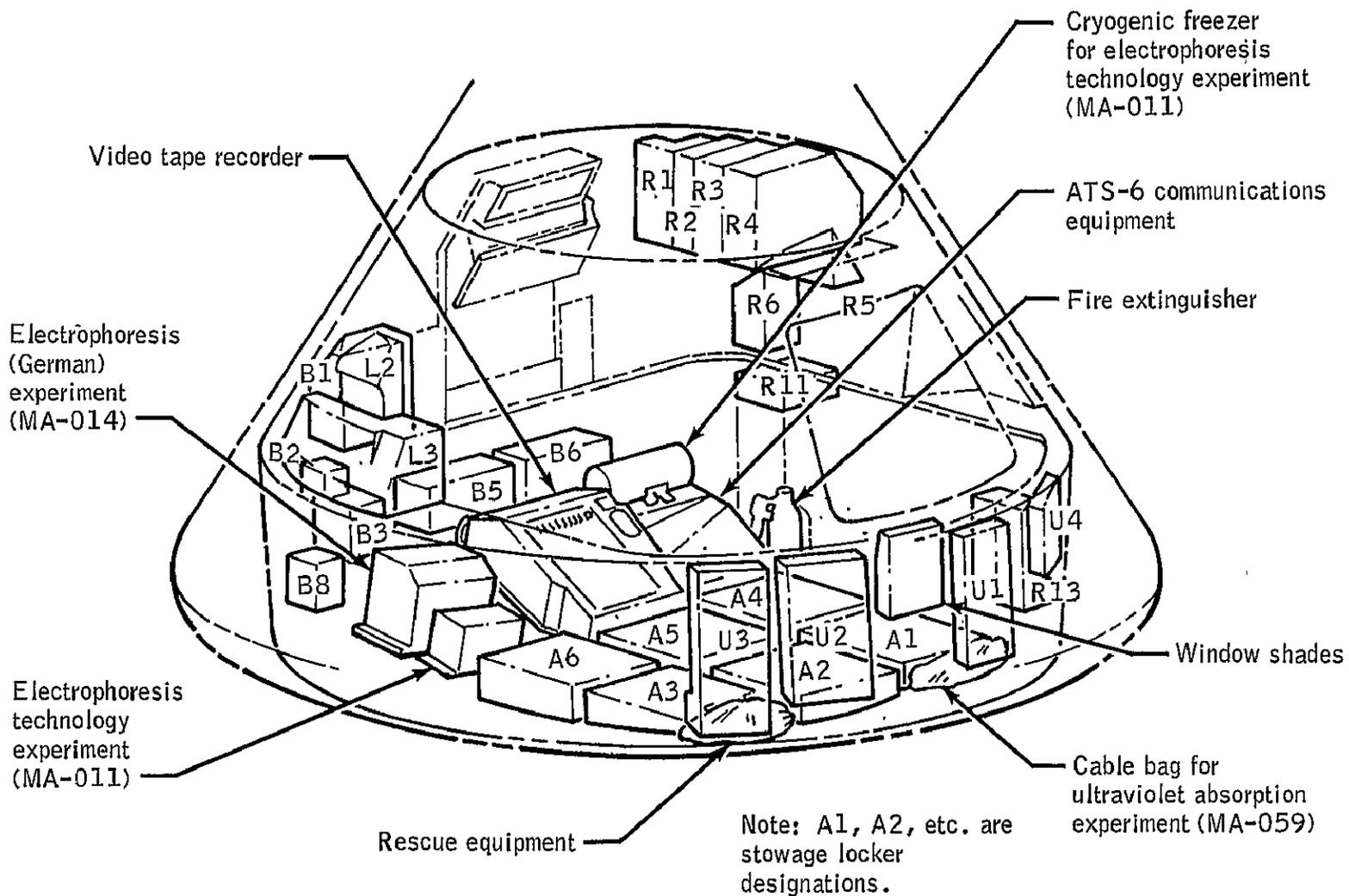


Figure A-26.- Command module launch stowage arrangement.

A.3.3.2 Command module rescue configuration.- Two options were developed for use in the event that rescue of the Soyuz crew became necessary. One option provided for rescue of the Soyuz crew in the event of a contingency arising subsequent to launch. The second option provided for rescue in the event of a Soyuz failure before the normal Apollo launch. For the first option, restraint harness assemblies, foot restraints, and helmets were stowed in the locations shown in figure A-26. Torso and buttock pads were stowed in the crew couches. For the second option, two lockers containing extra rescue equipment and survival gear would have replaced lockers A-2 and A-3 (fig. A-26).

A.3.3.3 Docking module.- The general docking module stowage arrangement is shown in figure A-27.

A.4 APPLICATIONS TECHNOLOGY SATELLITE 6

Applications Technology Satellite 6 (ATS-6) is a communications relay satellite that was launched on May 30, 1974, and was placed in a geosynchronous orbit; that is, it remains above a fixed point on the earth unless intentionally moved. For the Apollo Soyuz mission, the satellite was located in an equatorial orbit at longitude 35° east over the eastern coast of Africa at an approximate altitude of 35 800 kilometers. Because it was in a synchronous orbit, ATS-6 was in view of a single Spaceflight Tracking and Data Network ground station at all times. The ground station was located at Buitrago, Spain. Two receivers were available at Buitrago - a 97-foot antenna which was used primarily for television reception and a 21-foot antenna for other communications (fig. A-28).

ATS-6 was used to expand continuous communications contact with the Apollo command and service module up to approximately 50 minutes per revolution. The ground station transmitted voice and spacecraft command data to the ATS-6 via C-band at 6152.4 megahertz; the ATS-6 then retransmitted to the spacecraft via S-band at 2077.4 megahertz. This was termed the forward link. On the reverse link, the spacecraft transmitted to the ATS-6 on S-band at 2256 megahertz and the ATS-6 retransmitted to the Spaceflight Tracking and Data Network ground station on C-band at 3956 megahertz. The reverse link was used to transmit voice, data, or television and interleaved voice, either in real time or played back.

A.5 EXPERIMENT HARDWARE

Figure A-29 illustrates the locations of the various items of experiment hardware discussed in the following paragraphs.

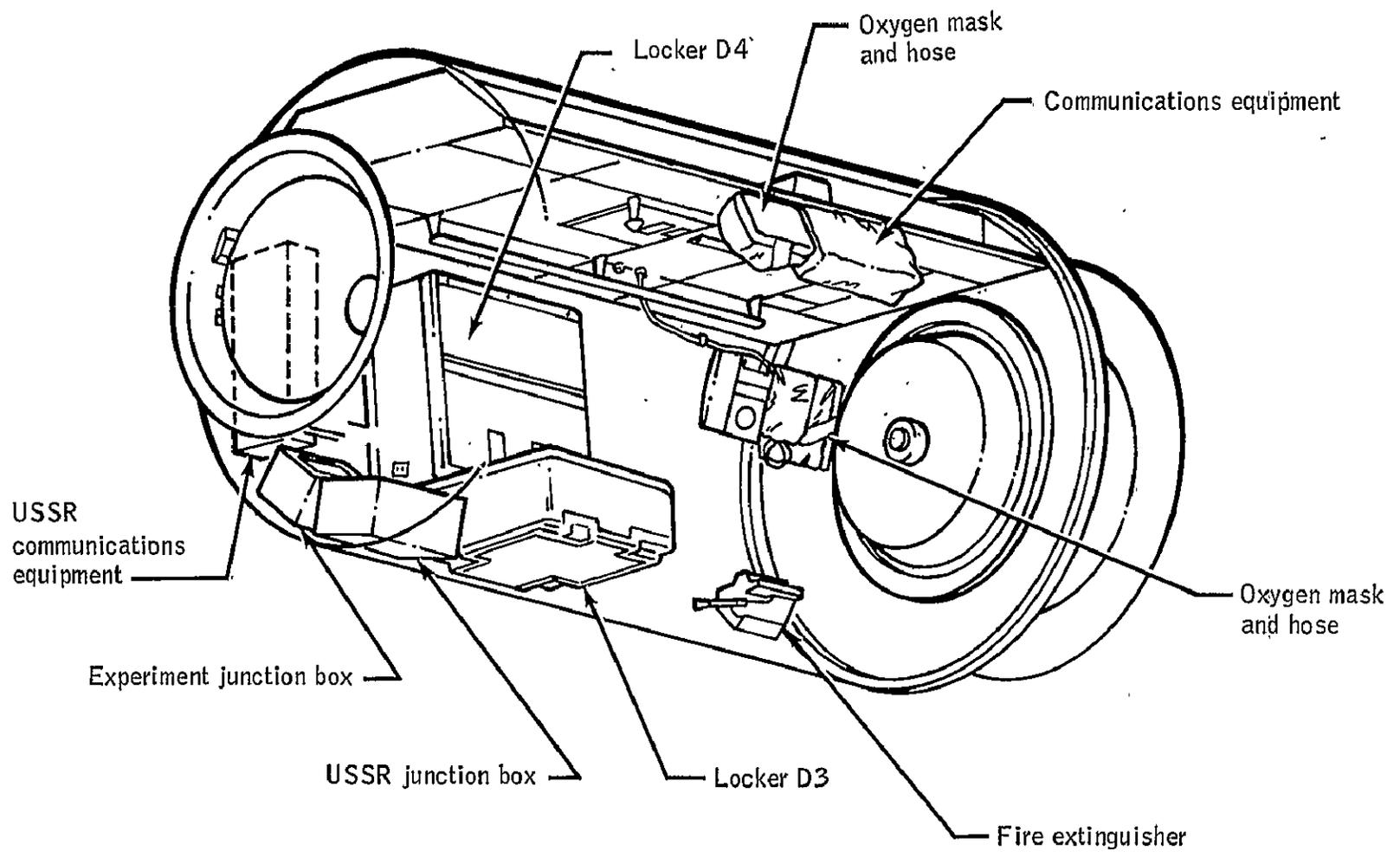


Figure A-27.- Docking module launch stowage arrangement.

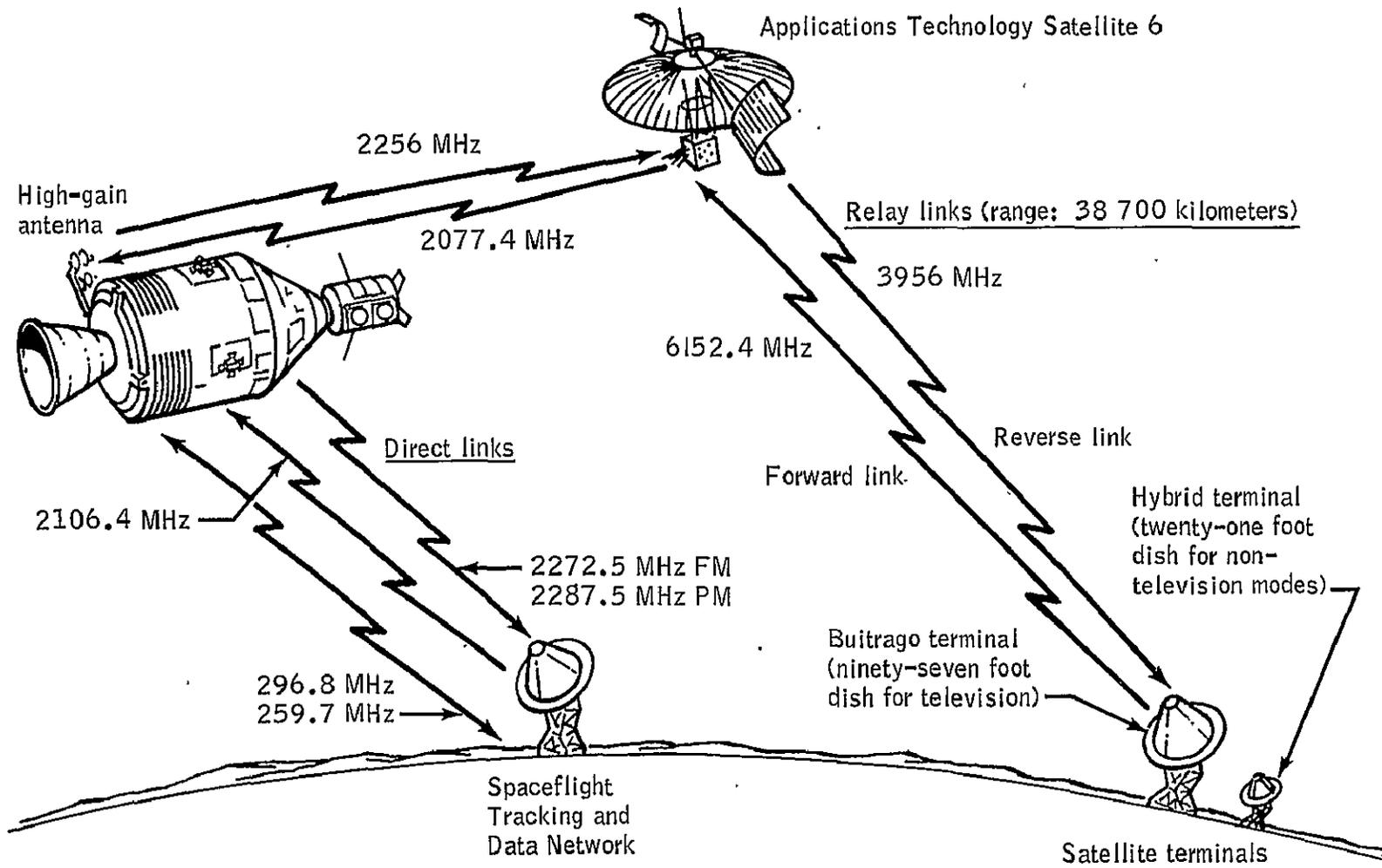


Figure A-28.- Apollo communication links.

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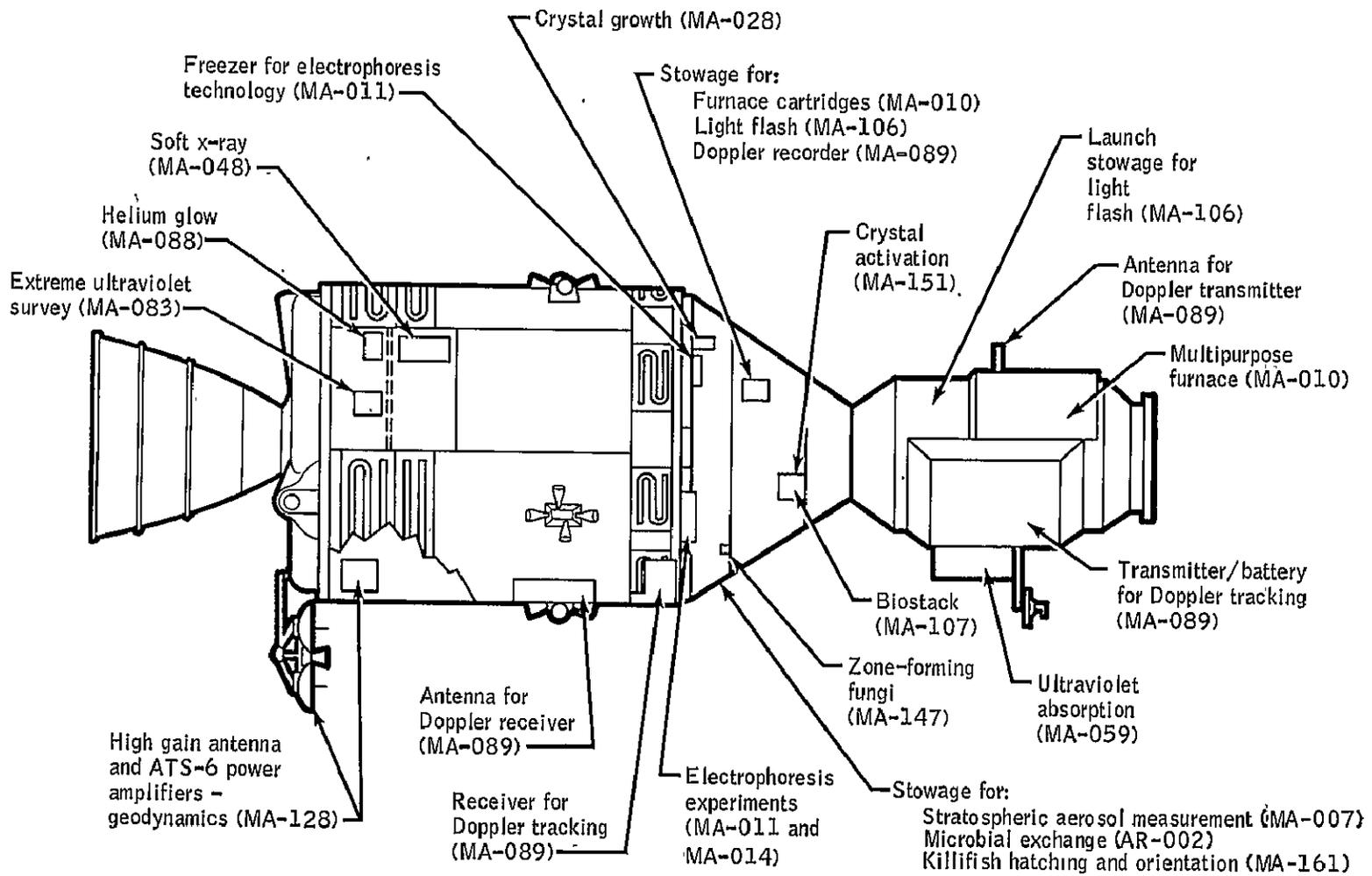


Figure A-29.- Experiment equipment locations.

A.5.1 AR-002 Microbial Exchange

The inflight equipment for the microbial exchange experiment consisted of 64 sample tubes supplied by the USSR and four sample kits supplied by the USA. The sample kits were launched and returned in the Soyuz spacecraft.

A.5.2 MA-007 Stratospheric Aerosol Measurement

The photometer used in the experiment measured solar intensity in the 1-micrometer wavelength (near infrared) region of the spectrum. The instrument also contained an electronics package which provided a signal that could be transmitted to the ground in real time or recorded and transmitted at a later time via the spacecraft telemetry system. The hand-held 70-mm reflex camera with the 250-mm lens and infrared filter were used to record the sun during photometer data takes.

A.5.3 MA-010 Multipurpose Furnace

The multipurpose furnace allowed high-temperature (up to 1423° K) processing of materials in space. Subsequent study of the processed materials will have implications in several fields.

The furnace system was comprised of three units - the furnace, a control package, and a rapid cooldown assembly. The furnace was used to simultaneously heat three cartridges containing sample materials. The control package provided programmable temperature control functions. The rapid cooldown assembly allowed the operator to inject helium gas into the furnace to accelerate cooling of the cartridges.

A.5.4 MA-011 Electrophoresis Technology

The electrophoresis technology experiment employed a static column of buffer solution with an electric field aligned along the column. Sample mixture was introduced at one end of the column, and the mixture separated into zones which traveled at different rates down the column. The samples were preserved before and after processing by freezing them in a cryogenic freezer containing liquid nitrogen. The electrophoresis unit was self-contained and required power connection and photographic coverage. The unit contained eight columns, thermoelectric units to cool and freeze the columns, and a pump and motor for buffer circulation.

A.5.5 MA-014 Electrophoresis - German

In this method of electrophoretic separation, the buffer solution flowed along a tube while an electric field was aligned perpendicularly to the tube. The sample mixture was inserted continuously at one end. Multiple streams of separated sample material were collected continuously at the other end. The deflection and density distribution of the deflected streams was detected opto-electronically, and the data were recorded on magnetic tape. No attempt was made to preserve these samples. The apparatus for the experiment was self-contained and required spacecraft electrical power and a coldplate for heat dissipation.

A.5.6 MA-028 Crystal Growth

The crystal growth experiment consisted of six transparent reactors, each having three compartments. The center compartment contained pure water and the two end compartments contained the reactants. The end compartments were opened so that the reactants could diffuse toward each other through pure water. The crew made observations and photographed the experiment at approximately 12 hour intervals after completion of joint activities with the Soyuz crew.

A.5.7 MA-031 Cellular Immune Response

The cellular immune response experiment required no flight hardware.

A.5.8 MA-032 Polymorphonuclear Leukocyte Response

The polymorphonuclear leukocyte response experiment required no flight hardware.

A.5.9 MA-048 Soft X-Ray

The soft X-ray experiment consisted of a proportional counter detector, an X-ray window, an electronics assembly, and a gas (90-percent argon, 10-percent methane) supply. The experiment was permanently mounted in service module bay 1. A remotely controlled cover shielded the experiment from solar radiation and particle contamination until it was opened for experiment operations.

A.5.10 MA-059 Ultraviolet Absorption

The ultraviolet absorption experiment required the use of both the Apollo and the Soyuz spacecraft following separation. The experiment hardware consisted of a spectrometer assembly on the docking module and three retroreflector arrays on the Soyuz spacecraft. Ultraviolet light was transmitted from the docking module to one of the retroreflector arrays on the Soyuz and was returned to a detector on the docking module. The separation of the two spacecraft was varied (150 m, 500 m, and 1000 m) to eliminate absorption effects of contamination around either spacecraft. The location of the spectrometer assembly on the docking module is shown in figure A-5.

A.5.11 MA-083 Extreme Ultraviolet Survey

The instrument for this experiment was an extreme ultraviolet telescope mounted in service module bay 1. The telescope consisted of an optical system, a filter wheel assembly, a detector assembly, and an electronics and signal conditioning module. A remotely controlled cover protected the instrument from contamination until it was opened for data collection.

A.5.12 MA-088 Helium Glow

The helium glow experiment consisted of a detector, including a helium gas system, and electronics. The detector subassembly contained four channel electron multiplier detectors. Two of these observed the 584-angstrom (He I) spectral emission line and the other two observed the 304-angstrom (He II) line. The experiment was permanently mounted in service module bay 1. A remotely controlled cover protected the detection system from contamination until it was opened for data collection operations.

A.5.13 MA-089 Doppler Tracking

The use of high-stability VHF transmission between spacecraft can theoretically yield doppler determinations of relative speeds less than 1/2 mm/sec, and such signals integrated for 10 seconds are capable of detecting anomalous accelerations of 10 millionths of one g. The MA-089 experiment investigated a technique in which the two orbiting bodies were in the same orbit. To do this, the command and service module and the docking module were separated to a distance of approximately 300 kilometers and the command and service module received transmitted signals

from the docking module. With this configuration, mass anomalies of approximately 200- to 350-kilometer size were expected to be resolved. (A second technique is described in sec. A.5.16.)

A.5.14 MA-106 Light Flash

The light flash experiment hardware consisted of individually fitted, light-tight masks; a power control box with an integral digital tape recorder; a silicon detector spectrometer-telescope; and silver chloride (cadmium doped) thin crystals. A comparison between flux detected by the instrumentation and events observed by the crewmen will be made.

A.5.15 MA-107 Biostack

The Biostack experiment hardware consisted of two cylindrically shaped canisters containing biological materials and cosmic ion track detectors. The packages were placed in a location where radiation shielding was as low as possible. One canister was completely passive and required no crew participation. The other required the crew to turn it on and off at specified intervals.

A.5.16 MA-128 Geodynamics

The technique used in the geodynamics experiment employed a low orbiting spacecraft and a synchronous satellite in a high orbit (ATS-6). Doppler data were collected throughout preselected Apollo passes during times of ATS-6 coverage. Forward-link signals originated from the Spaceflight Tracking and Data Network ground station at Madrid, Spain, and reverse-link signals were relayed through the ATS-6 to the ground. Measurements between the spacecraft and the ATS-6 were expected to resolve mass anomalies of approximately the same size as those of the doppler tracking experiment (MA-089).

A.5.17 MA-136 Earth Observations and Photography

Visual observations were made using 20-power binoculars, a spotting telescope (20 to 45 power) and specially prepared maps and charts. Photographs were taken with the handheld 70-mm reflex camera. Stereoscopic photographs were taken to support mapping objectives using the 70-mm data camera. Long-strip documentary photography was accomplished with the 16-mm data acquisition camera. Details of the camera systems are given in section A.3.2.

A.5.18 MA-147 Zone Forming Fungi

The zone-forming fungi experiment consisted of four containers - two located in the Soyuz spacecraft and two in the Apollo spacecraft. Each container held two Petri dishes having microorganism cultures. During the flight, the cultures were photographed periodically, and one container from each spacecraft was exchanged during the joint operations.

A.5.19 MA-148 Artificial Solar Eclipse

The artificial solar eclipse experiment utilized a 50-mm camera (90-mm focal length lens) in the Soyuz docking hatch window to obtain photographs of the solar corona while the Apollo spacecraft eclipsed the sun. Scattered light was controlled by a baffle mounted on the outside of the hatch window. The camera was to be operated automatically with repeated sequences of 0.1, 0.3, 1.0, 3.0, and 10.0 second exposures.

The Apollo shadow on the Soyuz was photographed with the 16-mm data acquisition camera (75-mm focal length lens) mounted in a window of the command module to provide information on separation distance and umbra location as a function of time.

A.5.20 MA-151 Crystal Activation

Hermetically sealed stainless steel cylinders containing crystals of sodium iodide and germanium were carried aboard the Apollo spacecraft and were returned for postflight analysis.

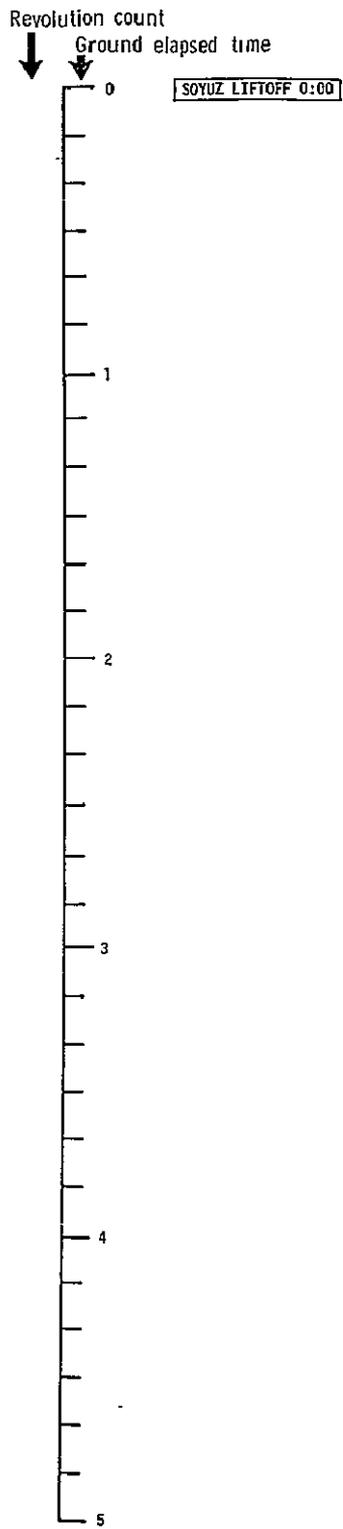
A.5.21 MA-161 Killifish Hatching and Orientation

The killifish hatching and orientation experiment consisted of two plastic film packages, one containing preconditioned killifish (*Fundulus*) fry, and the other, a graded series of embryonated *Fundulus* eggs representing key development stages. The fry in the first package were subjected to visual cues and disturbances and observed for anomalous behavior. The behavior was also recorded photographically and televised to allow observation by ground personnel. Both packages were returned and samples were killed and fixed for microscopic analysis.

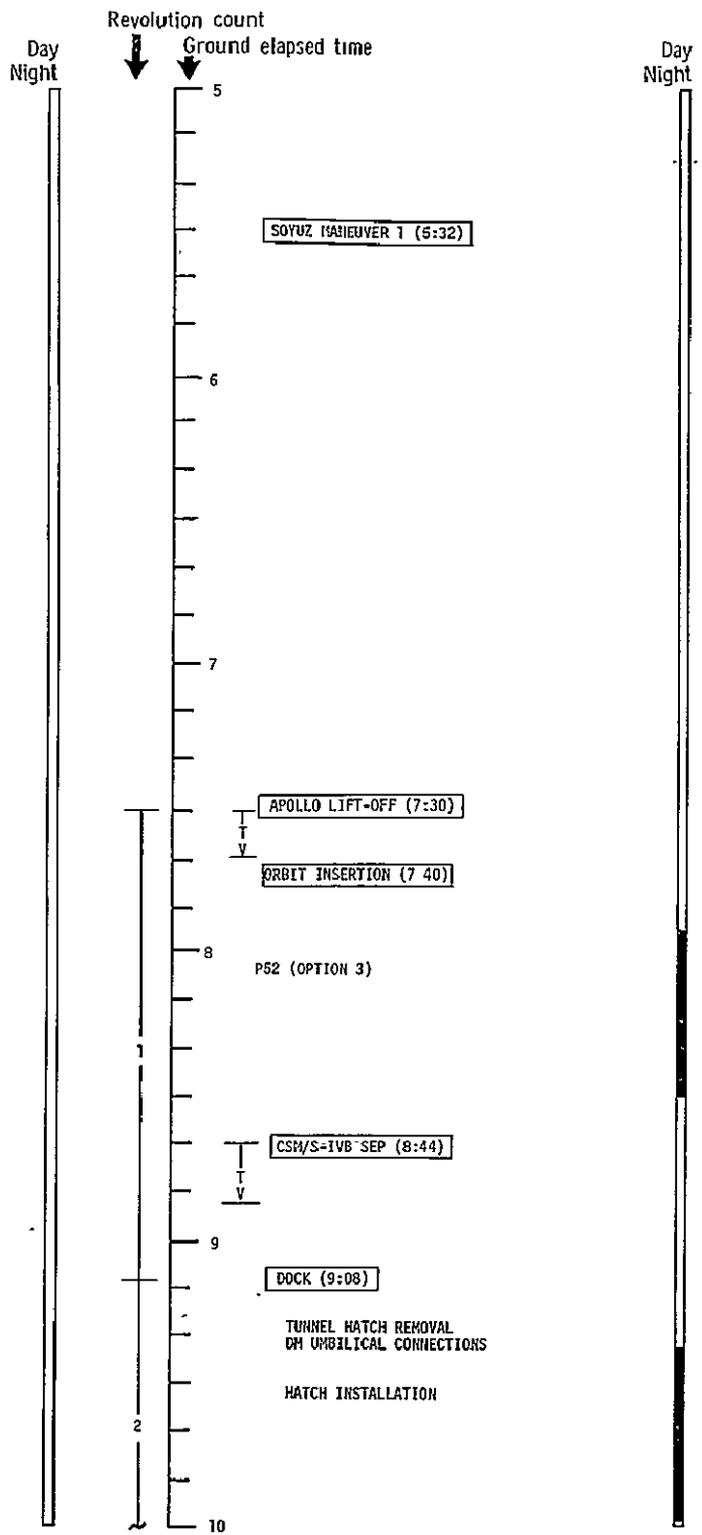
APPENDIX B - ASTP SUMMARY AS-FLOWN FLIGHT PLAN

This appendix consists of a timeline which summarizes the mission as it was accomplished. Acronyms and abbreviations used in the timeline are defined in table B-I which follows the timeline.

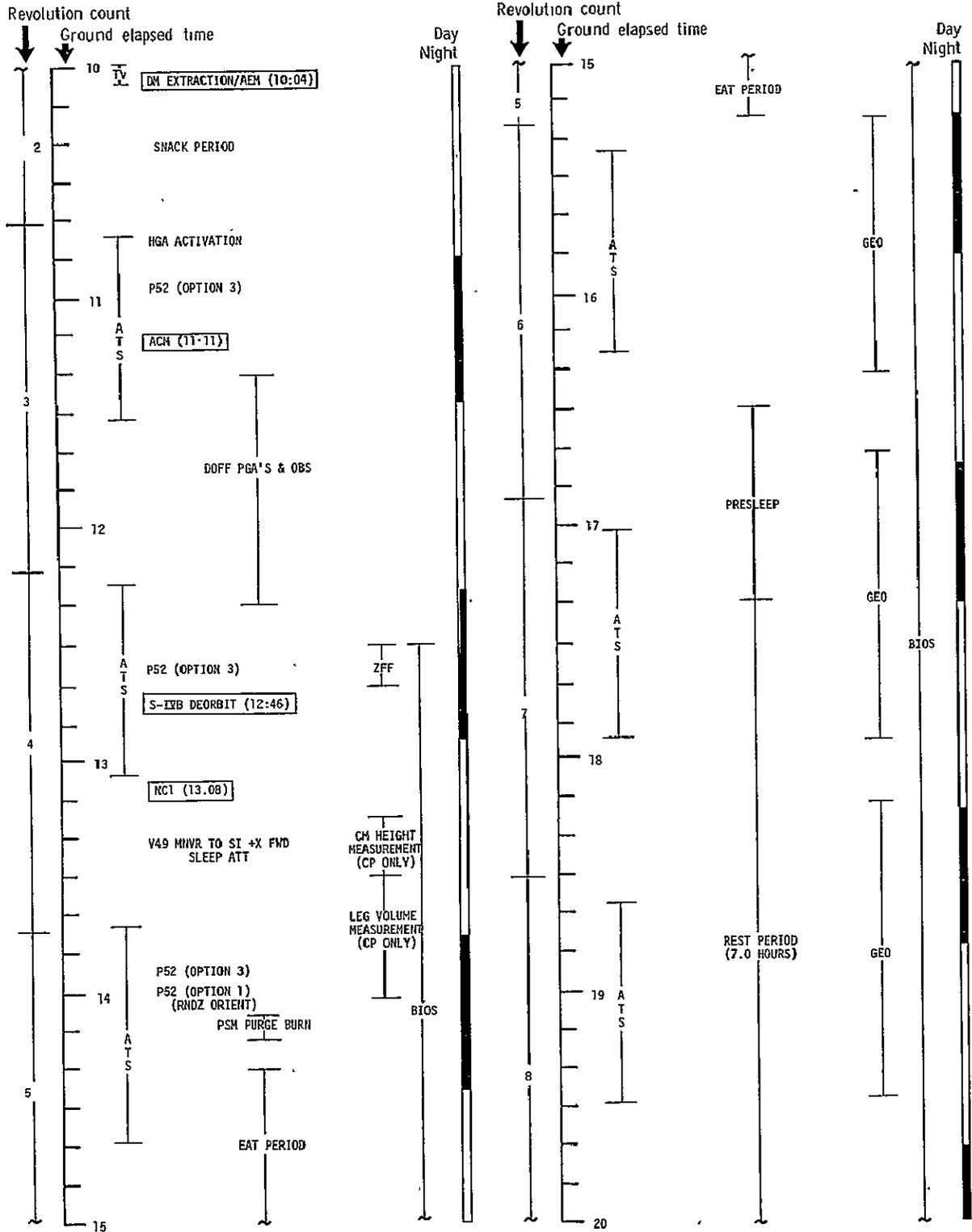
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN



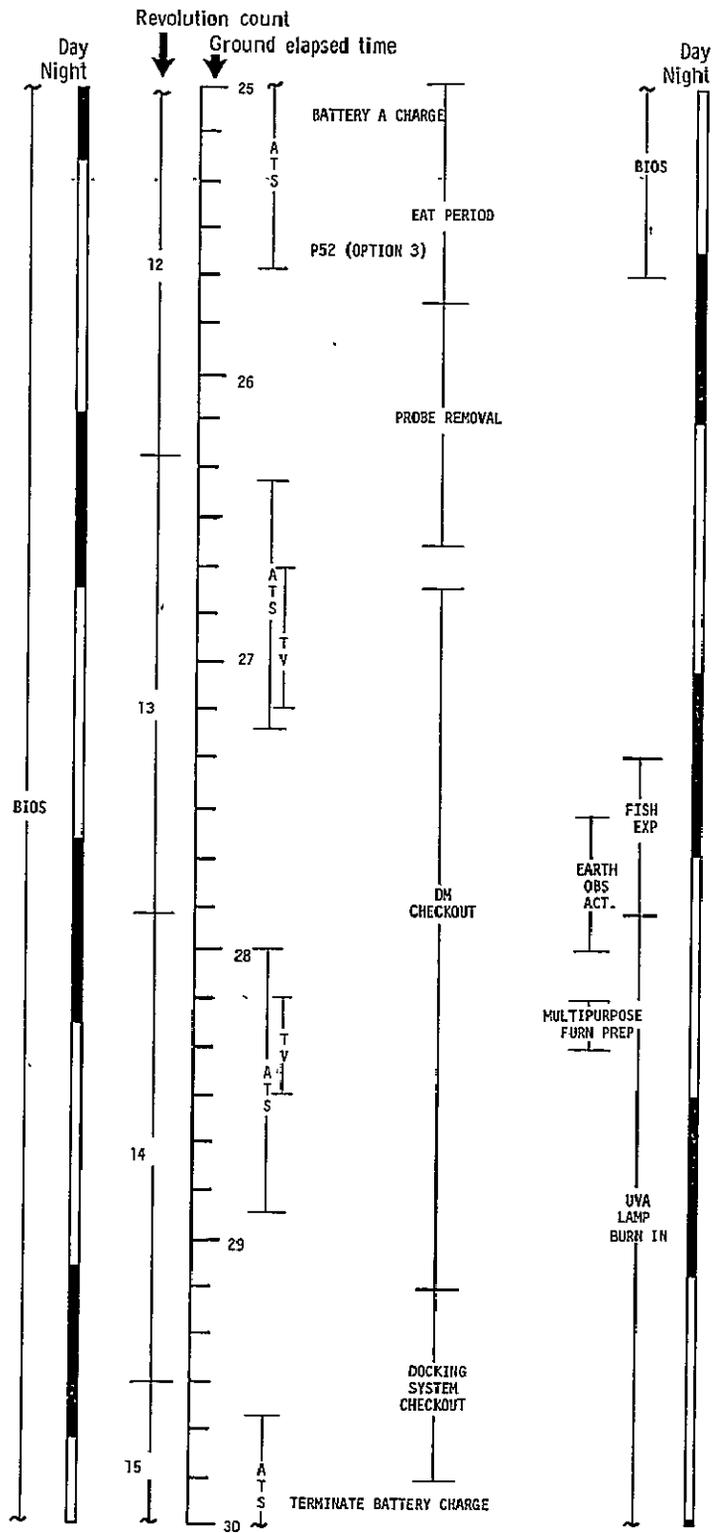
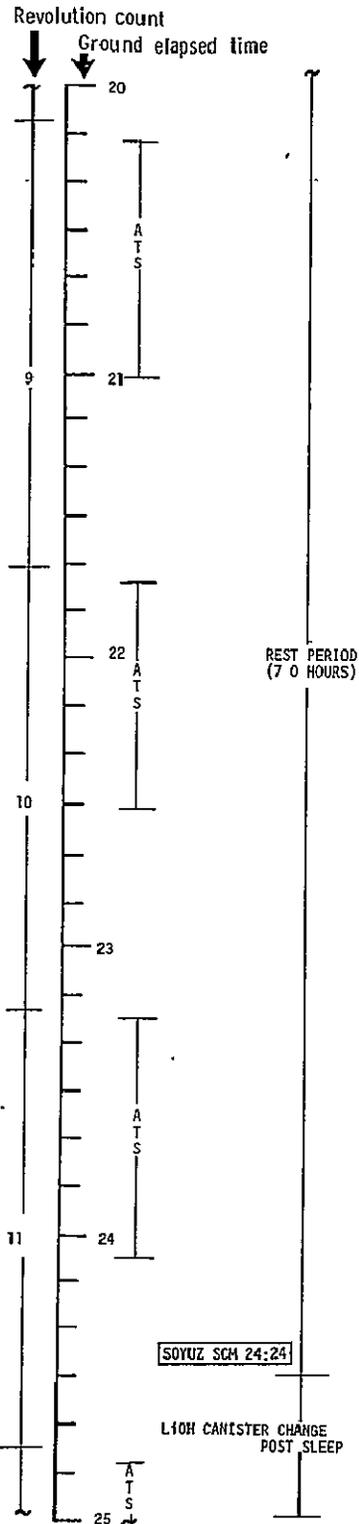
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FLIGHT PLAN



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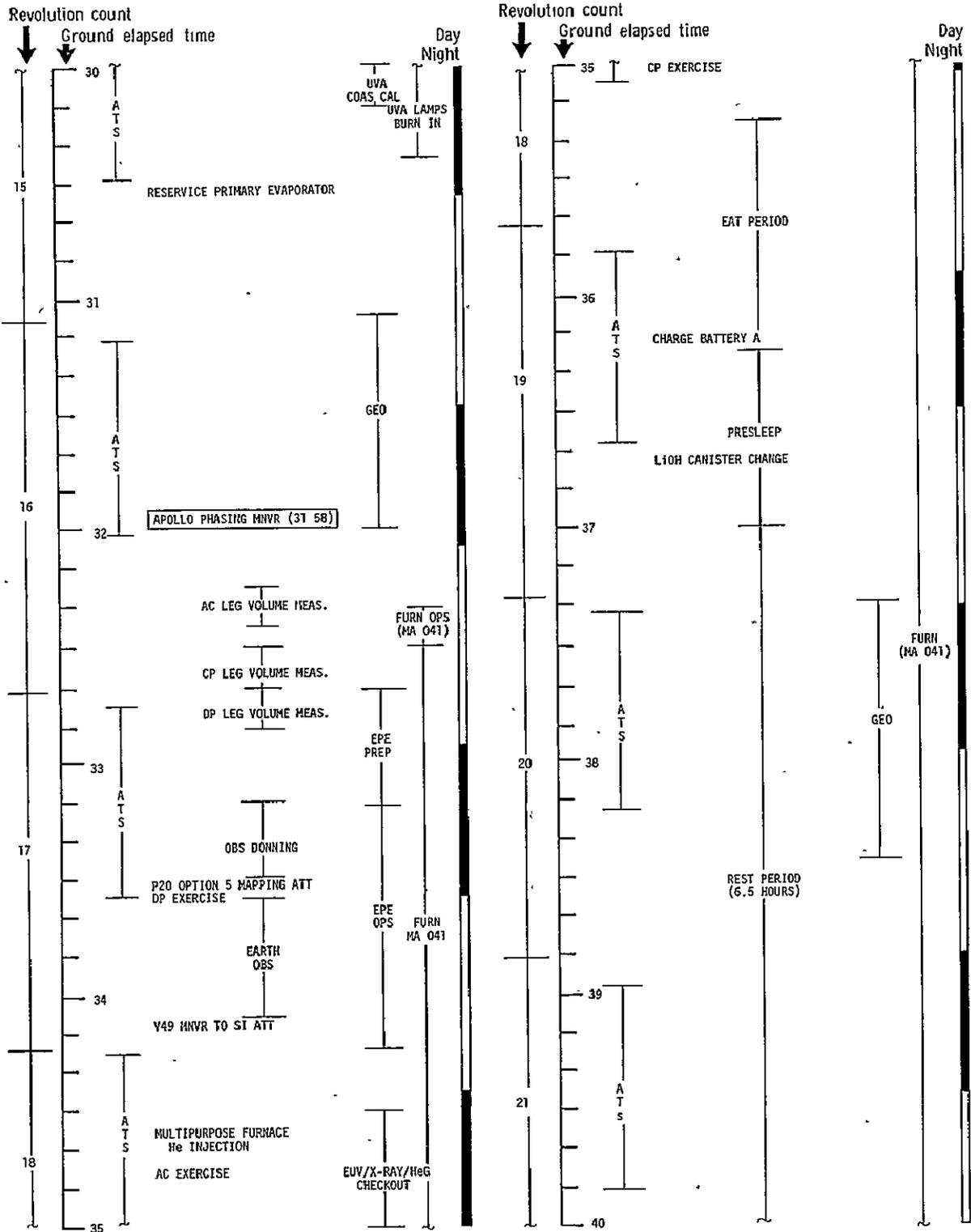
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN



ASTP SUMMARY AS FLOWN
FLIGHT PLAN

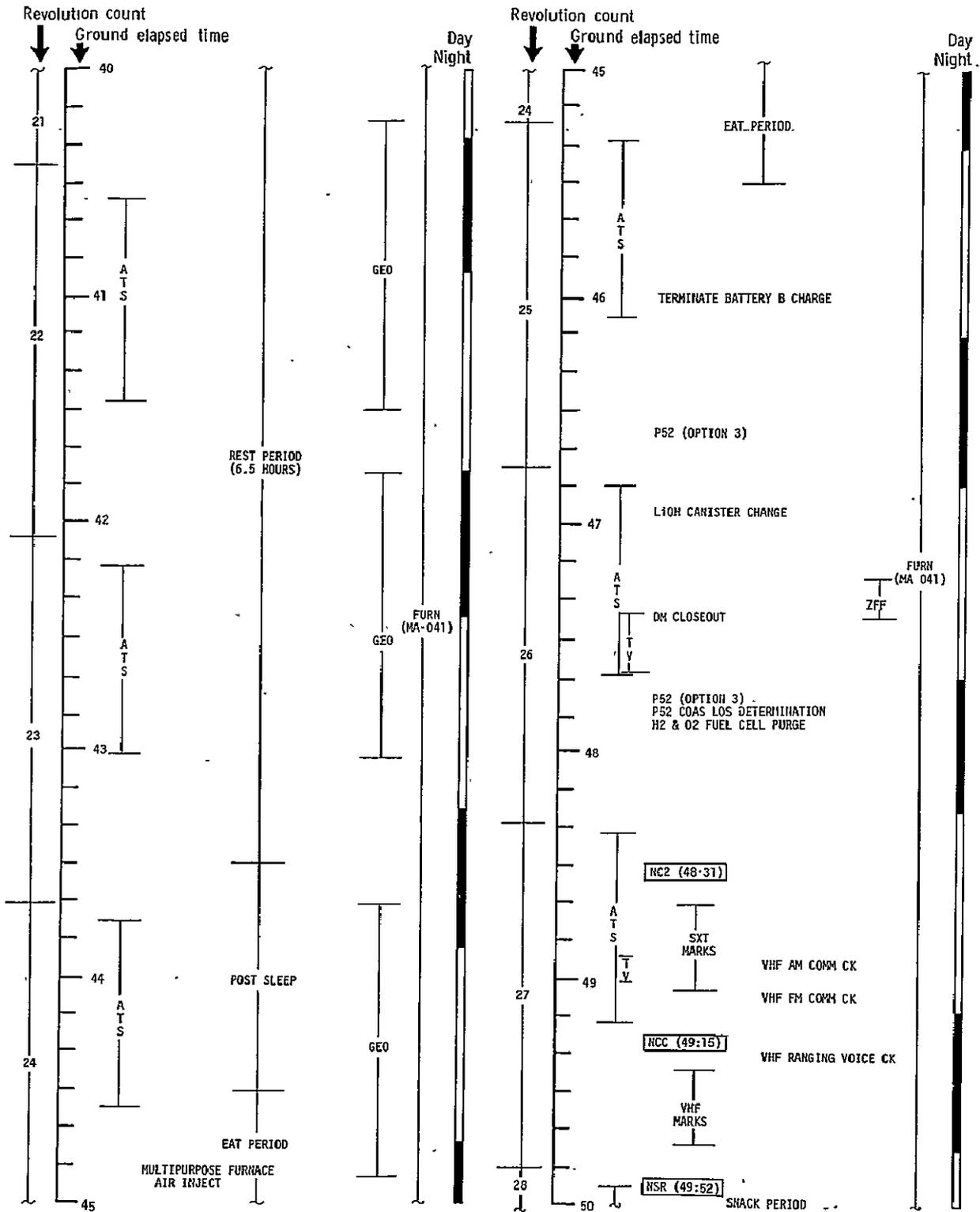
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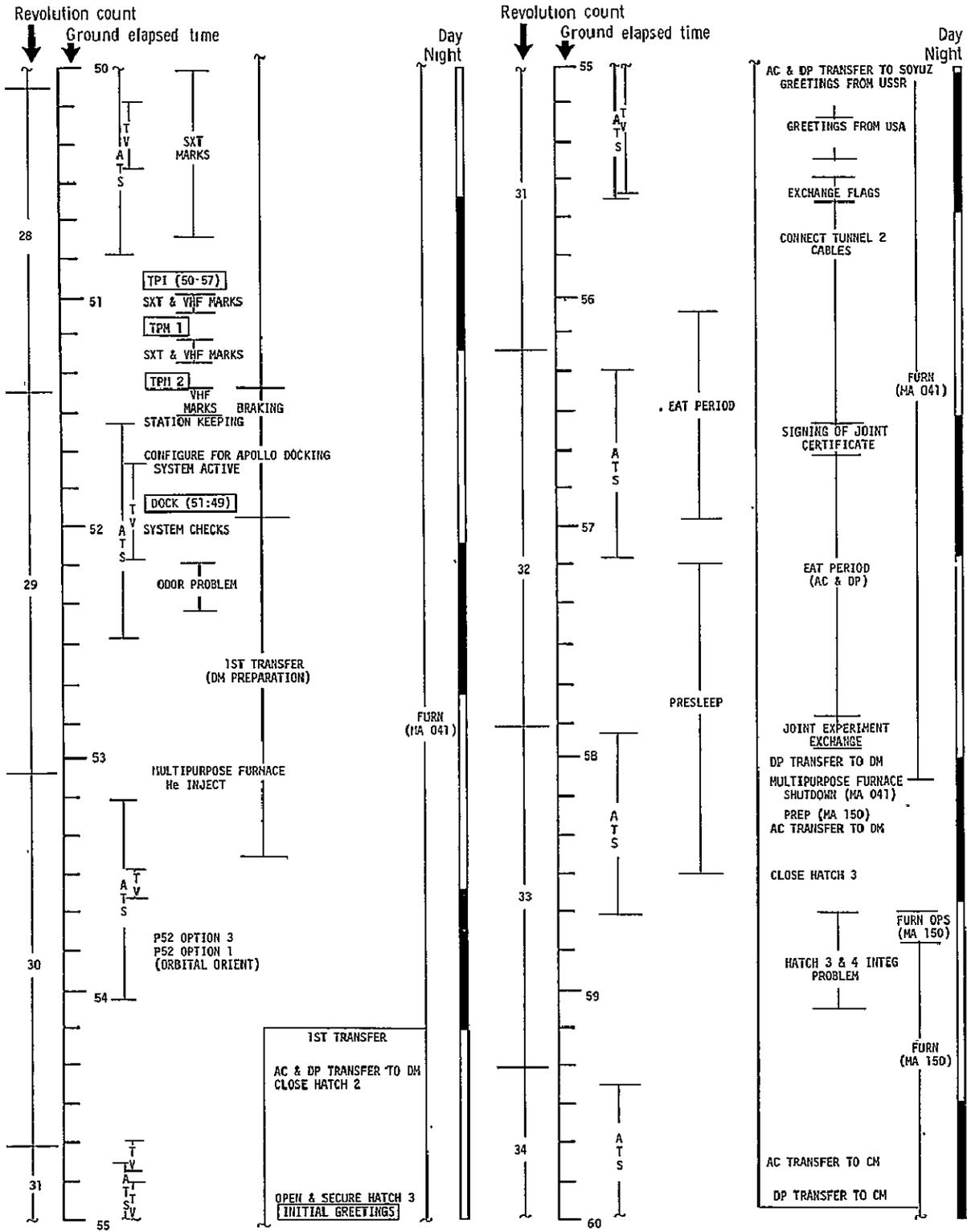
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN



ASTP SUMMARY AS FLOWN
FLIGHT PLAN (APOLLO)

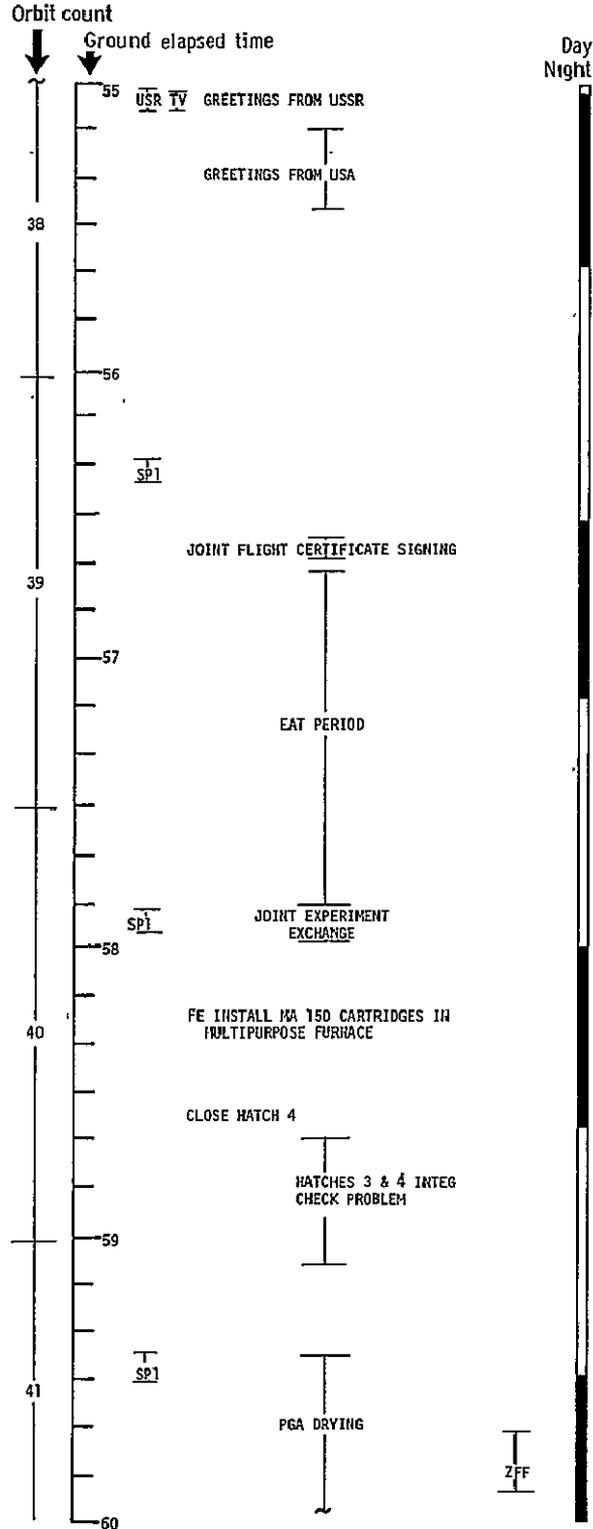
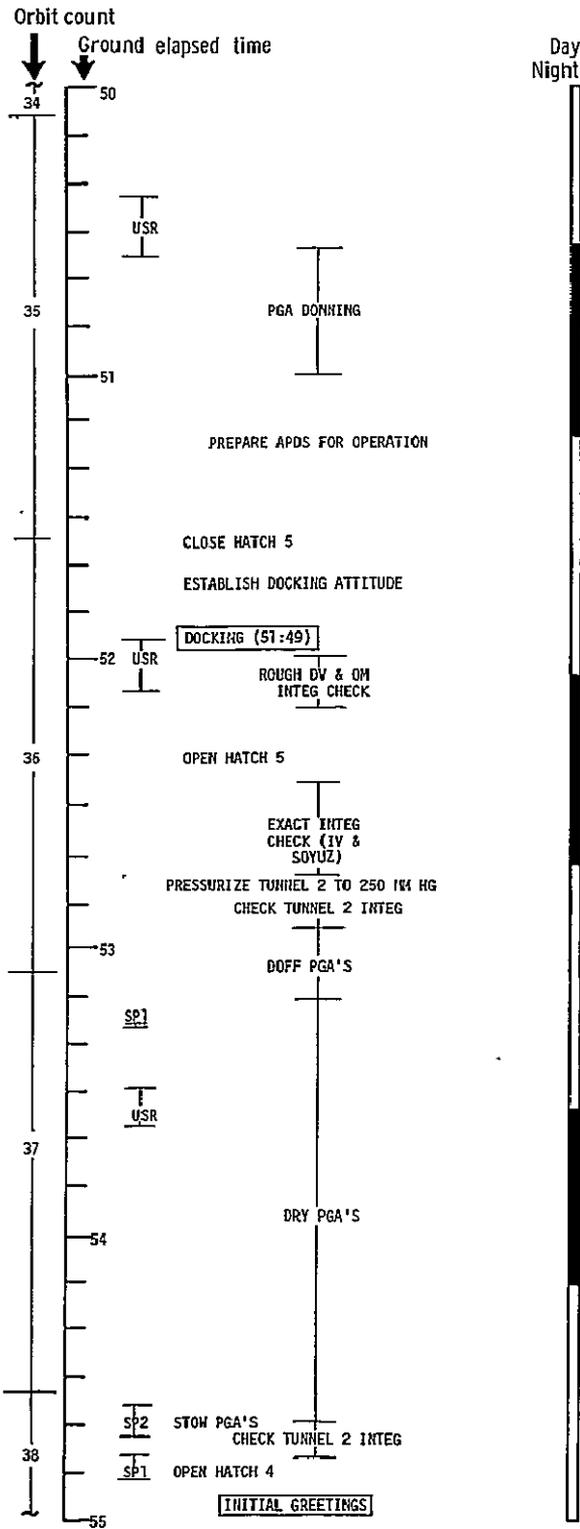
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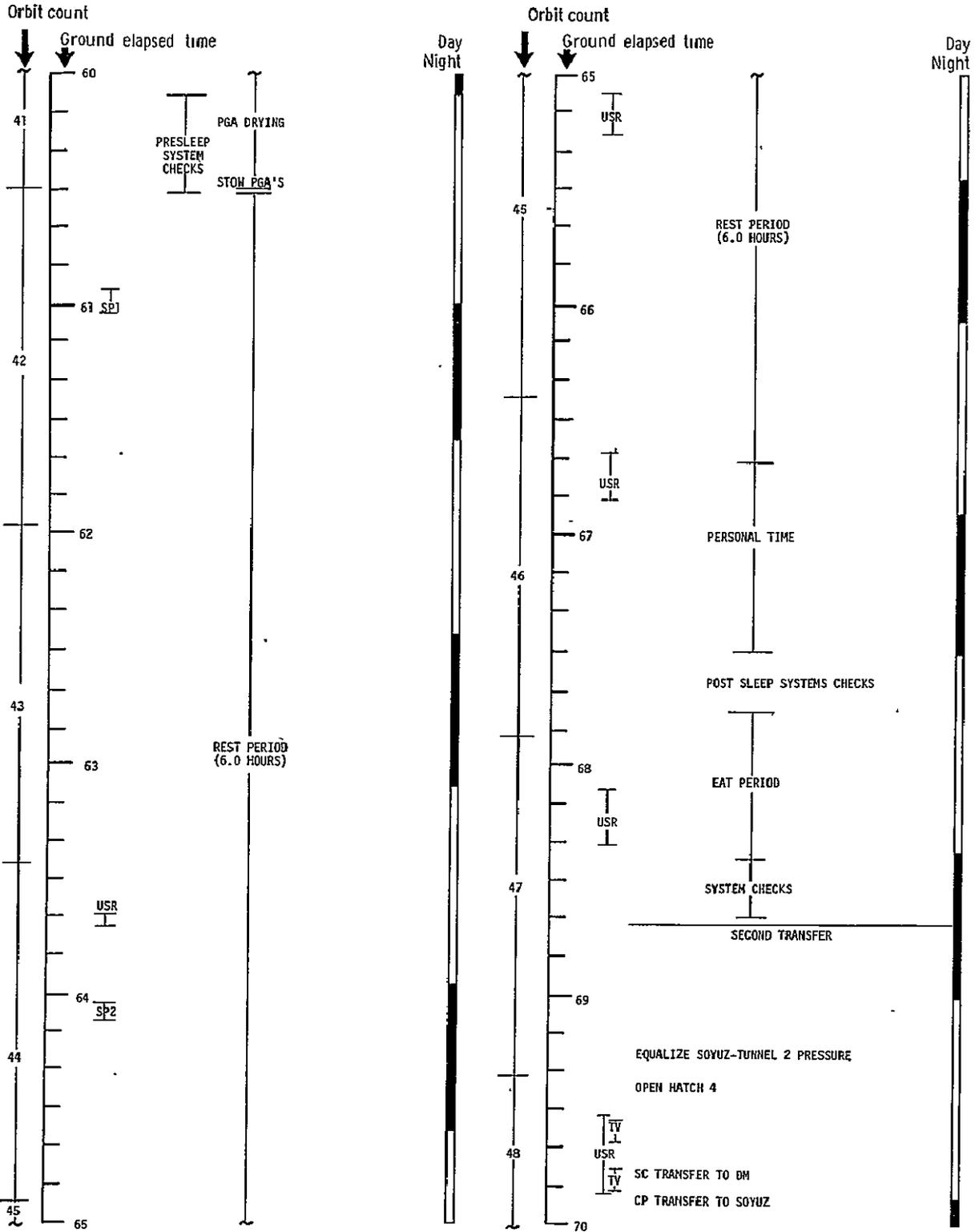
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN (SOYUZ)



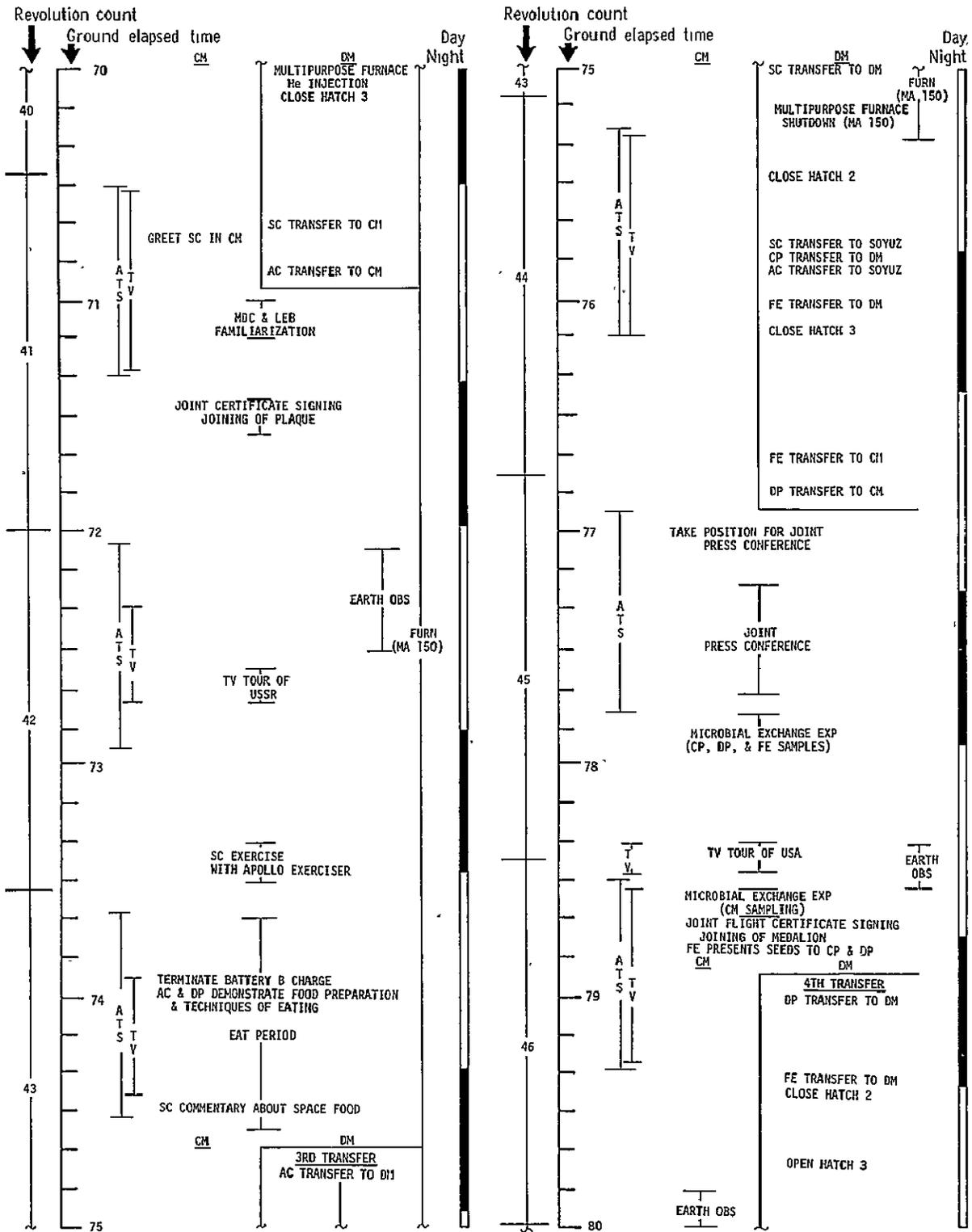
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN (SOYUZ)



ASTP SUMMARY AS FLOWN
FLIGHT PLAN (APOLLO)

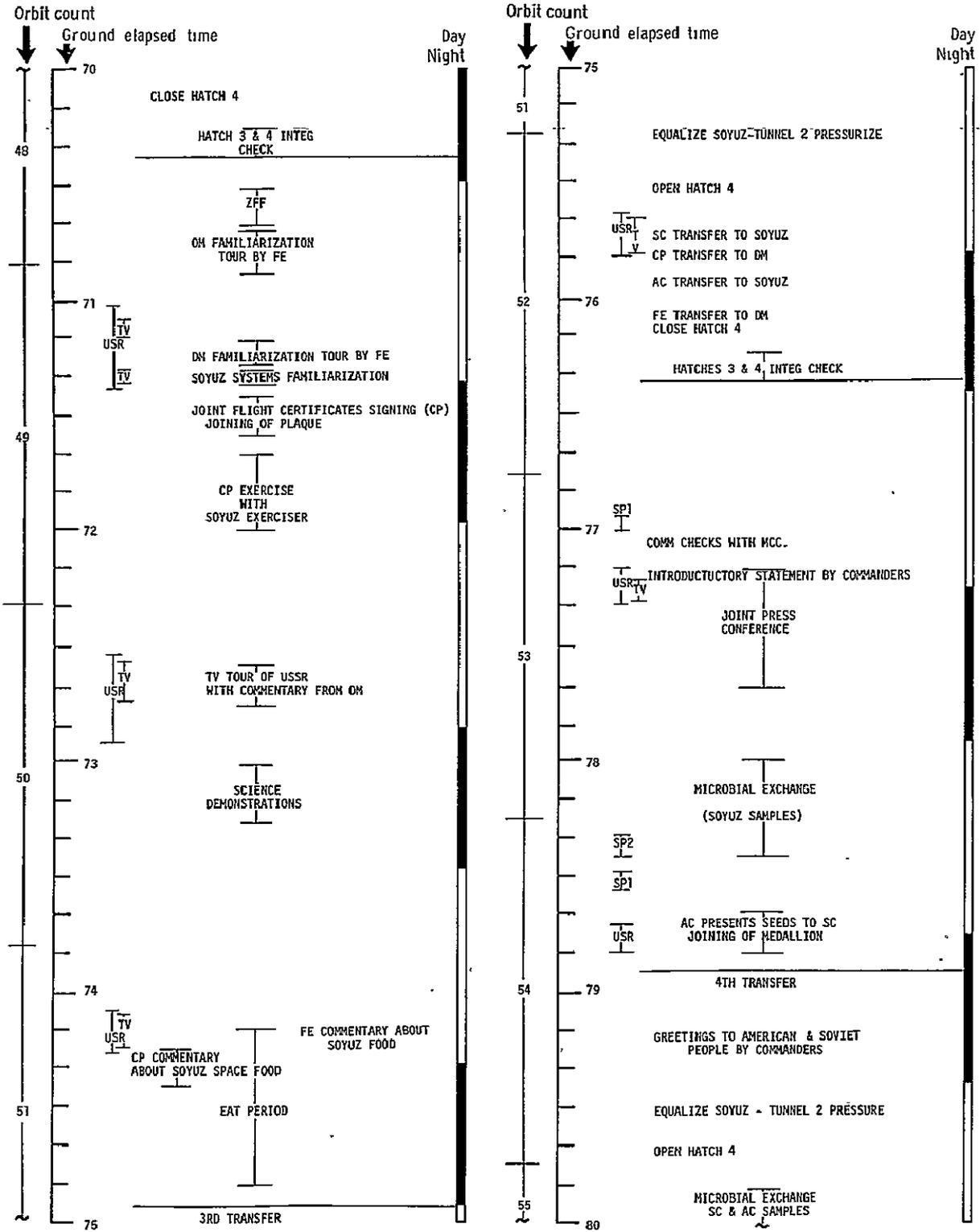
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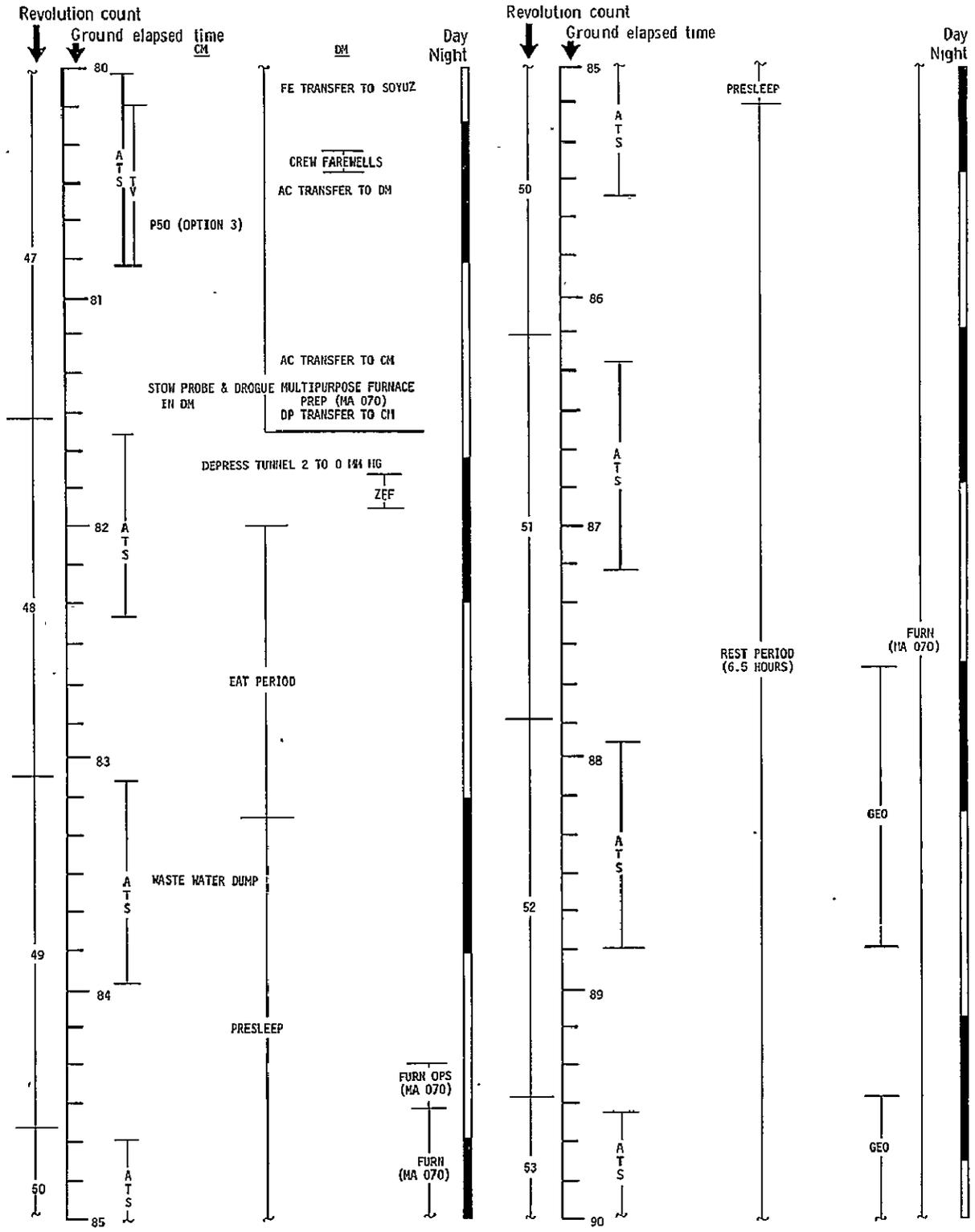
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FLIGHT PLAN (SOYUZ)

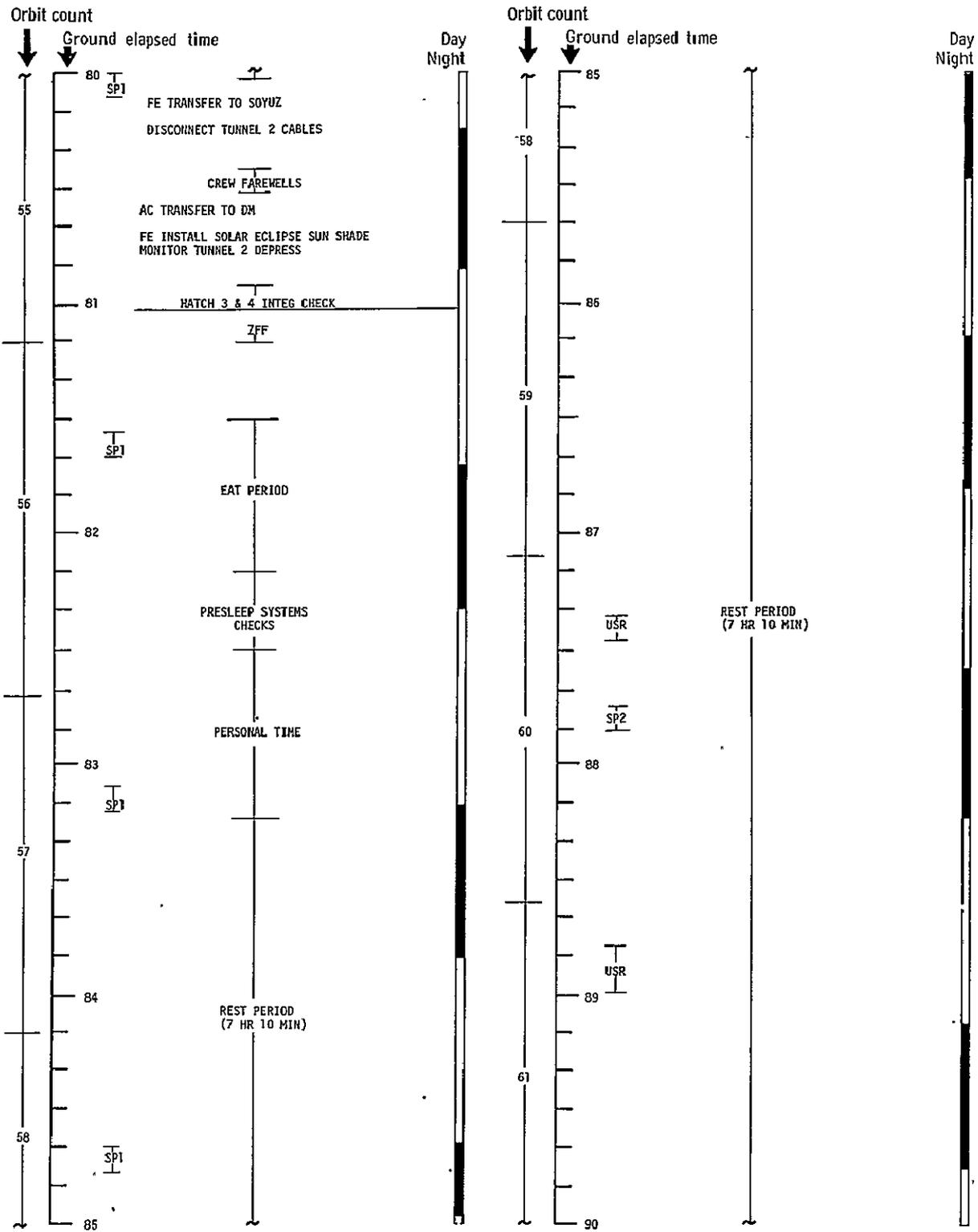


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FLIGHT PLAN (APOLLO)

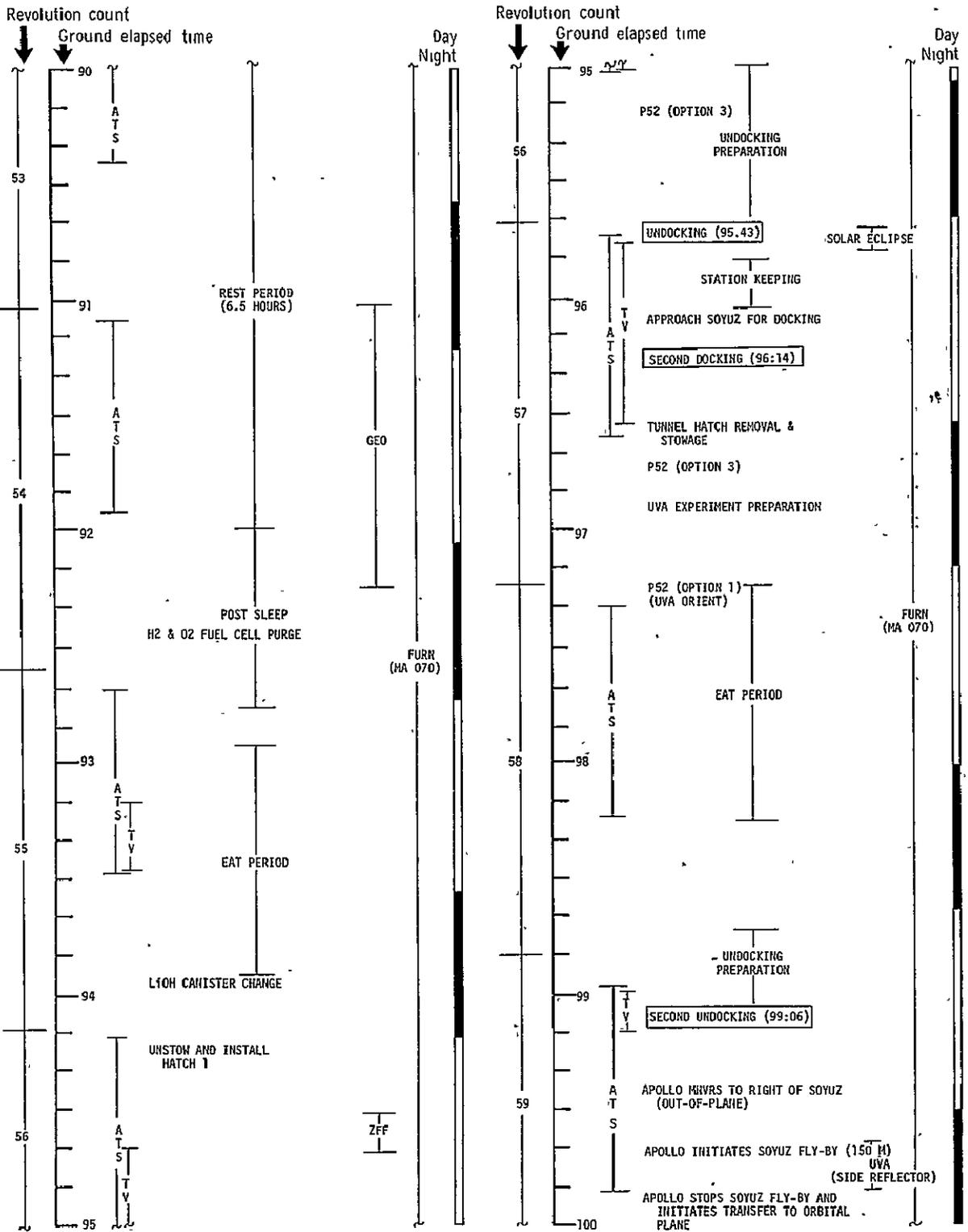


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FLIGHT PLAN (SOYUZ)



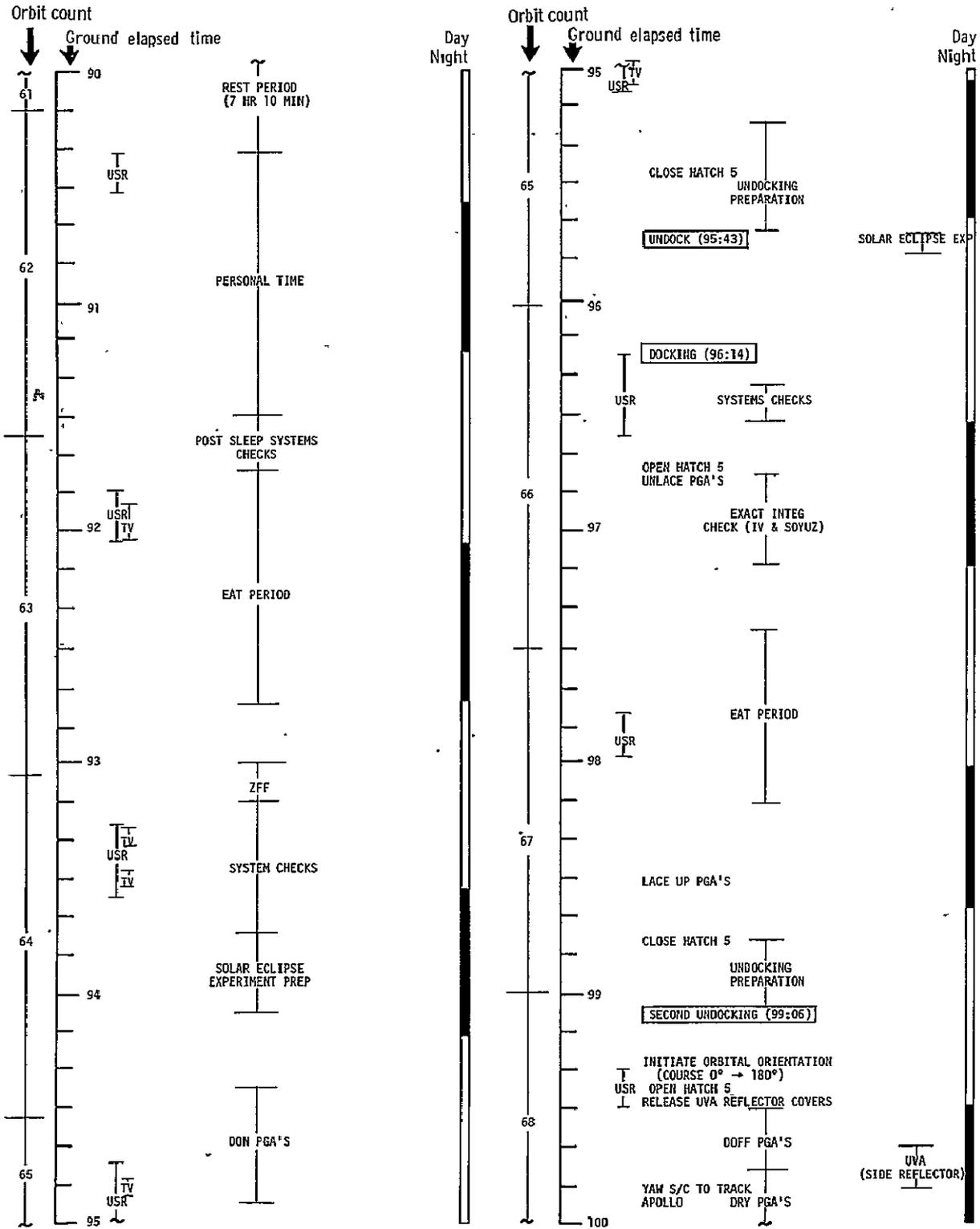
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FLIGHT PLAN (APOLLO)



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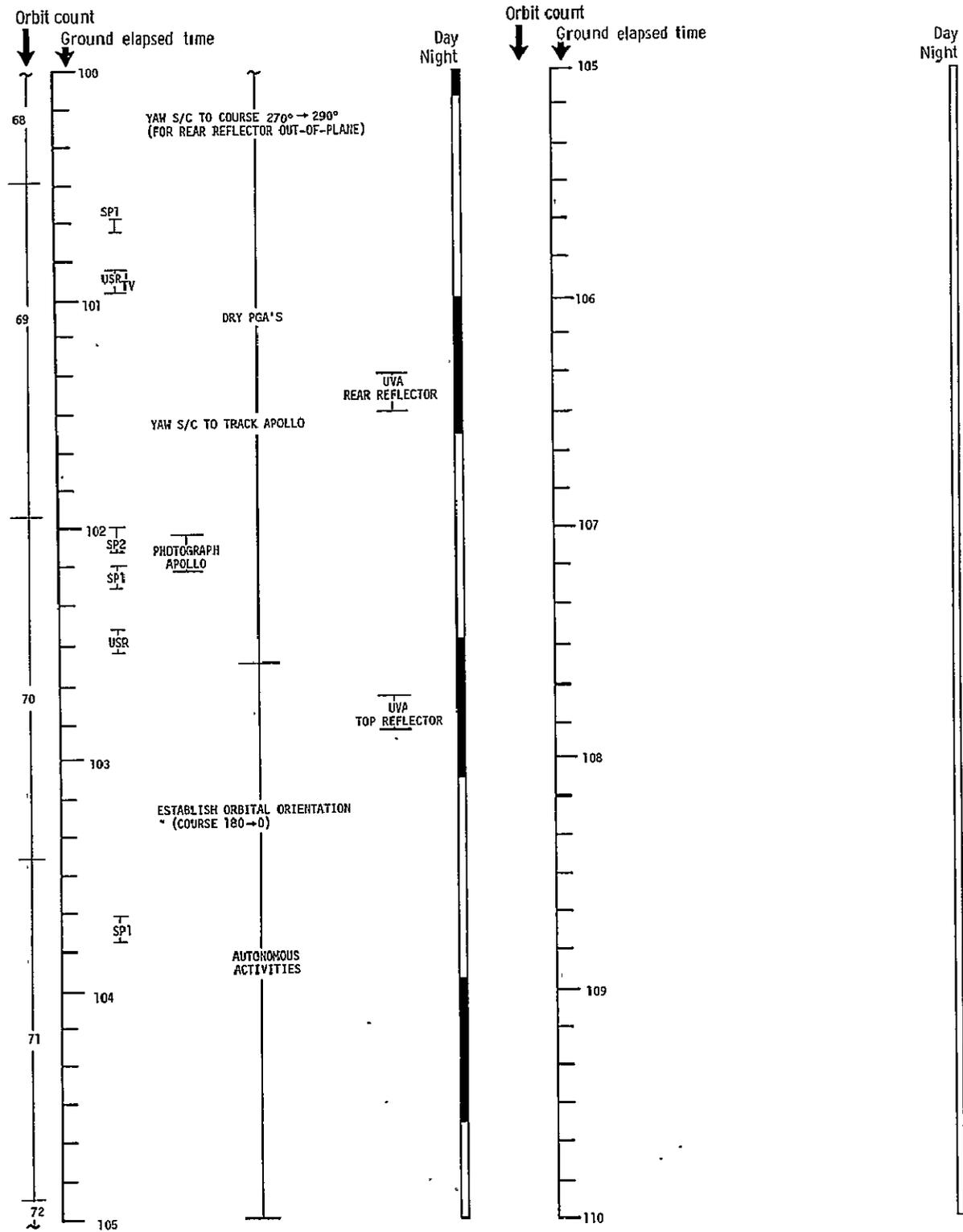
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN (SOYUZ)

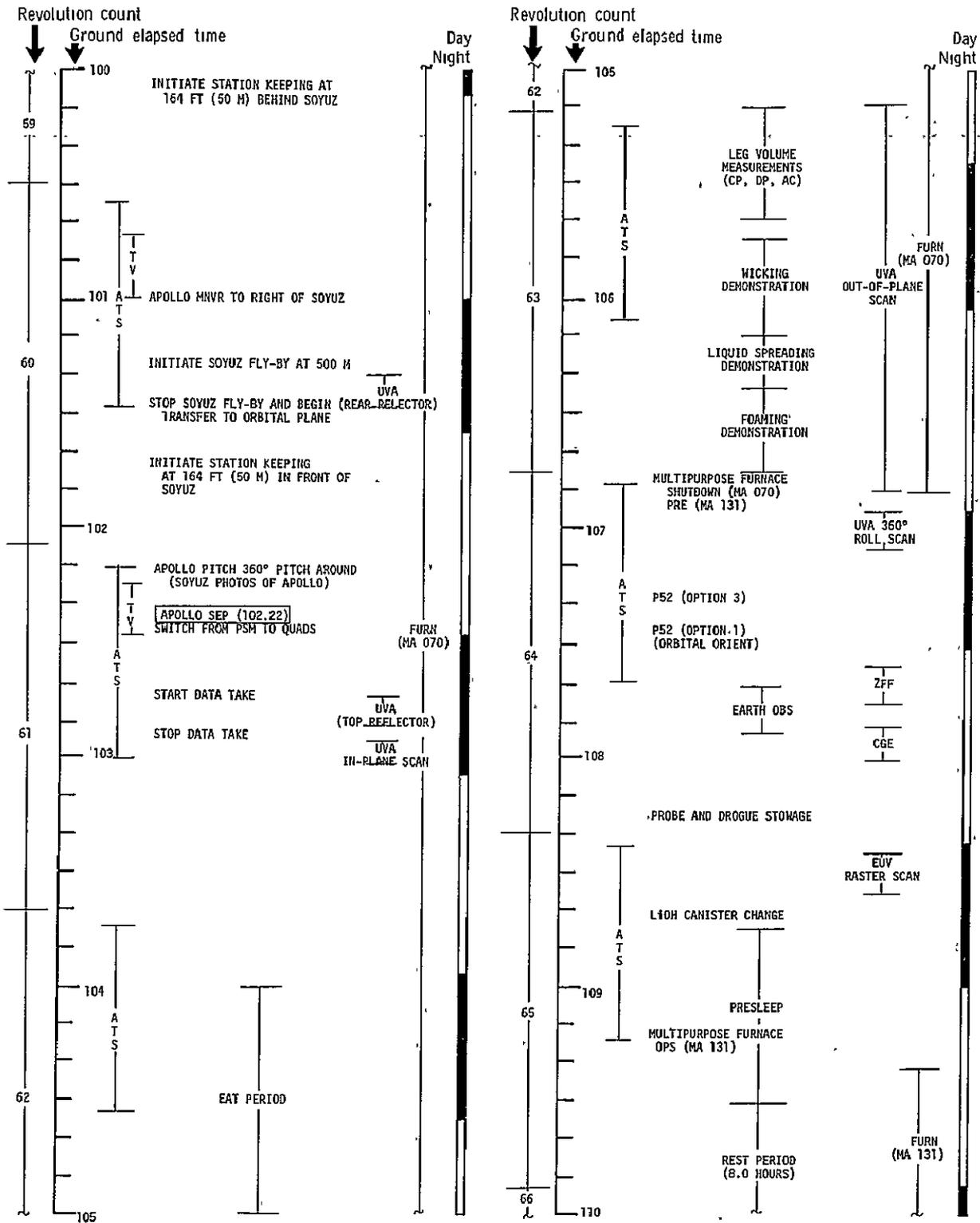


ASTP SUMMARY AS FLOWN
FLIGHT PLAN (SOYUZ)

B-17



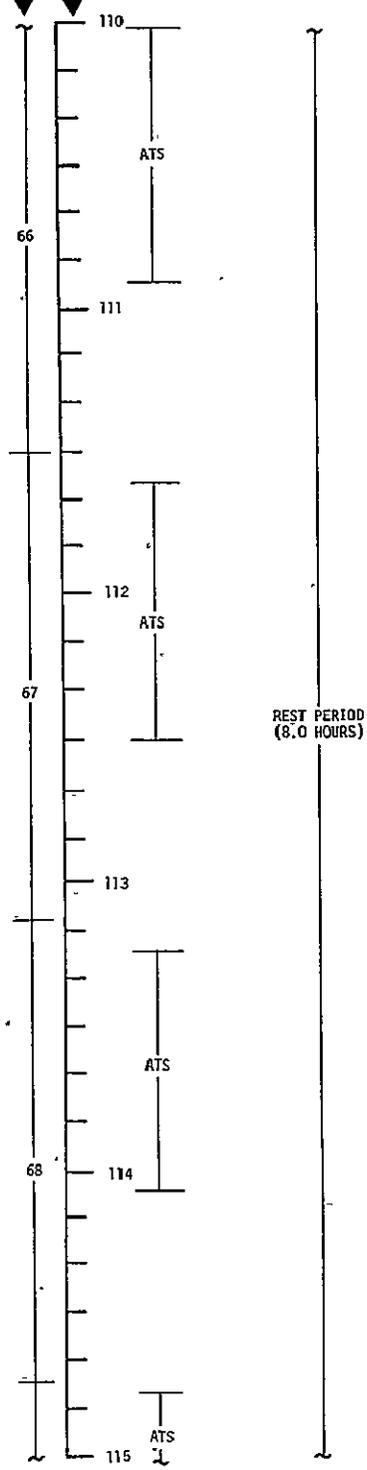
ASTP SUMMARY AS FLOWN
FLIGHT PLAN (APOLLO)



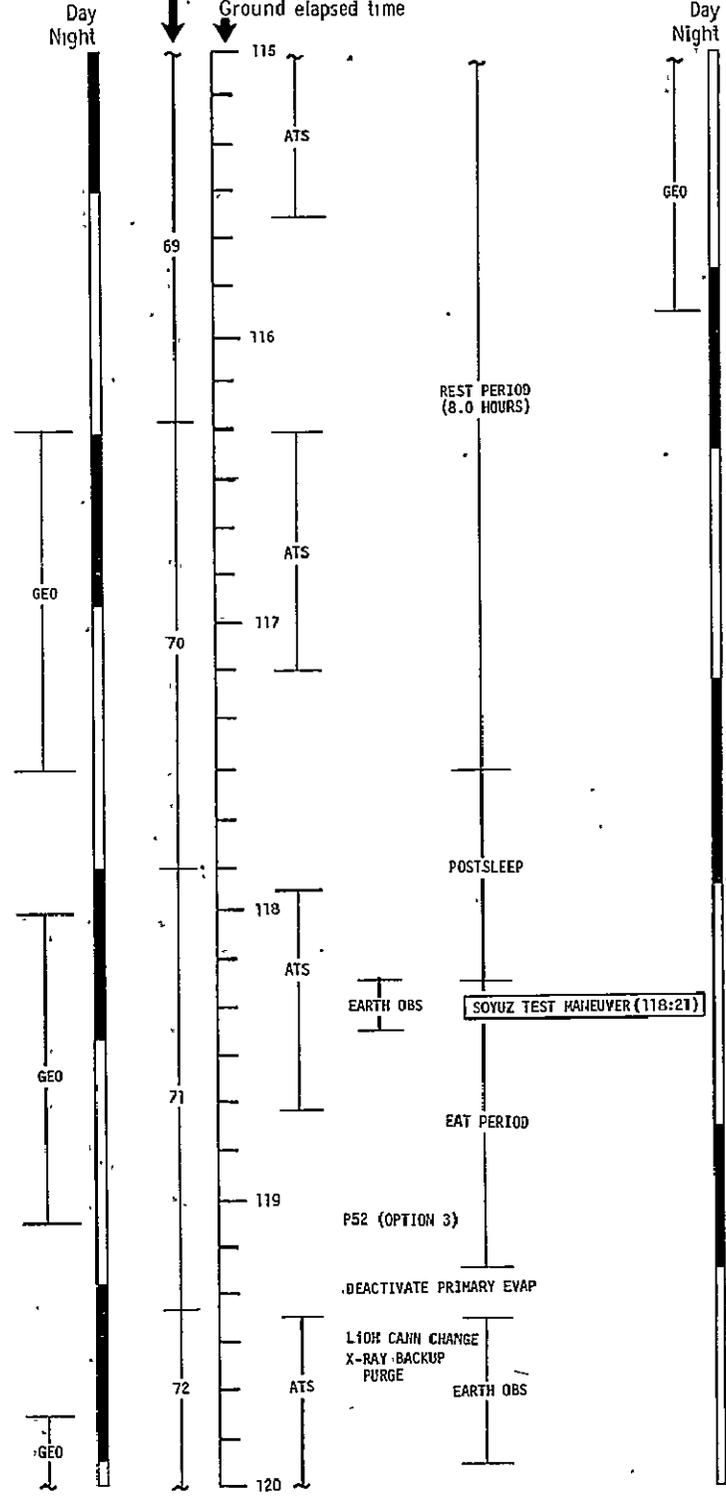
ASTP SUMMARY AS FLOWN
FLIGHT PLAN

B-19

Revolution count
Ground elapsed time



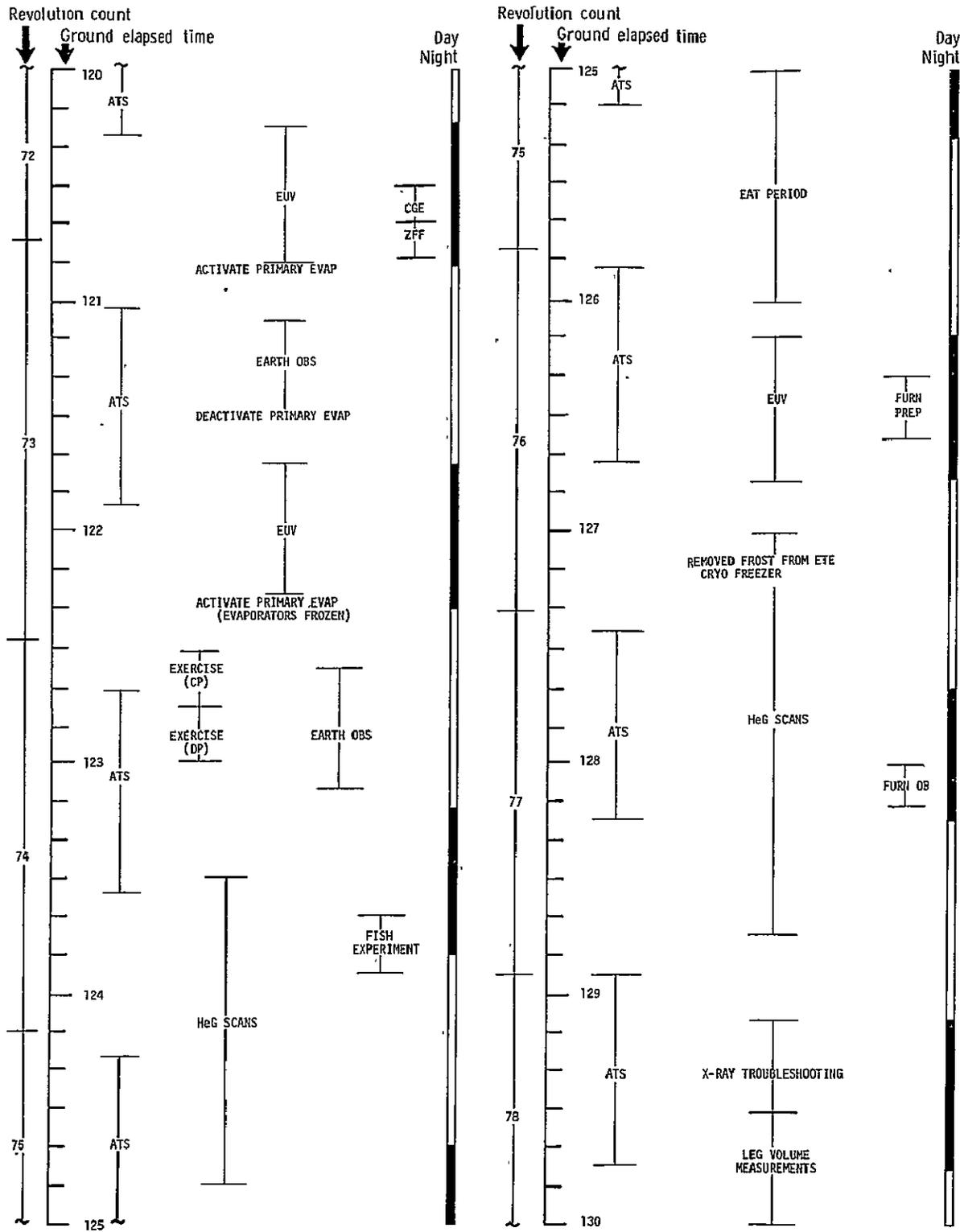
Revolution count
Ground elapsed time



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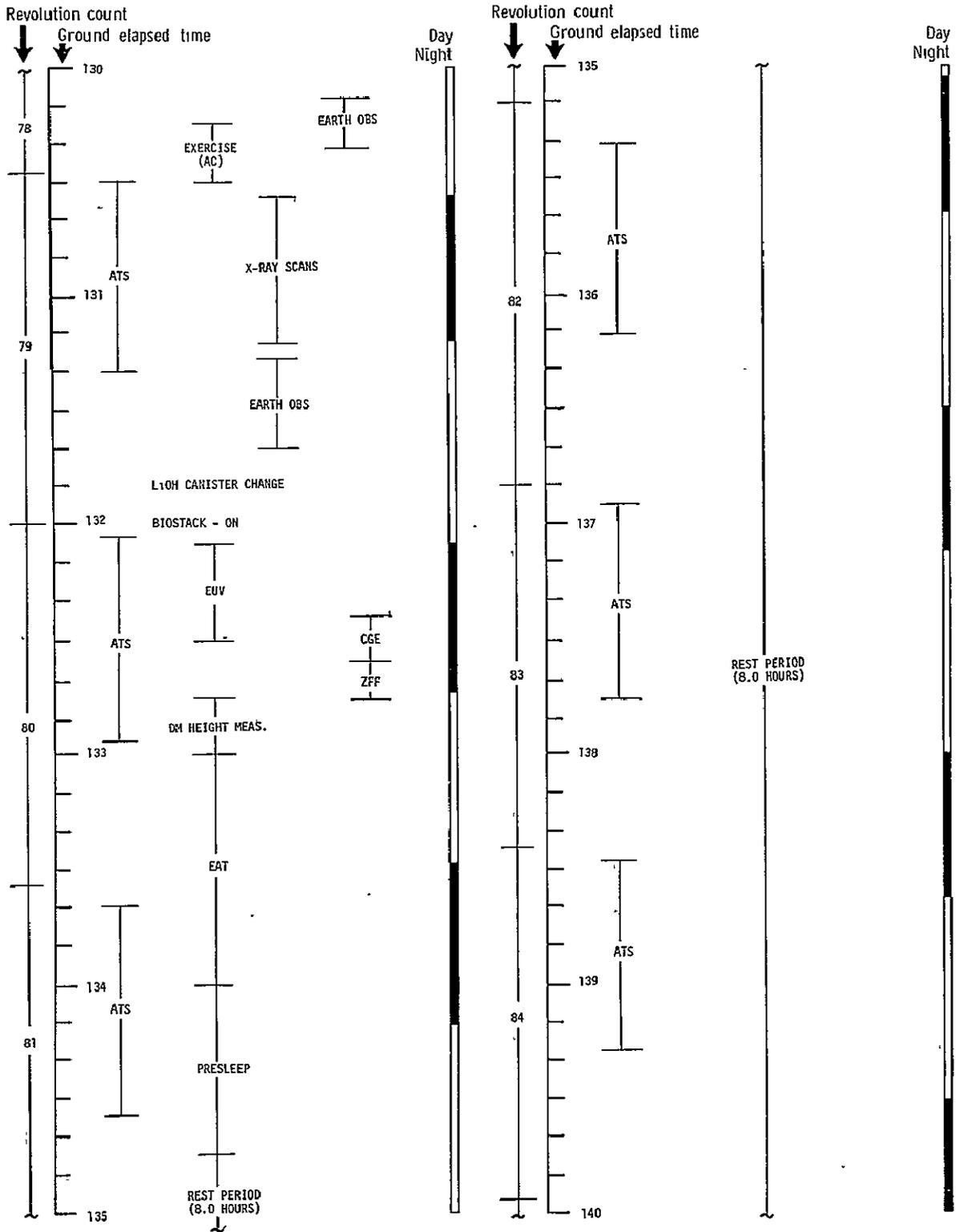
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ASTP SUMMARY AS FLOWN
FLIGHT PLAN

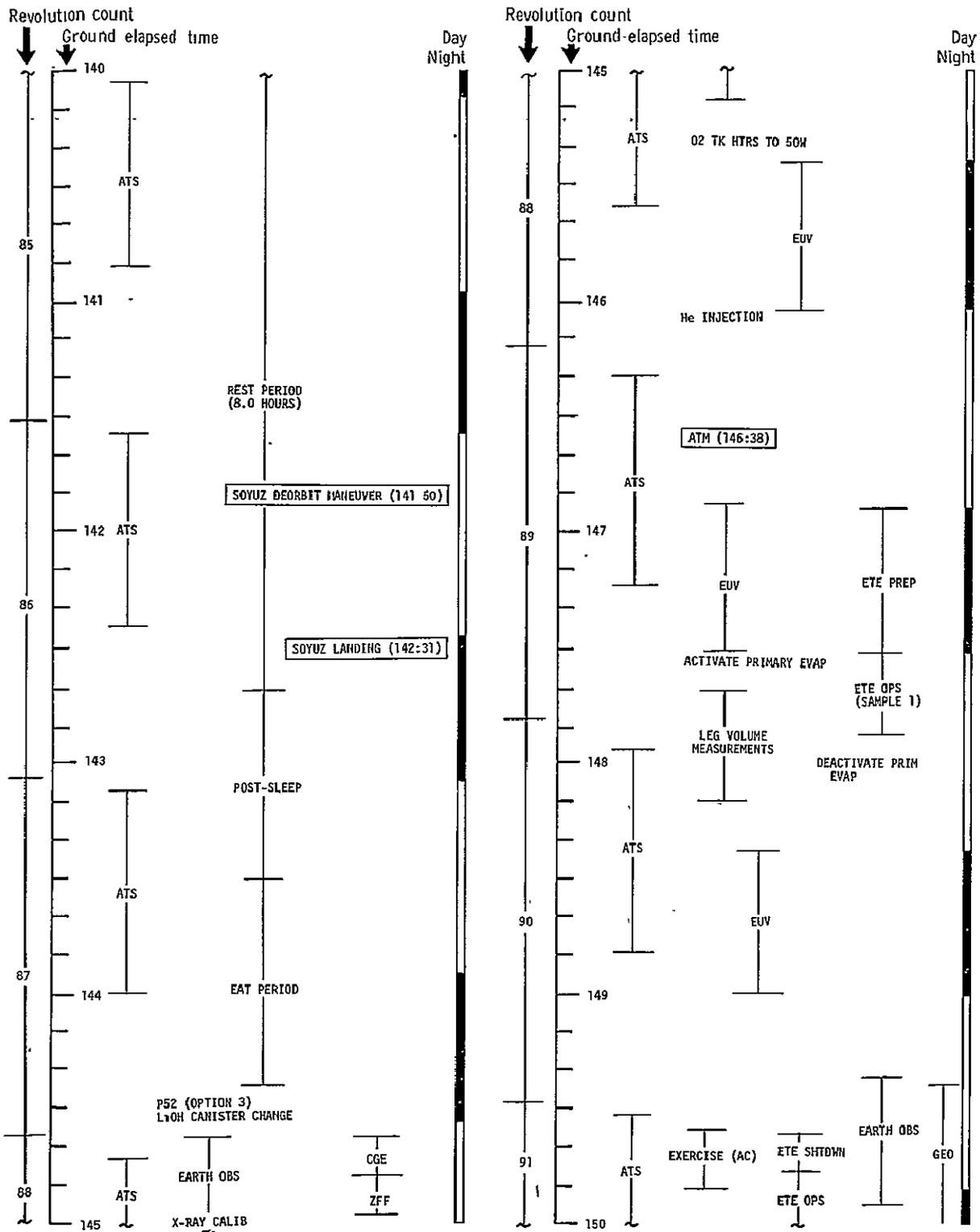


ASTP SUMMARY AS FLOWN
FLIGHT PLAN

B-21

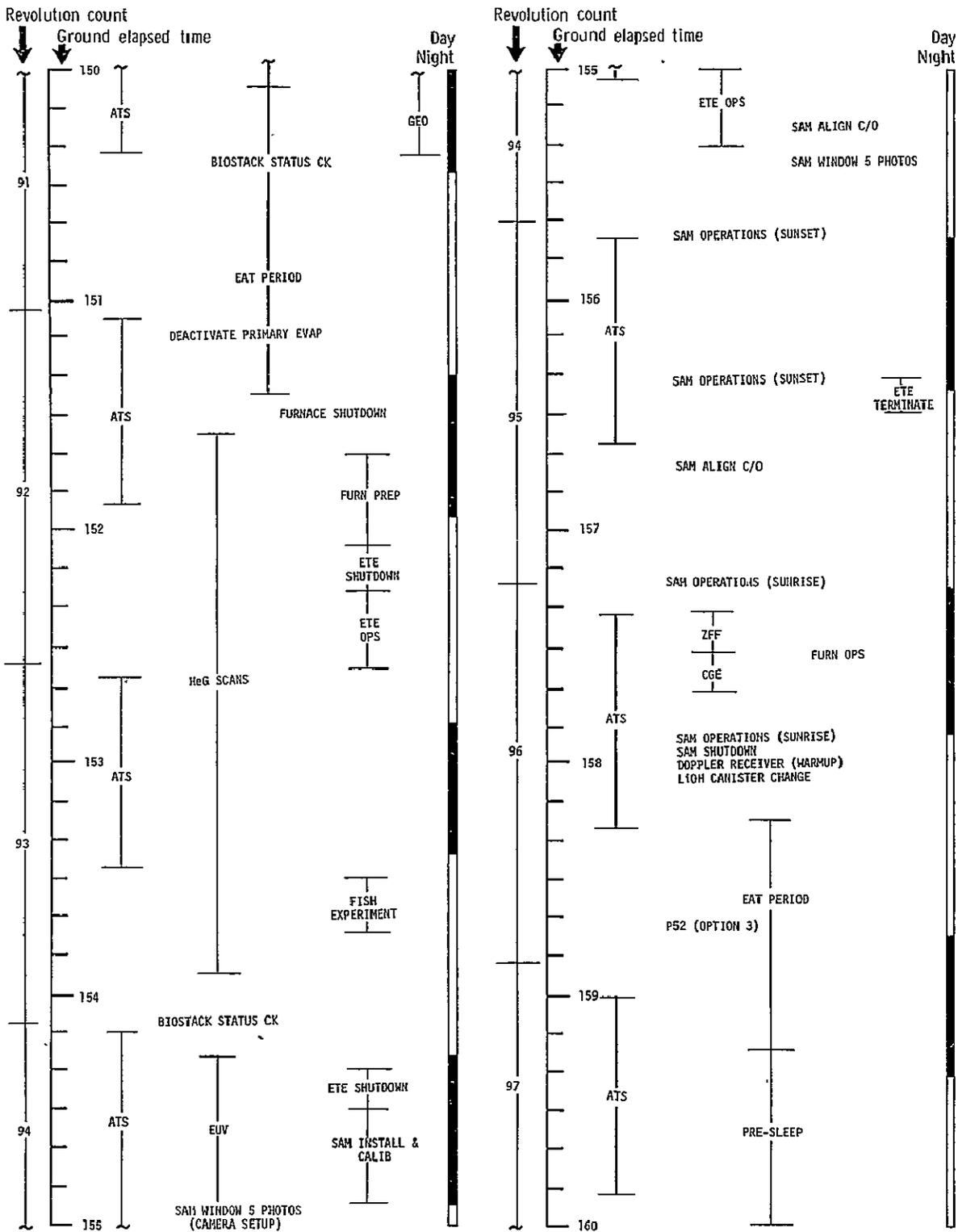


ASTP SUMMARY AS FLOWN
FLIGHT PLAN



ASTP SUMMARY AS FLOWN
FLIGHT PLAN

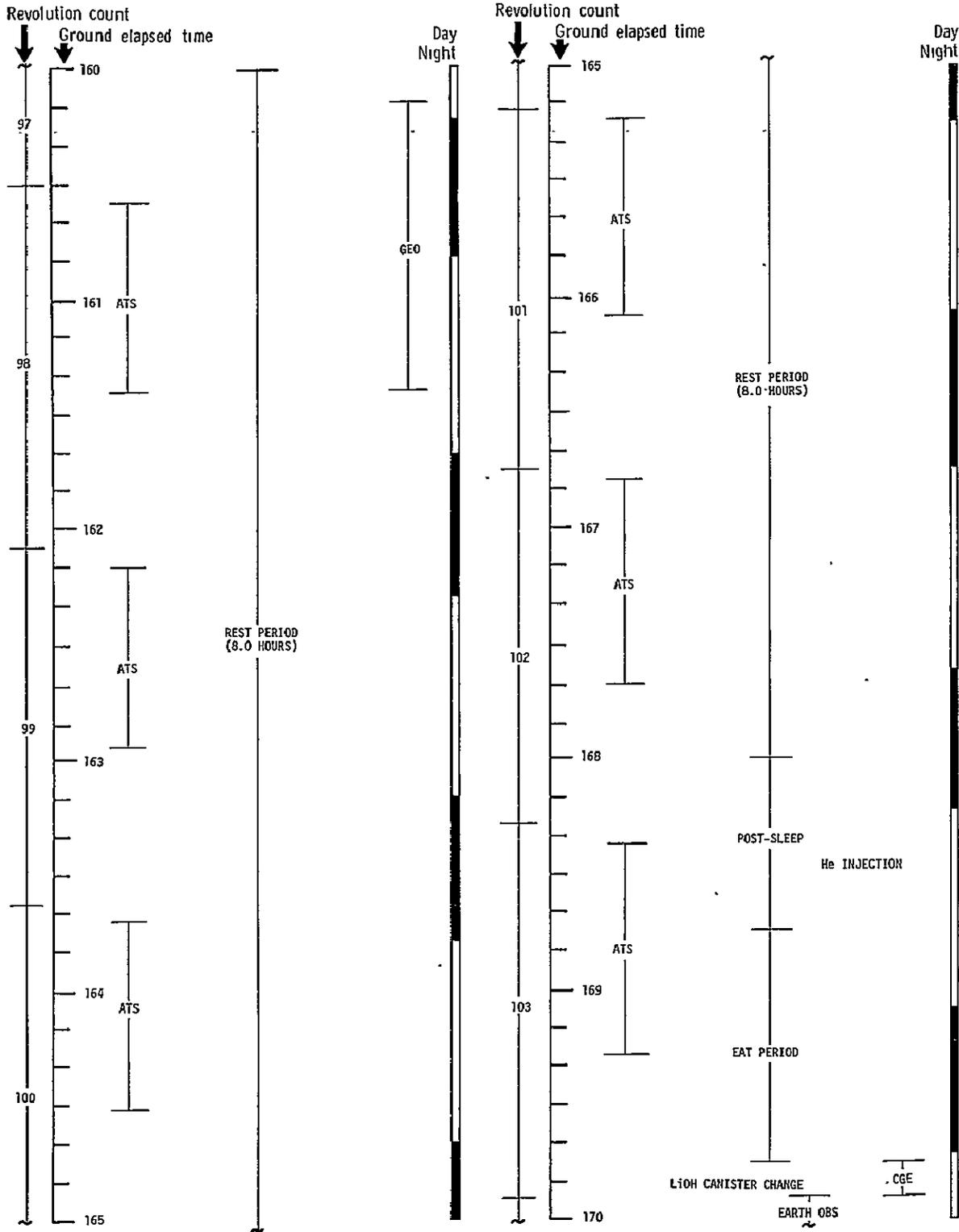
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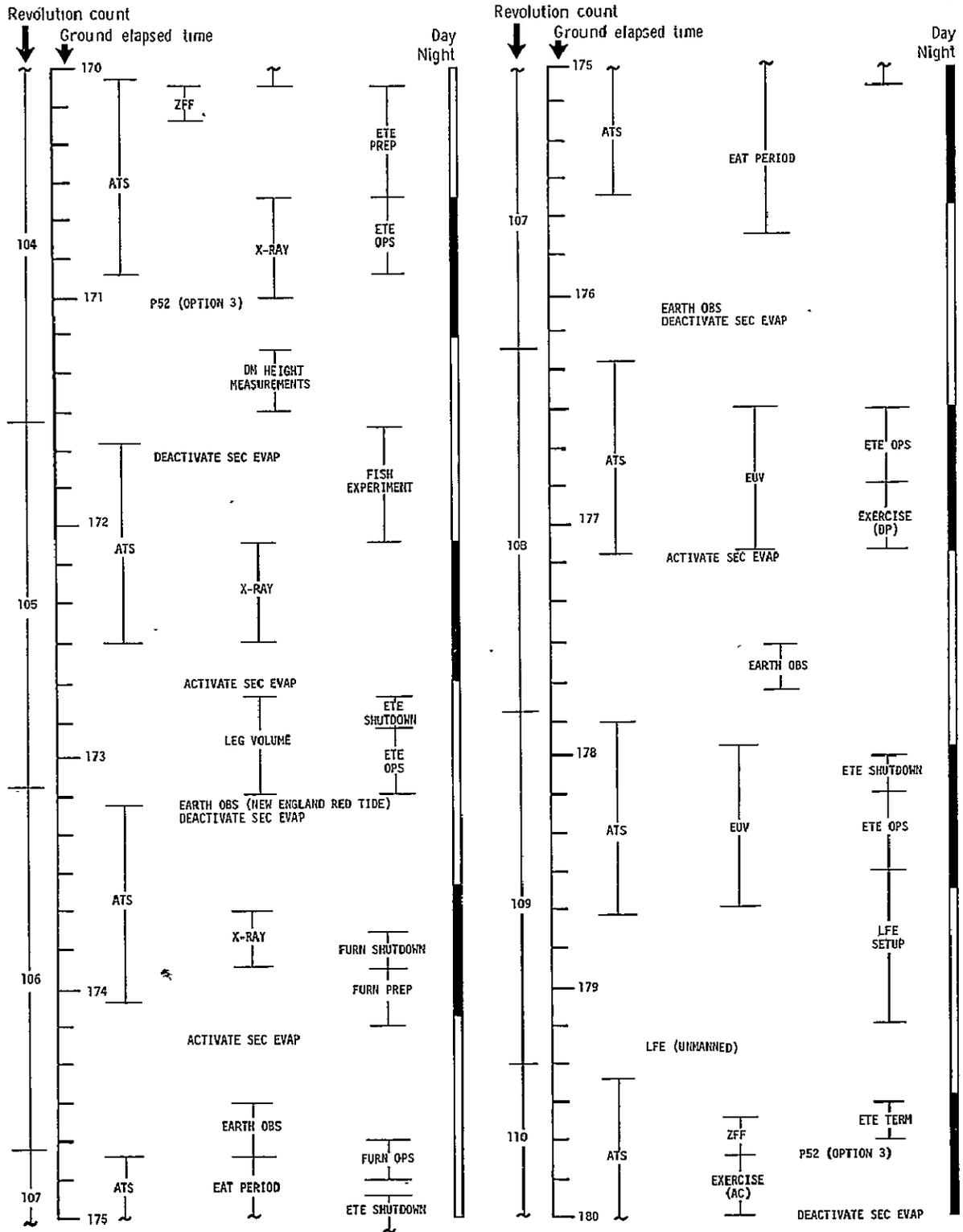
B-24

ASTP SUMMARY AS FLOWN FLIGHT PLAN

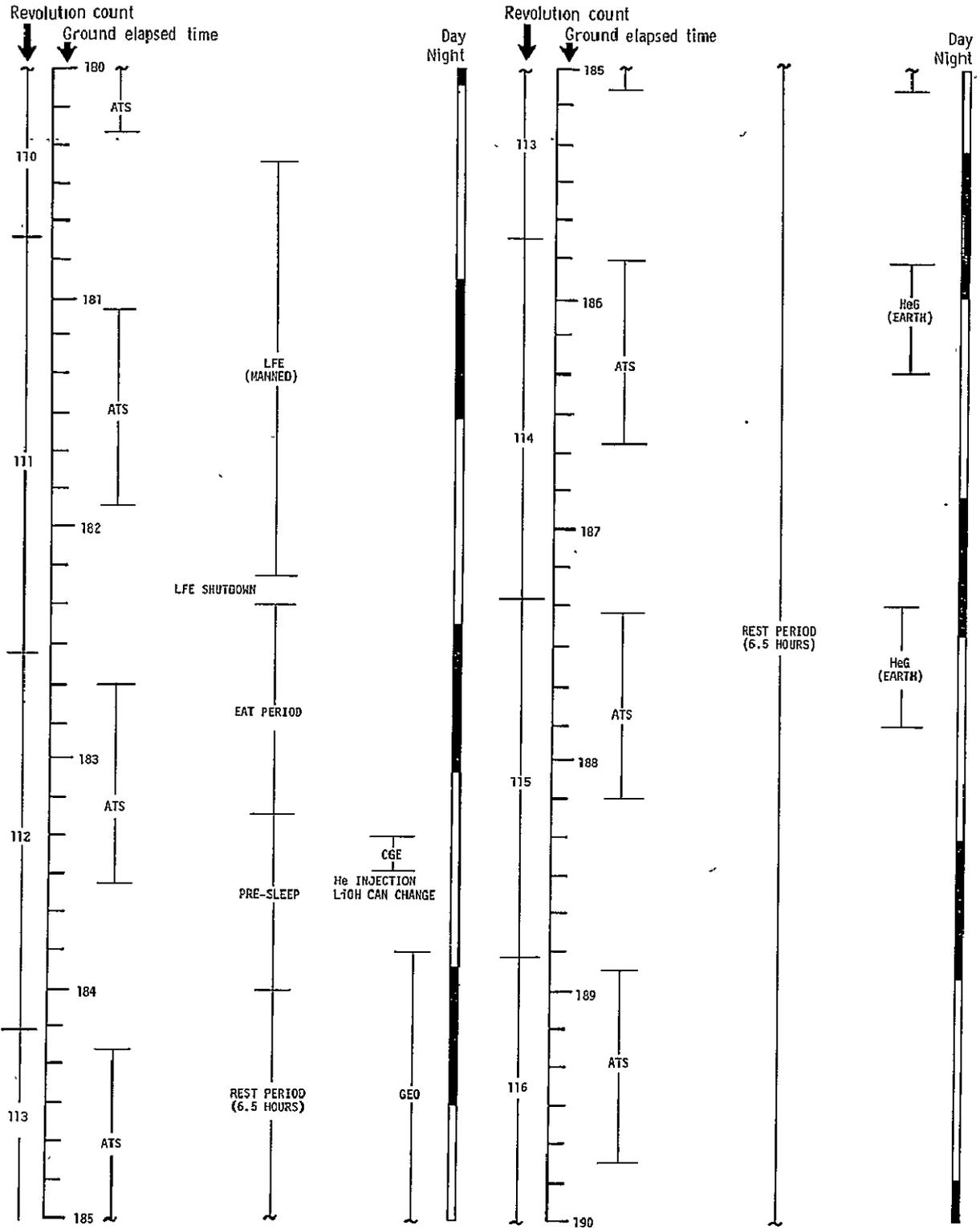


ASTP SUMMARY AS FLOWN
FLIGHT PLAN

B-25

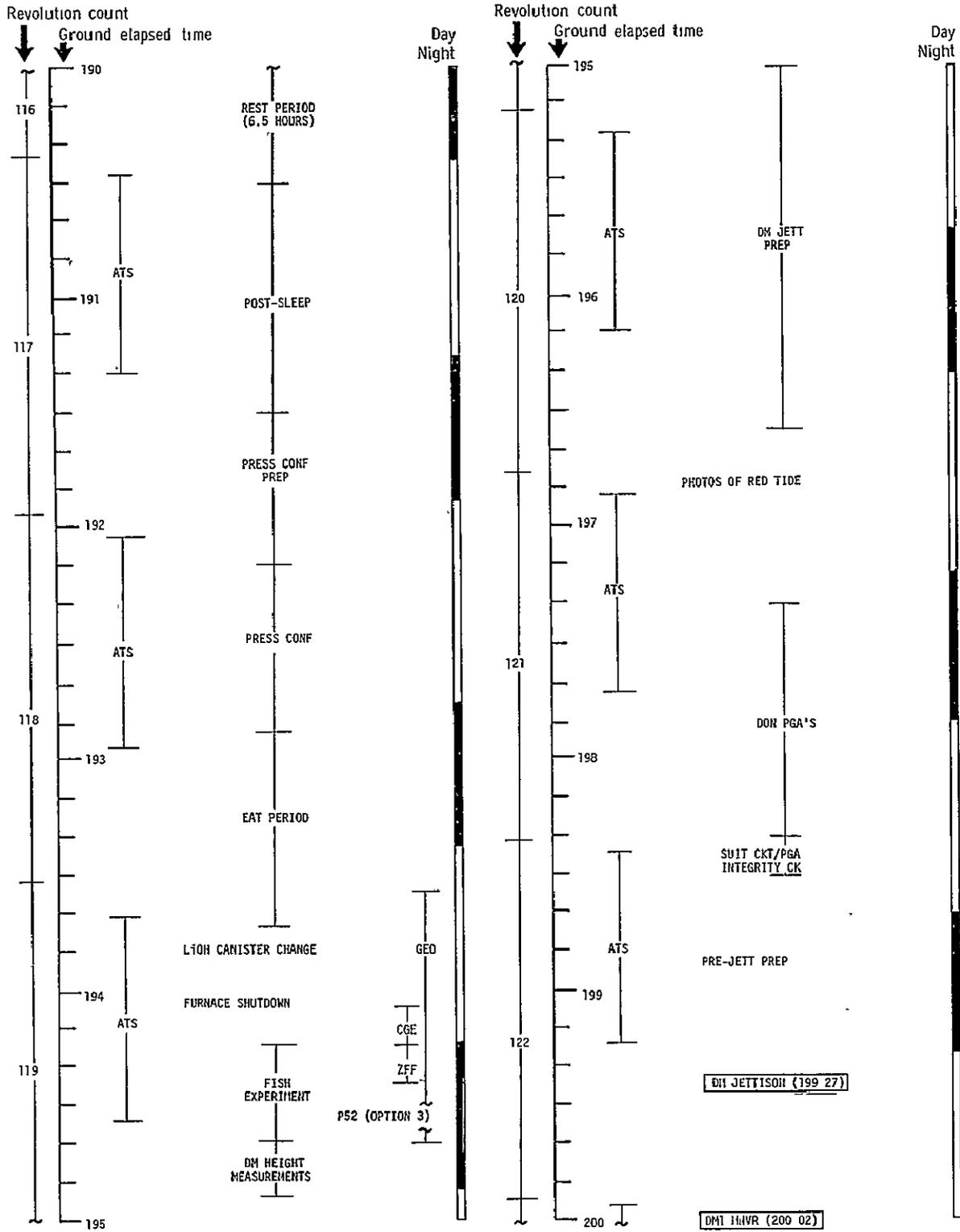


ASTP SUMMARY AS FLOWN
FLIGHT PLAN



ASTP SUMMARY AS FLOWN
FLIGHT PLAN

B-27



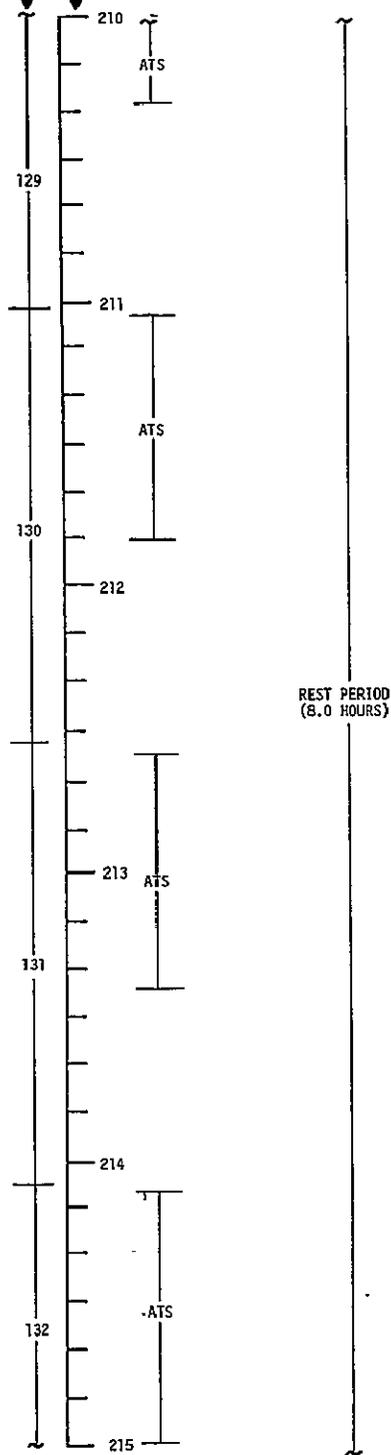
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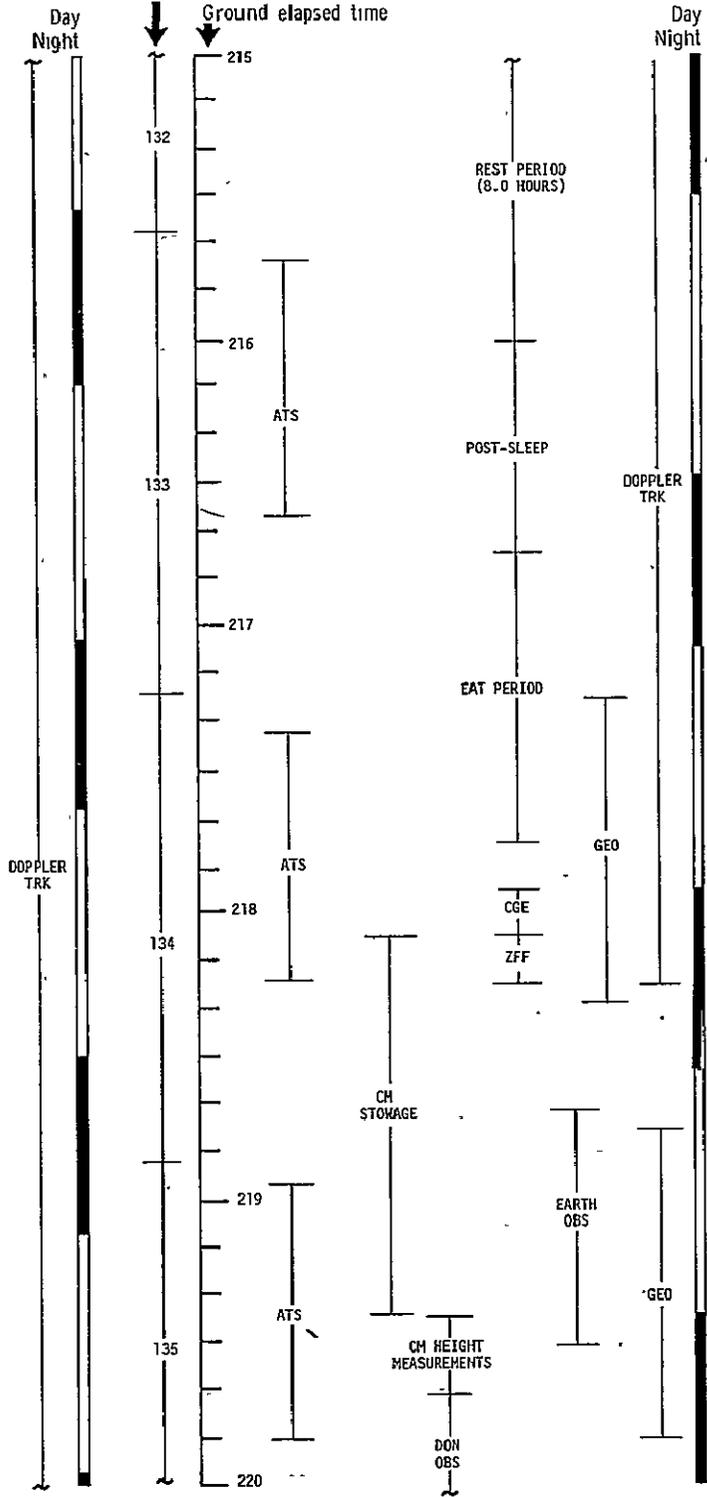
ASTP SUMMARY AS FLOWN
FLIGHT PLAN

B-29

Revolution count
Ground elapsed time



Revolution count
Ground elapsed time



B-30

ASTP SUMMARY AS FLOWN
FLIGHT PLAN

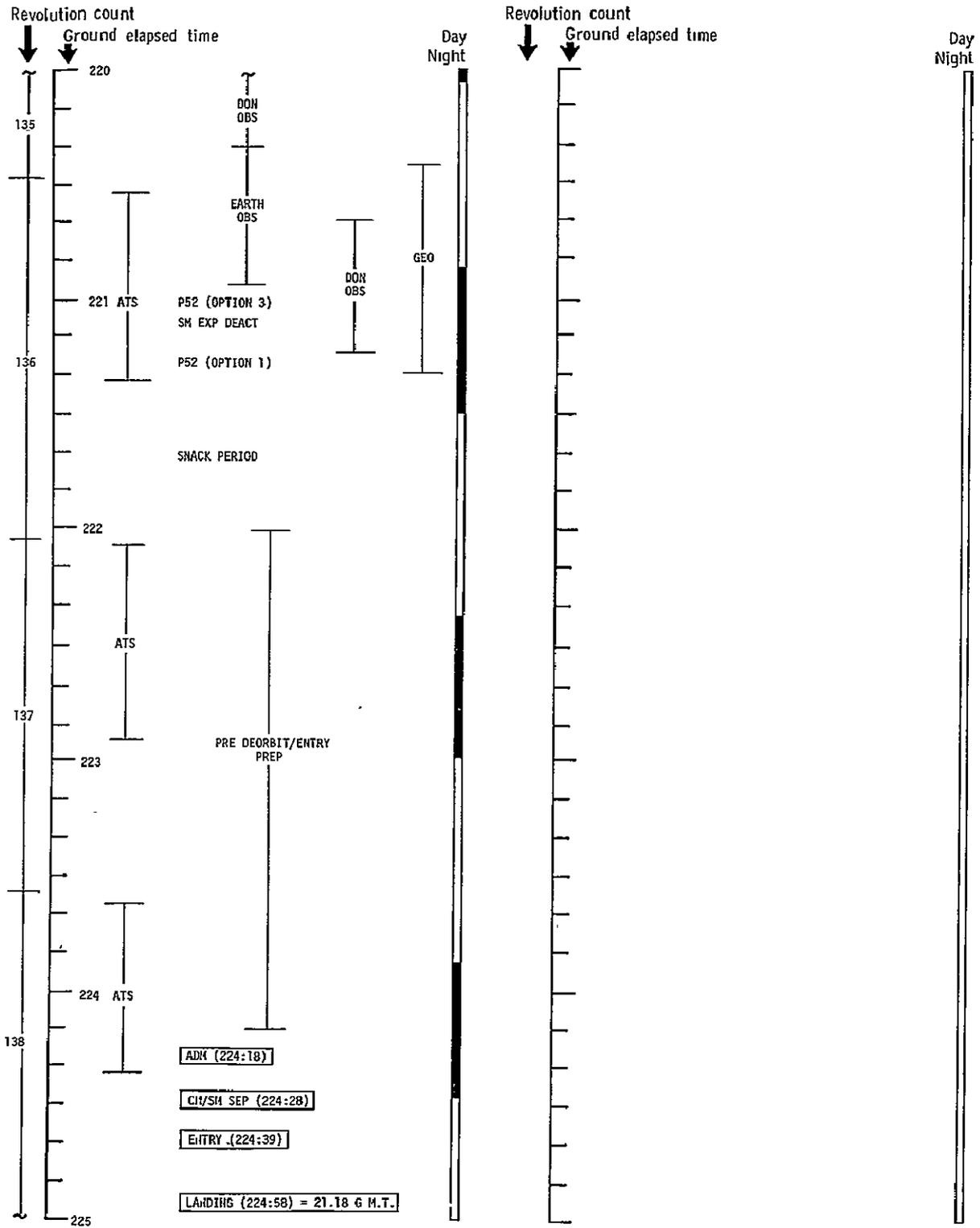


TABLE B-I.- ACRONYMS AND ABBREVIATIONS
USED IN AS-FLOWN FLIGHT PLAN

<u>Acronym/abbreviation</u>	<u>Definition</u>
AC	Apollo Commander
ACM	Apollo circularization maneuver
ADM	Apollo deorbit maneuver
AEM	Apollo evasive maneuver
APDS	Androgynous peripheral docking system
ASTP	Apollo Soyuz Test Project
ATM	Apollo trim maneuver
ATS	Applications Technology Satellite
ATT	Attitude
BIOS	Biostack Experiment (MA-017)
CAL	Calibration
CGE	Crystal Growth Experiment (MA-028)
CK	Check
CKT	Circuit
CM	Command module
C/O	Checkout
COAS	Crew optical alignment sight
CP	Command Module Pilot
CSM	Command and service modules
DM	Docking module (Apollo)
DM1	1st shaping maneuver for doppler tracking experiment
DM2	2nd shaping maneuver for doppler tracking experiment
DP	Docking Module Pilot
DV	Descent vehicle (Soyuz)
EPE	Electrophoresis, German Experiment (MA-014)
ETE	Electrophoresis Technology, USA Experiment (MA-011)
EUV	Extreme Ultraviolet Telescope Experiment (MA-083)
EXP	Experiment

TABLE B-I.- ACRONYMS AND ABBREVIATIONS
USED IN AS-FLOWN FLIGHT PLAN - Continued

<u>Acronym/abbreviation</u>	<u>Definition</u>
FE	Flight Engineer (Soyuz)
FURN	Multipurpose Furnace Experiment (MA-010)
FWD	Forward
GEO	Geodynamics Experiment (MA-128)
H2	Hydrogen
HeG	Helium Glow Experiment (MA-088)
HGA	High-gain antenna
IV	Interface volume
LEB	Lower equipment bay
LFE	Light Flash Experiment (MA-106)
LiOH	Lithium hydroxide
MCC	Mission Control Center
MDC	Main display console
MNVR	Orientation maneuver
NC1	First phasing maneuver
NC2	Second phasing maneuver
NCC	Corrective combination maneuver.
NSR	Coelliptic maneuver
O2	Oxygen
OM	Orbital module (Soyuz)
P20	Command module computer program - universal tracking
P52	Command module computer program - inertial measurement unit realignment
PGA	Pressure garment assembly
PSM	Propellant storage module
RNDZ	Rendezvous
SAM	Stratospheric Aerosol Measurement Experiment (MA-007)
SC	Soyuz Commander

TABLE B-I.- ACRONYMS AND ABBREVIATIONS
USED IN AS-FLOWN FLIGHT PLAN - Concluded

<u>Acronym/abbreviation</u>	<u>Definition</u>
S/C	Spacecraft
SCM	Soyuz circularization maneuver
SEP	Separate or separation
S-IVB	Saturn IV-B (launch vehicle second stage)
SM	Service module
SP1	USSR tracking ship (off Nova Scotia)
SP2	USSR tracking ship (off Honduras)
SI	Solar inertial
SXT	Sextant
TK	Tank
TPI	Terminal phase initiation
TPM	Terminal phase midcourse
TRK	Track or tracking
TV	Television report (to control centers only)
USSR or USSR	Union of Soviet Socialist Republics
UVA	Ultraviolet Absorption Experiment (MA-059)
X-RAY	Soft X-Ray Experiment (MA-048)
ZFF	Zone-Forming Fungi Experiment (MA-147)

APPENDIX C - SPACECRAFT HISTORY

The manufacturing and checkout history of the command and service module (CSM-111), the docking module (DM-2), and the compatible docking system (DS-5) is given in figures C-1 and C-2. Figure C-1 shows the operations at the contractor's facility and figure C-2 shows the test operations conducted in the USSR and the checkout operations conducted at the Kennedy Space Center.

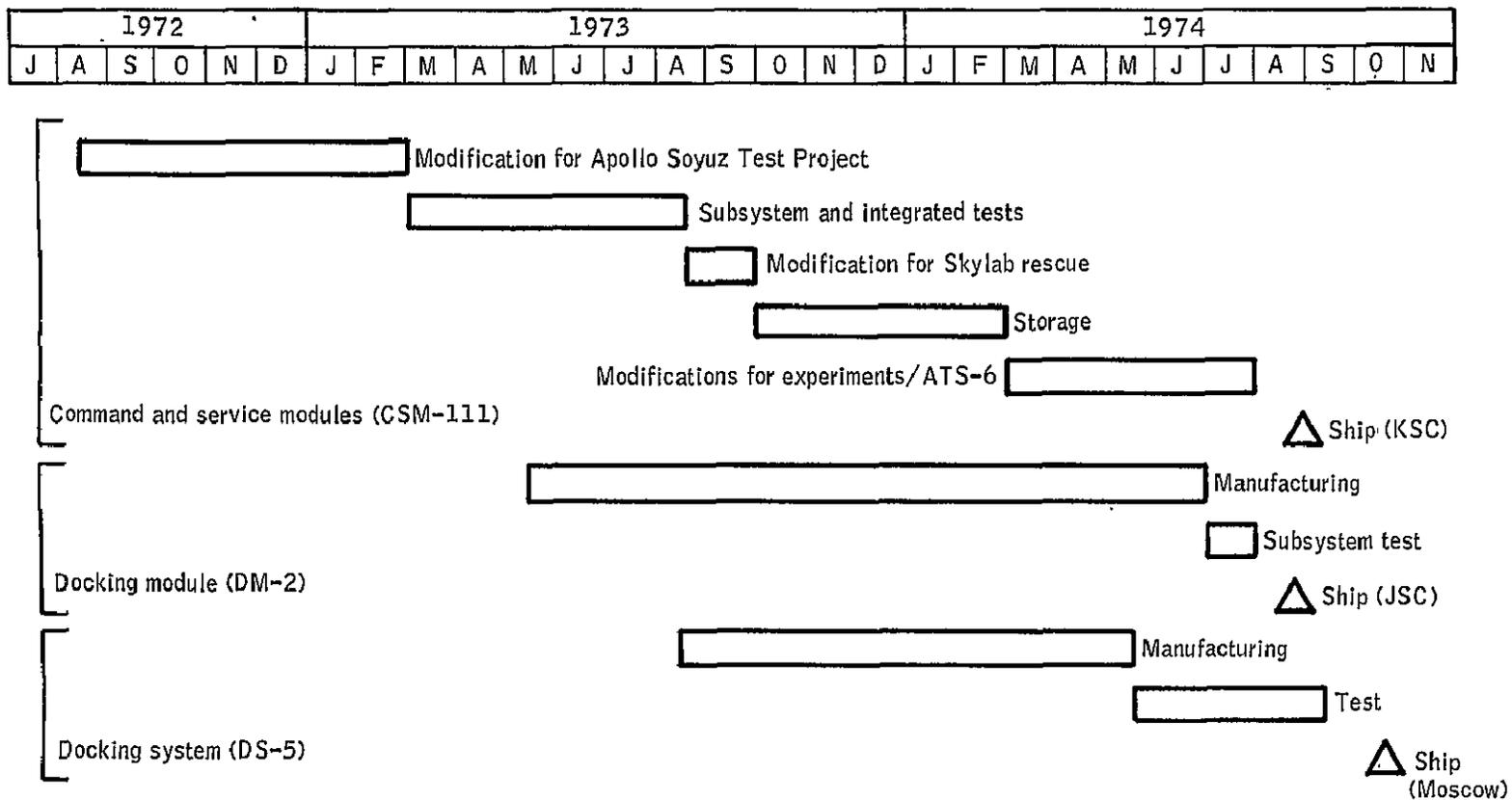


Figure C-1.- ASTP flight hardware manufacturing and checkout history at Contractor's facility.

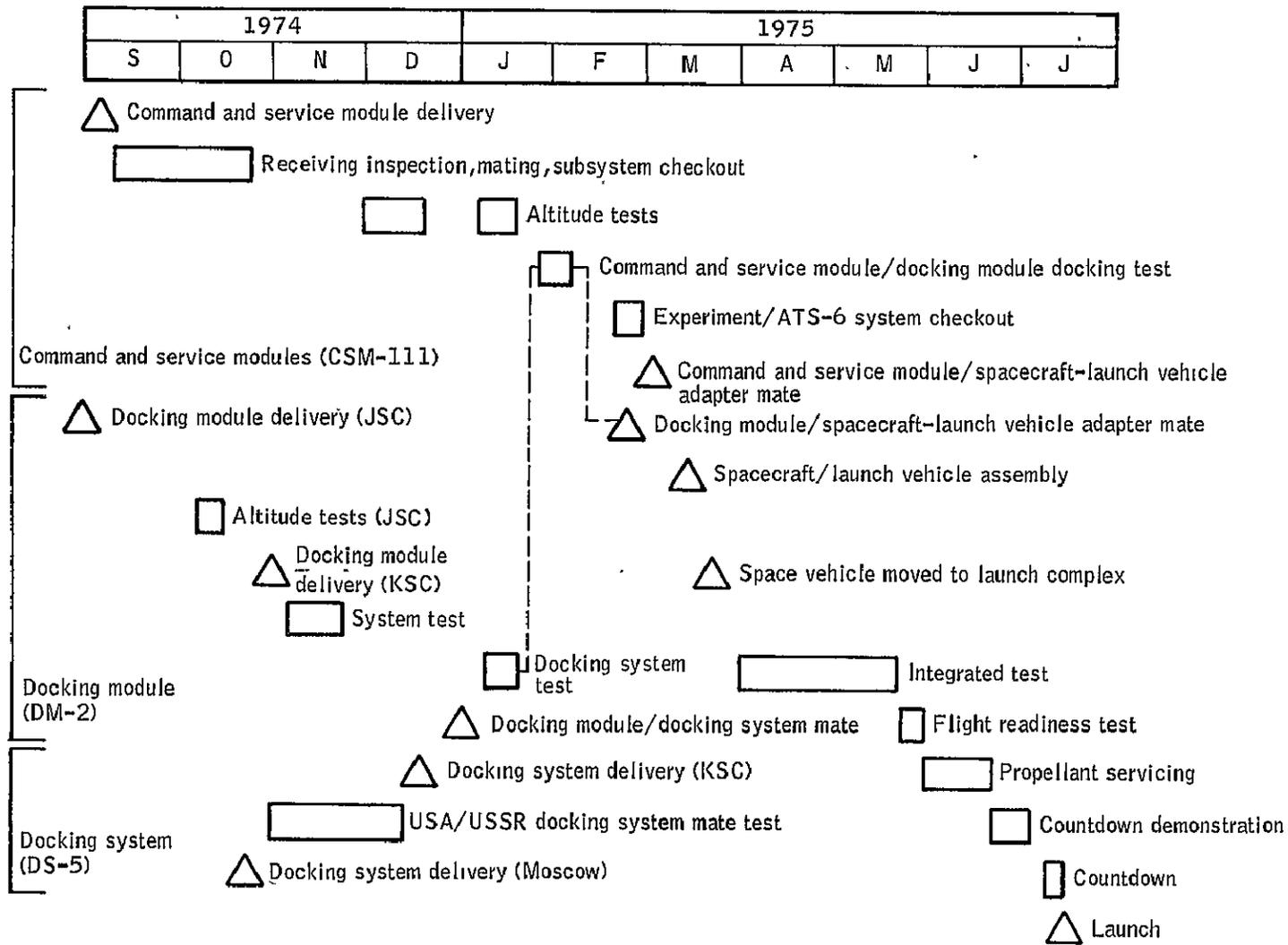


Figure C-2.- ASTP flight hardware checkout history at test and launch sites.

APPENDIX D - SPACECRAFT MASS PROPERTIES

Spacecraft and docking module mass properties are summarized in table D-I. These data represent conditions as determined from analyses of expendable loadings and usage. Mass properties were determined for each significant mission phase from lift-off through landing. Expendables usage is based on real-time data. The weights and centers of gravity of the individual modules were measured prior to flight and the inertia values calculated. All changes incorporated after the actual weighing were monitored and the mass properties updated.

TABLE D-I.- MASS PROPERTIES

Event	Weight, kg	Center of gravity, cm			Moment of inertia, kg m ²			Product of inertia, kg m ²		
		X _{A6}	Y _{A6}	Z _{A6}	I _{XX}	I _{YY}	I _{ZZ}	P _{ZY}	P _{ZZ}	P _{YZ}
Lift-off	21 138.8	2 466.9	2.3	5.9	39 027	707 415	706 847	-1 259	4 666	-2 122
Command and service modules	12 908.1	2 460.8	3.8	9.9	22 249	65 712	65 295	-1 189	1 306	-2 206
Docking module	2 006.3	1 635.8	-1.3	-8.4	1 089	2 193	2 426	-14	-69	-4
Spacecraft/launch vehicle adapter	2 044.4	1 607.6	1.3	-0.3	14 466	17 945	17 674	-39	107	65
Launch escape system	4 180.0	3 304.5	0.3	2.5	1 140	38 712	38 666	-22	1 104	0
Earth orbit insertion	16 955.2	2 260.4	3.1	6.6	37 873	303 075	302 560	-270	5 007	-2 123
Command and service modules	12 904.5	2 460.8	3.8	9.9	22 249	65 712	65 295	-1 189	1 306	-2 206
Docking module	2 006.3	1 635.8	-1.3	-8.4	1 089	2 192	2 426	-14	-69	-4
Spacecraft/launch vehicle adapter	2 044.4	1 607.6	1.3	-0.3	14 466	17 945	17 674	-39	107	65
Command and service module/docking module docked	14 872.6	2 532.9	3.3	9.7	23 226	116 846	116 717	-1 656	1 098	-2 165
Command and service module/docking module circularization	14 861.3	2 532.6	3.1	9.9	23 166	116 686	116 560	-1 652	1 134	-2 157
Command and service module/docking module first phasing maneuver	14 817.8	2 533.4	3.1	9.9	23 076	116 229	116 100	-1 628	1 124	-2 152
Command and service module/docking module second phasing maneuver	14 578.7	2 537.0	2.8	10.2	22 677	113 966	113 806	-1 513	1 079	-2 076
Command and service module/docking module corrective combination maneuver	14 528.4	2 538.0	2.8	10.2	22 596	113 399	113 196	-1 464	1 051	-2 069
Command and service module/docking module coelliptic maneuver	14 452.1	2 539.5	2.5	10.4	22 471	112 581	112 327	-1 384	987	-2 057
Command and service module/docking module terminal phase initiation maneuver	14 424.9	2 540.3	2.5	10.4	22 429	112 185	111 889	-1 355	980	-2 062
First docking	20 920.5	2 870.5	1.0	8.1	26 859	602 532	603 256	-1 504	-5 018	-1 912
Command and service module/docking module	14 183.2	2 543.3	1.5	11.9	22 052	110 987	110 776	-1 109	640	-1 927
^a Soyuz	6 737.3	3 558.0	0.5	-0.5	4 804	21 571	22 551	-49	294	-20
Final undocking	20 562.1	2 875.8	0.8	8.9	26 315	594 702	595 633	-854	-6 013	-1 679
Command and service module/docking module	13 824.8	2 543.6	0.8	13.5	21 396	106 566	106 616	-671	94	-1 649
^a Soyuz	6 737.3	3 558.0	0.5	-0.5	4 804	21 571	22 551	-49	294	-20
Docking module jettison	13 569.0	2 551.7	1.3	13.0	20 797	108 383	108 468	-819	171	-1 477
Command and service module	11 521.0	2 472.9	1.5	14.0	19 685	59 201	59 040	-704	868	-1 482
Docking module	2 048.0	2 991.9	0	7.1	1 083	2 277	2 561	16	-41	1
Command and service module at docking module separation maneuver	11 455.2	2 474.5	1.3	14.2	19 547	58 720	58 502	-673	860	-1 471
Command and service module at stable orbit maneuver	11 410.3	2 475.5	1.3	14.2	19 454	58 382	58 115	-647	852	-1 467
Command and service module at deorbit maneuver	11 350.9	2 476.0	1.5	14.2	19 309	58 222	57 965	-651	857	-1 420
Command module/service module at separation	11 128.2	2 482.1	1.0	14.2	18 947	55 916	55 365	-469	789	-1 447
Command module	5 848.3	2 640.1	0.8	14.2	7 920	7 170	6 385	95	-465	-41
Service module	5 279.9	2 307.1	1.3	14.2	11 026	17 952	18 187	-504	1 262	-1 406
Command module at entry interface	5 842.8	2 640.3	0.8	14.2	7 912	7 161	6 381	94	-462	-41
Command module at drogue parachute deployment	5 654.6	2 637.3	0.8	14.0	7 727	6 742	5 986	96	-428	-38
Command module at landing	5 415.5	2 631.7	0.8	14.5	7 615	6 270	5 510	98	-403	-37

^a Apollo spacecraft reference system.ORIGINAL PAGE IS
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APPENDIX E - POSTFLIGHT TESTING

Postflight testing and inspection of the command module and crew equipment for evaluation of the inflight performance and investigation of flight problems were conducted at the contractor's and vendor's facilities and at the Johnson Space Center in accordance with approved Apollo Spacecraft Hardware Utilization Requests (ASHUR's). The tests performed as a result of inflight problems are described in table E-I and discussed in the appropriate systems performance sections of this report. Tests being conducted for other purposes in accordance with other ASHUR's and the basic contract are not included.

TABLE E-I.- POSTFLIGHT TESTING SUMMARY

ASHUR number	Purpose	Test performed	Results
Environmental Control			
111022	Determine status of postlanding ventilation system.	Operate vent valves and fan.	Valve operation normal. Fan was inoperative.
Communications			
111024	Investigate crew reported squeal/noise from approximately 50 000 feet to splashdown and while on water.	Operate system in entry and postlanding configurations.	Postlanding noise caused by salt-water shorting the docking ring connector pins.
Guidance and Navigation			
111012	Investigate inertial coupling data unit failure indications.	Remove equipment and perform analysis and test.	No abnormal conditions found.
Caution and Warning			
111023	Investigate low caution and warning tone volume reported by crew.	Check out possible configurations of audio systems.	Tone volume normal.
Bioinstrumentation			
111015	Investigate cause for intermittent biomedical data.	Perform inspection and checks on biomedical belts and harnesses.	Problem duplicated by separating electrodes from crewman's skin.

TABLE E-I.- POSTFLIGHT TESTING SUMMARY - Concluded

ASHUR number	Purpose	Test performed	Results
Color Television System			
111021	Investigate cause for color breakup and sporadic luminance modulation on television camera SN9 and loss of color on camera SN4002.	Perform failure analysis.	Faceplate arcing of high voltage photocathode bias supply found in camera SN9. Small piece of cable bundle lacing found in camera SN4002 color wheel housing.
Experiments			
111013	Determine cause for inoperative reels on MA-089 recorder 2.	Perform failure analysis.	Movement of reels prevented by tape sticking to recorder head as a result of residual cleaning fluid.

APPENDIX F - DATA AVAILABILITY

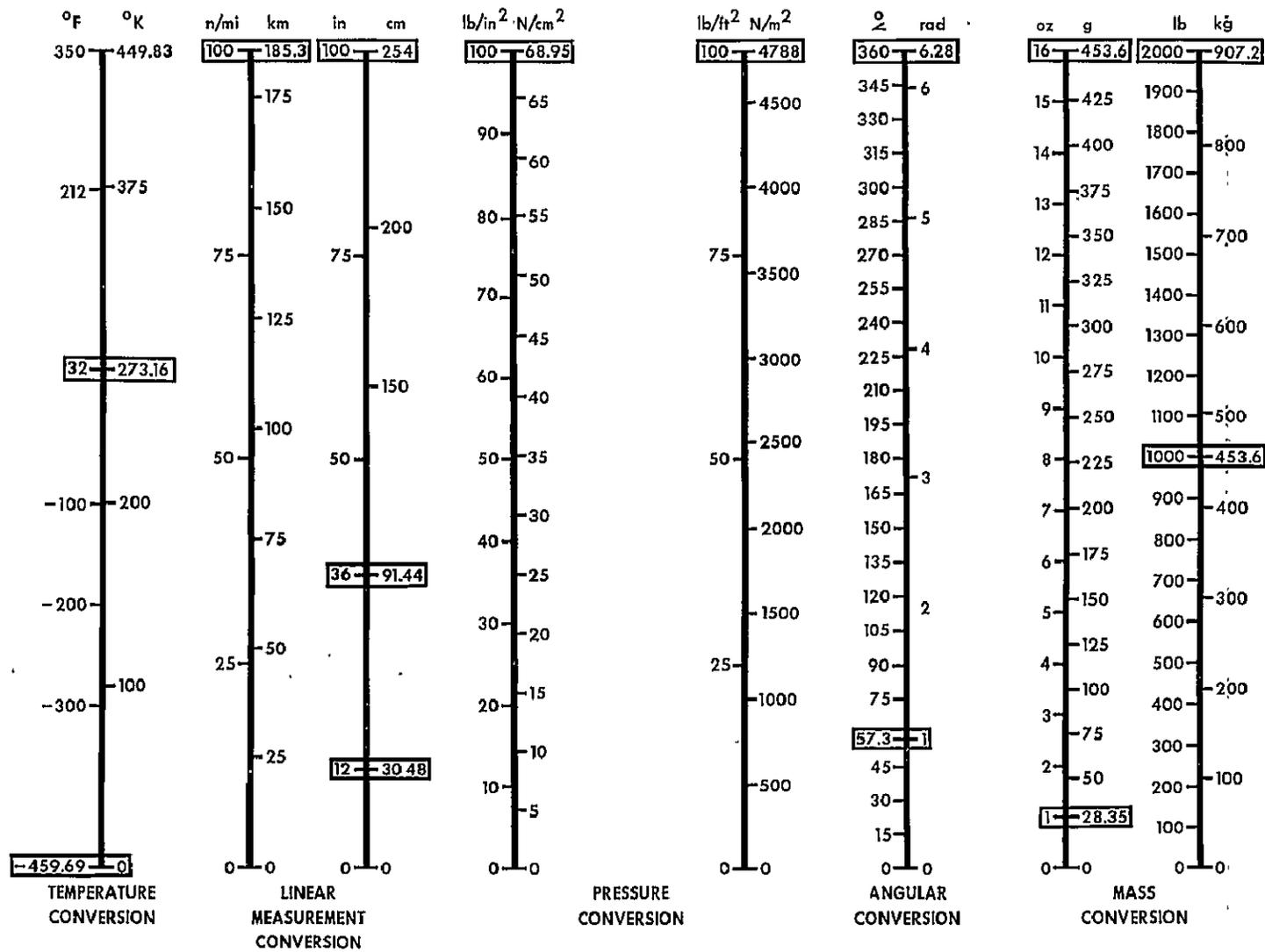
No data were processed in support of postflight evaluation of the Apollo-Soyuz mission. The data processing that was conducted in support of the Apollo-Soyuz mission experiments for scientific analysis are published in JSC-09994, Apollo Soyuz Test Project Data Availability Timeline.

APPENDIX G - CONVERSION SCALES

Relationships between SI units and conventional units of measurement are shown in figure G-1.

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Figure G-1. - Conversion scales.



APPENDIX H - GLOSSARY

Arrhythmia	Any variation from the normal series rhythm of the heart.
Autoflora	The population of microorganisms normally residing in or on the subject's body.
Chemical pneumonitis	Lung tissue reaction to a toxic chemical agent.
Footprint	Outline of a predicted landing area.
Gravity anomaly	A small perturbation of the general gravitational field.
Irradiated	An application of ionizing radiation.
Kilocalorie	The unit used in the study of metabolism known as the large calorie being the amount of heat required to raise 1 kilogram of water from 288° to 289° K.
Magellanic clouds	Two small galaxies which are close neighbors to our own "Milky Way" galaxy. They are seen from the Southern Hemisphere.
POGO	A sustained vibration in liquid fueled rockets, sometimes exceeding structural or human tolerance limits, caused by a regenerative dynamic interaction of propulsion system propellant flow and thrust oscillations with launch vehicle structural vibrations.
Posigrade	In the direction of the velocity vector.
Raster scan	To scan an area with an instrument using a pattern of back and forth traverses.
Roentgenogram	An X-ray or gamma ray photograph.
South Atlantic anomaly	A region of trapped protons and electrons of the Van Allen Belt located over a large portion of South America, the South Atlantic and the southern tip of Africa and dipping down to a relatively low altitude.

APPENDIX I - MISSION REPORT SUPPLEMENT

The following report supplements the Apollo-Soyuz Mission Evaluation Report.

<u>Supplement number</u>	<u>Title</u>	<u>Publication status</u>
1	Summary of Lightning Activities by NASA for ASTP	Planned for publication

JSC-10607
Supplement 1

SUMMARY OF LIGHTNING ACTIVITIES
BY NASA FOR THE
APOLLO SOYUZ TEST PROJECT

SUPPLEMENT NO. 1
TO
APOLLO SOYUZ MISSION EVALUATION REPORT



National Aeronautics and Space Administration
LYNDON B. JOHNSON SPACE CENTER
Houston, Texas
July 1976

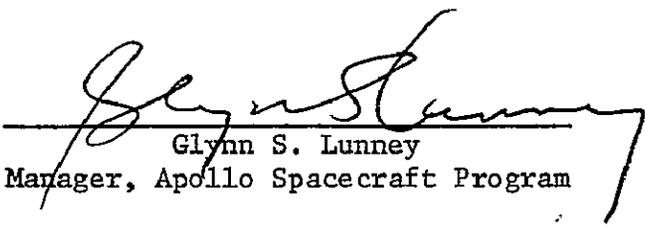
SUMMARY OF LIGHTNING ACTIVITIES BY NASA
FOR THE
APOLLO SOYUZ TEST PROJECT

Supplement No. 1
to
Apollo Soyuz Mission Evaluation Report

PREPARED BY
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS
July 1976

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INTRODUCTION

The Apollo Soyuz Test Project launch was scheduled for approximately 3 o'clock e.d.t. on the afternoon of July 15, 1975, the time of the statistical peak of thunderstorm activity at the Kennedy Space Center (KSC). The launch window for the Apollo launch was unusually short (5 to 8 minutes). Any hold for weather would force the postponement of the launch until the next day. Since the existing launch rules were based primarily on meteorological observations, the presence of a nearby storm could cause the postponement of the launch even though the electric field intensity along the track of the vehicle was sufficiently low that triggering of lightning by the launch vehicle could not occur.

To avoid the possibility of an unnecessary launch delay, NASA initiated a special program to provide aircraft measurements of electric fields at various altitudes over the Apollo vehicle launch pad. Eight aircraft, each equipped with electric field meters, were used in the program. This report discusses this program and some of the more important findings. Also included is a summary of the history of manned space vehicle involvement with lightning (appendix A), a brief description of the lightning instrumentation in use at KSC at the time of the Apollo Soyuz mission (appendix B), and a discussion of the airborne instrumentation and related data (appendix C).

LIGHTNING RELATED PREPARATIONS FOR THE APOLLO SOYUZ TEST PROJECT MISSION

Prelaunch Assessment

During the year preceding the Apollo Soyuz Test Project mission, it became evident that the launch would occur during the period of highest thunderstorm incidence at KSC (fig. 1). Consequently, simulated lightning tests were performed on a non-flight launch vehicle and the backup spacecraft to assess the damage that could be caused by lightning. The results of these tests indicated that the vehicle could be lost if it experienced an inflight lightning strike. Because of this hazard and in light of the past experiences with lightning (appendix A), the existing launch rules (fig. 2) were reexamined to determine if tighter restrictions should be imposed. This activity, however, indicated quite the opposite. It was found that the restrictions could, in fact, be relaxed if certain conditions were met. Prior experience indicated that the risk of triggering lightning during launch would be low if the electric field intensities measured by the ground instrumentation did not exceed 1000 volts per meter and if the electric field intensities along the vehicle flight path did not exceed 3000 volts per meter to 15 000 volts per meter.

In the past, the weather office at KSC used electric field intensity measurements from a large network of field mills (see appendix B for a description of field mills and the network) to identify impending lightning hazards to space vehicles. Very little data were available to correlate the field intensities aloft to those measured at ground level. A special research program was conducted by KSC and the National Oceanic and Atmospheric Administration (NOAA) to obtain simultaneous ground and airborne field intensity measurements. Four aircraft participated in this effort. (See table I for the aircraft types and appendix C for a discussion of a typical airborne system.) The results of the KSC/NOAA research activity indicated that, below an altitude of 3050 meters, the airborne data compared favorably with the data obtained using ground instrumentation. Above this altitude, however, the ground-based instrumentation could not be relied upon to provide an accurate measurement of the electric field intensities aloft. The conclusion was that airborne electric field data would be required in support of the Apollo launch.

In order to measure field levels along the complete track of the launch vehicle, four additional high-altitude aircraft were obtained and equipped with electric field meters. Thus, on launch day, eight aircraft would be available for data acquisition at different altitudes just before the Apollo lift-off (fig. 3). In conjunction with ground readings, the data from these aircraft would be used to implement a set of revised launch rules that were based on acceptable/unacceptable risk factors associated with real-time measurement of electric field intensities (figs. 4 and 5). With this approach, it would be possible to launch safely even though electrified clouds were in the launch vicinity.

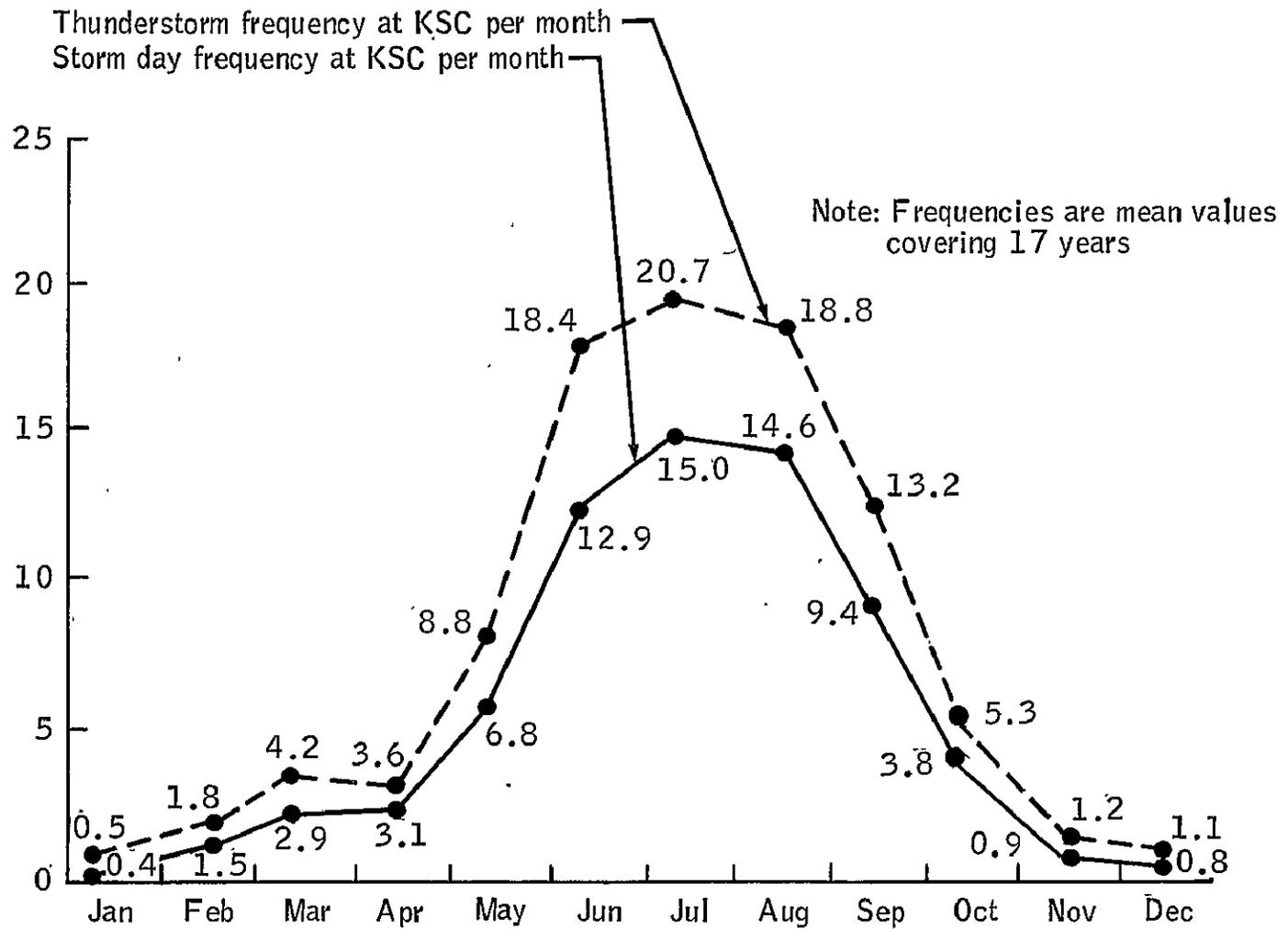


Figure 1.- Lightning history at KSC

SEVERE WEATHER CONDITIONS

LAUNCH RULES:

1. THE SPACE VEHICLE WILL NOT BE LAUNCHED IF THE NOMINAL FLIGHT PATH WILL CARRY THE VEHICLE:
 - A. WITHIN 5 STATUTE MILES OF A CUMULONIBUS (THUNDERSTORM) CLOUD OR WITHIN 3 STATUTE MILES OF AN ASSOCIATED ANVIL.
 - B. THROUGH COLD-FRONT OR SQUALL-LINE CLOUDS WHICH EXTEND ABOVE 10,000 FEET.
 - C. THROUGH MIDDLE CLOUD LAYERS 6,000 FEET OR GREATER IN DEPTH WHERE THE FREEZE LEVEL IS IN THE CLOUDS.
 - D. THROUGH CUMULUS CLOUDS WITH TOPS AT 10,000 FEET OR HIGHER.

Figure 2.- Launch mission rules for Skylab

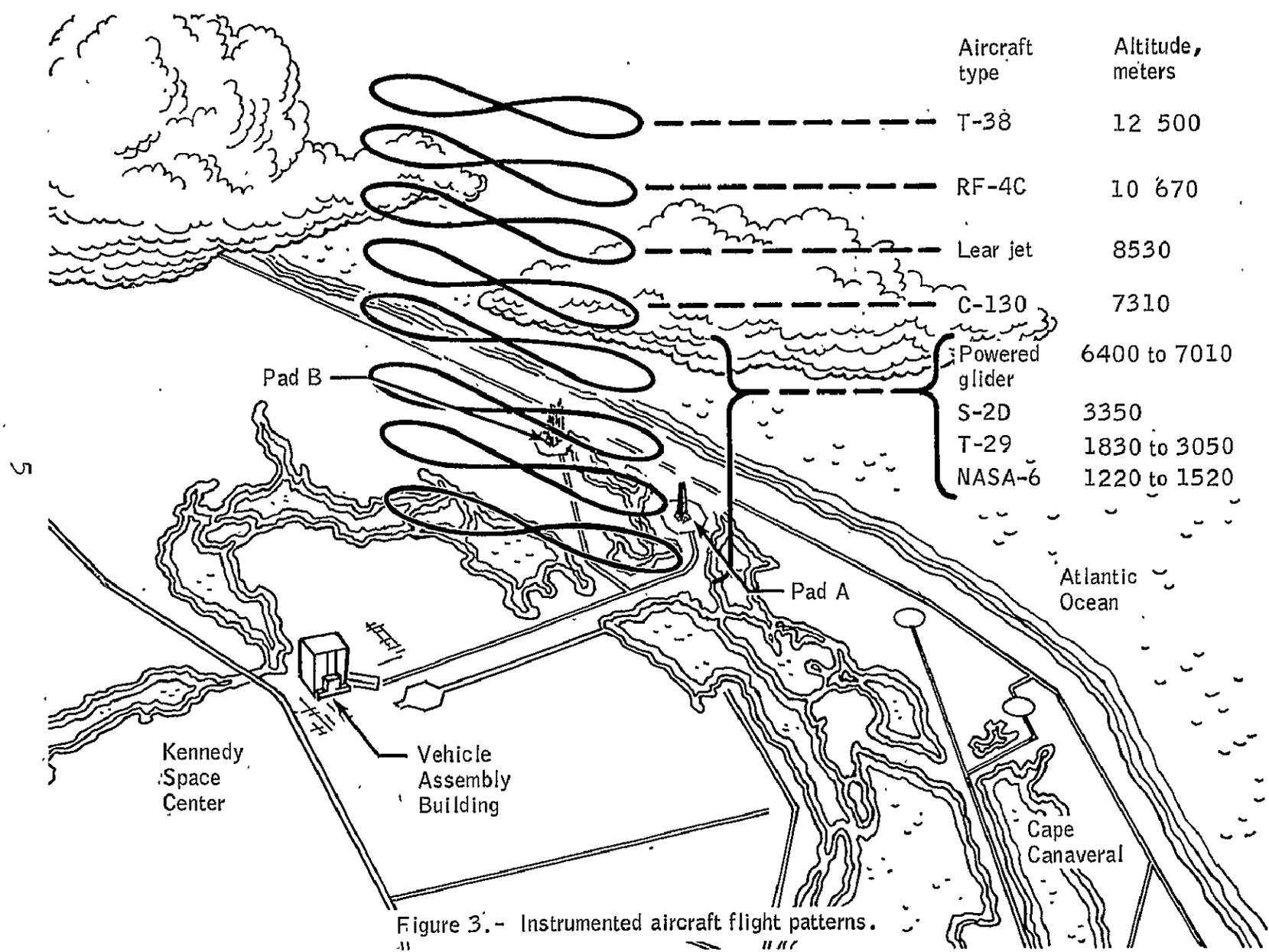


Figure 3.- Instrumented aircraft flight patterns.

JOHN F. KENNEDY SPACE CENTER, NASA - LAUNCH MISSION RULES

REV	ITEM	DESCRIPTION			
		<u>WEATHER RESTRICTIONS (CONTINUED):</u>			
3	1-406	<p><u>SEVERE WEATHER CONDITIONS:</u></p> <p>1. THE SPACE VEHICLE WILL NOT BE LAUNCHED IF THE NOMINAL FLIGHT PATH WILL CARRY THE VEHICLE:</p> <ul style="list-style-type: none"> A. THROUGH A CUMULONIMBUS (THUNDERSTORM) CLOUD. B. WITHIN 5 STATUTE MILES OF A CUMULONIMBUS (THUNDERSTORM) CLOUD OR WITHIN 3/4 STATUTE MILES OF AN ASSOCIATED ANVIL. THIS RULE MAY BE RELAXED AT THE DISCRETION OF THE LAUNCH DIRECTOR IF THE ELECTRIC FIELD AT THE LAUNCH PAD IS LESS THAN 1 KILOVOLT PER METER. C. THROUGH COLD-FRONT OR SQUALL-LINE CLOUDS WHICH EXTEND ABOVE 10,000 FEET. D. THROUGH MIDDLE CLOUD LAYERS 6,000 FEET OR GREATER IN DEPTH WHERE THE FREEZE LEVEL IS IN THE CLOUDS. E. THROUGH CUMULUS CLOUDS WITH THE FREEZE LEVEL IN THE CLOUDS. F. RULES C, D, AND E ABOVE MAY BE RELAXED AT THE DISCRETION OF THE LAUNCH DIRECTOR WHEN ELECTRIC FIELD MEASUREMENTS IN THE LAUNCH PAD AREA ARE STABLE AND MEASURE LESS THAN 1 KILOVOLT PER METER. G. RULES C, D, AND E ABOVE MAY BE FURTHER RELAXED PROVIDED THAT AIRBORNE AND GROUND ELECTRIC FIELD MEASUREMENTS MEET THE CRITERIA DEFINED IN FIGURE 1. <p>2. FOR FLIGHT CREW EGRESS RULES RELATIVE TO SEVERE WEATHER CONDITIONS, SEE "FLIGHT CREW SAFETY RULES" SUBSECTION, ITEMS 1-609 (9) AND 1-610 (6).</p>			
	1-407	<p><u>PAD OR LAUNCH ABORT RECOVERY WEATHER CONDITIONS:</u></p> <p>THE LAUNCH OPERATIONS MANAGER WILL OBTAIN A GO/NO-GO DECISION FROM THE FLIGHT DIRECTOR BASED UPON THE PAD OR LAUNCH ABORT RECOVERY WEATHER CONDITIONS.</p>			
MISSION	SECTION	STAGE SUBSECTION	SYSTEM	COMPARISON NO	PAGE
ASTP	SPACE VEHICLE	WEATHER RESTRICTIONS		JULY 14, 1975	1-18

KSC FORM 4-47A (REV. 1-68)

Figure 4.- Launch mission rules for launching during severe weather.

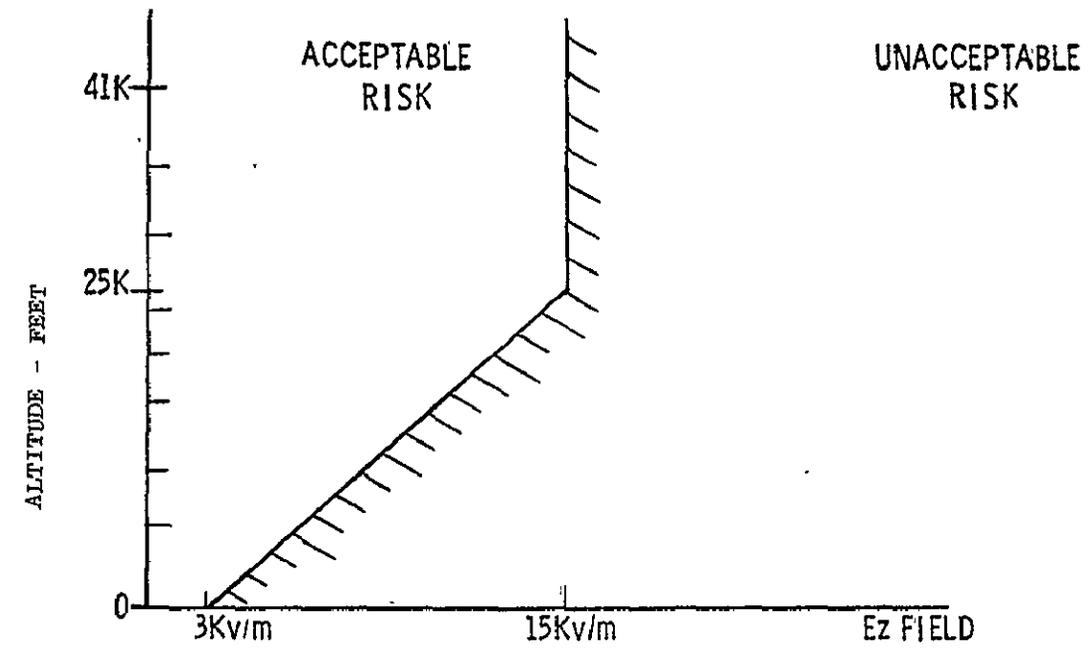
REV	ITEM	DESCRIPTION			
3		<p style="text-align: center;">ASTP LAUNCH GUIDE LINES WITH AIRBORNE AND GROUND INSTRUMENTATION</p>  <ul style="list-style-type: none"> • VERTICAL FIELD MEASUREMENTS ALONG FLIGHT PATH IN A 3 MILE AREA ARE WITHIN THE ABOVE ENVELOPE • AND NO RAPID FLUCTUATION OF ABOUT 3 Kv/m AT ABOUT 1 MIN INTERVALS WITHIN THE 3 MI AREA MEASURED BY THE GROUND MILLS. 			
MISSION ASTP	SECTION SPACE VEHICLE	STAGE SUBSECTION WEATHER RESTRICTIONS	SYSTEM FIGURE 1	DATE JULY 14, 1975	PAGE 1-18A

Figure 5.- Launch guidelines with airborne and ground instrumentation.

KSC FORM 4-47A (REV. 1 68)

TABLE I.- INSTRUMENTED AIRCRAFT

Type of aircraft	NASA or other designation	Remarks
<p>Aircraft for KSC/NOAA research:</p> <p>Schweitzer powered glider</p> <p>T-29</p> <p>S-2D</p> <p>C-45</p>	<p>---</p> <p>----</p> <p>---</p> <p>NASA 6</p>	<p>Owned by U.S. Navy, bailed to New Mexico Tech and based at Socorro, New Mexico.</p> <p>NOAA aircraft.</p> <p>Naval Research Laboratory aircraft under contract to KSC and NOAA.</p> <p>KSC aircraft stationed at Patrick AFB. Used as basis for comparison of data.</p>
<p>Additional aircraft for launch vehicle flight path measurements:</p> <p>Lear Jet</p> <p>T-38</p> <p>C-130</p> <p>RF-4C</p>	<p>NASA 705</p> <p>NASA 902</p> <p>MIKE 70</p> <p>AGAR 22</p>	<p>Ames Research Center aircraft stationed at Moffitt Field, Calif. Instrumented by the Stanford Research Institute.</p> <p>JSC aircraft stationed at Ellington AFB. Instrumented by the Stanford Research Institute.</p> <p>U.S. Air Force aircraft stationed at Kirtland AFB. Instrumented by the Air Force Cambridge Research Laboratory.</p> <p>U.S. Air Force aircraft stationed at Kirtland AFB. Instrumented by the Air Force Weapons Laboratory.</p>

Launch Day Aircraft Operations

The planned launch time for the Apollo vehicle was 15:50:00 e.d.t., July 15, 1975. A KSC Operations Task Team had been formed to control the aircraft that would fly over the pad area. This area was designated R-2902 as shown in figure 6. The aircraft were scheduled to arrive on station in area R-2902 at T minus 35 minutes with a planned departure at T minus 5 minutes.

The four lower aircraft (fig. 3) were to relay data directly to the KSC weather office while the data acquisition point for the higher aircraft was to be the Range Control Center of the Air Force Eastern Test Range. At the Range Control Center, the data were to be plotted and analyzed by a weather specialist and subsequently transmitted to the KSC weather office. The KSC weather office would then compile the data from all of the aircraft and the ground instrumentation. These data would form the basis to implement the revised launch mission rules for lightning.

Because of the number of aircraft planned to be in the R-2902 area on launch day, prime importance was to be placed on safety and weather avoidance. The Federal Aviation Agency would participate in the planning of aircraft control and safety. All aircraft were to be under the control of the Miami Center and aircraft positions were to be monitored. The aircraft were to enter and depart the R-2902 area from predetermined staging areas controlled by the Range Control Center. In addition, weather advisories were to be relayed to the aircraft on a periodic basis so that the flight patterns could be changed to avoid unsafe weather conditions.

The entire operation went quite smoothly on launch day due mainly to the sequence of weather events leading up to the actual launch time. The deep southwesterly flow pattern that is strongly conducive to afternoon thunderstorm development over the Cape Canaveral area had previously existed, but had ended the day before launch. A change in direction of the upper winds caused a drastic decrease in thunderstorm probability (the probability was 22 percent). Thus, on launch day, conditions were good with scattered cumulus and thin cirrus clouds at the launch site. Thunderstorms were visible to the west and some isolated cumulonimbus clouds were off shore. As a precaution however, the field measuring aircraft were deployed about an hour before launch. After confirming the presence of normal electric fields (fair weather values are 200 volts per meter to 300 volts per meter), the aircraft were released before the scheduled time for them to clear the area.

Conclusions

The lightning related preparations for the Apollo Soyuz Test Project mission highlighted a number of significant points to be considered for future space programs.

Launch delays due to the presence of clouds in the launch area can be avoided by determination of the electric field intensity in the flight path.

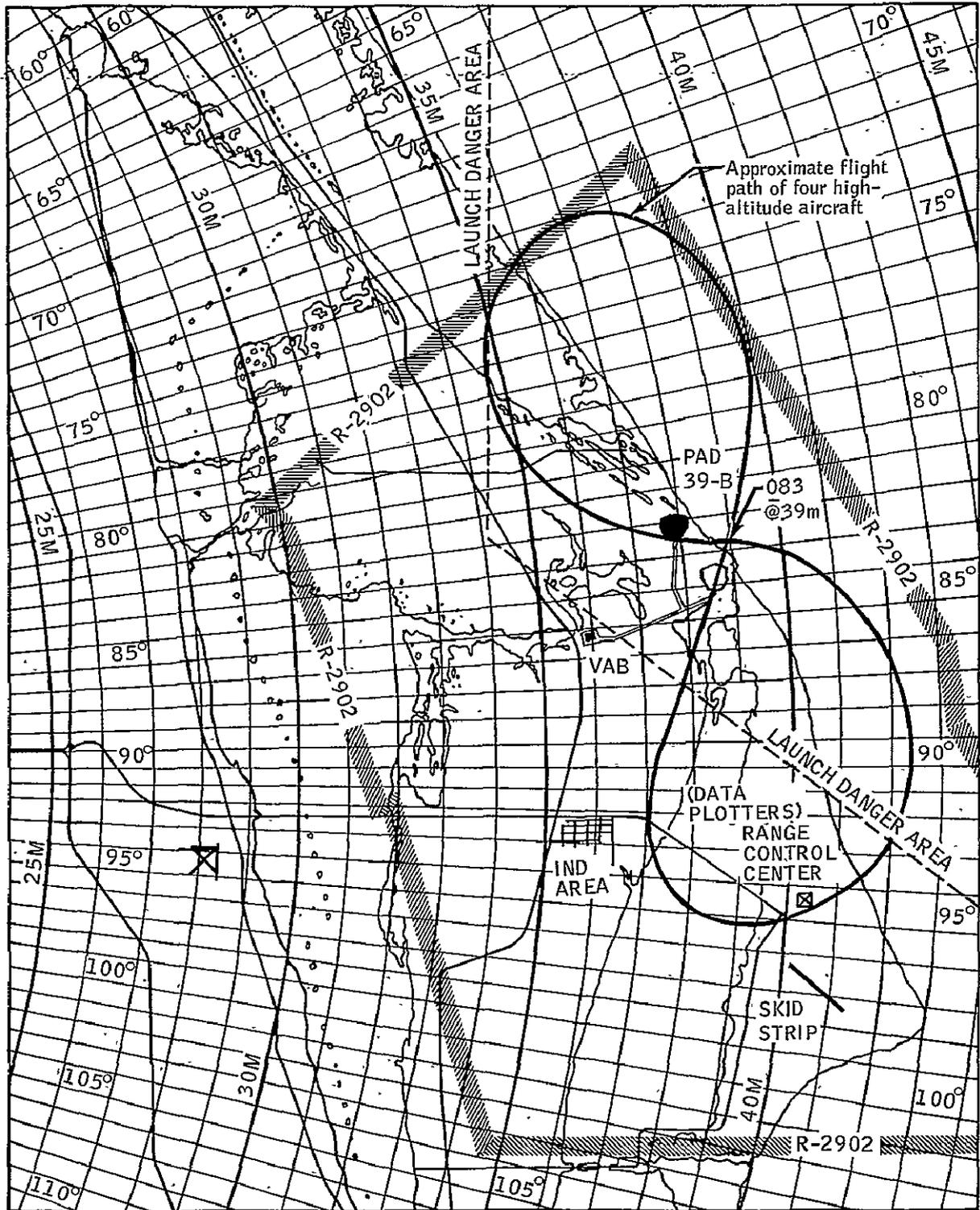


Figure 7.- Area R-2902 and typical flight path.

Further research is required to characterize the electrical properties of various cloud types. This activity should be directed toward reducing the cost and operational complexity of obtaining the electric field intensity data in and near the flight path.

Multiple-instrumented aircraft can be used to measure the electric fields aloft just prior to launch.

Lightning protection should be a major consideration in the design of future vehicles.

POST APOLLO SOYUZ TEST PROJECT ACTIVITIES

Viking Spacecraft Launch

The value of having airborne electric field measuring capability was fully demonstrated in the month following the Apollo Soyuz Test Project mission. Three of the monitoring aircraft supported the two Viking space vehicle launches; one on August 20, 1975, and the other on September 9, 1975.

Since both ground and airborne instrumentation indicated acceptable electric field levels over the launch site, the first vehicle was launched through a thick deck of middle clouds even though the internal cloud temperatures were approaching the freeze level.

For the second launch, large convective storms had developed off shore and were moving toward the launch site. Again, the ground and airborne instrumentation indicated that the electric fields were not dangerously high over the launch site ahead of the storm, and the vehicle was launched through opaque clouds that were estimated to be at an altitude of 2450 meters. Within minutes after launch, these clouds moved on shore and unacceptable high electric fields were present over the launch site.

Neither of these launches would have been permitted without the real-time measurement of electric field intensities aloft.

Cloud Electrification Studies

Of the aircraft that participated in the real-time measurement of electric fields during the Apollo-Soyuz and Viking launches, the Lear Jet had unique capabilities. During one of the prelaunch calibration flights, it was found that the aircraft's 2073-meter-per-minute climb rate made it possible to carry out successive flights over the top of a developing thunderstorm that was growing at a rate of 610 meters per minute.

In recognition of the unique capabilities of the Lear Jet system, provisions were made for it to remain at Patrick Air Force Base following the Apollo-Soyuz launch to make high-altitude field measurements in the vicinity of storm cells over KSC. In particular, a cooperative program was arranged in which the electric fields of cells chosen by NOAA for experiments in seeding with radar chaff were monitored by the Lear Jet. When conditions over KSC were not appropriate for the chaff seeding studies, the Lear Jet was flown around developing thunderstorm cells to study the fields in their vicinity - with particular attention to the region of the anvil. In the course of this flying, electric field data were gathered from ten different storm cells.

Most of the measurements started when a cell was very young. The measurements followed the cell's development throughout the growth period and continued until it began to disintegrate. Photographs or sketches were made of most of the storms studied and the chart records of electric field data were annotated. Accordingly, the data constitutes a unique source of information on the electrical behavior of developing thunderstorms. (See appendix C).

APPENDIX A

HISTORY OF MANNED SPACE VEHICLE INVOLVEMENT WITH LIGHTNING

The Apollo 12 Incident

On November 14, 1969, the Apollo 12 spacecraft was affected by two triggered lightning strikes during launch. Major electrical disturbances were caused at 36.5 seconds and again at 52 seconds. In addition to many temporary effects noted in both the launch vehicle and the spacecraft, some permanent damage resulted - the loss of nine nonessential instrumentation sensors. All upsets and permanent damage were associated with solid-state electrical circuits.

Follow-on investigations determined that the Apollo launch vehicle could, because of its electrical length, trigger lightning if launched through certain electrified cloud configurations. Thus, launch restrictions were imposed to prevent a recurrence of the Apollo 12 incident.

Other Apollo and Skylab Vehicle Lightning Damage

Apollo.- While Apollo 13 and 14 did not experience damage to spacecraft systems or associated ground support equipment, the Apollo 14 launch was delayed for 40 minutes as a result of thunderstorm conditions near the launch site. Prior to the Apollo 15 mission, the launch pad was struck by lightning on five occasions during June and July 1971. Two instrumentation sensors on the spacecraft were damaged on the first occasion and 10 ground support equipment units were damaged during three of the five strikes. Corrective measures were taken to insure proper equipment grounding after the first three strikes and no equipment damage resulted from the last two strikes. The launch facility was hit by lightning twice in March 1973 while the Apollo 16 vehicle was being prepared for launch. Neither the spacecraft nor the ground equipment was affected.

Skylab.- The Skylab 2 spacecraft (CSM 116) experienced damage to four instrumentation sensors in the first of two strikes to the launch facility structure in May 1973. The ground support equipment was not affected. The Skylab 3 spacecraft (CSM 117) was at the launch site in June and July 1973. The first of two strikes to the launch facility resulted in damage to 10 spacecraft instrumentation sensors, 8 ground instrumentation sensors, and 3 ground support equipment units. The Skylab 4 spacecraft (CSM 118) experienced the largest of all the strikes (200 000 amperes) in August 1973. Four spacecraft instrumentation sensors were damaged. In addition, the guidance and navigation system inertial measurement unit, a signal conditioner, and the coupling data unit indicated anomolous operation and were replaced. The ground equipment was not affected.

Lightning Research Activity

As a result of the electrical disturbances experienced during the Apollo 12 launch and the lightning damage to spacecraft and ground equipment on other Apollo missions, the value of further research in this area was recognized. Several experiments were performed prior to, during, and subsequent to the Apollo 13 and Apollo 14 launches to study some of the launch phase electrical phenomena. The Apollo 13 experiments were designed primarily to study the effects of the spacecraft on the atmospheric electrical fields during launch. Other experiments were conducted in connection with Apollo 14 to better define the electrical charge and the triggering mechanism of the discharge. Additional measurements were made subsequent to Apollo 14 to acquire data for the derivation of peak lightning strokes.

Other general lightning research activity was conducted at KSC, including additional measurements to determine parameters of lightning in the launch area, measurement of cloud charge transfer during discharge, stroke characteristics and field changes (magnetic and electric), airborne electric field measurements correlated to ground field measurements, and correlation of cloud charge growth and transfer with associated meteorological phenomena. Also chaff seeding tests were conducted on charged clouds to observe the effect on electric field decay and lightning activity.

APPENDIX B

LIGHTNING INSTRUMENTATION SYSTEMS AT KSC

KSC involvement with lightning instrumentation began with the construction of the Saturn mobile launcher and other high structures such as the mobile service structure and the vehicle assembly building. It was necessary to provide advance thunderstorm warning for the safety of outdoor work crews and to initiate measures that would protect any equipment or instruments that could be damaged or destroyed by a direct or nearby lightning strike. In later launch operations, it was necessary to locate and determine the magnitude of lightning strikes so that a proper damage assessment could be made. Thus, a lightning warning instrumentation system and a lightning data acquisition system existed at the time of the Apollo Soyuz mission.

Lightning Warning Instrumentation System

The lightning warning instrumentation system has two major elements - an electric field measuring and display system and an instrumented aircraft. The electric field measuring and display system consists of 25 remote sites (fig. B-1) which measure the electric field of the atmosphere. The field mill is the instrument that is used for detection and measurement of electric fields. "Mill" is jargon for a capacitively coupled electric field pickup or probe and the terminology comes from the fact that the detector has a rotating shutter over the sensing plate (fig. B-2). The grounded rotating plate shutter causes the electric field applied to the sensing plate to fluctuate. This fluctuation generates an a-c signal across the sensing plates which, in turn, is amplified and converted into a voltage that is proportional to the electric field. The range of the field-measuring instruments is plus 15 kilovolts per meter to minus 15 kilovolts per meter with a resolution of 30 volts per meter.

Signals from the individual field mills are digitized and transmitted to a central data processing facility. The entire network of field mill sites is interrogated simultaneously from this data processing area at a once-a-minute rate. After the interrogation signal, each field mill station reports back sequentially to a central receiver system. The computer then generates a page listing of electric field data and a map display showing the equipotential lines of the electric field. The centers of these closed-loop patterns indicate the high-charge centers under the clouds above KSC.

The instrumented aircraft (NASA-6) was obtained to extend the range of electric field measurements and to help determine the relationship of the ground-measured electric field with actual cloud electrification. The instrumentation measures and records the electric field, air temperature, and dew point. Radiometric temperature measurements and photographs are also obtained. The aircraft is especially useful for detecting the initial charge generating stages of clouds approaching the launch area. Electric field values of 10 volts per meter to 800 000 volts per meter can be measured.

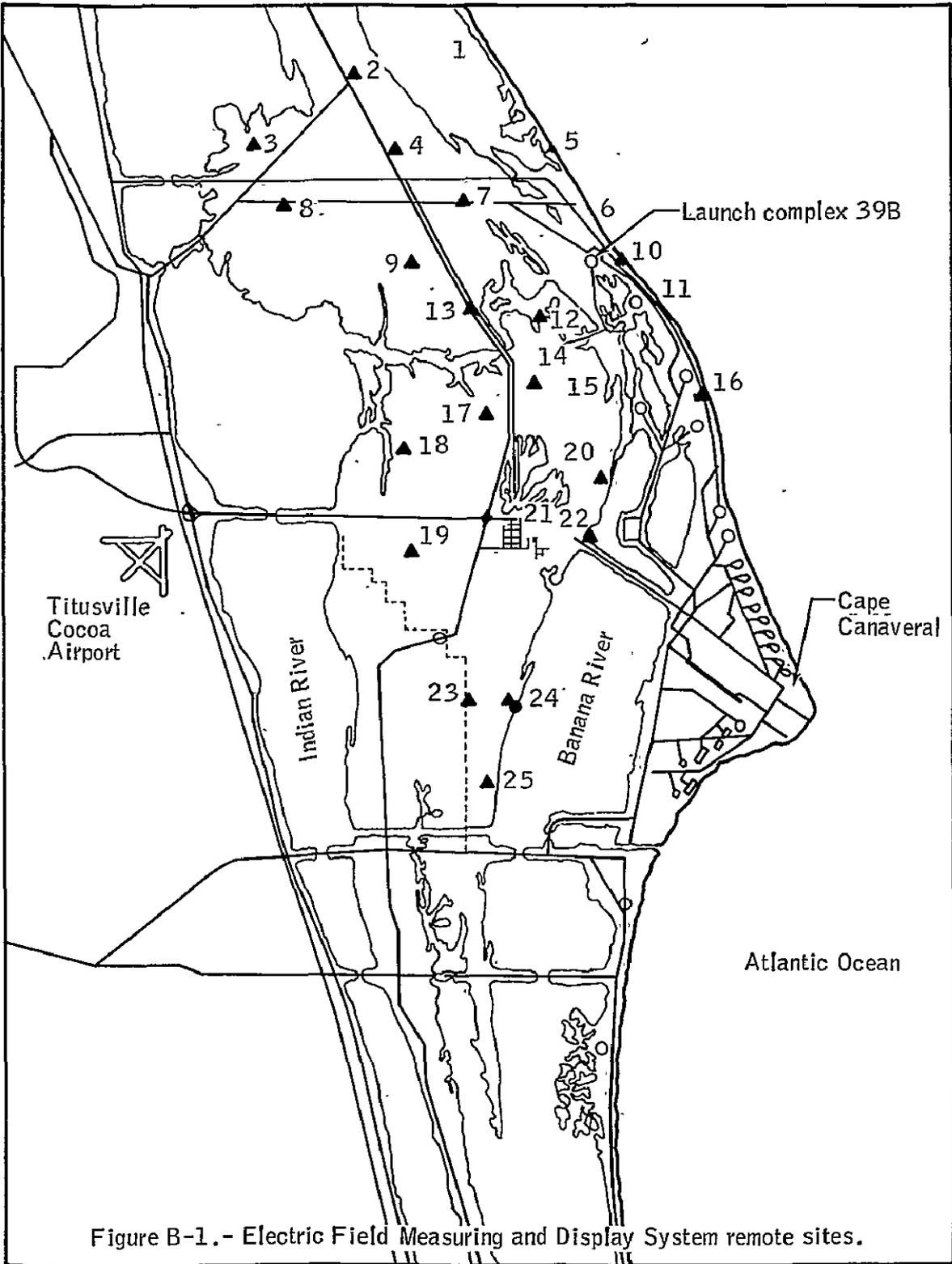


Figure B-1.- Electric Field Measuring and Display System remote sites.

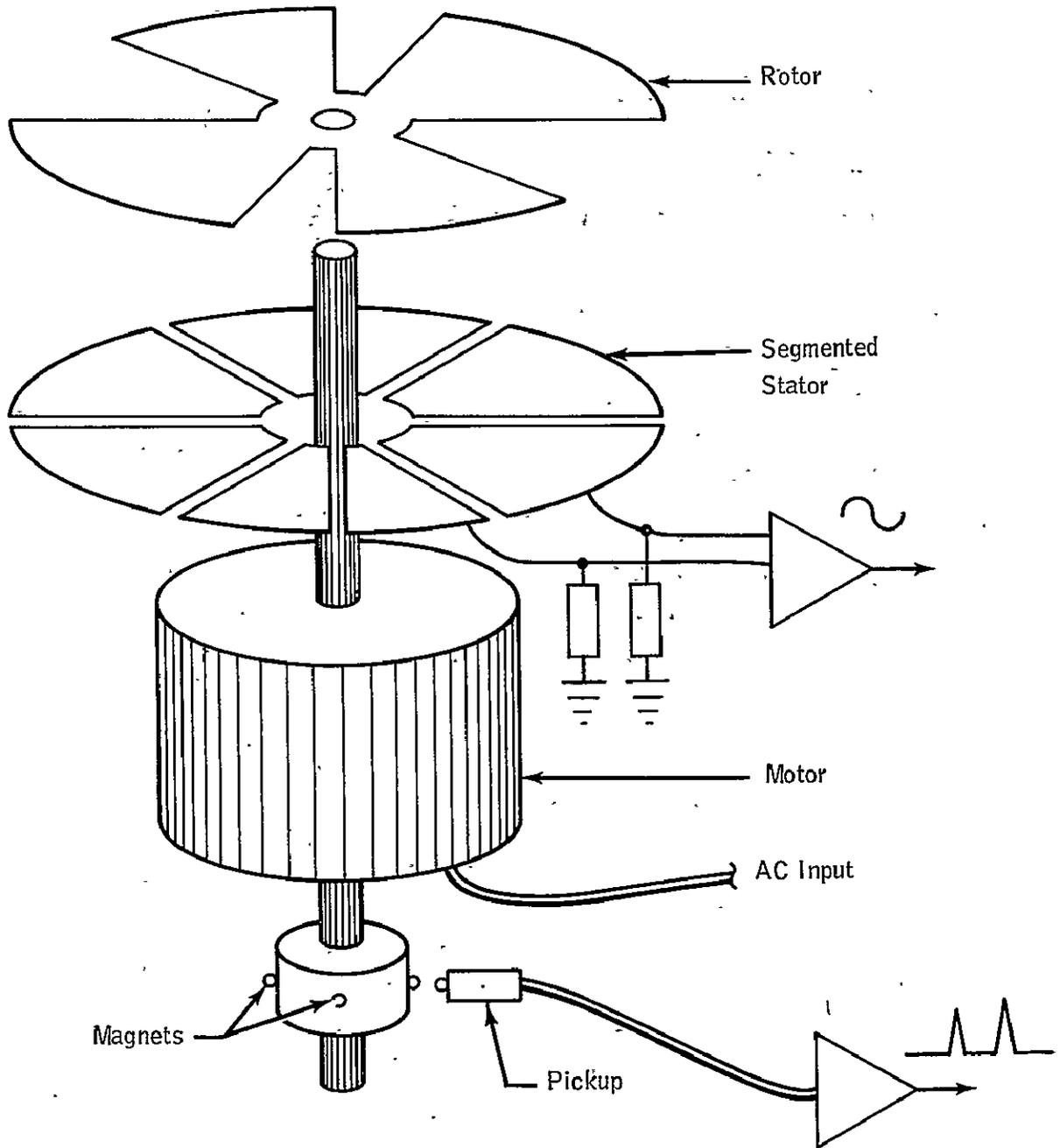


Figure B-2.- Field mill.

The electric field readings obtained with the field mills and the NASA 6 aircraft play an important part in the adverse weather warnings issued by the KSC weather office. For long range monitoring of weather systems at or approaching KSC, two weather surveillance radars are used - an FPS 77 unit with a range of 370 kilometers located at Patrick Air Force Base, and a WSR-57 unit with a range of 460 kilometers located at Daytona Beach, Florida. For local KSC monitoring, a WSR 72X radar with a range of 55 kilometers is used. The display of the FPS 77 radar is available at the KSC weather office through a closed-circuit television loop. The radars are used principally to determine cloud location, size, movement, and rate of growth.

Depending upon these data, and when the field mill readings start increasing above the fair weather values (100 to 300 volts per meter positive) to approximately 2000 volts per meter, conditions exist that are conducive to cloud-to-ground lightning. Adverse weather warnings are issued before the ground field readings exceed 3000 volts per meter. If the clouds are high, then the electrification measured by the ground field mills becomes less reliable and the NASA 6 aircraft is dispatched to measure the electric fields aloft. Large and fluctuating fields aloft are of concern during launch because of the lightning triggering possibility. The NASA 6 aircraft is also used outside the KSC area to monitor developing clouds that are approaching the KSC area.

A research and development Lightning Detection and Ranging System is also being developed for future use. The system operation is based upon measuring the precise time-of-arrival of r-f emission from lightning discharge channels in both azimuth and elevation. The range to the discharge is determined by processing the time-of-arrival with a minicomputer which provides the inputs to transient recorders. These data are then transmitted to remote displays and, thus, provide location of clouds with electrical activity.

Lightning Data Acquisition System

The lightning data acquisition systems for Apollo, Skylab and the Apollo Soyuz Test Project were closely associated with the lightning protection systems of the launch pad structure. There was a lightning mast atop the mobile service structure and, for the Apollo Soyuz Test Project, a special launch tower lightning protection system was installed to protect the mobile launcher. This system consisted of a catenary steel cable that was supported above the mobile launcher by a 25-meter fiberglass tower (fig. B-3). The cable was anchored to earth at two places 275 meters from the structure base. The lightning peak current and the current waveform was measured at the anchor points. The output of a current transformer around the cable grounding wire was fed to a transient recorder which obtained the digital equivalent of electrical waveforms. Currents surging through current transformers located on the ground down wires were transformed into voltages and fed to individual memory voltmeters and recorded on strip charts.

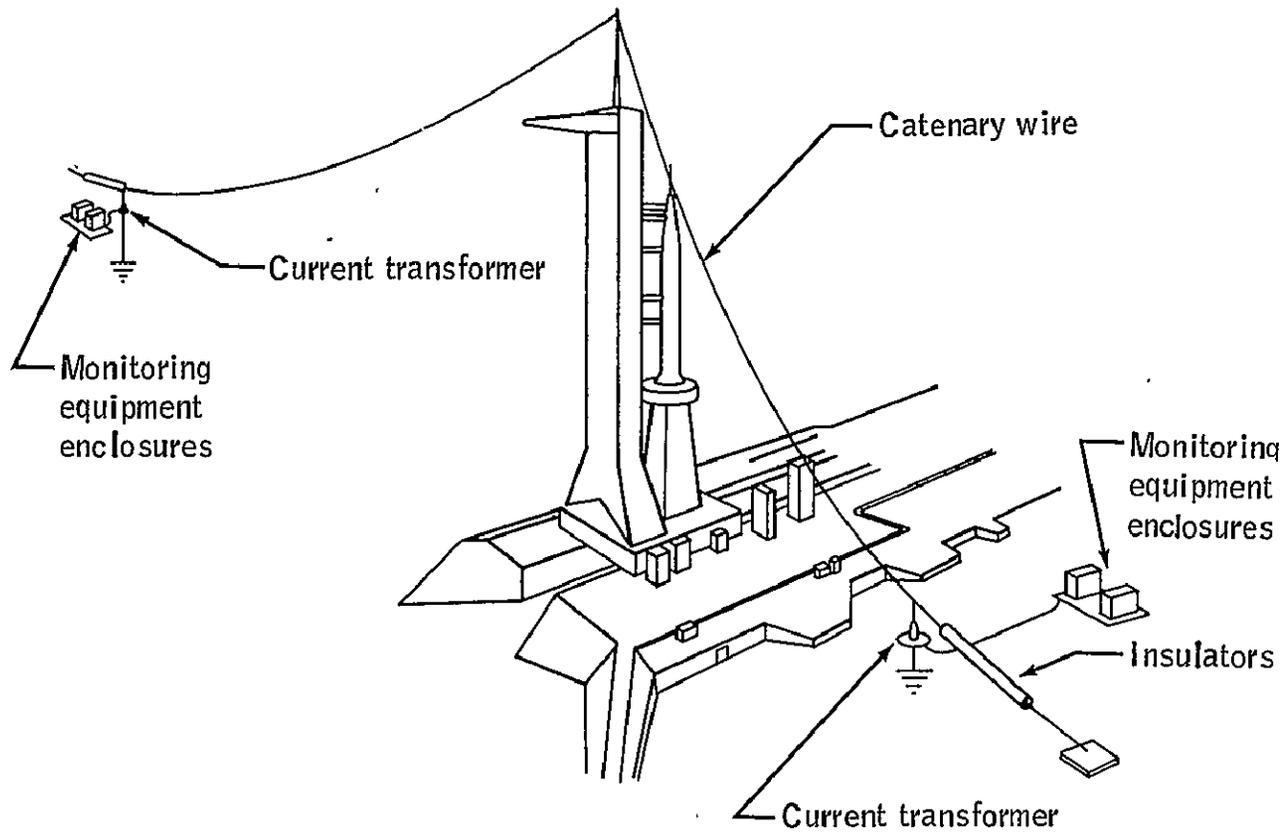


Figure B-3.- Launch Complex Lighting Protection Systems for the Apollo Soyuz Test Project.

The memory voltmeter is a wideband solid-state voltmeter having amplitude memory. This instrument measures voltages ranging from dc to rf, holding the maximum amplitude indications of continuous voltages or spiked transient pulses as short as 50 nanoseconds. The sensed voltage is held in electronic circuitry and indicated on a meter until reset by the operator. Amplitudes greater than those stored cause the indications to increase. The instrument indications are of the maximum amplitude detected between reset periods. This memory voltmeter has an automatic reset circuit that is adjustable over a wide range with an automatic recovery time of 5 seconds arbitrarily selected.

Another method of monitoring lightning strikes to launch-associated structures is by the use of 1.27 by 0.95 cm cobalt alloy laminated slugs. These slugs (called magnetic links) retain residual magnetism induced by the current flow of a nearby lightning strike. The magnetic links were mounted on the highest points of structures and weather towers and on both the high and low ends of the pad slide-wire escape system at Launch Complex 39. When installed, the slide-wire itself acted as the conductor for lightning strikes. Magnetic links were located on wooden arms or masts on each mobile launcher and on the crane cabs. Three magnetic links were inserted in drilled holes on the side of the arm assembly and retained by phenolic tabs. They were mounted 12.7, 25.4 and 60.9 cm from the conducting structure for exposure to magnetic fields generated by lightning strikes (fig. B-4).

The magnetic links are magnetized when lightning strikes the structure near them. The flux density is dependent on the distance of each link from the centerline of the lightning rod, the magnitude of the current induced by the strike, and the magnetic characteristics of that particular batch of links. Residual magnetism in a link is measured with a unigalvanometer. A calibration curve supplied by the manufacturer is then used to convert the unigalvanometer indication to the peak kiloampere current flow in the lightning strike. The properties of the links are such that they can measure currents from 3000 to 100 000 amperes. In the field, an ordinary pocket compass held near each link is enough to determine if a link has been magnetized.

An automatic lightning photography system augments the measurement system. Three cameras around the periphery of the launch area (Complex 39) photograph lightning strikes. Each camera station of the system consists of a motorized Nikon camera with a 180° (fisheye) lens and a triggering circuit as shown in figure B-5. The lens is aimed such that a horizon-to-horizon picture in all directions is obtained. The camera systems are self-contained giving complete freedom of placement. The cameras are routinely serviced once a week with a film load of 250 frames. Camera performance figures include a shutter opening of 40 milliseconds, exposure time of 0.5 second, and a reset interval of 1 second. The electric field pulse from a lightning strike triggers the electronic circuitry, which trips the camera shutter and photographs the lightning. The film then automatically advances and the camera is automatically reset for the next lightning strike. Although the leader lightning strike is not photographed, succeeding strikes which immediately follow the leader's path were photographed. This method permits photographing 90 percent of the cloud-to-ground flashes. The photographs are used to locate the lightning impact point, either by showing actual location or by triangulation.

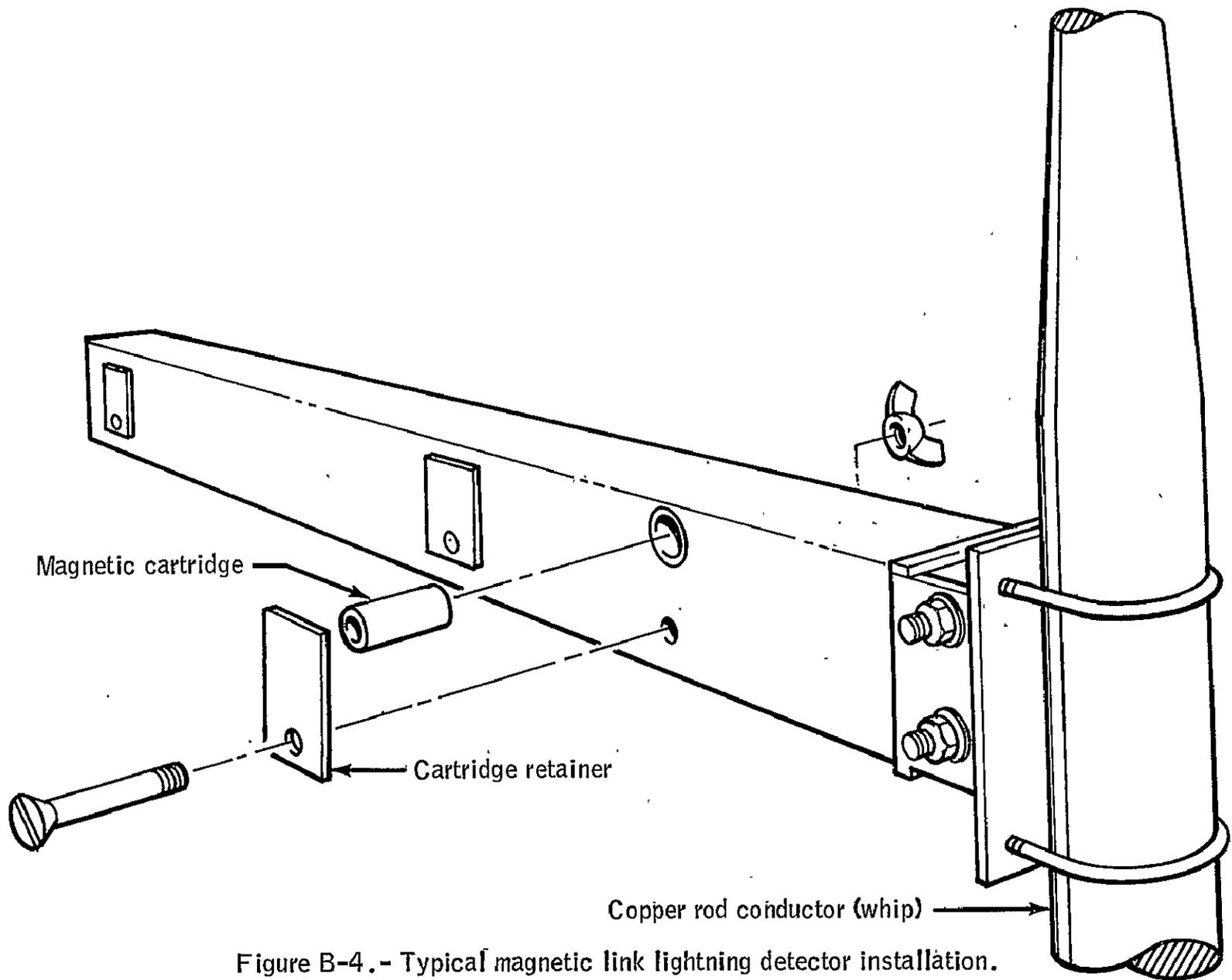


Figure B-4.- Typical magnetic link lightning detector installation.

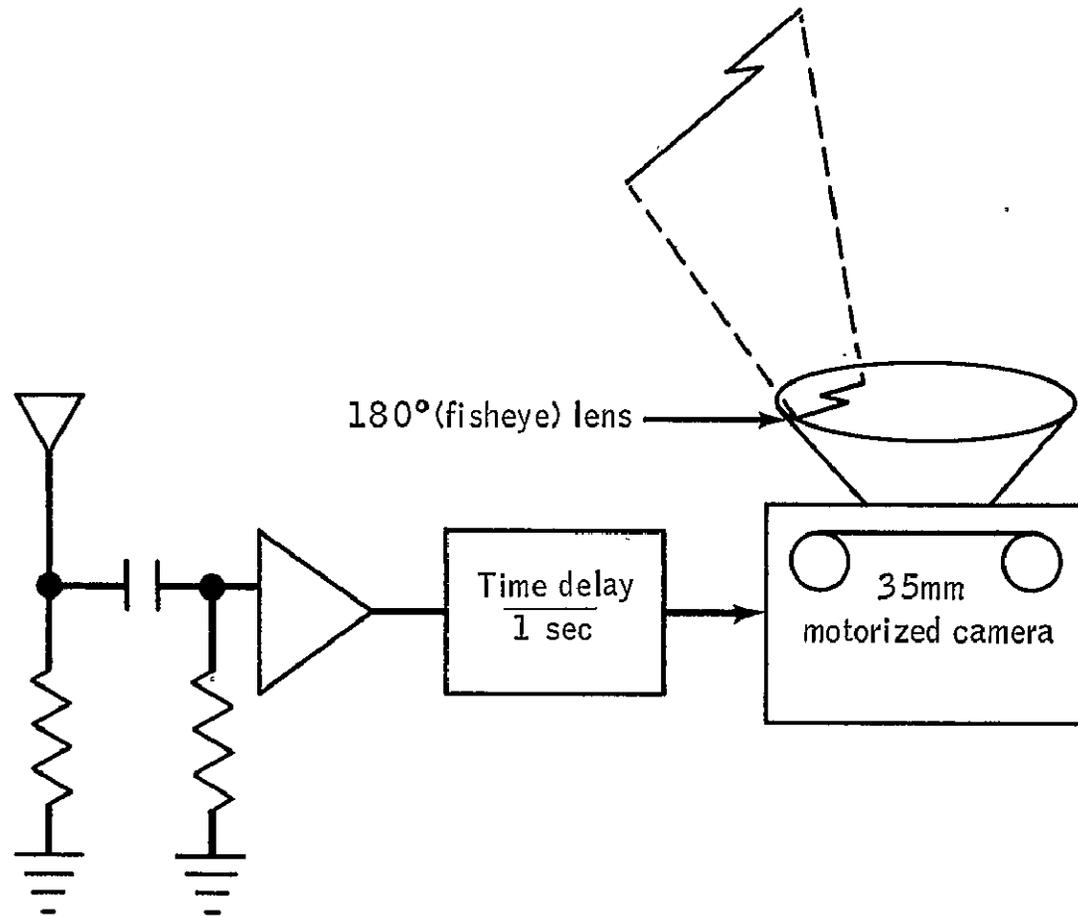


Figure B-5.- Automatic motorized camera.

To fully assess the damage to the vehicle and supporting ground equipment caused by a lightning strike to or near the launch facility, it is important to know the strike impact location, current waveshape, and peak magnitudes. The cameras provide a gross indication of the strike point. The readings obtained with the various magnetic links, while giving the peak currents, also help to locate the strike point. The memory voltmeters provide voltage levels on key spacecraft and ground equipment circuits from which the induction on other circuits is estimated. From the lightning waveform monitoring equipment, the current rate-of-rise is determined and is used to determine gross induction levels. These indications coupled with spacecraft and ground support system downlink telemetry measurements are used to determine if anomalous conditions exist in the spacecraft and ground equipment systems.

APPENDIX C

MEASUREMENT OF ELECTRIC FIELDS FROM AIRCRAFT

The airborne electric field meters that were available for the Apollo-Soyuz field measuring activity were of two basic types. Three aircraft were equipped with a cylindrical type that employed a rotating sense plate. The other five aircraft were equipped with a fixed-plate type with a grounded rotating shutter like the ground instruments described in appendix B. The airborne system discussed in this appendix used the fixed sense plate type.

The measurement of electric fields aloft is complicated by certain factors. The conductive surface of the aircraft distorts the ambient electric field such that its direction is normal to the aircraft surface (fig. C-1). This must be considered in the design of the measurement system so that the ambient field can be reconstructed from the field components measured at the surface of the aircraft. To do this, the Cartesian components of the ambient field are measured with three orthogonally oriented field meters. These are located such that each will respond strongly to one field component but will have a minimum response to the other two components (fig. C-2).

Another factor to be considered in the design of the airborne system is the electric field caused by the self charge of the aircraft. This field is superimposed upon the ambient field when measured at the aircraft surface. The effect is determined by measuring the aircraft potential with a fourth field meter and applying an enhancement factor. Thus, there are four quantities to be measured; the ambient field components E_x , E_y , E_z , and the potential, V_a , of the aircraft. Figure C-3 is a matrix describing the relationship of these four unknowns. The coefficients a , b , and c are constants associated with the location and orientation geometry of each field meter. The constant d is a factor indicating the relative enhancement of the measured electric field due to the self charge on the aircraft.

These constants are determined empirically by using a metallic model of the aircraft in an electrostatic cage (fig. C-4). Electrostatic probe measurements are taken while rotating the model in pitch, yaw and roll. These data are then used to determine the constants for the inverted matrix (fig. C-5) that gives the ambient field components and the aircraft self charge enhancement factor in terms of the field meter readings. This procedure thus calibrates the field meters to the aircraft.

In the airborne systems, the ambient field components are obtained by feeding the field sensor outputs to an electronic analog processor (fig. C-6). These quantities are then recorded on strip charts for future use. The outputs of the field sensors can be recorded before processing, if desired.

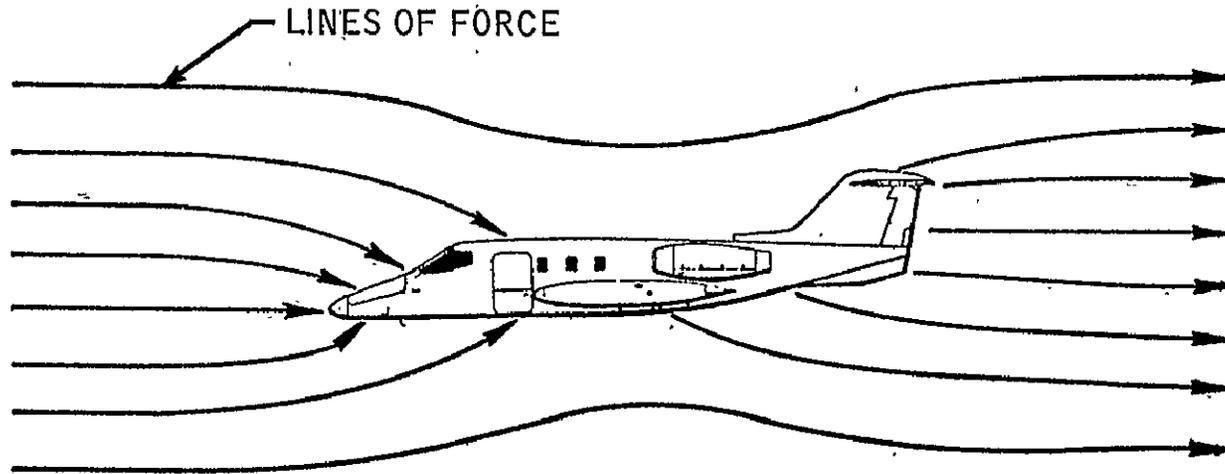


Figure C-1.- Distortion of uniform electric field caused by an aircraft.

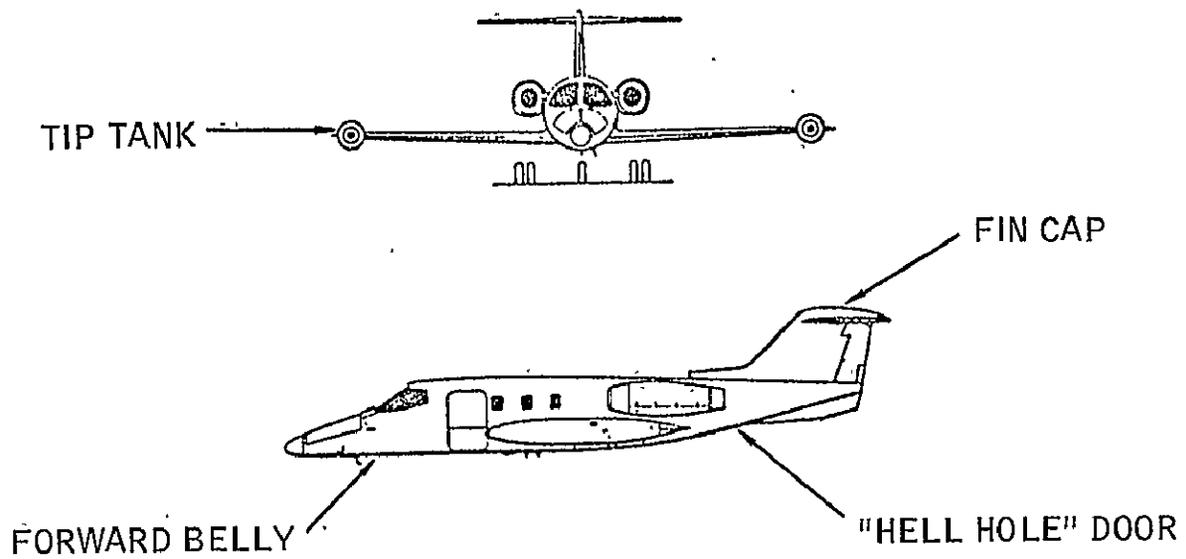


Figure C-2.- Location of field meter sensors on test aircraft.

$$\begin{aligned} E_{FIN} &+ a_1 E_x + b_1 E_y + c_1 E_z + d_1 V_a \\ E_{AFT} &+ a_2 E_x + b_2 E_y + c_2 E_z + d_2 V_a \\ E_{FWD} &+ a_3 E_x + b_3 E_y + c_3 E_z + d_3 V_a \\ E_{WING} &+ a_4 E_x + b_4 E_y + c_4 E_z + d_4 V_a \end{aligned}$$

Figure C-3.- Matrix of field calibrations on test aircraft.

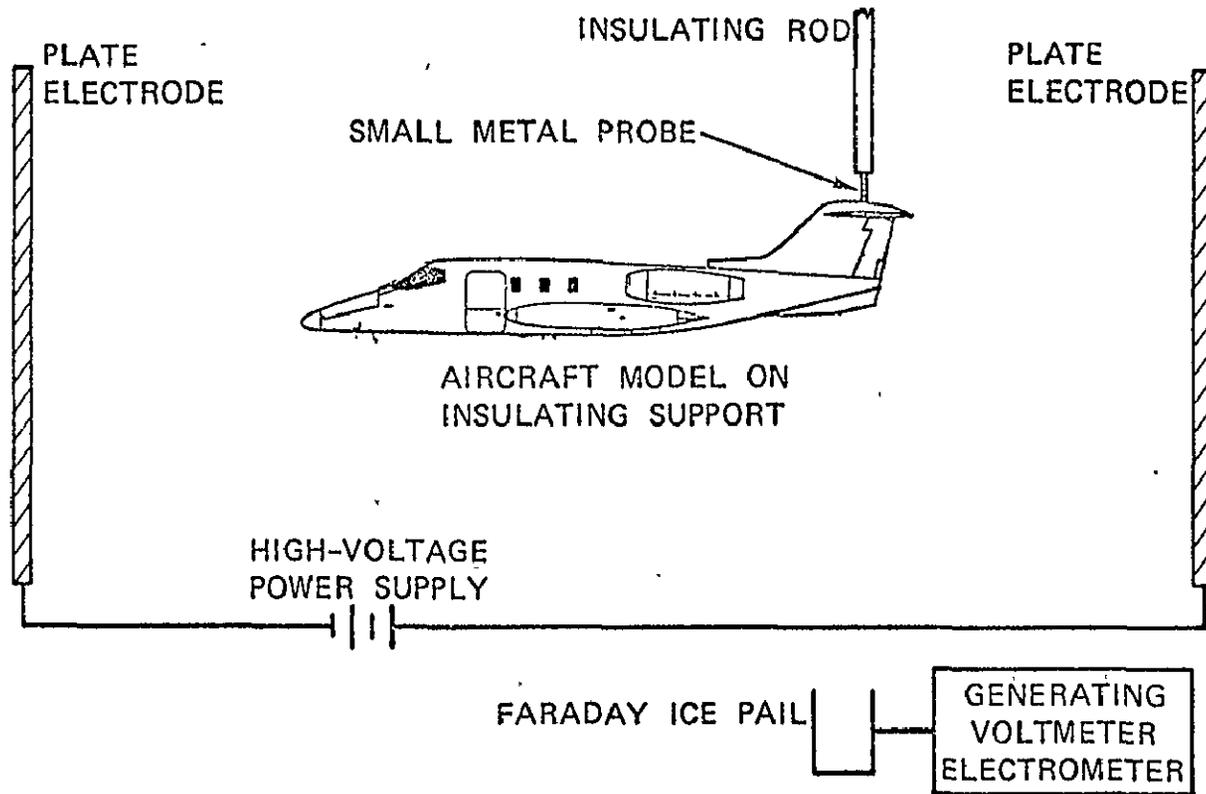


Figure C-4.- Electrostatic cage used for aircraft calibration.

$$\begin{aligned}E_x &= 0.012 E_{FIN} - 0.52 E_{AFT} + 0.56 E_{FWD} + 0.00 E_{WING} \\E_y &= -0.14 E_{FIN} + 0.36 E_{AFT} - 0.63 E_{FWD} + 0.11 E_{WING} \\E_z &= 0.180 E_{FIN} - 0.48 E_{AFT} + 0.21 E_{FWD} + 0.00 E_{WING} \\V_a &= -1.15 E_{FIN} + 0.86 E_{AFT} - 3.10 E_{FWD} + 0.00 E_{WING}\end{aligned}$$

Figure C-5.- Inverted matrix of aircraft field meter calibration.

C-7

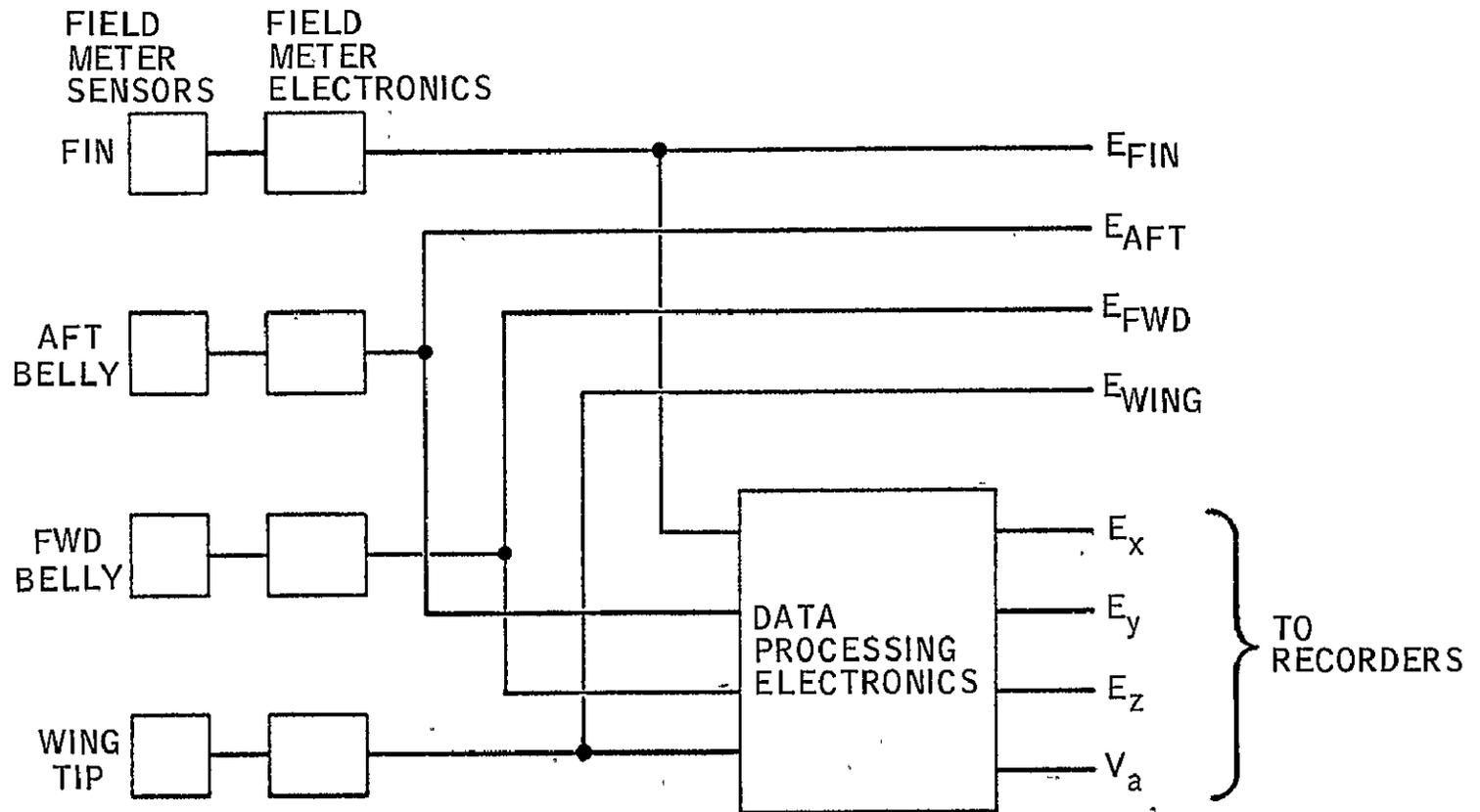


Figure C-6.- Block diagram of aircraft field measuring electronics.

The airborne system that has been discussed was used on the NASA Lear Jet for cloud electrification studies in the summer of 1975. The aircraft possessed some important characteristics (fig. C-7) that allowed an acquisition of unique data on the electrical properties of developing thunderstorms. Of particular importance was the service ceiling of 13.7 kilometers and the climb rate of 2073 meters per minute. These characteristics made possible the measurement of electric fields around the storm cell during development of the turret (main vertical body) and subsequent anvil.

Isolated storm cells were selected so that the measured electric fields would not be affected by nearby storms, to avoid unsafe flying conditions, and to insure a level flight attitude when taking data. The position of isolated cells could also be more readily fixed with ground-based radar when available. The aircraft was flown at various altitudes around the thunderstorm while recording electric field data. Several passes, with different headings, were flown for each altitude (fig. C-8).

The onboard weather radar provided an indication of the aircraft position relative to the cloud. During each pass the strip charts were annotated with time hacks indicating the start, midpoint, and end of the run. The aircraft position coordinates, airspeed, heading, altitude, and attitude (if not horizontal flight) were also noted. In addition, the unusually high field levels were flagged. These flags help to establish priorities for subsequent data reduction activity.

Postflight, the data were used to establish ground tracks from which the electric field vectors along the flight path of the aircraft could be ascertained. A summary of some of the data is shown in figure C-9. The electric field vectors that were calculated for two passes over an anvil at the 12.5-kilometer altitude is shown in figure C-10. It can be seen that charts such as this, showing the electric field vectors for each altitude, can provide a fairly complete description of the electric fields surrounding various cloud types.

From a limited analysis of the statistically small amount of data taken by the Lear Jet, an interesting observation is made. It appears that the high electric fields observed in the vicinity of anvils are most likely associated with charge concentrations in the thunderstorm main vertical body. Additional data and further analysis of existing data will be necessary to confirm that this is indeed the case.

SERVICE CEILING	13.7 km (45,000 ft)
CRUISING SPEED	239 meters/sec (464 knots)
RATE OF CLIMB	2073 meters/min (6800 ft/min)
ACCOMMODATIONS	Three passengers + instrumentation
CABIN PRESSURE	Sea level below 7010 meters (23,000 ft) aircraft altitude
CREW	Pilot, Copilot
CABIN ACOUSTIC NOISE	Low
VIBRATION LEVEL	Low

Figure C-7.- Some important characteristics of Lear Jet Aircraft.

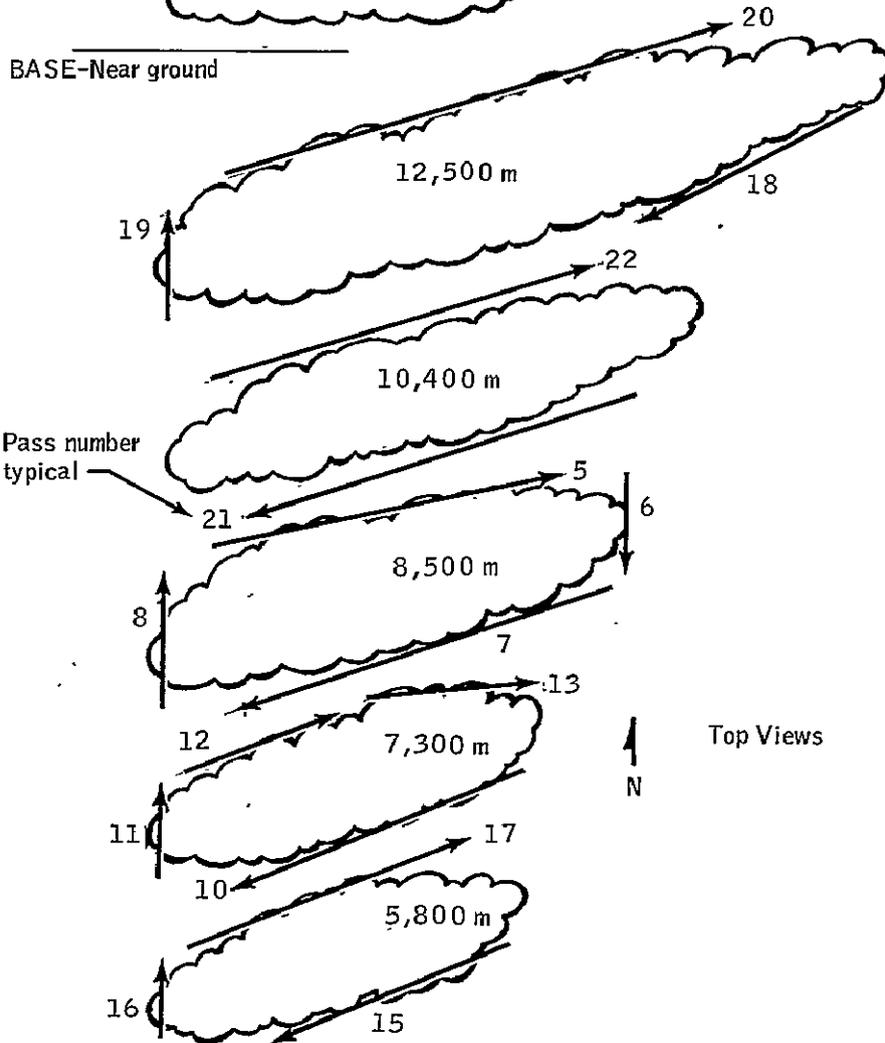
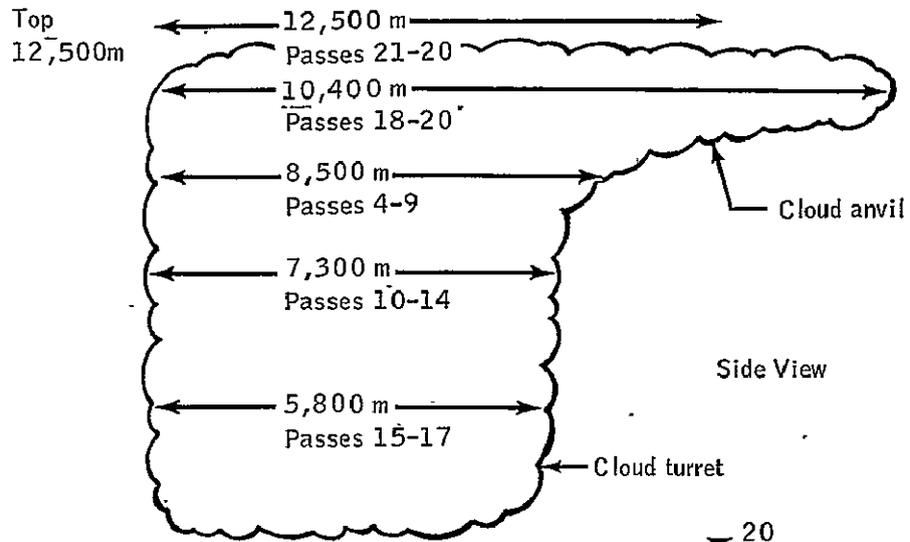
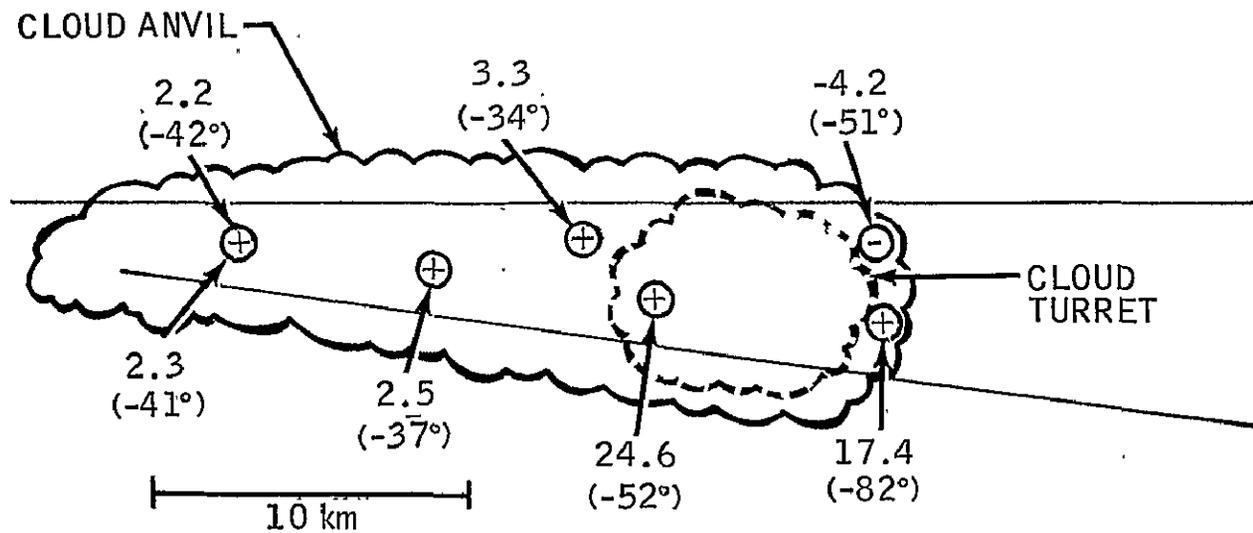


Figure C-8.- Flight path of aircraft around storm cell.

DATE	HEIGHT (km)	PASSES	MAXIMUM FIELD (kV/m)				COMMENT
			E	E _Z	E _X	E _Y	
7/17/75	9.1	1	Q-mill	-21	+29	-29	Building cloud; at anvil height
	10.1	1	Q-mill	-8	+10	-5	
7/21/75	8.8	3	47	+41	-23	-4	Under anvil
	10.4	3	33	-31	-6	-11	In anvil at times
	11.9	3	25	-21	-8	-11	Anvil middle
	13.1	3	17	-17	-1	+0.3	Near top of anvil
7/22/75	10.4	3	31	-30	+4	-8	Skirting anvil base
	12.5	2	38	-38	0	0	Over anvil top
7/22/75	12.5	2	3.4	-2	+2	-1.8	Data? (trapped charge) Over anvil top
7/25/75	5.5	2	2.7	+2	-1.4	+0.3	Building cloud with turret and anvil
7/26/75	4.9	2	15	+10	-11	+3	Under pseudo-anvil
	6.4	1	51	+50	0	-8.3	Thru side-pseudo anvil

Figure C-9.- Electric field levels in and around thunderstorm anvils.



Note.

Top number is electric field vector magnitude in kilovolts/meter

Number in parenthesis is the vector dip angle

Figure C-10.- Electric field vectors above an anvil at 12.5 km altitude.