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SATELLITE SYSTEMS DATA BOOK

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DISCOVERER PROGRAM

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LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION
SUNNYVALE, CALIFORNIA

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LMSD-6164B
Revision 1
1 June 1961

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INSERT				REMOVE			
PAGE NO.	REV.	DATE	CLASS	PAGE NO.	REV.	DATE	CLASS
Title Page	B-1	6/1/61	S	Title Page	B	5/15/61	S
c-1	B-1	"	U	c-1	B	"	U
SECTION I							
*3. 6. 7	B	5/15/61	C	3. 6. 7	B	5/15/61	C
3. 6. 9	B-1	6/1/61	C				
3. 6. 11	B-1	"	C				
3. 6. 13	B-1	"	C.				

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LMSD-6164B
15 May 1961

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REPORT CHANGE RECORD
FOR
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The following additions, revisions, or errata corrections, should be incorporated into the document identified above. This Report Change Record page should be inserted as the first page of the affected report preceding the title page. If a page in the original document is eliminated and/or replaced by the instructions which follow, the page must be destroyed according to the Air Force directive governing such destruction.

CONTRACT NUMBER

ADDENDUM PAGE	REVISION		ERRATA		REVISION OR ERRATA CORRECTION (CORRECT IN INK)	CORRECTION MADE	
	REMOVE PAGE	INSERT PAGE	REMOVE PAGE	INSERT PAGE		INITIAL	DATE
					Section I, p. 6.6.6, Change LMSD-6165B to read LMSD-6164B		
					Section II, p. 1.6.7, Change Section I to read Section II		

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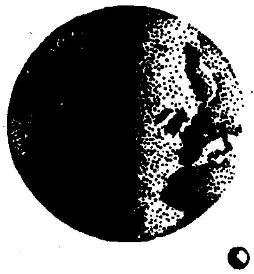

SATELLITE SYSTEMS DATA BOOK

DISCOVERER PROGRAM

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INSERT				REMOVE			
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The material contained in this revision represents a complete replacement of material presently appearing in the Discoverer Data Book. Remove all pages in the current data book and replace with the attached pages.							

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INTRODUCTION

The SATELLITE SYSTEMS DATA BOOK is a collection of documents, each of which functions as an authoritative reference source for a particular Satellite Systems Program as indicated below. Each document's binder carries the title "SATELLITE SYSTEMS DATA BOOK" and the sub-title applicable to the binder's content material. Each Program data book carries on its title page, but not on the binder thereof, the LMSD report number assigned to the document as noted below.

Sub-title	LMSD Report No.	Content Material
Discoverer Program	6164	Program, flight, and design parameters and descriptions of Discoverer Satellite.
MIDAS Program	6165	Program, flight, and design parameters and descriptions of MIDAS Satellite and Ground Data Handling Equipments.
Samos Program	445641	Program, flight, and design parameters and descriptions of Samos Satellite and Ground Data Handling and Recovery/Retrieval Equipments.
NASA/Agena Program	447000	Program, flight, and design parameters and descriptions of NASA/Agena vehicle and associated LMSD activities.
Advent/Agena Program	448168	Program, flight, and design parameters and descriptions of Advent/Agena space vehicle and associated LMSD activities
Samos II Program	448169	Program, flight, and design parameters and descriptions of the Agena vehicle and associated LMSD activities in support of the Samos II Program.
Terms and Definitions	447376	Terms encountered in the space effort and definitions thereof; charts and tables of technical data applicable to the space effort.

Each SATELLITE SYSTEMS DATA BOOK for a specific Program is maintained on a continuing basis for the life of the Program to reflect the "as is" condition of the flight parameters and of the space and ground hardware. To implement this maintenance, each data book will be re-issued periodically, such re-issuances to be identified as a lettered change, e.g. LMSD-6165A. Revised data will be issued between lettered changes, with instructions as to the insertion of new pages and the deletion of any previously issued pages which might no longer be applicable and/or effective. These pages are identified as numbered revisions to the current change, e.g. LMSD-6165A, Revision 1. A succeeding lettered re-issuance will incorporate the interim numbered revisions as well as new material.

It is to be noted that any data, information, or illustration contained in any SATELLITE SYSTEMS DATA BOOK document for a specific Program is solely for internal reference purposes and does not by its presentation therein, constitute a contractual obligation or commitment.

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INTRODUCTION

The SATELLITE SYSTEMS DATA BOOK - DISCOVERER PROGRAM, LMSD-6164, is a reference document for the Discoverer Satellite System, including the overall System objectives, the flight and design parameters centered in the Discoverer satellite, the delineation of the inter-related structure and systems of the satellite, and a description of the ground handling equipment applicable to the Biomedical Recovery Capsule (BRC).

The SATELLITE SYSTEMS DATA BOOK - DISCOVERER PROGRAM will be revised on a continuing basis to maintain it concurrently with the overall Program philosophy, the flight parameters, and the design configurations. Major revisions will be disseminated periodically as "re-issues," identified as a lettered change. Revised data will be issued between lettered changes, with instructions as to the insertion of new pages and the deletion of old pages. These revisions are numbered, e.g. "Rev. 1, Rev. 2, Rev. 3, Rev. 4," etc. A succeeding lettered change will incorporate the interim revisions as well as any new material. It is the intent of the Data Book Staff that the SATELLITE SYSTEMS DATA BOOK - DISCOVERER PROGRAM contain, at any given time, only material pertinent to current and projected Discoverer satellites and their affiliated equipments. Since LMSD Report No. 6164 is an "unbound" document, each page thereof is classified with the degree of security determined by the content of the page.

The SATELLITE SYSTEMS DATA BOOK - DISCOVERER PROGRAM, LMSD-6164, is divided into major sections of material, which are then subdivided into pages. The first digit of a page number indicates the major grouping within the section, such as "Missions and Objectives" or "Spaceframe;" the second digit of a page number indicates the flight configuration to which the material on that given page refers. The third digit of the page number is the sequential number of that page for a given write-up or description, which page may contain either textual material, illustrative material, or both. An example from the Table of Contents follows:

SECTION II

1.0	STRUCTURE AND SYSTEMS, SPACE VEHICLE
1.3	Spaceframe
1.3.1	Flight Configuration III
1.3.2	General Arrangement
1.3.3	Inboard Profile
1.3.4	*
1.3.5	Fwd. Equipment Racks
	*

* Text Material

It is intended that each write-up and/or description be a self-contained unit; therefore, a minimum of cross referencing is used.

To assist the Data Book user in locating specific system, subsystem, component, or other descriptions, a general index is provided at the back of the book and detailed indexes are included at the end of each write-up. The general index contains the basic write-up number in which given reference material may be located. Each such basic write-up is subdivided in accordance with the flight configurations, as indicated above, and these individual write-ups have the detailed indexes appended as the last paragraph in each write-up. An example of this somewhat unusual indexing follows:

GENERAL INDEX (main - at back of Data Book)

. . . . .
Meter, Velocity — Section II, Page 4.0 ff
. . . . .
Spin Gas Bottle — Section II, Page 8.0 ff

"INDEX" (supplemental - at back of individual write-ups)

4.6 Guidance and Control, Flight Configuration VI

. . . .

8.0 INDEX

Velocity Meter, Mod II — Para. 3.1.3

It is to be noted that the SATELLITE SYSTEMS DATA BOOK - DISCOVERER PROGRAM, LMSD-6164, is an internal information document, and any data or information contained herein does not, by that fact, constitute a contractual obligation or commitment.

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104	A. N. Allred, Maj.		AFSSD-WDZY-2	
105	F. S. Buzard, Maj.		AFSSD-SSZD	
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SATELLITE NUMBER / Flight Configuration	DISC. NUMBER	DATE	Did Satellite attain orbit?	VELOCITY Feet/Sec.	APOGEE Naut. Miles	PERIGEE Naut. Miles	ECCEN-TRICITY	PERIOD Minutes	USEFUL ORBITAL LIFE	ACTIVE ORBITAL LIFE	Was capsule recovered?	REMARKS
3205-1019 FC I	*	1/21/59	No	-	-	-	-	-	-	-		Instrumented payload. Malfunction during countdown caused ullage rockets, retro-rockets, separation bolts, and horizon scanner fairing pin pullers to fire when hydraulic motor was turned on. Launch was aborted.
3205-1022 FC I	I	2/28/59	Yes	N/A	N/A	N/A	N/A	N/A	-	13	N/R	Instrumented payload. No telemetry or radar orbit contact made. Sporadic CWAT contact reported. Satellite believed to have been damaged structurally and/or thermally at injection or during first pass.
2205-1018 FC II	II	4/13/59	Yes	25,850	195.3	135.4	0.008	90.5	N/A	5-28	No	Biomedical research capsule containing four mechanical mice. Engine shutdown by command (source unknown, but believed to be due to relay malfunction). Capsule ejected and believed to have landed near Spitzbergen, Norway; was not recovered (by us).
2205-1020 FC II	III	6/3/59	No	-	-	-	-	-	-	-	No	Biomedical research capsule, containing four live mice. Premature engine burnout due to fuel exhaustion resulted in insufficient satellite velocity to attain orbit.
2205-1023 FC IV	IV	6/25/60	No	-	-	-	-	-	-	-	No	AET Payload. Premature engine burnout, resulting in insufficient velocity to attain orbit.
2205-1029 FC IV	V	8/13/59	Yes	26,448	399.3	118	0.0373	94.1	N/A	46	No	AET payload. Burnout due to propellant exhaustion. Orbit achieved. Capsule separated but not recovered. Unexplained internal heat loss affected mercury battery; recovery sequence not accomplished.
2205-1028 FC IV	VI	8/19/59	Yes	26,079	454.8	122.4	0.047	95.3	N/A	63	No	AET payload. Burnout due to propellant exhaustion. Orbit achieved. Capsule separated but not recovered. Recovery sequence believed not accomplished. Radar interference experienced on VAFB tracking net. Same thermal problems as in Discoverer V.
2205-1051 FC IV	VII	11/7/59	Yes	26,235	379.3	104.2	0.049	94.65	2	15	No	AET payload. An apparent failure of the satellite 400-cycle power supply occurred during the first orbit, resulting in the disablement of the satellite guidance system, which includes the D-timer which controls the capsule ejection sequence. The satellite was assessed to be tumbling throughout its orbital passes.

N/A - not available
N/R - not required

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SATELLITE SYSTEMS DATA BOOK

SATELLITE NUMBER / Flight Configuration	DISC. NUMBER	DATE	Did Satellite attain orbit injection?	VELOCITY Feet/Sec.	APOGEE Naut. Miles	PERIGEE Naut. Miles	ECCEN-TRICITY	PERIOD Minutes	USEFUL ORBITAL LIFE	ACTIVE ORBITAL LIFE	Was capsule recovered?	REMARKS
2205-1050 FC V	VIII	11/20/59	Yes	26,951	909	100	N/A	103.7	N/A	90+	No	AET payload. Burnout due to propellant exhaustion following accelerator-integrator malfunction. Excessive injection velocity resulted in a relatively high eccentricity and an abnormally long period. Programmer was adjusted on Pass 2 to allow for this longer period. Separation occurred; re-entry sequence was normal. Recovery was not effected.
2205-1052 FC V	IX	2/4/60	No	-	-	-	-	-	-	-	-	AET payload. Satellite's helium quick-disconnect failed to function properly, resulting in separation of part of the fitting and loss of helium gas for propellant pressurization. Booster MECO occurred approximately 19 seconds early at 145.06 seconds after liftoff. Satellite's pitch gimbal actuator failed to function properly with resultant uncontrolled tumbling of satellite shortly after satellite engine start.
2205-1054 FC V	X	2/19/60	No	-	-	-	-	-	-	-	-	AET payload. Booster began to oscillate in the pitch plane immediately after liftoff. Discoverer was destroyed by ground command 56.36 seconds after liftoff.
2205-1055 FC V	XI	4/15/60	Yes	26,015	329	91	0.032	92.32	2 days	11 days	No	AET payload. Booster and satellite performance most successful to date. Capsule separated but was not recovered. Effective retro velocity of capsule was determined to be approximately half the predicted value for proper re-entry, due to capsule spin deficiency.
2205-1053 FC VII	XII	6/29/60	No	-	-	-	-	-	-	-	-	Diagnostic payload. Launch and ascent were normal except that due to a malfunction in the pitch channel of the satellite horizon scanner, the satellite arrived at burnout with an attitude that resulted in a flight-path angle of -8.3 degrees. The satellite failed to achieve orbit and impacted downrange of the launch facility.
2205-1057 FC VII	XIII	8/10/60	Yes	25,786	379	137	0.0326	94.1	2 days	84 days	Yes	Diagnostic payload. Very successful operation, culminating in the recovery of a space capsule. New cold gas spin system utilized for capsule spin-de-spin. Capsule was retrieved from the water by helicopter from USNS Haiti Victory.

N/A - not available
N/R - not required

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SATELLITE NUMBER / Flight Configuration	DISC. NUMBER	DATE	Did Satellite attain orbit injection?	VELOCITY Feet/Sec.	APOGEE Naut. Miles	PERIGEE Naut. Miles	ECCENTRICITY	PERIOD Minutes	USEFUL ORBITAL LIFE	ACTIVE ORBITAL LIFE	Was capsule recovered?	REMARKS
2205-1056 FC V	XIV	8/18/60	Yes	26,126	441	103.5	0.046	94.54	1-1/2 days	30 days	Yes	AET payload. Performance was good except for instability for a period after injection and prior to capsule ejection. The latter resulted in an impact over 430 miles south of that predicted. The first in-flight recovery of a space capsule was achieved by a C-119 aircraft.
2205-1058 FC V	XV	9/13/60	Yes	26,015	416	114.2	0.040	94.2	1 day	44 days	No	AET payload. Satellite attitude perturbations resulted in control gas depletion prior to reorientation for capsule ejection. Capsule ejected, however, and was located 960 miles southeast of predicted impact point. Inclement weather prevented capsule retrieval before capsule sank.
6205-1061 FC VI	XVI	10/26/60	No	-	-	-	-	-	-	-	No	AET payload. Functions controlled by D-timer after satellite-booster separation did not occur. Satellite failed to attain orbit.
6205-1062 FC VI	XVII	11/12/60	Yes	26,280	538	103.1	0.0582	96.44	2-1/2 days	28 days	Yes	AET payload. Most successful flight to date. Recovered payload capsule after two days of orbit. The second in-flight recovery of a space capsule was accomplished, utilizing the C-119 aircraft. Satellite, with payload, orbited during strongest solar flare ever recorded. BREHMSTRAHLUNG effects noted on biological elements in lead shielded containers.
6205-1103 FC VI	XVIII	12/7/60	Yes	25,820	380	136	0.031	93.67	5 days	55 days	Yes	AET-L payload. Capsule ejection occurred on the third day of orbit (Pass 48). An in-flight recovery was effected by a C-119 aircraft. Post-ejection reorientation of the satellite to a horizontal tail-first attitude was accomplished.
6205-1101 FC VIII	XIX	12/20/60	Yes	25,860	348	116	0.033	92.9	4-1/2 days	17 days	N/R	Radiometer payload, to determine infrared background radiation of the earth. In the period following launch between loss of telemetry and acquisition by the Kodiak Tracking Station the control gas was lost. Consequently the satellites orbital stability was lost. Nevertheless considerable payload data was obtained.

N/A - not available
N/R - not required

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LMSD-6164B   15 May 1961

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SATELLITE NUMBER / Flight Configuration	DISC. NUMBER	DATE	Did Satellite attain orbit injection?	VELOCITY Feet/Sec.	APOGEE Naut. Miles	PERIGEE Naut. Miles	ECCEN-TRICITY	PERIOD Minutes	USEFUL ORBITAL LIFE	ACTIVE ORBITAL LIFE	Was capsule recovered?	REMARKS
6205-1104 FC VI	XX	2/17/61	Yes	25,690	435	162	0.0366	95.31	4-1/2 days	184 days	No	AET-H payload. Attitude stability, power, and communications were satisfactory through Pass 32. Subsequently S-Band beacon and telemetry were lost. Such loss was not due to loss of primary power. No recovery attempted. BTL guidance system carried on booster for developmental testing: command and control thereby only simulated. Performance of the system adjudged satisfactory.
6205-1102 FC VIII	XXI	2/18/61	Yes	25,984* 25,999	N/R 583	127* 138	0.0383* 0.0589	93.87* 97.84	3 days	177 days	N/R	Radiometer payload; to determine infrared background radiation of the earth. Satellite was first to use the re-start capability of the engine. Under positive control of Kodiak Tracking Station, rocket engine was re-started and maintained full thrust for approximately one (1) second. No problems in performance were apparent through Pass 2. On next contact (Pass 6), the 400-cps single-phase power amplifier had failed, resulting in loss of attitude stability. Performance of the payload was good. 4.3 micron data was as anticipated; 2.7 micron data appeared questionable. Interpretation of payload data is difficult due to attitude instability. Payload data was in agreement with that from DISC XIX (Satellite 1101). Cooling gas pressure for the 4.3 micron detector dropped by Pass 15 to point where radiometer was not sufficiently cooled. 4.3 micron data subsequent is not expected to be useable. Telemetered satellite instrumentation and payload data was received until power depletion which occurred on Pass 54.
6205-1105 FC VI	XXII	3/30/61	No	-	-	-	-	-	-	-	No	AET-L payload. Failure of the hydraulic control system occurred approximately 20 seconds before engine cutoff. Attitude control was lost. Engine thrust duration was shorter than predicted, probably as a result of attitude variations and resultant propellant sloshing. Orbital velocity was not achieved.

* at end of 1st burn

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SATELLITE NUMBER / Flight Configuration	DISC. NUMBER	DATE	Did Satellite attain orbit injection?	VELOCITY Feet/Sec.	APOGEE Naut. Miles	PERIGEE Naut. Miles	ECCEN-TRICITY	PERIOD Minutes	USEFUL ORBITAL LIFE	ACTIVE ORBITAL LIFE	Was capsule recovered?	REMARKS
6205-1106 FC VI	XXIII	4/8/61	Yes	25,600	351	162	0.0257	94.874	5 days	124 days	No	AET-H payload. Injection into orbit was satisfactory. Operation thru Pass 6 were normal. Between Pass 6 and Pass 9 horizon scanner anomalies appears. Between Passes 9 and 10 all control gas was expended. This loss prevented further damping of attitude variations. Capsule ejection was advanced to Pass 32. A spurious skip command was registered, and ejection of the capsule occurred on Pass 31. The capsule was injected into a new and higher orbit because of the incorrect attitude of the satellite at time of ejection.

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As defined in the "Terms and Definitions Data Book," a configuration, as pertaining to a satellite, is:

> "the structure and installed system equipments which make a space vehicle, or associated ground equipment and item, identifiable in itself and distinct from others with which it is not completely interchangeable."

Flight configuration is primarily determined by the nature or distinction of the respective payload. Secondarily, a change in configuration may be predicated upon a major difference in the space vehicle structure and/or systems, viz. spaceframe and/or propulsion system.

Flight Configuration	Series	Serial No.	Payload	Spaceframe	Propulsion System
I	2206	1019, 1022	Instrumented	1X Tank	BAC 8048
II	2206	1018, 1020	Biomedical Research Capsule, MKI	1X Tank	BAC 8048
III (Not flown as yet)	2206	1025	Biomedical Research Capsule, MK II	1X Tank	BAC 8048
IV	2206	1023, 1029, 1028, 1051	AET	1X Tank	BAC 8048
V*	2206	1050, 1052, 1054, 1055, 1056, 1058	AET	1X Tank	BAC 8048
VI	6206	1061, 1062	AET	2X Tank	BAC 8081
VII	2206	1053, 1057	Diagnostic	1X Tank	BAC 8048
VIII	6206	1101, 1102	Radiometer	2X Tank	BAC 8081
VI	6206	1103–1120	AET-H or AET-L	2X Tank	BAC 8096

+ no dual

BAC 8048 = USAF YLR 81-BA-5
BAC 8081 = USAF XLR 81-BA-7
BAC 8096 = USAF XLR 81-BA-9

*Incorporates structure and systems changes from FC IV

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I GENERAL

1.0 **DISCOVERER PROGRAM**

The Discoverer Program is an experimental effort centered in the development of a space vehicle capable of carrying diverse payloads in either satellite or space probe operations. The Discoverer Program has thus far produced the Discoverer Satellite System which is based upon the Agena space vehicle and which includes the components listed in Para. 2.0 below. Evidence of the success of the Discoverer Satellite System lies in the degree to which the Agena space vehicle is being adapted to other space programs. System components designed for the Discoverer Satellite System are being adapted to other satellite and probe programs. Meanwhile, the scientific spatial research for which the Discoverer Satellite System was designed is continuing, utilizing advanced configurations of the Discoverer satellite.

2.0 **DISCOVERER SATELLITE SYSTEM COMPONENTS**

The Discoverer Satellite System includes the following:

2.1 The Discoverer Vehicle, consisting of: (a) the Discoverer satellite which is composed of the LMSD-developed Agena space vehicle as the carrier and a Discoverer payload; and (b) the Discoverer booster which is the Douglas Aircraft-developed Thor (DM-21) satellite launching vehicle, modified to function as the first stage thrust device for the Discoverer satellite.

2.2 The Discoverer Payload, which may be: (a) the Heavyweight Advanced Engineering Test (AET-H) package; (b) the Lightweight Advanced Engineering Test (AET-L) package; (c) the Biomedical Recovery Capsule; and/or (d) specialized equipment, such as a radiometer package or Oscar.

2.3 The Discoverer launch complex, including the launch pad and associated facility(s) and integral equipments, the operations ground equipments (OGE); the maintenance ground equipments (MGE); the Vandenberg Tracking Station (VTS) in its launch functions, the telemetry ship(s), and the Satellite Test Center (STC) in its launch control functions.

2.4 The Discoverer ground station complex, including the satellite tracking and acquisition (T&A) stations, satellite instrumentation and payload data read-in and satellite system command (RI/C) stations, and the STC in its orbital control functions.

2.5 Communications Network, consisting of the equipments in the facilities for communication between the ground complex stations and the STC for the exchange of satellite performance data and commands to satellite and for inter-station administrative communications.

2.6 Recover/Retrieval Equipments, consisting of those airborne and surface equipments used for air recovery and sea retrieval of the payload capsules.

2.7 Human Engineering, for the support of design and operation of the satellite, booster, and ground equipments.

Note: (1) Geophysical Research Directorate (GRD) Equipments, which are not components of the Discoverer Satellite System but which may be carried, on a non-interference basis, for additional scientific research purposes. Satellite weight contingency usually determines number of GRD equipments which may be carried on a given Discoverer satellite.

(2) Special Equipments "piggy-backed" on a non-interference basis.

3.0 **PROGRAM OBJECTIVES**

The objectives of the Discoverer Program are as follows:

3.1 Develop and demonstrate the equipment, techniques, and procedures for launching the Discoverer vehicle.

3.2 Demonstrate the capability in the propulsion system of the single-burn, extended duration operation and the re-start (dual-burn) operation, in their respective configurations.

3.3 Demonstrate the capability of the Discoverer satellite to attain the planned orbit.

3.4 Demonstrate the capability of the Discoverer satellite to reorient itself and to maintain the programmed attitude throughout the useful orbital life of the satellite.

3.5 Develop and demonstrate the equipment, techniques, and procedures by which the Discoverer satellite obtains the scientific data, including the environmental data, for which it is placed in orbit; demonstrate the capability to obtain such data on orbit.

3.6 Develop and demonstrate the equipment, techniques, and procedures for the transmission of scientific data and data pertaining to satellite systems performance and environment to RI/C stations.

3.7 Develop and demonstrate the equipment, techniques, and procedures for acquiring, tracking, and commanding the satellite during ascent-to-orbit and in orbit.

3.8 Develop and demonstrate the equipment, techniques, and procedures for utilizing supplemental tracking aids, such as Doppler beacons and satellite-borne tracking lights.

3.9 Develop and demonstrate the equipment, techniques, and procedures for processing data received from orbiting satellite and the preparation of ephemerides therefrom.

3.10 Develop and demonstrate capsule recovery by ejecting capsules from satellites in orbit, propelling them in an appropriate descent trajectory, enabling recovery for examination and evaluation.

3.11 Develop and demonstrate equipment, techniques, and procedures for in-flight recovery of ejected payload capsules and for sea retrieval of the capsules in the event that air recovery is not effected.

3.12 Evaluate satellite, system, and subsystem performance as the basis for refinements in the current equipments and to determine the parameters for more advanced design configurations, including spaceborne structures and equipments and associated ground equipments.

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	DISCOVERER PROGRAM OBJECTIVES	
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1.0 **GENERAL**

Launch Site	Vandenberg Air Force Base (VAFB)	
Launch Azimuth	172°	
Flight Configuration	VI	
Payload	Recoverable Advanced Engineering Test Package Light (AET-L) Heavy (AET-H)	
Satellite Series and Serial Numbers	6205-1061, 1062 6205-1103 thru 1120*	
Orbit Parameters#	AET-L	AET-H
Injection Altitude (statute miles)	150	190
Eccentricity	.033	.020
Orbital Period (nominal minutes)	93.8	94.4
Orbital Plane Inclination (degrees)	81.8	81.8
Useful Orbital Life (hours)	100	100

*Satellite 1120 incorporates a payload which represents a departure from the alternate payloads available for the remainder of the satellites in the flight configuration.

#Parameters applicable to Flight Configuration VI Satellites 1107 and subsequent, except Satellite 1120.

2.0 **FLIGHT MISSION**

2.1 Demonstrate the compatibility of the booster and satellite, as components of the Discoverer Program Flight Configuration VI vehicle, to place the satellite at an altitude, velocity, and attitude requisite to achievement of satellite injection.

2.2 Demonstrate the capability of the satellite: (1) to separate from the booster with a minimum of ascent trajectory disturbance, and (2) to provide the second stage thrust, supplemental to the booster-induced first stage thrust, to achieve the planned orbital injection altitude and velocity.

2.3 Demonstrate the capability of the satellite to carry the Advanced Engineering Test Package in either the lightweight (AET-L) or the heavyweight (AET-H) configuration, as determined 19 days prior to launch readiness date for the individual flight test.

2.4 Demonstrate the capability of the satellite: (1) to orbit in the planned attitude at an acceptable altitude for the planned period of useful orbital life; (2) to accomplish the events requisite to ejection of the payload; and (3) to eject the payload into an acceptable re-entry trajectory for recovery/retrieval.

2.5 Demonstrate the capability of the satellite to telemeter to Discoverer read-in and command (RI/C) stations data relative to environmental conditions internal and external to the satellite structure and to satellite system and payload equipment performance.

2.6 Secure scientific data and information by means of the AET payload.

2.7 Demonstrate the capability of the satellite to telemeter scientific data secured by the AET package.

2.8 Demonstrate the capability of the recoverable capsule, after ejection from the satellite: (1) to interrupt the orbital altitude and velocity induced by the satellite; (2) to stabilize itself, decelerate in the planned re-entry trajectory; (3) to re-enter the atmosphere; (4) to identify itself to Discoverer recovery/retrieval forces and equipments while trajecting to the planned target area; and (5) to maintain its contents in the planned internal environment from countdown through recovery/retrieval.

2.9 Demonstrate the capability of the Discoverer recovery/retrieval forces and equipments to air-recover the payload package in its capsule housing or to retrieve it from the ocean target area.

2.10 Demonstrate the capability of the payload capsule to maintain the affected AET package in a controlled environment from preparation for launch until recovery/retrieval.

2.11 Demonstrate the compatibility of the payload-associated and general ground support equipments in supporting the satellite and its payload in preparation-for-launch, countdown, and launch.

2.12 Demonstrate the capability of the Discoverer ground station complex, including associated stations, as applicable to individual flight tests, to acquire and track the satellite, predict its orbital path and period, command and control certain satellite system and payload equipment operations, accept data telemetered from the satellite, and provide pertinent data to the Satellite Test Center.

2.13 Demonstrate the capability of the Discoverer communications network to convey from each of the Discoverer integral ground stations and the associated ground stations to the Satellite Test Center data and information pertinent to the affected flight test.

2.14 Demonstrate the capability of the Satellite Test Center: (1) to maintain an overriding control (i) of the vehicle countdown, launch, and ascent to separation, (ii) of the satellite final ascent to orbit, orbit, and ejection of the payload capsule, (iii) of the recovery/retrieval forces and equipment, (iv) of the ground station operations, and (v) of the communications network; (2) to compile ephemeris tables; and (3) to process, compile, and disseminate to affected agencies data and information from and about the recovered payload package.

3.0 **PRIMARY TEST OBJECTIVES**

3.1 Discoverer Vehicle. Test and evaluate the capability of the Discoverer Flight Configuration VI vehicle, composed of a Discoverer booster and a Discoverer satellite, to provide for injection of the satellite into the planned polar orbit, including:

(a) The structural and aerodynamic conformation of the booster and satellite in countdown and during the launch phase.

(b) The nature of any separation perturbations induced in the satellite by the booster.

3.2 Discoverer Booster. Test and evaluate the capability of the Discoverer booster, a Douglas Aircraft-developed Thor (DM-21) satellite launching vehicle, to function as the first stage thrust device, including:

(a) Structural integrality under launch phase conditions.

(b) Capability of the booster's guidance and control system to: (1) to execute programmed commands, including roll and pitch routines; (2) to accept commands from Douglas ground operations for control of booster system equipment operations, including hard shutdown of the main engine and of the vernier engines; and (3) to accept commands from Douglas ground operations for transmission to the satellite, including SS/D timer brake control holding and, in the case of satellites 6-205-1105 and subsequent, integrator minus velocity gain corrections.

(c) Capability to maintain stability of the successively programmed or commanded attitudes for the required periods of time.

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(d) Capability to achieve the planned apogee within acceptable tolerances of altitude, velocity, and attitude.

3.3 Discoverer Satellite. Test and evaluate the capability of the Discoverer satellite as the second stage thrust device and payload carrier to orbit at the nominal altitude in the tailfirst, horizontal attitude, including:

(a) Structural, aerodynamic, and spatiodynamic integrality under launch, coast, orbital boost, and orbit phase conditions, including reorientation.

(b) Capability to withstand and damp out separation perturbations.

(c) Capability to achieve an orbit of acceptable eccentricity by use of a relatively extended propulsion system operating time.

(d) Capability to damp out possible reorientation perturbations within acceptable time tolerances.

(e) Capability to maintain a stable attitude on orbit.

(f) Capability to orient to the proper attitude and to eject the payload capsule with a minimum of disturbance to the capsule.

3.4 Discoverer Satellite Spaceframe. Test and evaluate the capability of the Discoverer satellite spaceframe to house, support, and/or carry satellite system and payload equipments, including:

(a) Capability to withstand and/or modify within acceptable limits shear loads, torques, and stresses induced by Discoverer booster thrust, booster separation, orbital rocket engine thrust, nose section housing separation, and programmed attitude changes.

(b) Capability to maintain center of pressure in consonance with the center of gravity within acceptable tolerances.

(c) Capability of the external paint and color code pattern to provide a means for optical tracking in the launch phase.

3.5 Discoverer Satellite Propulsion System. Test and evaluate the capability of the Discoverer satellite porpulsion system to provide the orbital altitude and velocity thrust incremental to the booster-induced thrust to assure injection, including:

(a) Capability to start on programmed signal from the SS/D timer, burn for a relatively extended period of time, and shut down on command signal from the SS/D integrator.

(b) Capability of the (USAF XLR-81-BA9) (BAC Model 8096) engine to properly utilize the propellant supply under applicable flow rate and pressurization conditions.

(c) Capability to vent the unexpended propellant supply and propellant pressurization gas supply within acceptable time limits after propulsion system command shutdown.

(d) Capability to provide turbine power to hydraulic control system motor.

(e) Capability of the propellant subsystem to provide oxidizer and fuel at the required flow rates and pressures.

3.6 Discoverer Satellite Electrical Power System. Test and evaluate the capability of the Discoverer satellite electrical power system to supply electrical power to satellite system and payload equipments for the ascent-to-orbit and the planned useful orbital life of the satellite, including:

(a) Capability of the primary battery installation to supply basic power, directly or via the power distribution subsystem to satellite system and payload equipments.

(b) Capability of the power distribution subsystem, including the 400 cps and 2000 cps inverters, 400 cps power amplifier, load limiters, and voltage regulators: (1) to add an increment to the battery input dc voltage to assure an output of 28.5 volts dc to affected equipments; (2) to convert battery input dc voltage to 115 volts ac, three-phase, 400 cps and one-phase, 2000 cps as applicable to affected equipments; (3) to convert battery input +dc voltage to -dc voltage output to affected equipments; and (4) to control these output voltages and frequencies within acceptable tolerances.

3.7 Discoverer Satellite Guidance and Control System. Test and evaluate the capability of the Discoverer satellite guidance and control system to provide programmed navigational and attitudinal control of the satellite from separation through capsule ejection and to initiate, modify, and/or terminate certain satellite system equipment operations at selected times during this period, including:

(a) Capability of the horizon scanner: (1) to establish the local vertical after separation and to maintain it until reorientation;(2) to establish the local vertical after reorientation and to maintain it until initiation of attitude change for the payload capsule ejection; (3) to detect satellite attitude deviations resulting from drag or from gyro drift; and (4) to supply error signals to torque the affected IRP gyro(s) proportionate to the deviation(s).

(b) Capability of the inertial reference package: (1) to establish satellite attitude references from the Discoverer vehicle orientation at separation; (2) subsequently to accept gyro torquing signals from the horizon scanner, the SS/D timer, and the SS/D integrator, as applicable; (3) to supply corresponding differential error signals for attitude correction and definitive attitude change, as applicable, to the flight control electronics system, from separation to payload capsule ejection; and (4) to supply satellite acceleration signal to the SS/D integrator during orbital boost.

(c) Capability of the SS/D (sequence) timer to start, modify, and/or stop satellite system equipment operations from separation through reorientation and during the payload capsule ejection sequence, as listed in the Timed Sequence of Events, Section IV, Page 2.9 ff.

(d) Capability of the SS/D integrator: (1) to supply IRP pitch gyro torquing signal(s) to induce a residual satellite pitch rate during the coast phase; (2) to accept minute velocity gain corrections for final setting of the satellite velocity to be gained during orbital boost: (i) from the Discoverer ground operations, in the case of 6205-1104, and (ii) from the DAC ground operations BTL guidance equipments via the booster guidance system and umbilical to the satellite, in the case of 6205-1105 and subsequent; (3) to accept satellite acceleration signals from the IRP longitudinal accelerometer; (4) to determine satellite achievement of the desired orbital velocity gain; and (5) to supply a corresponding hard shutdown command to the propulsion system to terminate orbital boost.

(e) Capability of the resistive voltage networks to supply voltages required for torquing of the IRP gyros on the basis of signals from the horizon scanner, SS/D timer, and SS/D integrator, as applicable.

(f) Capability of the secondary junction box to convert guidance and control system function signals to a form suitable for telemetry.

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(g) Capability of the flight control electronics system: (1) to accept differential error signals from the IRP pitch, yaw, and roll attitude (displacement) gyros, rate signals from its own pitch, yaw, and roll rate gyros, and, during propulsion system operation, piston position signals from the hydraulic control linear transducer; (2) process them; and (3) supply control signals to the pneumatic control system from separation through capsule ejection, to the hydraulic control system during propulsion system operation.

(h) Capability of the pneumatic control system to provide the required thrust impulse(s), on the basis of control signals from the flight control electronics system, for satellite pitch, yaw, and roll control during the coast, reorientation, and orbit phases and payload capsule ejection and for satellite roll control during the orbital boost phase.

(i) Capability of the hydraulic control system to provide, on the basis of control signals from flight control electronics system, engine gimballing torque(s) for pitch and yaw control during the propulsion system operated.

3.8 Discoverer Satellite Communications System. Test and evaluate the capability of the Discoverer satellite communications system: (1) to accept commands from Discoverer RI/C stations; (2) to start, stop, and/or modify satellite system equipment operations; and (3) to transmit pertinent data to affected Discoverer ground stations, including:

(a) Capability of the command subsystem: (1) to accept real-time commands from Discoverer RI/C stations via the S-Band transponder for adjusting operation of the SS/H orbital timer; (2) to program satellite system orbital events by means of the SS/H orbital timer, subject to the modifications via the transponder.

(b) Capability of the identification subsystem: (1) to supply CW identification transmitter signals to Discoverer T&A and RI/C stations for satellite acquisition, and tracking, and orbit prediction purposes; (2) Doppler signals to associated Transit stations for acquisition and tracking purposes.

(c) Capability of the telemetry subsystem to transmit to Discoverer RI/C stations data relative to satellite system, and special equipment and instrumentation performance and internal and external environmental conditions, either directly or by tape recording and reproduction.

(d) Capability of the antenna subsystem to accept commands and transmit data for the affected subsystems.

(e) Capability of the tracking light to provide T&A, RI/C, and associated ground stations with a means of tracking and orbit prediction.

3.9 Discoverer Satellite Payload. Test and evaluate the capability of the Discoverer satellite payload: (1) to achieve the end results of the advanced engineering tests; (2) determine the parameters for subsequent research.

3.10 Discoverer Ground Station Complex. Test and evaluate the capability of the Discoverer ground station complex to: acquire, track, command, and interrogate the satellite; receive satellite structure and system equipment data; record, process, and compile pertinent data in each station; and send such data to the Satellite Test Center over the associated communications network, including:

(a) Capability of the launch site complex, consisting of the Vandenberg Air Force Base (VAFB), VAFB Auxiliary (Point Mugu), and the telemetry ship(s), to track the satellite from lift-off through the initiation of the orbital boost phase and to receive data telemetered from the satellite reflecting structure and system environment and performance.

(b) Capability of the Discoverer tracking and acquisition (T&A) stations to acquire and track the satellite via the CW identification transmitter during active orbital life and the S-band transponder and Doppler during useful orbital life for purposes of orbit prediction and antenna positioning.

(c) Capability of the Discoverer read-in and command (RI/C) stations to: (1) insert real-time commands to change the position and velocity of the SS/H timer tape to conform the tape period to the satellite orbital period; (2) to receive data from the VHF communications equipments; and (3) to perform the tracking and acquisition operations noted in (b) above.

(d) Capability of the Discoverer-associated ground stations, including the Smithsonian Astronomical Observatories (SAO) and the Doppler Tracking Stations (cf. Sec. V, Page 1.1.2), to track the satellite by means of the tracking light and the APL Doppler signals, respectively.

3.11 Communications Network. Test and evaluate the capability of the communications network to: (1) provide communication between the several stations and the STC for purposes of ascent-to-orbit and orbit monitoring and controlling of the satellite and its systems; (2) channel other data and information relative to the flight test from the T&A and RI/C and Discoverer-associated stations to the STC.

3.12 Ground Support Equipments. Test and evaluate the capability of the general and payload-associated ground support equipments (GSE), including ground handling and service, subsystem and system checkout, and launch control and items, to support the satellite and its payload during preparation-for-launch, countdown, and launch operations.

3.13 DAC Guidance Ground Operations. For 6205-1105 and subsequent, test and evaluate the capability of the BTL guidance equipments to compute booster trajectory and supply appropriate commands via the booster guidance system to the satellite for SS/D integrator minus velocity gain corrections and for SS/D timer brake control hold prior to orbital boost operations.

3.14 Recovery/Retrieval Forces and Equipments. Test and evaluate the capability of the recovery and retrieval forces and equipments: (1) to patrol and monitor the predicted payload capsule target area; (2) to vector on the descending capsule by means of radar picket aircraft; (3) to air recover the capsule or to retrieve it from the sea, depending on local conditions; and (4) to return it to the STC while maintaining it in the necessary controlled environment.

3.15 Satellite Test Center (STC). Test and evaluate the capability of the Satellite Test Center (STC). (1) to control satellite countdown and launch policy; (2) to coordinate and control Discoverer launch site, ground station, and Discoverer-associated ground station performance, including launch "go", launch, ascent-to-orbit, and orbit, in matters requiring top echelon approval; (3) to control satellite orbit operations within the limitations of the satellite command subsystem; (4) to perform, or have performed, necessary computations relative to (1), (2), and (3) to provide RI/C station(s) with necessary data relative to adjustment of the SS/H orbital timer to match the satellite orbital period(s) so as to control sequential events, including initiation of payload capsule ejection activity; and (5) to perform advanced engineering test data analysis and to disseminate to affected agencies and the Company pertinent data and information relative to the flight test.

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4.0 SECONDARY TEST OBJECTIVES

4.1 Experiment to determine environmental conditions affecting satellite system and/or payload equipment performance and integrity under orbital operating conditions, including ionospheric, heterospheric, and thermospheric actions and reactions.

4.2 Obtain geophysical data, as equipment is available on a non-interference basis with primary objectives.

4.3 Demonstrate the capability of the Discoverer Satellite to "piggy-back" special instrumentation and equipment, such as scientific research components (including "Vela Hotel") and Oscar.

4.4 Determine possible spurious radiations and irradiations on, between, and among satellite system equipments for purposes of design refinements.

4.5 Determine signal-to-noise ratio(s) of satellite system equipments for purposes of design refinements.

4.6 Compile data relative to recovery/retrieval activity for application to other Programs.

5.0 TERTIARY TEST OBJECTIVES

5.1 Determine parameters for satellite orbital control leading to further refinement of recovery/retrieval facilities, equipments, forces, techniques, and procedures.

5.2 Determine parameters for a more expanded global network of ground stations and related communications for this and other Programs with which LMSD Satellite Systems is, or may become, affiliated.

5.3 Study causes of satellite internal build-up of pressures which may contribute to satellite orbital attitude and path distrubances, including sublimation of propellant and equipment surface paints, battery conditions, venting of propellants and gases, etc..

5.4 Demonstrate the capability, after capsule ejection, to re-orient the satellite to the tail first, horizontal attitude to maintain the stability of the satellite in that attitude until electrical power and/or pneumatic control gas depletion.

5.5 Test and evaluate the capability of the sun position indicator to supply to RI/C station(s) and/or the telemetry ship(s) via the FM/FM telemetry/data relative to satellite attitude during programmed attitude changes.

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1.0 GENERAL

The Discoverer Flight Configuration (FC) VI vehicle consists of: (a) the Discoverer booster, which is a Douglas-developed Thor (DM-21) satellite launching vehicle modified to mate with and provide orbital altitude for the Discoverer satellite; and (b) the Discoverer satellite, which is an LMSD-developed Agena B space vehicle modified to function as the carrier for the Discoverer payload. The Discoverer payload for FC VI is an Advanced Engineering Test (AET) light (-L) or heavy (-H) payload installed in a recoverable capsule which is mounted in the nose section of the satellite. The Discoverer satellite adapter is the mating structure that joins the Discoverer satellite and booster. At satellite - booster separation, the adapter, which is permanently attached to the booster during preparation for launch, remains with the booster.

2.0 COUNTDOWN PHASE

The countdown phase begins nominally 8.5 hours before launch. Certain systems in the Discoverer Satellite are turned on during countdown for warm-up and checkout by Discoverer Ground Operations (DiscOps) per the applicable Development Test Order (DTO). The Discoverer booster is similarly checked out by Douglas Ground Operations (DacOps). Propellant loading operations are accomplished. Satellite guidance and flight control electronics are turned on about 350 minutes before launch and remain on thereafter.

2.1 Terminal Countdown. At T-9.5 minutes (T-0 being the time of launch), DacOps turns on the automatic launch sequencer to start the terminal countdown. From this point on, much of the launching procedure - including the initiation of Discoverer booster ignition - is automatically controlled by the sequencer. Failure in any system controlled or monitored by the sequencer results in an automatic hold on the terminal countdown. The hold remains in effect until the hold is located and corrected, or until the decision is made to recycle or terminate the countdown. At T-8.5 minutes, DacOps turns on the booster telemetry and the command destruct radio receivers. An operational hold on the terminal countdown is scheduled at this point. When booster telemetry and the command destruct radio receivers are verified ON by the Flight Safety Officer, BMD Launch Control waives or releases the operational hold. At T-7 minutes, DiscOps turns on the satellite identification transmitter, dual-frequency tracking beacon,* radar transponder, and telemetry. Tracking of the satellite by all means except the radar transponder begins at this time. Once turned on, the identification transmitter and the dual-frequency tracking beacon operate continuously through launch and throughout the life of the satellite until depletion of the satellite electrical power supply. At T-2.5 minutes, electrical power for all booster and satellite requirements is switched from the ground power source used during countdown to the respective electrical power systems. At T-2 minutes, DiscOps: (a) arms the satellite flight termination system via the auxiliary umbilical from the blockhouse to the satellite; (b) closes a circuit to enable release of the main umbilical; and (c) arms the capsule battery circuit. At T-90 seconds, DiscOps turns on the SS/D timer motor and the horizon scanner. A safety circuit holds the SS/D timer brake ON via the auxiliary umbilical. If terminal countdown status at this time is normal, the Flight Safety Officer verifies CLEAR TO LAUNCH. At T-30 seconds, BMD Launch Control imposes a technical hold on the terminal countdown if CLEAR TO LAUNCH has not been received from the Flight Safety Officer by that time. The hold remains in effect until the problem is located and corrected, or until the decision is made to recycle or terminate the countdown. At about T-4 seconds, DacOps automatically initiates Discoverer booster ignition. At T-2 seconds, a signal initiated by the burning through of a wire across the booster nozzle disconnects the main umbilical to the satellite.

3.0 LAUNCH PHASE

The launch phase covers the time from lift-off (T-0) through separation of the Discoverer satellite from the adapter - booster combinations.

3.1 Booster Operation. Lift-off occurs when the booster engine approaches steady-state operation and lifts the vehicle two inches up from the launching pad. At launch, the booster pitch programmer initiates a 0° pitch program that maintains the vehicle's vertical flight path for approximately 10 seconds. At about T + 1 second (varies approximately from 0.5 to 1.5 seconds), the auxiliary umbilical pulls away from the satellite, opening the SS/D timer safety circuit to release the SS/D timer brake and start the programmed sequencing of satellite events. At the same time, the satellite radar transponder is interrogated by DiscOps via ground radar. Shortly thereafter (this event can occur at any time from lift-off to T+20 seconds), DacOps begins tracking the vehicle via the ground radar associated with the booster closed-loop guidance system. At T+2 seconds, the booster roll programmer initiates a 7-second program that rolls the vehicle to move its yaw (XX) axis from the launching orientation (259.5° for Pad 1) (181.5° for Pad 4; 218.4° for Pad 5) to an azimuth of 172°, in the plane of the intended final trajectory. At T + 10 seconds, the booster pitch programmer initiates a program that pitches the vehicle - 61.2° for satellites carrying AET-L payloads or 54.0° for satellites carrying AET-H payloads in about 120 seconds. At T + 20 seconds, DacOps synchronizes the booster closed-loop guidance system computer time reference. At T + 90 seconds, the closed-loop guidance systems begins to generate booster steering commands to correct vehicle deviations from a reference trajectory set into the computer. The ground based radar tracks the vehicle, and computer relates vehicle position to a coordinate system, determines the vehicle's velocity component along each axis of the system, compares these velocity components with inset reference conditions, and generates corrective steering commands as required. When the vehicle reaches an altitude of approximately 70,000 feet above sea level (at about T + 93 seconds), a barometric switch in the satellite closes to arm the recoverable capsule ejection system. At about T + 130 seconds, the booster pitch program ends, and the booster closed-loop guidance system begins sensing for booster main engine cut-off (MECO). MECO can occur in two ways: (a) by command from the closed-loop guidance system when predetermined vehicle altitude and velocity conditions are satisfied; or (b) at propellant depletion. About 4 seconds before MECO, booster steering via the closed-loop guidance system ends. Booster vernier engine cut-off (VECO) is commanded by a timer in the booster 9 seconds after MECO occurs.

3.2 Satellite SS/D Timer Delay, Velocity Correction, and Separation. At T + 151 seconds (about the time of MECO command), the satellite SS/D timer: (a) disarms the satellite flight termination system (b) supplies a redundant (back-up) signal to turn on satellite flight control electrical power; (c) opens the satellite pneumatic control gas supply valve; (d) starts the satellite's orbital timer; (e) uncages the satellite's accelerometer integrator to permit reset of the integrator dial reading for velocity-to-be-gained; (f) arms the separation pin puller squibs and retro rockets; and (g) closes a circuit to enable booster closed-loop guidance system commands to the satellite. The MECO command from the booster closed-loop guidance system supplies a redundant signal to disarm the booster and satellite flight termination systems. About 1 second later, a signal from the booster guidance system arms the satellite SS/D timer hold and velocity correction circuitry. After MECO, the closed-loop guidance system computer continues for

* When installed

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about 6.5 seconds to compare vehicle position and velocity with its own inset reference data. On the basis of these comparisons, it computes satellite SS/D timer hold and velocity correction values required in order for the Discoverer satellite to achieve the predetermined orbital injection conditions, and then transmits this information in the form of command signals via radar to the vehicle. The effect of the SS/D timer hold is to delay the start of satellite propulsion system operation, and the effect of the velocity correction is to reduce the velocity-to-be-gained setting of the satellite's accelerometer integrator, thus reducing the duration of orbital boost engine operation. The timer hold and velocity correction values are represented by two sequential command signals. The velocity correction information is represented by the duration of the first signal, and the timer hold information is represented by the combined duration of both signals. The first signal is also used to uncage the satellite inertial reference package (IRP) gyros. Thus, within 8.5 seconds after MECO command, the booster closed-loop guidance system: (a) uncages the satellite IRP pitch, yaw, and roll gyros; and (b) initiates satellite SS/D timer hold and velocity correction. When the time required for velocity correction has elapsed, the first of the two sequential commands is terminated and the second, for additional SS/D timer hold, is initiated. When the time required for SS/D timer hold has also elapsed, the second command is terminated. If the duration of the velocity correction required is greater than that of the SS/D timer hold required, the computed velocity correction is transmitted. Shortly after completion of the SS/D timer hold, but not before 13 seconds after MECO, the booster guidance system transmits a separation command signal to fire the separation pin puller squibs and ignite the two adapter-mounted retro rockets, initiating separation of the satellite from the adapter and booster. The retro rocket thrust (burn time = 0.5 + 1.0/-0.0 second) slows the adapter - booster combination and permits the satellite to separate from the adapter and booster. At almost the same time, the SS/D timer supplies a redundant signal to uncage the IRP pitch, yaw, and roll gyros.

3.3 Flight Termination. In the event of any Discoverer booster or satellite mal- or misfunctioning that might compromise range safety requirements, the Flight Safety Officer can command the destruction of the vehicle by means of a signal via the booster UHF range safety radio receivers to the booster and satellite flight termination systems. In the event of premature separation of the adapter from the satellite, the adapter's motion pulls adapter-mounted lanyard-type switches from lanyards installed on the aft equipment rack, completing the circuit that fires the destruct charge. In either commanded or automatic termination, the destruct charge, when ignited, penetrates the propellant tanks, permitting the oxidizer and fuel to mix indiscriminately. The hypergolic reaction between the two immedi-tely results in explosive combustion that terminates the flight.

4.0 COAST PHASE

The coast phase covers the time from vehicle separation to the firing of the ullage control rockets to initiate orbital boost engine operation. At separation, actuation of the separation switch between the satellite and booster activates satellite pitch, yaw, and roll pneumatic control. Several seconds after the separation command from the booster guidance system, the SS/D timer: (a) supplies a redundant signal to fire the separation pin puller squibs and ignite the retro rockets, assuring separation of the satellite from the adapter - booster combination; and (b) fires the horizon scanner fairing squib to jettison the fairing. Seconds later, the SS/D timer: (a) turns off power to the pneumatic control gas supply valve; and (b) switches telemetry Channel 18 from booster vibration measurement to satellite engine turbine speed measurement. Shortly thereafter, the SS/D timer: (a) supplies a redundant signal to activate pitch, yaw, and roll pneumatic control; (b) activates an integrator potentiometer circuit that torques the IRP pitch gyro to pitch the satellite at an initial -1.65°/min (increases to a maximum of -4.19°/min) in order to compensate for the satellite's geocentric angular motion; and (c) torques the IRP pitch gyro to pitch the satellite -28.8° for satellites carrying AET-L payloads or -36.0° for satellites carrying AET-H payloads at a -3.6°/sec rate in order to achieve a horizontal satellite attitude. After the pitch-down maneuver has been effected (either 8 seconds or 10 seconds later), the SS/D timer: (a) removes the -3.6°/sec pitch rate; and (b) connects the horizon scanner pitch signal to the IRP pitch gyro and horizon scanner roll signal to the IRP roll gyro.

5.0 ORBITAL BOOST PHASE

The orbital boost phase covers the time from the uncaging of the SS/D integrator through the reactivation of pitch and yaw pneumatic control after orbital boost engine shutdown. Several seconds after removal of the -3.6°/sec pitch rate, the SS/D timer: (a) torques the IRP pitch gyro to pitch the satellite at the rate required in order to achieve the desired pitch attitude during orbital boost (varies from flight to flight); and (b) fires the ullage control rocket squibs. The two ullage control rockets have a nominal burn time capability of 18.8 seconds, long enough to provide positive overlap with orbital boost engine thrust in order to maintain ullage control during engine ignition. Several seconds before the end of the ullage control rocket burn time, the D timer: (a) turns on +28v dc to the engine relay box, activating a circuit that shuts off pitch and yaw pneumatic control; (b) ungrounds the integrator input circuit, and connects the IRP accelerometer to the integrator; and (c) fires the helium valve squibs to pressurize the orbital boost engine propellant subsystem, fires the gas generator squib to initiate engine operation. One second later, the D timer: (a) provides a redundant signal to turn off pitch and yaw pneumatic control; (b) deactivates the gas generator and helium valve squib firing circuits and the gas generator arming circuit; (c) disarms the pitch and yaw pneumatic control OFF circuit to permit later reactivation of the pneumatic controls; and (d) locks telemetry on through a by-pass circuit. The orbital boost engine achieves steady state thrust 1.3 to 1.9 seconds after firing of the helium valve and gas generator squibs. The nominal engine burn time is 240 seconds (+ 16/-8, depending on propellant loading). A few seconds before engine shutdown, the D timer: (a) deactivates the engine cut-off safety switch, which has been preventing receipt of a possible premature shutdown signal from the integrator; and (b) arms the pitch and yaw pneumatic control ON circuit. When the programmed velocity gain, as corrected by the booster closed-loop guidance system, is achieved, a signal from the SS/D velocity integrator to the engine relay box cuts off electrical power to the engine and propellant subsystems, affecting shutdown of the orbital boost engine. At the same time a switch in the integrator disconnects the IRP accelerometer from the integrator and grounds the integrator input circuit, and the engine relay box activates the pitch and yaw pneumatic control ON circuit.

6.0 REORIENTATION PHASE

The reorientation phase covers the time from initiation of the reorientation maneuver through the turning off of communication equipment at the end of ascent-to-orbit operations. At the beginning of the phase (a few seconds after orbital boost engine shutdown), the D timer: (a) supplies a back-up signal to turn on pitch and yaw pneumatic

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control; (b) disconnects the integrator pitch rate potentiometer to remove the pitch rate that compensates for satellite geocentric angular motion; (c) torques the IRP pitch gyro to remove the additional pitch rate applied during orbital boost; (d) torques the IRP yaw gyro to yaw the satellite 180° counter-clockwise at a -40°/min rate for 4.5 minutes in order to achieve a tail-first horizontal attitude for orbital operation; and (e) fires the oxidizer and helium vent valve squibs. The helium pressure spheres and lines immediately begin to lose pressure, and the oxidizer tank begins to vent residual oxidizer through thrust nullifiers. When the satellite reorientation maneuver has been completed, the D timer: (a) removes the -40°/min yaw rate, and connects the horizon scanner roll signal to the IRP yaw gyro to achieve 3-axis control of the satellite from the horizon scanner; (b) torques the IRP pitch gyro to pitch the satellite at a -3.86°/min rate in order to compensate for the satellite's geocentric angular motion; (c) turns off reg. +28v dc and reg. 1ϕ 400 cps 115v ac to the accelerometer (d) turns on + 28v dc to the payload and recoverable capsule; and (e) starts telemetry calibration. After ten seconds, the telemetry index stopping switch stops telemetry calibration and switches telemetry instrumentation functions from ascent operation to orbit operation. At the same time, the D timer: (a) supplies a redundant signal to stop telemetry calibration and switch telemetry instrumentation functions from ascent operation to robit operation; (b) switches telemetry and identification transmitter operation from the VHF exit antenna to the VHF orbit antenna; (c) opens the engine shutdown circuit, and turns off power to the integrator; (d) switches the flight control system to low gain; and (e) switches IRP yaw and roll gyro telemetry signals to high gain (since smaller positional deviations than experienced during ascent are expected in orbital operation). A few seconds later, the D timer: (a) switches the telemetry hold-in-bypass circuit off, ending D timer control of telemetry operation and permitting telemetry turn-off by the orbital timer; (b) switches the horizon scanner signals to low gain; (c) fires the fuel vent valve squib; (d) arms the horizon scanner OFF circuit; and (e) turns itself off. The fuel tank begins to vent residual fuel through thrust nullifiers. When the satellite has passed beyond the range of communication with DiscOps facilities used during the ascent to orbit, the orbital timer turns off power to the radar transponder-decoder and to the telemetry.

7.0 ORBIT PHASE

During the orbit phase, the satellite makes a succession of revolutions about the earth in an orbital plane of approximately 81.7° inclination. The first south-to-north equatorial crossing, approximately 60 minutes after launch, marks the starting point of the first orbital period. With each orbit, the satellite crosses the equator at a point farther west than for the previous one, so that the 16th orbit starts at roughly the same longitude as the first orbit. During orbital operation, the IRP pitch, yaw, and roll position gyros operate continuously, and the horizon scanner feeds local vertical error signals to them (pitch error signals to the pitch gyro, and roll error signals to the yaw and roll gyros). The guidance and control electronics operate continuously to supply the pneumatic control system with pitch, yaw, and/or roll error signals determining the thrust impulse necessary to maintain the satellite in a steady-state attitude. The identification transmitter and the dual-frequency tracking beacon operate continuously.

7.1 Event Sequence Control. Satellite equipment ON-OFF functions during the orbit phase are controlled by the perforated-tape-type orbital timer in a programmed event sequence occurring in a succession of subcycles. A subcycle can be defined as that part of the programmed event sequence which corresponds to one orbit's operations. At launch, the orbital timer is phased such that Subcycle 1 corresponds to Orbit 1. If the nominal orbit is achieved, the operations programmed within each subcycle will be those required by the territory under the satellite. During orbits in which the satellite is within communication range of the DiscOps read-in and command (RI/C) stations in New Hampshire, California, Alaska, and Hawaii, the radar transponder-decoder and the telemetry are turned on and off by the orbital timer. Subcycles in which this is programmed to occur are designated as active subcycles. On these FC VI satellites that carry a tracking light installation, the orbital timer also turns this installation on and off in a sequence programmed to correspond with the satellite's night-time passes over optical tracking stations equipped with Baker-Nunn cameras. (For orbital timer programming and ground station locations, ref. Sect. I, pp 3.6 ff and Sect. V, p 1.1.2, respectively).

7.2 Subcycle Period Adjustment. The orbital timer program is prepared for a satellite orbital period capability of from 90.0 to 108.15 minutes. A subcycle period is represented on the tape by 5400 tape time units, and the real-time span of the 5400 tape units can be expanded or contracted to correspond with the satellite's actual orbital period by adjusting the tape drive speed. From a comparison of telemetry signals with radar tracking information, DiscOps determines whether the orbital timer's programmed subcycle period should be maintained, increased, or decreased to match the real time duration of the satellite's orbital period. If a change is required, DiscOps then signals the timer via radar transponder-decoder Channel 2 to increase or decrease the timer period as necessary in from one to ninety-nine 11-second increments. "Increase" and "decrease" are selected alternately via radar transponder-decoder Channel 1 (this channel is kept in the "decrease" position except when an "increase" command is required). For a nominal real time orbital period of 94.0 minutes (5640 seconds), the corresponding subcycle period setting would be 22 steps above minimum (5642 seconds). This adjustment (and those described in Par. 7.3, 7.4, and 7.5) can be accomplished during any operating subcycle.

7.3 Orbital Timer Reset. The orbital timer is also provided with a reset capability that permits adjustment of the programmed event sequence forward or backward with respect to time and latitude. For each active subcycle, a reset latitude is chosen on the basis of the nominal orbital period, and a corresponding reset point is programmed on the timer tape (at 240 tape time units after transponder and telemetry ON). When necessary, DiscOps signals the timer via radar transponder-decoder Channel 3, resetting the timer tape to the programmed reset point for that subcycle, so that the reset point corresponds with the verified reset latitude established on the first orbit and substantiated on the second orbit (the direction and magnitude of tape displacement depend on the position of the tape's reset point at the moment the reset command is given). A subcycle identification mark is programmed on the timer tape at a coded time interval after each reset point. The intervals are coded in multiples of 20 seconds and are allocated so that each subcycle can be distinguished from those preceding and following it, since the interval registers on the reset monitor lamp at ground operations. This lamp, which lights at the reset point, is temporarily switched off after the coded time interval, permitting identification of the subcycle in which the orbital timer is operating.

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7.4 **Ejection Programming.** Re-entry can be programmed on any active subcycle during the useful life of the satellite. The programmed re-entry subcycles are arranged in two series, one alternately enabled and disabled by DiscOps signals to the timer via transponder-de-coder Channel 5, and the other alternately enabled and disabled by signals via Channel 6. Thus, re-entry on a subcycle in one series is enabled by a signal via the transponder-decoder channel controlling that series and disabled by the next signal via the same channel. The enabling and/or disabling signals are effective on the next programmed re-entry in the series to which they apply.

7.5 **Recoverable Capsule Ejection.** The capsule ejection phase covers the time from D timer restart by the orbital timer through D timer shutoff after completion of capsule ejection sequence events. On the orbit preceding that on which capsule ejection occurs, the programmed orbital timer tape is reset by DiscOps to assure that on the following orbit the timer initiates the ejection sequence (by restarting the D timer) at the optimum point. The orbital timer initiates the ejection sequence at E-94.5 seconds (E-0 being the time of ejection) by turning off the horizon scanner and restarting the D timer. At E-72.0 seconds, the D timer (arming signal): (a) activates the capsule's Blossom telemetry battery; (b) turns on power to the Blossom telemetry filaments and plates; (c) turns on the capsule radio beacon; (d) ignites thermal relays to arm the capsule's re-entry programmer; (e) energizes the capsule's recovery system arming relay; and (f) torques the IRP pitch gyro to pitch the satellite (from its tail-first, -1.5° pitch attitude on orbit) at a $+45^{\circ}$/min rate (relative to the satellite's direction of travel), rotating the satellite's nose downward in order to achieve a -130.5° pitch angle for capsule ejection. At E-2.5 seconds, the D timer (transfer signal): (a) changes the IRP pitch gyro rate to a residual -3.86°/min; (b) actuates the capsule's re-entry system thermal batteries; (c) backs up the recovery system arming signal; and (d) actuates a pyrotechnic delay switch to initiate firing of the satellite - capsule electrical disconnect squib. At approximately E-1.5 seconds, the disconnect squib activated by the D timer transfer signal fires, disconnecting the satellite - capsule electrical cable. Cable disconnection switches capsule equipment from the satellite electrical power system to capsule-borne electrical power, and lifts a ground loop to start the re-entry programmer. At E-0, the D timer (separation signal) ignites the satellite - capsule separation pin puller squibs. The squibs fire (0 to 7 milliseconds delay), releasing the physical connection between the satellite and capsule, and four compressed springs push the capsule away from the satellite at a velocity of about 1.7 ft/sec. At E + 1.9 seconds, the re-entry programmer (Event 1) actuates spin jets that spin the capsule to about 78 rpm in approximately 0.8 seconds. At E + 3.15 seconds, the re-entry programmer (Event 2) ignites the capsule's retro rocket, which imparts approximately 4 g acceleration for about 9 seconds. At E + 13.9 seconds, the re-entry programmer (Event 3) actuates de-spin jets that reduce the capsule spin rate to about 10 rpm. At E + 15.4 seconds, the re-entry programmer (Event 4): (a) ignites the capsule - thrust cone electrical disconnect squib; and (b) ignites the capsule - thrust cone explosive separation bolts. These fire, and the capsule separates from its thrust cone. At E + 59.5 seconds, the D timer: (a) reactivates the horizon scanner; (b) switches the horizon scanner to low gain; and (c) turns itself off. (For further detail on capsule re-entry, ref. Sect. II pp 8.9 ff, "Capsule Descent System".)

7.6 **Orbit Phase -- Resumption.** After the horizon scanner is reactivated, as noted above, it senses the satellite's -130.5° pitch angle and signals the control system to return the satellite to its horizontal attitude with respect to the earth's surface. Programmed operation of satellite communications equipment (ref. Par. 7.1, above) continues. The total number of subcycles programmed on the orbital timer tape varies from one satellite to the next. At the end of the programmed sequence, all items of satellite communications equipment are turned on for continuous operation until depletion of the satellite electrical power supply.

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NOTE: Back-up signal for certain commands are provided as indicated below in the event that the original command is not received. Compliance with the back-up signal vice the original signal will affect neither the order of nor the time separation between subsequent events.

1.0 GENERAL

Timed Sequence of Events for Discoverer Satellites 6205-1107 & 1108, Flight Configuration VI. Launch Site: Vandenberg AFB, Pad 1 (1107), Pad 4 (1108). Launch Azimuth = 172 degrees. Perigee altitude* = 130 nautical miles; apogee altitude* = 253 nautical miles. Eccentricity* = 0.017.

2.0 COUNTDOWN PHASE

The countdown phase begins nominally 8.5 hours before launch. Certain systems in the Discoverer satellite are turned on during countdown for warmup and checkout by Discoverer ground operations (DiscOps) per the applicable Development Test Order (DTO). The Discoverer booster is similarly checked out by Douglas Ground Operations (DacOps) and propellant loading operations are accomplished. Satellite guidance and flight control electronics are turned on about 380 minutes before launch and remain on thereafter. Terminal countdown is started 9.5 minutes before launch. From this point on, much of the launching procedure -- including the initiation of Discoverer booster ignition -- is controlled by an automatic launch sequencer. Failure in any system controlled or monitored by the sequencer results in an automatic hold on the terminal countdown. The hold remains in effect until the malfunction is located and corrected, or until the decision is made to recycle or terminate the countdown. Certain items of satellite guidance and communications equipment are turned on for flight operation during the terminal countdown. The identification transmitter and the dual-frequency tracking beacon operate continuously thereafter until depletion of the satellite electrical power supply. Other such items are subsequently controlled as noted in succeeding paragraphs of this sequence.

Seconds From Launch Time(T)	Orig. System	Event
T-570	DacOps	Turns on launch sequencer to start terminal countdown.
T-510	DacOps	Turns on booster telemetry and command destruct radio receivers.
T-510		Operational hold on terminal countdown scheduled.
T-510	BMD/LC; Range Safety	BMD Launch Control waives or releases operational hold when booster telemetry and command destruct radio receivers are verified ON by Flight Safety Officer.
T-420	DiscOps	Turns on satellite identification transmitter, dual-frequency tracking beacon, radar transponder, and telemetry, and tracking of satellite begins.
T-150	DacOps	Switches electrical power for all booster requirements from ground power source to booster electrical power system.
T-150	DiscOps	Switches electrical power for all satellite requirements from ground power source to satellite electrical power system.
T-120	DiscOps	Arms flight termination system via auxiliary umbilical from blockhouse to satellite.
		Enables release of main umbilical from blockhouse to satellite.
		Turns on nose cone battery.

* Specifically applicable to Satellite 1107.
Generally applicable to Satellite 1108.

Seconds From Launch Time(T)	Orig. System	Event
T-90	DiscOps	Turns on SS/D timer motor (safety circuit holds timer brake ON via auxiliary umbilical) and horizon scanner.
T-90	Range Safety	Flight Safety Officer verifies CLEAR TO LAUNCH has not been received from Flight Safety Officer; releases hold when CLEAR TO RELEASE HOLD is received from the Range Safety Officer and from the Vandenberg and Satellite Test Center (STC -- Sunnyvale) Control Centers, if hold has been imposed.
T-4	DacOps	Automatically initiates Discoverer booster ignition.
T-2	Booster	Signal initiated by burning through of wire across booster nozzle disconnects main umbilical to satellite.

3.0 LAUNCH PHASE

During the launch phase, the ground-guided Discoverer booster provides thrust and guidance to place the Discoverer vehicle at a predetermined altitude, to orient the vehicle in a predetermined attitude, and to impart to it a predetermined velocity. Guidance during this phase is provided by booster roll and pitch programs and by a closed-loop booster guidance system. The roll program, effective immediately after launch, rolls the vehicle to place its pitch plane in the plane of the intended final (orbital) trajectory. The pitch program, effective upon completion of the roll program, pitches the vehicle to move the vehicle yaw plane from its vertical orientation at launch to the elevation angle required at the end of boost. The closed-loop guidance system, which employs a ground-based radar and computer, generates pitch and yaw steering commands to correct vehicle deviations from a reference trajectory set into the computer. The ground-based radar tracks the vehicle, and the computer relates vehicle position to a coordinate system, determines the vehicle's velocity component along each axis of the system, compares these velocity components with the inset reference conditions, and generates corrective steering commands as required.

Booster main engine cut-off (MECO) occurs: (a) by command from the closed-loop guidance system when the predetermined altitude and velocity conditions are satisfied; or (b) at propellant depletion. After MECO, the computer continues to compare vehicle position and velocity with its inset reference data. On the basis of these comparisons, it computes satellite SS/D timer hold and velocity correction values required in order for the Discoverer satellite to achieve the predetermined orbital injection conditions and then transmits this information in the form of command signals via radar to the vehicle. The effect of the SS/D timer hold is to delay the start of satellite propulsion system operation, and the effect of the velocity correction is to reduce the velocity-to-be-gained setting of the satellite's accelerometer integrator, thus reducing the duration of orbital boost engine operation. The timer hold and velocity correction values are represented by two sequential command signals. The velocity correction information is represented by the duration of the first signal, and the timer hold information is represented by the combined duration of both signals. The first signal is also used to uncage the satellite inertial reference package (IRP) gyros.

When these commands have been transmitted, the booster guidance system transmits a separation signal which fires the explosive separation bolts and ignites the retro rockets on the adapter to effect separation of the Discoverer satellite from the adapter - booster combination.

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The sequence of events presented below is based on a nominal SS/D timer hold duration. If a hold of different length is employed, the times for subsequent events will vary accordingly, but the events themselves will follow in the same intervals.

Seconds From Launch Time(T)	Seconds From Phase Start	Orig. System	Event
T-0	0	Booster	Liftoff -- booster approaches full thrust, and vehicle moves two inches up from pad.
T-0	0	Booster	Pitch programmer commands 0° pitch rate in order to maintain vehicle's vertical flight path (for approximately 10 seconds)
T+1*	1*	D	Satellite auxiliary umbilical disconnects, opening SS/D timer safety circuit to release hold on DD/D timer and start programmed event sequencing.
T+1.1	1.1	H	Tracking of satellite via radar transponder begins.
T+2	2	Booster	Roll programmer initiates program that rolls vehicle from pad azimuth to launch azimuth.#
**	**	DacOps	Radar tracking of booster begins.
T+9	9	Booster	Roll program ends.
T+10	10	Booster	Pitch programmer initiates program that pitches vehicle -61.2° (requires approximately 120 seconds).
T+20	20	DacOps	Synchronizes booster closed-loop guidance system computer time reference.
T+90	90	DacOps	Begins steering booster via closed-loop guidance system.
***	***	L	Barometric switch closes to arm recoverable capsule ejection system
T+130	130	Booster	Pitch program ends.
T+147.9	147.9	DacOps	Steering of booster via closed-loop guidance system ends (4 seconds before MECO).
T+151	151	D timer	Starts orbital timer.
			Disarms satellite destruct system (opens flight termination battery circuit).
			Turns on satellite flight control electrical power (back-up signal)
			Uncages integrator to permit reset of integrator dial reading for velocity gain.**** 4.0
			Opens pneumatic control gas supply valve.
			Arms separation squib and retro rocket relays.
			Enables DacOps commands to satellite via booster guidance system.

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
T+151.2	151.2	Booster	Nominal time for MECO by propellant depletion.
T+151.7	151.7	DacOps	Nominal time for transmission of MECO command via booster closed-loop guidance system (command is transmitted even if MECO by propellant depletion has occurred previously).
			MECO command signal also serves as backup signal to disarm satellite destruct system.
T+158.8	158.8	DacOps	Arms satellite SS/D timer hold and velocity correction via booster closed-loop guidance system.
T+159.5	159.5	DacOps	Uncages satellite IRP pitch, yaw, and roll gyros via booster closed-loop guidance system.
			Initiates satellite SS/D timer hold and velocity correction via booster closed-loop guidance system.
T+160.1	160.1	Booster	Booster timer commands booster vernier engine cut-off (VECO)
T+162.6	162.6	DacOps	Terminates SS/D timer hold and velocity correction.
T+162.7	162.7	DacOps	Initiates additional SS/D timer hold via booster closed-loop guidance system.
T+167.6	167.6	DacOps	Terminates satellite SS/D timer hold.
T+168.0	168.0	DacOps	Back-up signal uncages IRP pitch, yaw, and roll gyros.
T+168.3		A	Transmits separation command signal via booster closed-loop guidance system to fire separation pin puller squibs and ignite retro rockets, initiating separation of satellite from adapter and booster.
T+167.8 to T+169.3	167.8 to 169.3	D timer	Adapter-mounted retro rockets (2) burn to complete separation of satellite from adapter and booster. Burn time = 0.5 + 1.0/-0.0 second.

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COAST PHASE

Seconds From Launch	Seconds From Phase Start	Orig. System	Event
171	0	A	Separation switch activates satellite pitch, yaw, and roll pneumatic control as satellite separates from adapter and booster.

* Nominal value - varies approximately from 0.5 to 1.5 seconds.
** Can occur at any time from lift-off to T+20.
*** Event occurs at approximately 70,000 feet above sea level, about 93 seconds after launch.
**** Integrator dial set at reading of 2128, representing a velocity-to-be-gained of 17,200 ±40 fps.
\# 1107 - Pad 1: -87° to 172°.
1108 - Pad 4: -9.5° to 172°.

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Seconds From Launch	Seconds From Phase Start	Orig. System	Event
183	12	D timer	Back-up signal fires separation pin puller squibs and ignites retro rockets to assure separation of satellite from adapter and booster.
			Fires horizon scanner fairing squib to jettison fairing.
186	15	D timer	Turns off power to pneumatic control gas supply valve.
			Switches Telemetry Channel 18 from booster vibration to satellite engine turbine speed.
191	20	D timer	Back-up signal activates pitch, yaw, and roll pneumatic control.
			Activates integrator potentiometer circuit that torques IRP pitch gyro to pitch satellite at an initial -1.66°/min (increases to a maximum of -4.19°/min) in order to compensate for satellite's geocentric angular motion.
			Torques IRP pitch gyro to pitch satellite -28.8° at -3.6°/sec rate for 10 seconds in order to achieve horizontal attitude.
199	28	D timer	Removes -3.6°/sec pitch rate.
			Connects horizon scanner pitch signal to IRP pitch gyro and horizon scanner roll signal to IRP gyro.

5.0 ORBITAL BOOST PHASE

Seconds From Launch	Seconds From Phase Start	Orig. System	Event
212	0	D timer	Fires ullage control rocket squibs.
212 to 230	0 to 18	B	Ullage control rockets burn. Nominal burn time capability of 18.8 seconds provides positive overlap with engine thrust in order to maintain ullage control during engine ignition.
224	12	D timer B	Turns on +28v dc to engine relay box, activating circuit that shuts off pitch and yaw pneumatic control.
224	12	D timer	Arms gas generator squib; fires gas generator and helium valve squibs.
			Ungrounds integrator input and connects IRP accelerometer to integrator.
225	13	D timer	Back-up signal turns off pitch and yaw pneumatic control.
			Deactivates gas generator and helium valve squib firing circuits and gas generator arming circuit.

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
225	13	D timer	Disarms engine relay box pitch and yaw pneumatic control OFF circuit.
			Locks telemetry on through bypass circuit.
225.5	13.5	B	Engine achieves steady state thrust (1.3 to 1.9 seconds after firing of gas generator and helium valve squibs).
450	238	D timer	Arms pitch and yaw pneumatic control ON circuit.
			Deactivates engine cut-off safety switch, which has been preventing possible premature shutdown signal from integrator.
456	244	D velocity integrator	Engine shutdown signal from integrator to engine relay box cuts off electrical power to engine and propellant subsystems when programmed velocity gain, as corrected by DiscOps via transponder-decoder Channel 6, has been achieved.
			Integrator switch disconnects IRP accelerometer from integrator and grounds integrator input.
456	244	B	Engine relay box activates circuit turning on pitch and yaw pneumatic control

6.0 REORIENTATION PHASE

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
484	0	D timer	Shuts down hydraulic control system and supplied back-up signal to turn on pitch and yaw pneumatic control
			Shuts off ullage control rocket firing circuits.
			Disconnects integrator pitch rate potentiometer to remove geocentric angular pitch rate.
484	0	D timer	Torques IRP yaw gyro to yaw satellite 180° counterclockwise at -40°/min rate for 4.5 minutes to achieve tail-first horizontal attitude.
			Fires oxidizer and helium vent valve squibs.
484 up	0 up	B	Helium pressure spheres and lines lose pressure, and oxidizer tank vents through thrust nullifier.
754	170	D timer	Turns on +28v dc to payload and capsule.
			Starts telemetry calibration.

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Seconds From Launch Time(T)	Seconds From Phase Start	Orig. System	
754	270	D timer	Connects Horizon scanner roll signal to IRP yaw gyro (stops -40°/min yaw rate) to achieve 3-axis control from horizon scanner.
			Changes IRP pitch gyro rate to residual +3.86°/min.
			Shuts off regulated +28v dc and 0.1% regulated 19 400 cps 115 v ac to accelerometer.
764	280	H	Ledex stepping switch stops telemetry calibration and switches telemetry from ascent functions to orbit functions.
		D timer	Supplies redundant signal to stop telemetry calibration and switch telemetry from ascent to orbit functions.
			Switches telemetry and identification transmitter operation from VHF exit antenna to VHF orbit antenna.
			Opens engine shutdown circuit and shuts off power to integrator.
764	280	D timer	Switches flight control system to low gain; switches IRP yaw and roll gyro telemetry to high gain.
771	287	D timer	Switches telemetry bypass circuit off.
			Switches horizon scanner signals to low gain.
			Fires fuel vent valve squib.
			Arms horizon scanner OFF circuit.
			Shuts itself off.
771 up	287 up	B	Fuel tank vents through thrust nullifiers.
931	447	H timer	Orbital timer turns off power to transponder-decoder and telemetry plates and filaments.

7.0 ORBIT PHASE

During the orbit phase, the satellite makes a succession of revolutions about the earth in an orbital plane of approximately 81.7° inclination. The first south-to-north equatorial crossing, approximately 60 minutes after launch, marks the starting point of the first orbital period. With each orbit, the satellite crosses the equator at a point further west than for the previous one, so that the 16th orbit starts at roughly the same longitude as the first orbit.

Satellite equipment ON-OFF functions during the orbit phase are controlled by the orbital timer in a programmed event sequence occuring in a succession of subcycles. A subcycle can be defined as that part of the programmed event sequence which corresponds to one orbit's operations. Subcycles are consecutively numbered and begin at the ascending nodes. At launch, the orbital timer is phased such that Subcycle 1 corresponds to Orbit 1. If the nominal orbital period of 91.6 minutes is achieved, the operations within each subcycle will be those required by the territory under the satellite.

During orbits in which the satellite is within range of the read-in and command (RI/C) stations in New Hampshire, California, Alaska, and Hawaii, the transponder-decoder and telemetry are turned on and off by the orbital timer. Subcycles in which this is programmed to occur are designated as operating subcycles. The orbital timer also turns the satellite's tracking light installation on and off in a sequence programmed to correspond with the satellite's nighttime passes over optical tracking stations equipped with Baker-Nunn cameras. A summary of the programming for these events and the location of ground stations are presented in Tables I - III below (ref. also Sect. V, p. 1.1.2).

The orbital timer program is prepared for a satellite orbital period capability of from 90.3 to 100.15 minutes. A subcycle period is represented on the tape by 5400 tape time units, and the real time span of the 5400 tape units can be expanded or contracted to correspond with the satellite's actual orbital period by adjusting the tape drive speed. From a comparison of telemetry signals with radar tracking information DiscOps determines whether the orbital timer's programmed subcycle period should be maintained, increased, or decreased to match the real time duration of the satellite's orbital period. If a change is required, DiscOps then signals the timer via transponder-decoder Channel 2 to to increase or decrease the timer period as necessary in from one to ninety-nine 11-second increments. "Increase" and "decrease" are selected alternately via transponder-decoder Channel 1 (this channel is kept in the "decrease" position except when an "increase" command is required). This adjustment (and those described in the following paragraphs) can be accomplished during any active subcycle. The nominal real time orbital period for this flight is 91.6 minutes (5496 seconds), and the corresponding subcycle period setting is 9 steps above minimum (5499 seconds).

The orbital timer is also provided with a reset capability that permits adjustment of the programmed event sequence forward or backward with respect to time and latitude. When necessary, DiscOps signals the timer via transponder-decoder Channel 3, resetting the timer tape to the programmed reset point for that subcycle, so that the reset point corresponds with the verified reset latitude and time established on the first orbit and substantiated on the second orbit (the direction and magnitude of tape displacement depend on the position of the tape's reset point at the moment the reset command is given). A subcycle identification mark is programmed on the timer tape at a coded time interval after the reset point. This interval is coded in multiples of 20 seconds and is allocated so that each subcycle is coded differently than those near it. The reset monitor lamp at ground control, which lights at the reset point, is temporarily switched off after the coded time interval, permitting identification of the subcycle in which the timer is operating.

[redacted]

* Applies to Satellite 1106 only.

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Re-entry can be programmed on any active subcycle during the useful life of the satellite. The programmed re-entry subcycles are arranged in two series, one alternately enabled and disabled by DiscOps signals to the timer via transponder-decoder Channel 5, and the other alternately enabled and disabled by signals via Channel 6. For Satellite 1107 re-entry is programmed on Subcycles 9, 16, 18, 24, 31, 33, 40, 47, 49, 55, 63, 65, 79, and 81 -- controlled via Channel 5 -- and on Subcycles 10, 17, 25, 32, 41, 48, 56, 64, 65, 80, and 81 -- controlled via Channel 6.

TABLE I * OPERATIONAL PROGRAMMING

	Type of Pass						
	Day Southbound				Night Northbound		
Code	A	B	C	D	E	F	G
R/C Stations							
New Boston	x				x		
Vandenberg		x	x			x	
Kodiak			x	x			x
Hawaii				x			x
Event Latitudes							
Transp. & T/M Plates On	61°N	52°N	75°N	75°N	25°N	17°N	5°N
Reset Monitor	45°N	36°N	60°N	60°N	40°N	32°N	20°N
Transp. & T/M Plates Off	34°N	22°N	22°N	12°N	60°N	52°N	65°N

Subcycle Change Area: Between 60°S & 40°S on northbound pass

TABLE II TRACKING LIGHT PROGRAMMING **

Code	Station	ST. No.	Latitude (Deg.)	Longitude (°W)
N	Cadiz, Spain	4	36.459 N	6.207
P	Villa Dolores, Argentina	11	31.936 S	65.112
Q	Arequipa, Peru	7	16.462 S	71.491
R	Curacao, NWI	9	12.097 N	68.837
S	Jupiter, Florida	10	27.020 N	80.113
T	Organ Pass, N.M.	1	32.423 N	106.552
U	Maui, Hawaii	12	20.710 N	156.260
V	Woomera, Australia	3	31.102 S	223.217
W	Tokyo, Japan	5	35.670 N	220.459
X	Nainital, India	6	29.359 N	280.546
Y	Olifantsfontein, So. Africa	2	25.959 S	331.752
Z	Shiraz, Iran	8	29.645 N	307.474

For all codes: a 45-second flash is programmed directly over each station.
For Codes P & Q: a 45-second flash is programmed halfway between Stations P & Q.

* Specifically applicable to Satellite 1107; generally applicable to Satellite 1108.
** Applies to Satellite 1108 only.

TABLE III* SUBCYCLE ALLOCATION

Subcycle	Nominal Ascend. Node (°W)	Pass Code	Subcycle Ment. Mark (Sec.)	Tracking** Lights Code	Re-entry Command Channel
0	Ascent	VTS	40		
1	309	D	20		
2	332	D	60		
3	355			N	
4	18			N	
5	41			P-Q	
6	64	E	60	R	
7	87	E	20	S	
8	110	F	80	T	
9	133	F	40	U	5
10	156	G	60	U	6
11	179	G	20	V	
12	202			W	
13	225	A	60	W	
14	248	A	20		
15	271	B	80	X	
16	294	C	40	Y & Z	5
17	317	D	60		6
18	340	D	20	N	5
19	3			N	
20	26			P-Q	
21	49	E	60	R	
22	72	E	20	S	
23	95	E	80	T	
24	118	F	40		5
25	141	G	60	U	6
26	165	G	20	V	
27	188			W	
28	211	A	60	W	
29	234	A	20		
30	257	A	80	X	
31	280	C	40	Y & Z	5
32	303	D	60		6
33	326	D	20		5
34	349			N	
35	12		20	P-Q	
36	35		40	R	
37	58	E	60	S	
38	81	E	20	T	
39	104	F	80		
40	127	F	40	U	5
41	150	G	60	V	6
42	173	G	20	V	
43	196			W	
44	219	A	60		
45	242	A	20	X	
46	265	B	80	Y & Z	
47	288	C	40		5
48	311	D	60		6
49	334	D	20	N	5
50	357			P-Q	
51	20		20	R	
52	43	E	60	R & S	

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TABLE III SUBCYCLE ALLOCATION (Continued)

Subcycle	Nominal Ascend. Node (°W)	Pass Code	Subcycle Mont. Mark (Sec.)	Tracking Lights Code	Re-entry Command Channel
53	66	E			
54	89	E	20	T	
55	112	F	80		
56	135	G	40	U	5
57	158	G	60		6
58	181		20	V	
59	204	A		W	
60	227	A	40		
61	250	B	60	X	
62	273	D	20	Y, Z	
63	296	D	80	Y, Z	
	319	D	40		5
64	342	D	60		
			20	N	
66	5			P-Q	6
67	28			P-Q	5
68	51	E		R & S	
69	74	E	60	T	
70	97	E	20		
71	120	F	80	U	
72	143	G	40		
73	166	G	60	V	
74	189		20	W	
75	212				
76	236	A		X	
77	259	A	60	Y & Z	
78	282	B	20	Y & Z	
79	305	D	80		
80	328	D	40		
81	351	D	60	N	5
82	14		20	P-Q	6
83	37			P-Q	5
84	60	E		R & S	
85	83	E	60	T	
86	106	F	20	T	
87	129	F	80	U	
88	152	G	40	U	
89	175	G	60	V	
90	198		20	W	
91	221	A			
92	244	A	60		
93	267	B	20	X	
94	290	D	80		
95	313	D	40		
			60		

8.0 CAPSULE EJECTION AND RECOVERY PHASE

On the orbit preceding that on which capsule ejection occurs, the programmed orbital timer tape is reset by DiscOps to assure that on the following orbit the timer initiates the ejection sequence (by restarting the SS/D timer) at the optimum point. The ejection and recovery sequence events listed below are referenced to ejection time (E) or to altitude above sea level.

Time From Ejection (E)	Event
E - 94.5 sec (±0.5 sec)	Orbital timer initiates ejection and recovery sequence by turning off horizon scanner and restarting SS/D timer.
E - 81.5 sec (±0.1 sec)	D timer (arming signal): (a) activates Blossom telemetry battery; (b) turns on power to Blossom telemetry filaments and plates; (c) turns on capsule radio beacon; (d) ignites thermal relays to arm ejection programmer; (e) energizes recovery system arming relay; and (f) torques IRP pitch gyro to rotate satellite at +45°/min rate from tail-first horizontal position in order to achieve -130.5° pitch angle for capsule ejection.
E - 2.5 sec (±0.1 sec)	D timer (transfer signal): (a) changes IRP pitch gyro rate to residual - 3.86°/min; (b) actuates ejection system thermal batteries; (c) backs up recovery system arming signal; and (d) actuates pyrotechnic delay switch to initiate firing of satellite - capsule electrical disconnect squib.
E - 1.5 sec (-0.5/+0.32)	Disconnect squib, activated by D timer transfer signal, fires, disconnecting satellite - capsule electrical cable.
E - 1.5 sec (-0.5/+0.32)	Cable disconnection switches capsule equipment from satellite auxiliary power system to capsule-borne electrical power and lifts ground loop to start ejection programmer.
E - 0	D timer (separation signal) ignites satellite - capsule separation pin puller squibs (0 to 7 millisecond delay).
	Four compressed springs push capsule away from satellite, imparting a velocity of about 1.7 ft/sec.
E + 1.9 sec (±0.17 sec)	Ejection programmer (Event 1) actuates spin jets which spin capsule to about 78 rpm in approximately 0.8 second.
E + 3.15 sec (±0.27 sec cum. tol.)	Ejection programmer (Event 2) ignites retro rocket which imparts approximately 4 g acceleration for about 9 seconds (total impulse = 10,470 lb-sec).
E + 13.9 sec (±0.81 sec cum. tol.)	Ejection programmer (Event 3) actuates de-spin jets which reduce capsule spin rate to about 10 rpm.
E + 15.4 sec (±0.96 sec cum. tol.)	Ejection programmer (Event 4): (a) ignites thrust cone - capsule electrical disconnect squib; and (b) ignites thrust cone - capsule explosive separation bolts.
	Capsule separates from thrust cone.
E + 35.5 sec	D timer: (a) turns on horizon scanner; (b) switches horizon scanner to low gain; and (c) turns itself off.

Feet Above Sea Level (Approx.)	Event
350,000 ft	Re-entry friction heats recoverable capsule to point at which ionized layer forms on capsule surface, causing r.f. blackout.
225,000 ft	G switch closes (at 3 to 6.6 g) as re-entry deceleration increases.
135,000 ft	Re-entry deceleration reduces friction and permits recoverable capsule surface cooling to point at which ionized layer disappears, ending r.f. blackout.

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Feet Above Sea Level (Approx.)	Event
100,000 ft	G switch opens (at 3 g ±5%) as re-entry deceleration decreases, permitting recovery programmer timing circuit to begin operation.
55,000 ft (g switch opening + 47.0±1.7 sec)	Timer switch closes, and recovery programmer: (a) ignites ejection pistons; and (b) switches power to recovery system flashing light.
	Ejection pistons: (a) release ablative shell from capsule; and (b) blow parachute cover off.
	Ejected parachute cover pulls out pilot chute, which in turn pulls out the main parachute bag with the chute in reefed condition.
	Mechanically-actuated time-delay pyrotechnic cutter (4-second delay) disreefs main parachute, permitting chute deployment.
	Radar reflective chaff, packed with the parachute, falls free as the chute emerges from its bag.
	Parachute system decelerates recoverable capsule, and ablative shell falls clear of capsule.

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NOTE: Back-up signal for certain commands are provided as indicated below in the event that the original command is not received. Compliance with the back-up signal vice the original signal will affect neither the order of nor the time separation between subsequent events.

1.0 GENERAL†

Timed Sequence of Events for Discoverer Satellites 6205-1107 & 1108, Flight Configuration VI. Launch Site: Vandenberg AFB, Pad 1 (1107), Pad 4 (1108). Launch Azimuth = 172 degrees. Perigee altitude* = 165 nautical miles; apogee altitude* = 336 nautical miles. Eccentricity* = 0.023

2.0 COUNTDOWN PHASE

The countdown phase begins nominally 8.5 hours before launch. Certain systems in the Discoverer satellite are turned on during countdown for warmup and checkout by Discoverer ground operations(DiscOps) per the applicable Development Test Order (DTO). The Discoverer booster is similarly checked out by Douglas Ground Operations (DacOps) and propellant loading operations are accomplished. Satellite guidance and flight control electronics are turned on about 350 minutes before launch and remain on thereafter. Terminal countdown is started 9.5 minutes before launch. From this point on, much of the launching procedure -- including the initiation of Discoverer booster ignition -- is controlled by an automatic launch sequencer. Failure in any system controlled or monitored by the sequencer results in an automatic hold on the terminal countdown. The hold remains in effect until the malfunction is located and corrected, or until the decision is made to recycle or terminate the countdown. Certain items of satellite guidance and communications equipment are turned on for flight operation during the terminal countdown. The identification transmitter and the dual-frequency tracking beacon operate continuously thereafter until depletion of the satellite electrical power supply. Other such items are subsequently controlled as noted in succeeding paragraphs of this sequence.

Seconds From Launch Time(T)	Orig. System	Event
T-570	DacOps	Turns on launch sequencer to start terminal countdown.
T-510	DacOps	Turns on booster telemetry and command destruct radio receivers.
T-510		Operational hold on terminal countdown scheduled.
T-510	BMD/LC; Range Safety	BMD Launch Control waives or releases operational hold when booster telemetry and command destruct radio receivers are verified ON by Flight Safety Officer.
T-420	DiscOps	Turns on satellite identification transmitter, dual-frequency tracking beacon, radar transponder, and telemetry, and tracking of satellite begins.
T-150	DacOps	Switches electrical power for all booster requirements from ground power source to booster electrical power system.
T-150	DiscOps	Switches electrical power for all satellite requirements from ground power source to satellite electrical power system.
T-120	DiscOps	Arms flight termination system via auxiliary umbilical from blockhouse to satellite.
		Enables release of main umbilical from blockhouse to satellite.
		Turns on nose cone battery.

Seconds From Launch Time(T)	Orig. System	Event
T-90	DiscOps	Turns on SS/D timer motor (safety circuit holds timer brake ON via auxiliary umbilical) and horizon scanner.
T-90	Range Safety	Flight Safety Officer verifies CLEAR TO LAUNCH has not been received from Flight Safety Officer; releases hold when CLEAR TO RELEASE HOLD is received from the Range Safety Officer and from the Vandenberg and Satellite Test Center (STC -- Sunnyvale) Control Centers, if hold has been imposed.
T-4	DacOps	Automatically initiates Discoverer booster ignition.
T-2	Booster	Signal initiated by burning through of wire across booster nozzle disconnects main umbilical to satellite.

3.0 LAUNCH PHASE

During the launch phase, the ground-guided Discoverer booster provides thrust and guidance to place the Discoverer vehicle at a predetermined altitude, to orient the vehicle in a predetermined attitude, and to impart to it a predetermined velocity. Guidance during this phase is provided by booster roll and pitch programs and by a closed-loop booster guidance system. The roll program, effective immediately after launch, rolls the vehicle to place its pitch plane in the plane of the intended final (orbital) trajectory. The pitch program, effective upon completion of the roll program, pitches the vehicle to move the vehicle yaw plane from its vertical orientation at launch to the elevation angle required at the end of boost. The closed-loop guidance system, which employs a ground-based radar and computer, generates pitch and yaw steering commands to correct vehicle deviations from a reference trajectory set into the computer. The ground-based radar tracks the vehicle, and the computer relates vehicle position to a coordinate system, determines the vehicle's velocity component along each axis of the system, compares these velocity components with the inset reference conditions, and generates corrective steering commands as required.

Booster main engine cut-off (MECO) occurs: (a) by command from the closed-loop guidance system when the predetermined altitude and velocity conditions are satisfied; or (b) at propellant depletion. After MECO, the computer continues to compare vehicle position and velocity with its inset reference data. On the basis of these comparisons, it computes satellite SS/D timer hold and velocity correction values required in order for the Discoverer satellite to achieve the predetermined orbital injection conditions and then transmits this information in the form of command signals via radar to the vehicle. The effect of the SS/D timer hold is to delay the start of satellite propulsion system operation, and the effect of the velocity correction is to reduce the velocity-to-be-gained setting of the satellite's accelerometer integrator, thus reducing the duration of orbital boost engine operation. The timer hold and velocity correction values are represented by two sequential command signals. The velocity correction information is represented by the duration of the first signal, and the timer hold information is represented by the combined duration of both signals. The first signal is also used to uncage the satellite inertial reference package (IRP) gyros.

When these commands have been transmitted, the booster guidance system transmits a separation signal which fires the explosive separation bolts and ignites the retro rockets on the adapter to effect separation of the Discoverer satellite from the adapter - booster combination.

† The entire sequence of events following is specifically applicable to Satellite 1108 as distinguished from the sequence of events on the preceding pages which are applicable to Satellite 1107.

* Specifically applicable to Satellite 1108.

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The sequence of events presented below is based on a nominal SS/D timer hold duration. If a hold of different length is employed, the times for subsequent events will vary accordingly, but the events themselves will follow in the same intervals.

Seconds From Launch Time(T)	Seconds From Phase Start	Orig. System	Event
T-0	0	Booster	Liftoff -- booster approaches full thrust, and vehicle moves two inches up from pad.
T-0	0	Booster	Pitch programmer commands 0° pitch rate in order to maintain vehicle's vertical flight path (for approximately 10 seconds)
T+1*	1*	D	Satellite auxiliary umbilical disconnects, opening SS/D timer safety circuit to release hold on DD/D timer and start programmed event sequencing.
T+1.1	1.1	H	Tracking of satellite via radar transponder begins.
T+2	2	Booster	Roll programmer initiates program that rolls vehicle from pad azimuth to launch azimuth#
**	**	DacOps	Radar tracking of booster begins.
T+9	9	Booster	Roll program ends.
T+10	10	Booster	Pitch programmer initiates program that pitches vehicle -61.2° (requires approximately 120 seconds).
T+20	20	DacOps	Synchronizes booster closed-loop guidance system computer time reference.
T+90	90	DacOps	Begins steering booster via closed-loop guidance system.
***	***	L	Barometric switch closes to arm recoverable capsule ejection system
T+130	130	Booster	Pitch program ends.
T+147.9	147.9	DacOps	Steering of booster via closed-loop guidance system ends (4 seconds before MECO).
T+151	151	D timer	Starts orbital timer.
			Disarms satellite destruct system (opens flight termination battery circuit).
			Turns on satellite flight control electrical power (back-up signal)
			Uncages integrator to permit reset of integrator dial reading for velocity gain.**** 4.0
			Opens pneumatic control gas supply valve.
			Arms separation squib and retro rocket relays.
			Enables DacOps commands to satellite via booster guidance system.

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
T+151.7	151.7	Booster	Nominal time for MECO by propellant depletion.
T+151.7	151.7	DacOps	Nominal time for transmission of MECO command via booster closed-loop guidance system (command is transmitted even if MECO by propellant depletion has occurred previously).
			MECO command signal also serves as backup signal to disarm satellite destruct system.
T+156.2	156.2	DacOps	Arms satellite SS/D timer hold and velocity correction via booster closed-loop guidance system.
T+158.9	158.9	DacOps	Uncages satellite IRP pitch, yaw, and roll gyros via booster closed-loop guidance system.
			Initiates satellite SS/D timer hold and velocity correction via booster closed-loop guidance system.
T+160.7	160.7	Booster	Booster timer commands booster vernier engine cut-off (VECO)
T+161.9	161.9	DacOps	Terminates SS/D timer hold and velocity correction.
T+162.7	162.7	DacOps	Initiates additional SS/D timer hold via booster closed-loop guidance system.
T+167.0	167.0	DacOps	Terminates satellite SS/D timer hold.
T+167.5	167.5	DacOps	Transmits separation command signal via booster closed-loop guidance system to fire separation pin puller squibs and ignite retro rockets, initiating separation of satellite from adapter and booster.
T+167.8 to T+169.3	167.8 to 169.3	A	Adapter-mounted retro rockets (2) burn to complete separation of satellite from adapter and booster. Burn time = 0.5 + 1.0/-0.0 second.
T+168.0	168.0	DacOps	Back-up signal uncages IRP pitch, yaw, and roll gyros.

COAST PHASE

Seconds From Launch	Seconds From Phase Start	Orig. System	Event
170	0	A	Separation switch activates satellite pitch, yaw, and roll pneumatic control as satellite separates from adapter and booster.

* Nominal value - varies approximately from 0.5 to 1.5 seconds.
** Can occur at any time from lift-off to T+20.
*** Event occurs at approximately 70,000 feet above sea level, about 93 seconds after launch.
**** Integrator dial set at reading of 2128, representing a velocity-to-be-gained of 17,200 ±40 fps.
\# 1108 - Pad 4: -9.5° to 172°.

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Seconds From Launch	Seconds From Phase Start	Orig. System	Event
179	9	D timer	Back-up signal fires separation pin puller squibs and ignites retro rockets to assure separation of satellite from adapter and booster.
			Fires horizon scanner fairing squib to jettison fairing.
180	10	D timer	Turns off power to pneumatic control gas supply valve.
			Switches Telemetry Channel 18 from booster vibration to satellite engine turbine speed.
182	12	D timer	Back-up signal activates pitch, yaw, and roll pneumatic control.
			Activates integrator potentiometer circuit that torques IRP pitch gyro to pitch satellite at an initial -1.65°/min (increases to a maximum of -4.19°/min) in order to compensate for satellite's geocentric angular motion.
			Torques IRP pitch gyro to pitch satellite -36° at -3.6°/sec rate for 10 seconds in order to achieve horizontal attitude.
192	22	D timer	Removes -3.6°/sec pitch rate.
			Connects horizon scanner pitch signal to IRP pitch gyro and horizon scanner roll signal to IRP gyro.

5.0 ORBITAL BOOST PHASE

Seconds From Launch	Seconds From Phase Start	Orig. System	Event
209	0	D timer	Fires ullage control rocket squibs.
209 to 227	0 to 18	B	Ullage control rockets burn. Nominal burn time capability of 18.8 seconds provides positive overlap with engine thrust in order to maintain ullage control during engine ignition.
221	12	D timer B	Turns on +28v dc to engine relay box, activating circuit that shuts off pitch and yaw pneumatic control.
221	12	D timer	Arms gas generator squib; fires gas generator and helium valve squibs.
			Ungrounds integrator input and connects IRP accelerometer to integrator.
222	13	D timer	Back-up signal turns off pitch and yaw pneumatic control.
			Deactivates gas generator and helium valve squib firing circuits and gas generator arming circuit.

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
222	13	D timer	Disarms engine relay box pitch and yaw pneumatic control OFF circuit.
			Locks telemetry on through bypass circuit.
222.3	13.2	B	Engine achieves steady state thrust (1.3 to 1.9 seconds after firing of gas generator and helium valve squibs).
452	243	D timer	Arms pitch and yaw pneumatic control ON circuit.
			Deactivates engine cut-off safety switch, which has been preventing possible premature shutdown signal from integrator.
459.6	250.6	D velocity integrator	Engine shutdown signal from integrator to engine relay box cuts off electrical power to engine and propellant subsystems when programmed velocity gain, as corrected by DiscOps via transponder-decoder Channel 6, has been achieved.
			Integrator switch disconnects IRP accelerometer from integrator and grounds integrator input.
459.6	250.6	B	Engine relay box activates circuit turning on pitch and yaw pneumatic control

6.0 REORIENTATION PHASE

Seconds From Launch Time (T)	Seconds From Phase Start	Orig. System	Event
484	0	D timer	Shuts down hydraulic control system and supplies back-up signal to turn on pitch and yaw pneumatic control
			Shuts off ullage control rocket firing circuits.
			Disconnects integrator pitch rate potentiometer to remove geocentric angular pitch rate.
484	0	D timer	Torques IRP yaw gyro to yaw satellite 180° counterclockwise at -40°/min rate for 4.5 minutes to achieve tail-first horizontal attitude.
			Fires oxidizer and helium vent valve squibs.
484 up	0 up	B	Helium pressure spheres and lines lose pressure, and oxidizer tank vents through thrust nullifier.
494	10	D timer	Turns on +28v dc to payload and capsule.
			Starts telemetry calibration.

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Seconds From Launch Time(T)	Seconds From Phase Start	Orig. System	
754	270	D timer	Connects Horizon scanner roll signal to IRP yaw gyro (stops -40°/min yaw rate) to achieve 3-axis control from horizon scanner.
			Changes IRP pitch gyro rate to residual +3.86°/min.
			Shuts off regulated +28v dc and 0.1% regulated 1φ 400 cps 115 v ac to accelerometer.
764	280	H	Ledex stepping switch stops telemetry calibration and switches telemetry from ascent functions to orbit functions.
		D timer	Supplies redundant signal to stop telemetry calibration and switch telemetry from ascent to orbit functions.
			Switches telemetry and identification transmitter operation from VHF exit antenna to VHF orbit antenna.
			Opens engine shutdown circuit and shuts off power to integrator.
764	280	D timer	Switches flight control system to low gain; switches IRP yaw and roll gyro telemetry to high gain.
771	287	D timer	Switches telemetry bypass circuit off.
			Switches horizon scanner signals to low gain.
			Fires fuel vent valve squib.
			Arms horizon scanner OFF circuit.
			Shuts itself off.
771 up	287 up	B	Fuel tank vents through thrust nullifiers.
931	447	H timer	Orbital timer turns off power to transponder-decoder and telemetry plates and filaments.

7.0 ORBIT PHASE

During the orbit phase, the satellite makes a succession of revolutions about
the earth in an orbital plane of approximately 81.7° inclination. The first
south-to-north equatorial crossing, approximately 60 minutes after launch,
marks the starting point of the first orbital period. With each orbit, the sat-
ellite crosses the equator at a point further west than for the previous one, so
that the 16th orbit starts at roughly the same longitude as the first orbit.

Satellite equipment ON-OFF functions during the orbit phase are controlled by
the orbital timer in a programmed event sequence occuring in a succession of
subcycles. A subcycle can be defined as that part of the programmed event
sequence which corresponds to one orbit's operations. Subcycles are
consecutively numbered and begin at the ascending nodes. At launch, the
orbital timer is phased such that Subcycle 1 corresponds to Orbit 1. If the
nominal orbital period of 93.8 minutes is achieved, the operations within each
subcycle will be those required by the territory under the satellite.

During orbits in which the satellite is within range of the read-in and command
(RI/C) stations in New Hampshire, California, Alaska, and Hawaii, the
transponder-decoder and telemetry are turned on and off by the orbital timer.
Subcycles in which this is programmed to occur are designated as operating
subcycles. The orbital timer also turns the satellite's tracking light instal-
lation on and off in a sequence programmed to correspond with the satellite's
nighttime passes over optical tracking stations equipped with Baker-Nunn cameras.
A summary of the programming for these events and the location of ground
stations are presented in Tables I - III below (ref. also Sect. V, p. 1.1.2).

The orbital timer program is prepared for a satellite orbital period capability
of from 90.0 to 100.15 minutes. A subcycle period is represented on the tape by
5400 tape time units, and the real time span of the 5400 tape units can be ex-
panded or contracted to correspond with the satellite's actual orbital period by
adjusting the tape drive speed. From a comparison of telemetry signals with
radar tracking information DiscOps determines whether the orbital timer's
programmed subcycle period should be maintained, increased, or decreased to
match the real time duration of the satellite's orbital period. If a change is
required, DiscOps then signals the timer via transponder-decoder Channel 2 to
to increase or decrease the timer period as necessary in from one to ninety-nine
11-second increments. "Increase" and "decrease" are selected alternately via
transponder-decoder Channel 1 (this channel is kept in the "decrease" position
except when an "increase" command is required). This adjustment (and those
described in the following paragraphs) can be accomplished during any active
subcycle. The nominal real time orbital period for this flight is 93.8 minutes
(5628 seconds), and the corresponding subcycle period setting is 21 steps above
minimum (5631 seconds).

The orbital timer is also provided with a reset capability that permits adjust-
ment of the programmed event sequence forward or backward with respect to
time and latitude. When necessary, DiscOps signals the timer via transponder-
decoder Channel 3, resetting the timer tape to the programmed reset point
for that subcycle, so that the reset point corresponds with the verified reset
latitude and time established on the first orbit and substantiated on the second
orbit(the direction and magnitude of tape displacement depend on the position of
the tape's reset point at the moment the reset command is given). A subcylce
identification mark is programmed on the timer tape at a coded time interval
after the reset point. This interval is coded in multiples of 20 seconds and is
allocated so that each subcycle is coded differently than those near it. The
reset monitor lamp at ground control, which lights at the reset point, is tempo-
rarily switched off after the coded time interval, permitting identification of
the subcycle in which the timer is operating.

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Re-entry can be programmed on any active subcycle during the useful life of the satellite. The programmed re-entry subcycles are arranged in two series, one alternately enabled and disabled by DiscOps signals to the timer via transponder-decoder Channel 5, and the other alternately enabled and disabled by signals via Channel 6. For Satellite 1106, re-entry is programmed on Subcycles 9, 15, 17, 24, 30, 32, 39, 46, 48, 54, 56, 62, 64, 77, and 79 -- controlled via Channel 5 -- and on Subcycles 16, 25, 31, 33, 40, 47, 55, 63, 64, 78, and 79 -- controlled via Channel 6.

TABLE I* OPERATIONAL PROGRAMMING

Code	Type of Pass: Day Southbound A	B	C	D	Night Northbound E	F	G
RI/C Stations							
New Boston	x				x		
Vandenberg		x	x			x	
Kodiak			x	x			x
Hawaii				x			x
Event Latitudes							
Transp. & T/M Plates On	61°N	52°N	75°N	75°N	25°N	17°N	5°N
Reset Monitor	45°N	36°N	60°N	60°N	40°N	32°N	20°N
Transp. & T/M Plates Off	34°N	22°N	22°N	12°N	60°N	52°N	65°N

Subcycle Change Area Between 60°S & 40°S on northbound pass

TABLE II TRACKING LIGHT PROGRAMMING **

Code	Station	ST. No.	Latitude (Deg.)	Longitude (°W)
N	San Fernando, Spain	4	36.459 N	6.207
P	Villa Dolores, Argentina	11	31.936 S	65.112
Q	Arequipa, Peru	7	16.462 S	71.491
R	Curacao, NWI	9	12.097 N	68.837
S	Jupiter, Florida	10	27.020 N	80.113
T	Organ Pass, N.M.	1	32.423 N	106.552
U	Maui, Hawaii	12	20.710 N	156.260
V	Woomera, Australia	3	31.102 S	223.217
W	Mitaka, Japan	5	35.670 N	220.459
X	Nainital, India	6	29.359 N	280.568
Y	Olifantsfontein, So. Africa	2	25.959 S	331.752
Z	Shiraz, Iran	8	29.645 N	307.474

For all codes: a 45-second flash is programmed directly over each station.
For Codes P & Q: a 45-second flash is programmed halfway between Stations P & Q.

* Specifically applicable to Satellite 1106
** Applies to Satellite 1106 only.

TABLE III SUBCYCLE ALLOCATION

Subcycle	Nominal Ascend. Node (°W)	Pass Code	Subcycle Ident. Mark (Sec.)	Tracking** Lights Code	Re-entry Command Channel
0	Ascent	VTS	40		
1	309	D	20		
2	333	D	60		
3	356			N	
4	20			N	
5	43			P-Q	
6	67	E	60	R	
7	91	E	20	S	
8	114	F	80	T	
9	138	F	40		5
10	161	G	60	U	
11	185			V	
12	208			V	
13	232	A	60	W	
14	256	A	20		
15	279	C	80	X	5
16	303	D	40	Y & Z	6
17	326	D	60		5
18	350			N	
19	14			N	
20	14			P-Q	
21	61	E	60	R	
22	84	E	20	S	
23	108	F	80	T	
24	131	F	40		5
25	155	G	60	U	6
26	179	G	20	V	
27	202	A	60	V	
28	226	A	20	W	
29	249	A	80		
30	273	C	40	X	5
31	296	D	60	Y & Z	6
32	320	D	20	Y & Z	5
33	344	D	80		6
34	7			N	
35	31			P-Q	
36	54	E	60	R	
37	78	E	20	R & S	
38	102	F	80	T	
39	125	F	40		5
40	149	G	60	U	6
41	172	G	20		
42	196			V	
43	219	A	60	W	
44	243	A	20		
45	267	B	80	X	
46	290	D	40	Y & Z	5
47	314	D	60	Y & Z	6
48	337	D	20		5
49	1			N	
50	25			P-Q	

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TABLE III 1108 SUBCYCLE ALLOCATION (Continued)

Subcycle	Nominal Ascend. Node (°W)	Pass Code	Subcycle Ident. Mark (Sec.)	Tracking Lights Code	Re-entry Command Channel
51	48	E	60	P-Q	
52	72	E	20	R & S	
53	95	E	80	T	
54	119	F	40	T	5
55	142	G	60	U	6
56	166	G	20	U	5
57	190	G	80	V	
58	213	A	60	W	
59	237	A	20		
60	260	B	80		
61	284	C	40	X	
62	307	D	60	Y & Z	5
63	331	D	20		6
64	355	D	80	N	5 & 6
65	18			N	
66	42			P-Q	
67	65	E	60	R & S	
68	89	E	20	S	
69	112	F	80	T	
70	136	F	40		
71	160	G	60	U	
72	183			V	
73	207			W	
74	230	A	60	W	
75	254	A	20		
76	277	B	80		
77	301	D	40	X	5
78	325	D	60	Y & Z	6
79	348	D	20		5 & 6
80	12			N	
81	35			P-Q	
82	59	E	60	R	
83	82	E	20	S	
84	106	F	80	T	
85	130	F	40		
86	153	G	60	U	
87	177				
88	200			V	
89	224	A	60	W	
90	247	A	20		
91	271	B	80	X	
92	295	C	40	Y & Z	
93	318	D	60		

8.0 CAPSULE EJECTION AND RECOVERY PHASE

On the orbit preceding that on which capsule ejection occurs, the programmed orbital timer tape is reset by DiscOps to assure that on the following orbit the timer initiates the ejection sequence (by restarting the SS/D timer) at the optimum point. The ejection and recovery sequence events listed below are referenced to ejection time (E) or to altitude above sea level

Time From Ejection (E)	Event
E - 94.5 sec (±0.5 sec)	Orbital timer initiates ejection and recovery sequence by turning off horizon scanner and restarting SS/D timer.
E - 81.5 sec (±0.1 sec)	D timer (arming signal): (a) activates Blossom telemetry battery; (b) turns on power to Blossom telemetry filaments and plates; (c) turns on capsule radio beacon; (d) ignites thermal relays to arm ejection programmer; (e) energizes recovery system arming relay; and (f) torques IRP pitch gyro to rotate satellite at +45°/min rate from tail-first horizontal position in order to achieve -130.5° pitch angle for capsule ejection.
E - 25 sec (±0.1 sec)	D timer (transfer signal): (a) changes IRP pitch gyro rate to re-sidual - 3.86°/min; (b) actuates ejection system thermal batteries; (c) backs up recovery system arming signal; and (d) actuates pyrotechnic delay switch to initiate firing of satellite - capsule electrical disconnect squib.
E - 1.5 sec (-0.5/+0.32)	Disconnect squib, activated by D timer transfer signal, fires, disconnecting satellite - capsule electrical cable.
E - 1.5 sec (-0.5/+0.32)	Cable disconnection switches capsule equipment from satellite auxiliary power system to capsule-borne electrical power and lifts ground loop to start ejection programmer.
E - 0	D timer (separation signal) ignites satellite - capsule separation pin puller squibs (0 to 7 millisecond delay).
	Four compressed springs push capsule away from satellite, imparting a velocity of about 1.7 ft/sec.
E + 1.9 sec (±0.17 sec)	Ejection programmer (Event 1) actuates spin jets which spin capsule to about 78 rpm in approximately 0.8 second.
E + 3.15 sec (±0.27 sec cum. tol.)	Ejection programmer (Event 2) ignites retro rocket which imparts approximately 4 g acceleration for about 9 seconds (total impulse = 10,470 lb-sec).
E + 13.9 sec (±0.81 sec cum. tol.)	Ejection programmer (Event 3) actuates de-spin jets which reduce capsule spin rate to about 10 rpm.
E + 15.4 sec (±0.96 sec cum. tol.)	Ejection programmer (Event 4): (a) ignites thrust cone - capsule electrical disconnect squib; and (b) ignites thrust cone - capsule explosive separation bolts.
	Capsule separates from thrust cone.
E + 35.5 sec	D timer: (a) turns on horizon scanner; (b) switches horizon scanner to low gain; and (c) turns itself off.

Feet Above Sea Level (Approx.)	Event
350,000 ft	Re-entry friction heats recoverable capsule to point at which ionized layer forms on capsule surface, causing r.f. blackout.
225,000 ft	G switch closes (at 3 to 6.6 g) as re-entry deceleration increases.
135,000 ft	Re-entry deceleration reduces friction and permits recoverable capsule surface cooling to point at which ionized layer disappears, ending r.f. blackout.

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Feet Above Sea Level (Approx.)	Event
100,000 ft	G switch opens (at 3 g ±5%) as re-entry deceleration decreases, permitting recovery programmer timing circuit to begin operation.
55,000 ft (g switch opening + 47.0±1.7 sec)	Timer switch closes, and recovery programmer: (a) ignites ejection pistons; and (b) switches power to recovery system flashing light.
	Ejection pistons: (a) release ablative shell from capsule; and (b) blow parachute cover off.
	Ejected parachute cover pulls out pilot chute, which in turn pulls out the main parachute bag with the chute in reefed condition.
	Mechanically-actuated time-delay pyrotechnic cutter (4-second delay) disreefs main parachute, permitting chute deployment.
	Radar reflective chaff, packed with the parachute, falls free as the chute emerges from its bag.
	Parachute system decelerates recoverable capsule, and ablative shell falls clear of capsule.

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STRUCTURE/EQUIPMENT ITEM	Satellite No. 1025		
A SPACEFRAME	598		
Nose Section			43
Forward Body Section, including Forward Equipment Rack			92
Aft Body Section, Aft Equipment Rack, Thrust Cone			117
Adapter Section including Separation Rails			314
Midbody Fairing			8
Miscellaneous Attachments and Paint			8
Retrograde Rocket Installation		16	
Retro Rockets (2)			9
Attachments (2)			7
FLIGHT TERMINATION EQUIPMENT	7		
Destruct Charge			3
Initiator (safe & arm device)			2
Wiring			2
B PROPULSION EQUIPMENT	551		
Engine		267	
Engine (dry weight)			262
Starting Charge, Grease, Oil			2
Nozzle Closure			3
Propellant and Pressurization Equipment		284	
Tank Installation			165
Pressurization Equipment Installation			16
Pressure Spheres, Helium (2)			40
Attachments			6
Helium Gas			5
Propellant Load Equipment			12
Ullage Control Rocket Installation		40	
Ullage Control Rockets (2)			36
Attachments (2)			4
C GUIDANCE AND CONTROL EQUIPMENT	243		
Guidance Equipment		96	
Inertial Reference Package			38
Horizon Scanner and Fairing			14
Computer, including Timer and Integrator			32
Secondary Junction Box			4
Ground Cooling Installation			7
Sun Position Indicator			1
Flight Control Equipment		31	
Electronics and Rate Gyro Package			31
Pneumatic Control Equipment		79	
Thrust Valve Assembly (6)			8
Pressure Sphere, Control Gas			25
Attachments			3
Control Gas			37
Integrated Control Package			4
Plumbing and Fittings			2
Hydraulic Control Equipment		37	
Actuating Cylinders (2)			7
Electro Hydraulic Power Package			22
Hydraulic Fluid			2
Lines and Fittings			4
Start Relay, Hydraulic Motor			2
D AUXILIARY POWER EQUIPMENT	253		
Battery, Type IA Primary			108
Battery, Type IV Primary			16
Inverter, 115 v ac, 1Ø, 2000 cps			13
Load Limiter, 115 v ac, 1Ø, 2000 cps			2
Inverter, 115 v ac, 3Ø, 400 cps			16
Load Limiter, 115 v ac, 3Ø, 400 cps			6
Voltage Regulators, +28 v dc (2)			6
Regulated Power Supply, -28 v dc			3
Voltage Regulator, 115 v ac, 1Ø, 400 cps			5
Umbilical Connectors			3
Thermal Shielding, Wiring, connectors, and terminal strips			75
E COMMUNICATIONS EQUIPMENT	109		
Acquisition Transmitter, Diplexer, and Antenna Switch			4
Secondary Timer			9
Transponder-Decoder Installation		27	
Transponder			14
Decoder			6
Power Converter			6
Transponder Antenna and Leads			1
Telemetry Installation		54	
FM/FM Telemetry Unit Installation			24
Oscillator, Voltage-Controlled			8
Oscillator, Dual-Loop AC Bridge			5
VHF Antennas (2)			3
Leads, Connectors, Filters, etc.			3
Transducer Installation			11
Tape Recorder Installation		15	
AMR-100 Tape Recorder			11
Tape Recorder Modulator & Detector			2
Tape Recorder Programmer			2
L MARK II BIOMEDICAL RECOVERY CAPSULE	326		
GEOPHYSICS RESEARCH DIRECTORATE (GRD) EQUIPMENT	23		
Density Detector			9
Radiometer (3)			10
Cosmic Ray Detector			3
Power Programmer			1
MANUFACTURING TOLERANCE	12		
WEIGHT EMPTY	2,098		

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MATERIAL		Satellite No. 1025	
WEIGHT EMPTY	2,098		
PROPELLANTS	6,644		
Fuel, Impulse			1,799
Oxidizer, Impulse			6,713
Oxidizer Preflow			5
Trapped in Lines, Tank, Engine			113
Residuals for Mixture Ratio			14
GROSS WEIGHT, VEHICLE	8,742		
Less Expended and Jettisoned Material		-339	
Adapter and Attachments			-316
Retro Rockets			-16
Destruct System			-7
SEPARATION WEIGHT	8,403		
Less Expended and Jettisoned Material		-43	
Ullage Control Rockets			-38
Horizon Scanner Fairing			-2
Control Gas Expended during Coast Phase			-3
IGNITION WEIGHT, ORBITAL BOOST	8,360		
Less Expended and Jettisoned Material		-6524	
Propellants, Impulse			-6512
Oxidizer Preflow			-5
Engine Starting Charge			-1
Engine Nozzle Closure			-3
Control Gas Expended during Orbital Boost Phase			-3
BURNOUT WEIGHT	1,836		
Less Material Vented and Expended during Reorientation Phase		-132	
Propellants, Residual			-127
Helium Gas			-5
WEIGHT EMPTY ON ORBIT	1,704		
Remaining Control Gas			-31
WEIGHT EMPTY ON ORBIT - END OF USEFUL ORBITAL LIFE-(CONTROL GAS EXPENDED)	1,673		

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TITLE
OPERATIONAL WEIGHT FACTORS
Satellite 2205 - 1025
Flight Configuration III
DISCOVERER PROGRAM

PREPARED BY
J. Michael

APPROVED BY

#25

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LOCKHEED AIRCRAFT CORPORATION

6-16-61

STRUCTURE/EQUIPMENT ITEM	Satellite No. 1107		
SPACEFRAME	479		
Forward Body Section, including Forward Equipment Rack			80
Aft Body Section and Aft Equipment Rack			67
Engine Thrust Cone			32
Adapter Section, including Separation Rails			256
Midbody Fairing			23
Miscellaneous Attachments and Paint			8
Retrograde Rocket (2)			10
Pneumatic Bottle Heat Shields			3
FLIGHT TERMINATION EQUIPMENT	11		
Destruct Charge			7
Initiator (safe & arm device)			4
PROPULSION EQUIPMENT	694		
Engine		315	
Engine (dry weight)			296
Starting Charge, Grease, Oil			2
Nozzle Thermal Shield			16
Turbine Exhaust Heat Shield			1
Propellant and Pressurization Equipment		379	
Tank Installation			280
Pressurization Equipment Installation			17
Pressure Sphere, Helium			28
Helium Gas			3
Propellant Load Equipment			12
Ullage Control Rockets (2), and Attachments (2)			39
GUIDANCE AND CONTROL EQUIPMENT	339		
Guidance Equipment		87	
Inertial Reference Package			36
Horizon Scanner and Fairing			13
Computer, including Timer and Integrator			32
Secondary Junction Box			4
Ground Cooling Installation			1
Sun Position Indicator			1
Flight Control Equipment		31	
Electronics and Rate Gyro Package			31
Pneumatic Control Equipment		200	
Thrust Valve Assembly (6)			9
Pressure Spheres, Control Gas (2)			48
Attachments			5
Control Gas			132
Integrated Control Package			4
Plumbing and Fittings			2
Hydraulic Control Equipment		21	
Actuating Cylinders (2)			6
Turbo-Hydraulic Power Package			8
Hydraulic Fluid			1
Lines and Fittings			6

STRUCTURE/EQUIPMENT ITEM	Satellite No. 1107		
AUXILIARY POWER EQUIPMENT	575		
Batteries, Type IA Primary (4)			430
Inverter, 115 v ac, 1Ø, 2000 cps			13
Load Limiter, 115 v ac, 1Ø, 2000 cps			2
Inverter, 115 v ac, 3Ø, 400 cps			16
Load Limiter, 115 v ac, 3Ø, 400 cps			6
Power Amplifier, 115 v ac, 1Ø, 400 cps			8
Voltage Regulators, +28 v dc, Type III			6
Regulated Power Supply, -28 v dc			3
Voltage Regulator, 115 v ac, 1Ø, 400 cps			5
Filament Transformer			1
Umbilical Connectors			3
Thermal Shielding, Wiring, Connectors, & Terminal Strips			82
COMMUNICATIONS	112		
Acquisition Transmitter, & Diplexer		3	
Transponder-Decoder Installation		20	
Transistorized Transponder			12
Transistorized Decoder			7
Antenna			1
Telemetry Installation		74	
Unitized FM/FM Telemeter			44
Transducers & Brackets			22
Exit Antenna			2
Orbit Antenna			1
Leads, Connectors, Filter, Etc.			5
Orbital Timer		11	
Doppler Tracking Provisions		4	
SPECIAL INSTRUMENTATION	112		
AET-L PAYLOAD & FAIRING	492		
MANUFACTURING TOLERANCE	+13		
WEIGHT EMPTY	2827		

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	Satellite No. 1107		
WEIGHT EMPTY	2,827		
PROPELLANTS	13,213		
Fuel, Impulse			3,722
Oxidizer, Impulse			9,361
Oxidizer Preflow Expended			5
Trapped in Lines, Tank, Engine			106
Residuals for Mixture Ratio			19
GROSS WEIGHT, VEHICLE	16,040		
Less Expended and Jettisoned Material		279	
Adapter and Attachments			258
Retro Rockets			10
Destruct System			11
SEPARATION WEIGHT	15,761		
Less Expended and Jettisoned Material		46	
Ullage Control Rockets			38
Horizon Scanner Fairing			1
Control Gas Expended during Coast Phase			7
IGNITION WEIGHT, ORBITAL BOOST	15,715		
Less Expended and Jettisoned Material		13,108	
Propellants, Impulse			13,083
Oxidizer Preflow			5
Engine Starting Charge			1
Control Gas Expended during Boost Phase			19
BURNOUT WEIGHT	2,607		
Less Material Vented and Expended during Reorientation Phase		128	
Propellants, Residual			125
Helium Gas			3
WEIGHT EMPTY ON ORBIT	2,479		
Remaining Control Gas			106
WEIGHT EMPTY ON ORBIT - END OF USEFUL ORBITAL LIFE (CONTROL GAS EXPENDED)	2,373		

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Satellite No. 1108

STRUCTURE/EQUIPMENT ITEM	Total	Sub-Total	Item Wt.
SPACEFRAME	482		
Forward Body Section, including Forward Equipment Rack			80
Aft Body Section and Aft Equipment Rack			67
Engine Thrust Cone			32
Adapter Section, including Separation Rails			259
Midbody Fairing			23
Miscellaneous Attachments and Paint			8
Retrograde Rocket (2)			10
Pneumatic Bottle Heat Shields			3
FLIGHT TERMINATION EQUIPMENT	11		
Destruct Charge			7
Initiator (safe & arm device)			4
PROPULSION EQUIPMENT	694		
Engine		315	
Engine (dry weight)			296
Starting Charge, Grease, Oil			2
Nozzle Thermal Shield			16
Turbine Exhaust Heat Shield			1
Propellant and Pressurization Equipment		379	
Tank Installation			280
Pressurization Equipment Installation			17
Pressure Sphere, Helium			28
Helium Gas			3
Propellant Load Equipment			12
Ullage Control Rockets (2), and Attachments (2)			39
GUIDANCE AND CONTROL EQUIPMENT	339		
Guidance Equipment		87	
Inertial Reference Package			36
Horizon Scanner and Fairing			13
Computer, including Timer and Integrator			32
Secondary Junction Box			4
Ground Cooling Installation			1
Sun Position Indicator			1
Flight Control Equipment		31	
Electronics and Rate Gyro Package			31
Pneumatic Control Equipment		200	
Thrust Valve Assembly (6)			9
Pressure Spheres, Control Gas (2)			48
Attachments			5
Control Gas			132
Integrated Control Package			4
Plumbing and Fittings			2
Hydraulic Control Equipment		21	
Actuating Cylinders (2)			6
Turbo-Hydraulic Power Package			8
Hydraulic Fluid			1
Lines and Fittings			6

Satellite No. 1108

STRUCTURE/EQUIPMENT ITEM	Total	Sub-Total	Item Wt.
AUXILIARY POWER EQUIPMENT	518		
Batteries, Type IA Primary (3)			323
Batteries, Type VI Primary (2)			54
Inverter, 115 v ac, 1Ø, 2000 cps			13
Load Limiter, 115 v ac, 1Ø, 2000 cps			2
Inverter, 115 v ac, 3Ø, 400 cps			16
Load Limiter, 115 v ac, 3Ø, 400 cps			6
Power Amplifier, 115 v ac, 1Ø, 400 cps			8
Voltage Regulators, +28 v dc, Type III			6
Regulated Power Supply, -28 v dc			3
Voltage Regulator, 115 v ac, 1Ø, 400 cps			5
Filament Transformer			1
Umbilical Connectors			3
Thermal Shielding, Wiring, Connectors, & Terminal Strips			78
COMMUNICATIONS	132		
Acquisition Transmitter, Diplexer, & Antenna Switch		4	
Transponder-Decoder Installation		20	
Transistorized Transponder			12
Transistorized Decoder			7
Antenna			1
Telemetry Installation		81	
Unitized FM/FM Telemeter			44
Transducers & Bracketry			29
Exit Antenna			2
Orbit Antenna			1
Leads, Connectors, Filter, Etc.			5
Orbital Timer		11	
Doppler Tracking Provisions		10	
Optical Tracking Provisions		1	
Special Instrumentation Provision		5	
AKT-H PAYLOAD & FAIRING	522		
MANUFACTURING TOLERANCE	-1		
WEIGHT EMPTY	2697		

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EQUIPMENTS WEIGHT LIST
Satellite 6205-1108
Flight Configuration VI
DISCOVERER PROGRAM

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Satellite No. 1108

	Total	Sub-Total	Item Wt.
WEIGHT EMPTY	2,697		
PROPELLANTS	13,275		
Fuel, Impulse			3,704
Oxidizer, Impulse			9,441
Oxidizer Preflow Expended			5
Trapped and Residuals for Mixture Ratio			125
GROSS WEIGHT, VEHICLE	15,972		
Less Expended and Jettisoned Material		282	
Adapter and Attachments			261
Retro Rockets			10
Destruct System			11
SEPARATION WEIGHT	15,690		
Less Expended and Jettisoned Material		46	
Ullage Control Rockets			38
Horizon Scanner Fairing			1
Control Gas Expended during Coast Phase			7
IGNITION WEIGHT, ORBITAL BOOST	15,644		
Less Expended and Jettisoned Material		13,170	
Propellants, Impulse			13,145
Oxidizer Preflow			5
Engine Starting Charge			1
Control Gas Expended during Boost Phase			19
BURNOUT WEIGHT	2,474		
Less Material Vented and Expended during Reorientation Phase		128	
Propellants, Residual			125
Helium Gas			3
WEIGHT EMPTY ON ORBIT	2,346		
Remaining Control Gas			106
WEIGHT EMPTY ON ORBIT - END OF USEFUL ORBITAL LIFE (CONTROL GAS EXPENDED)	2,240		

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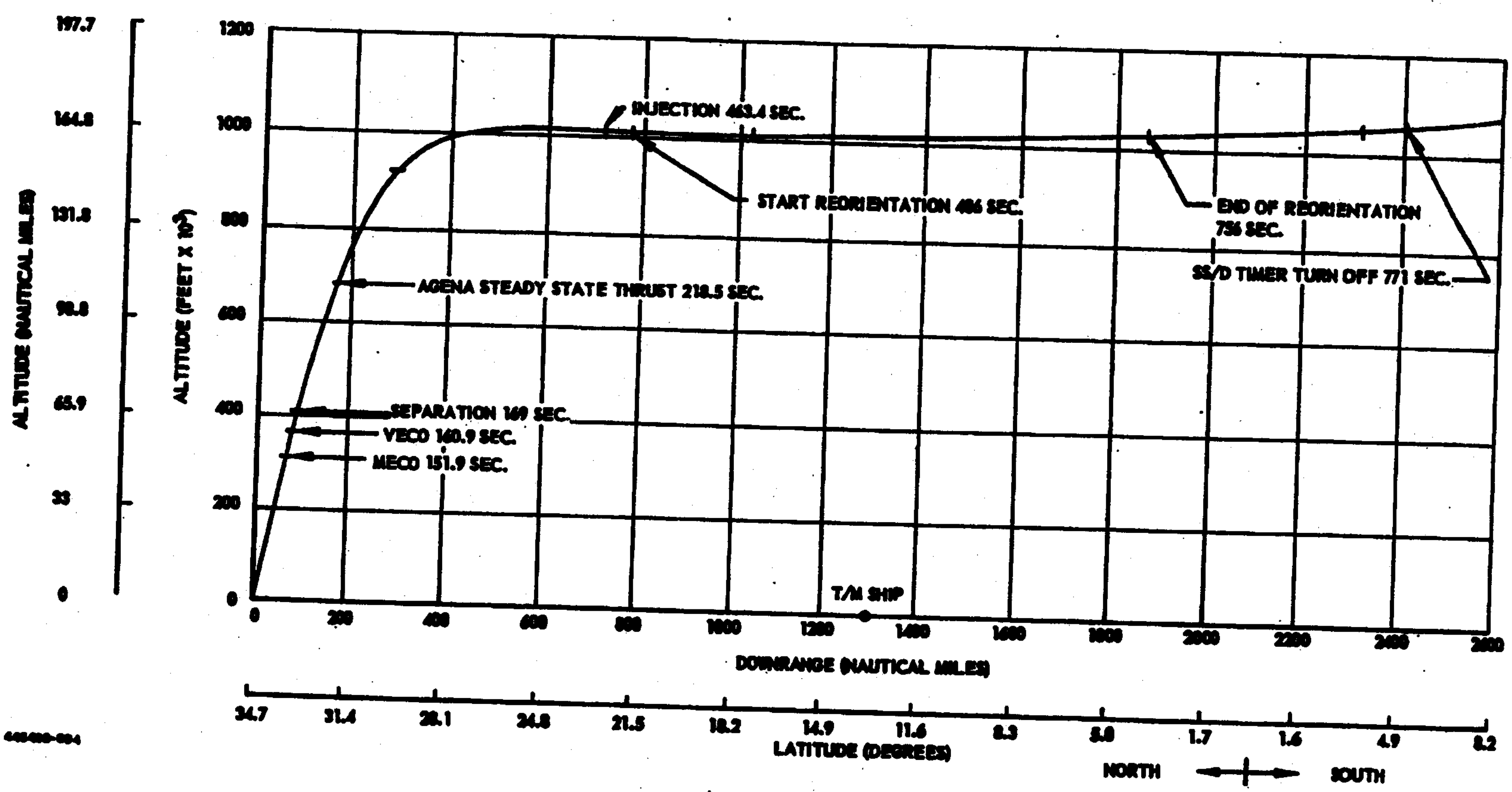

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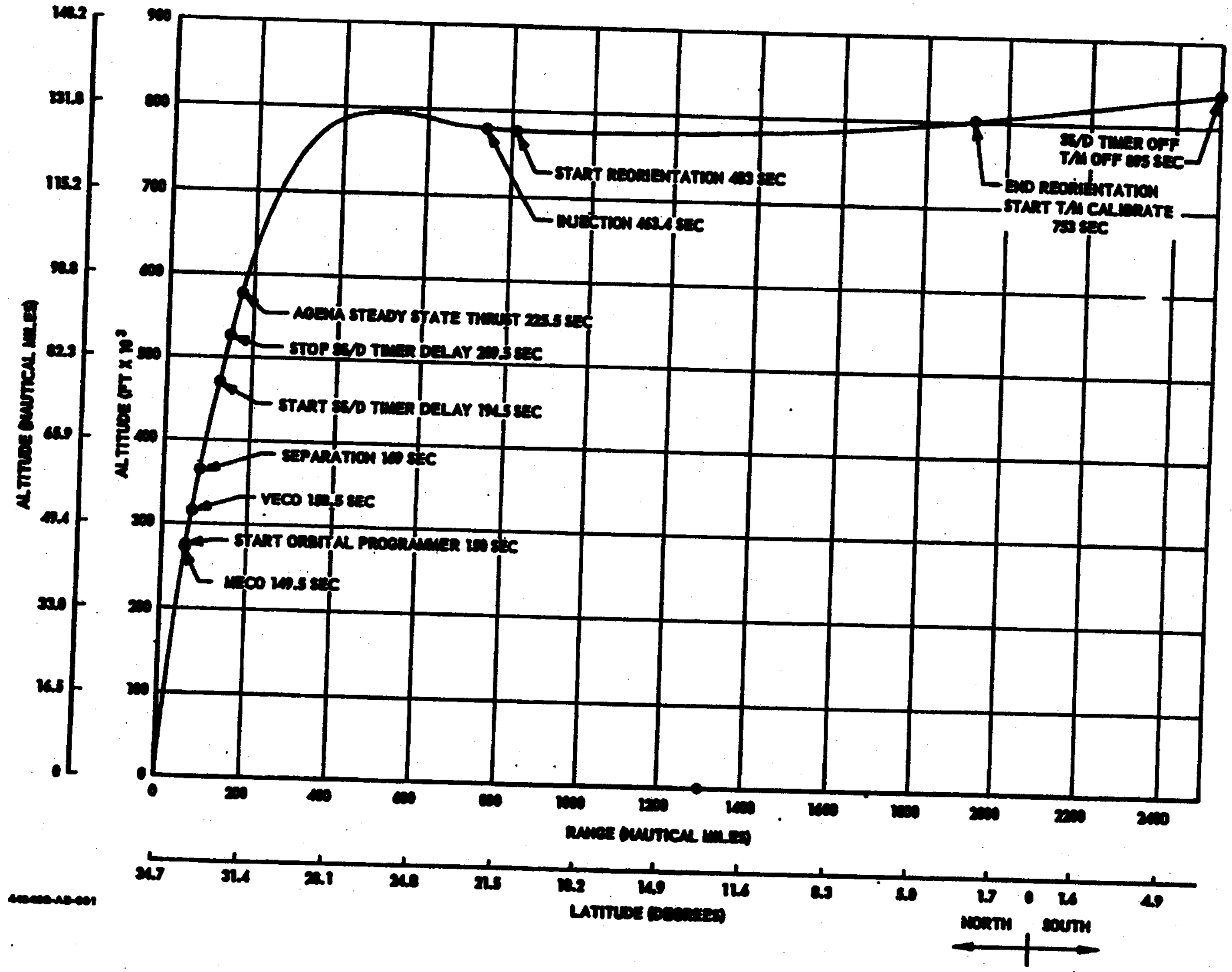

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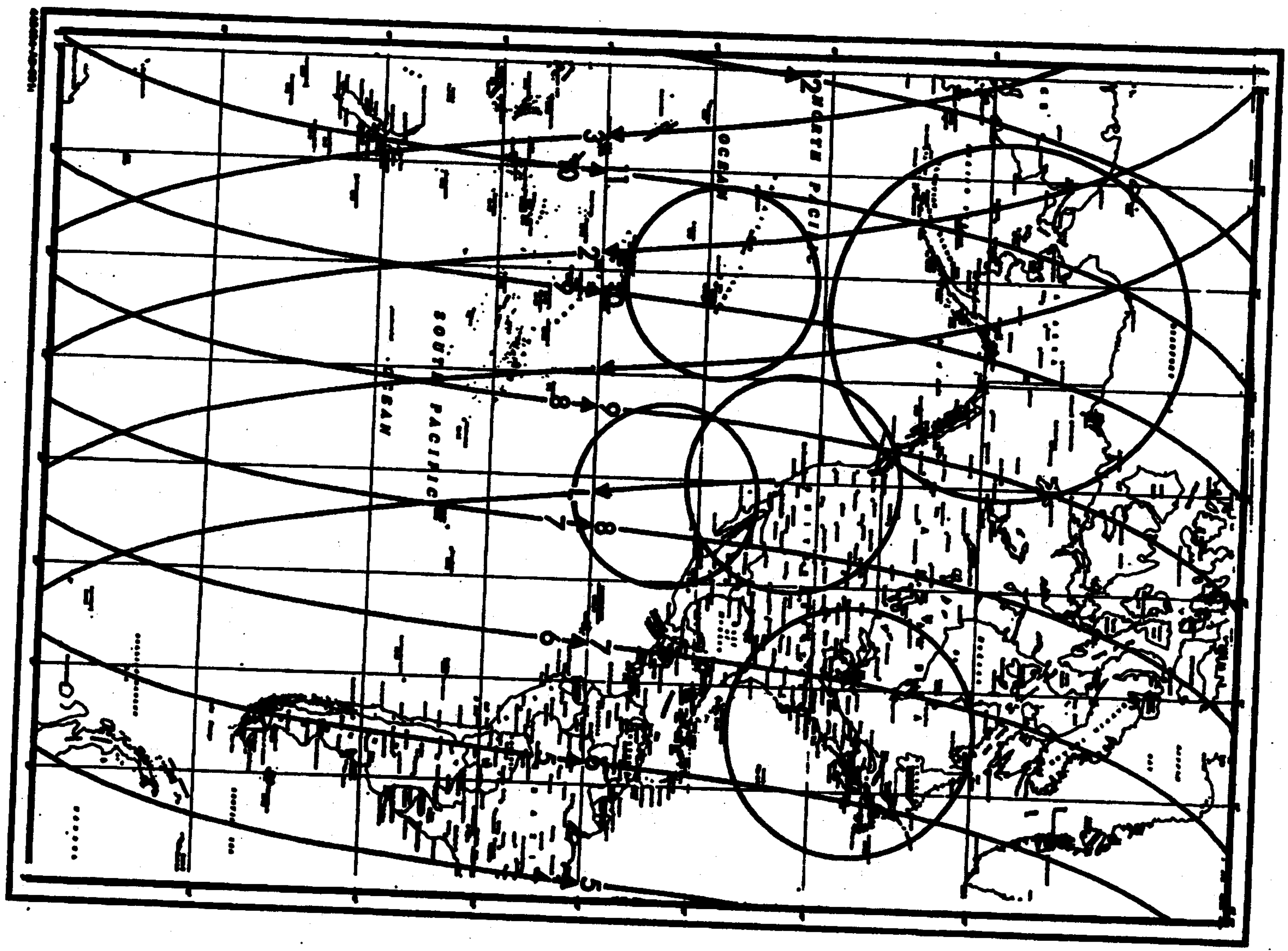

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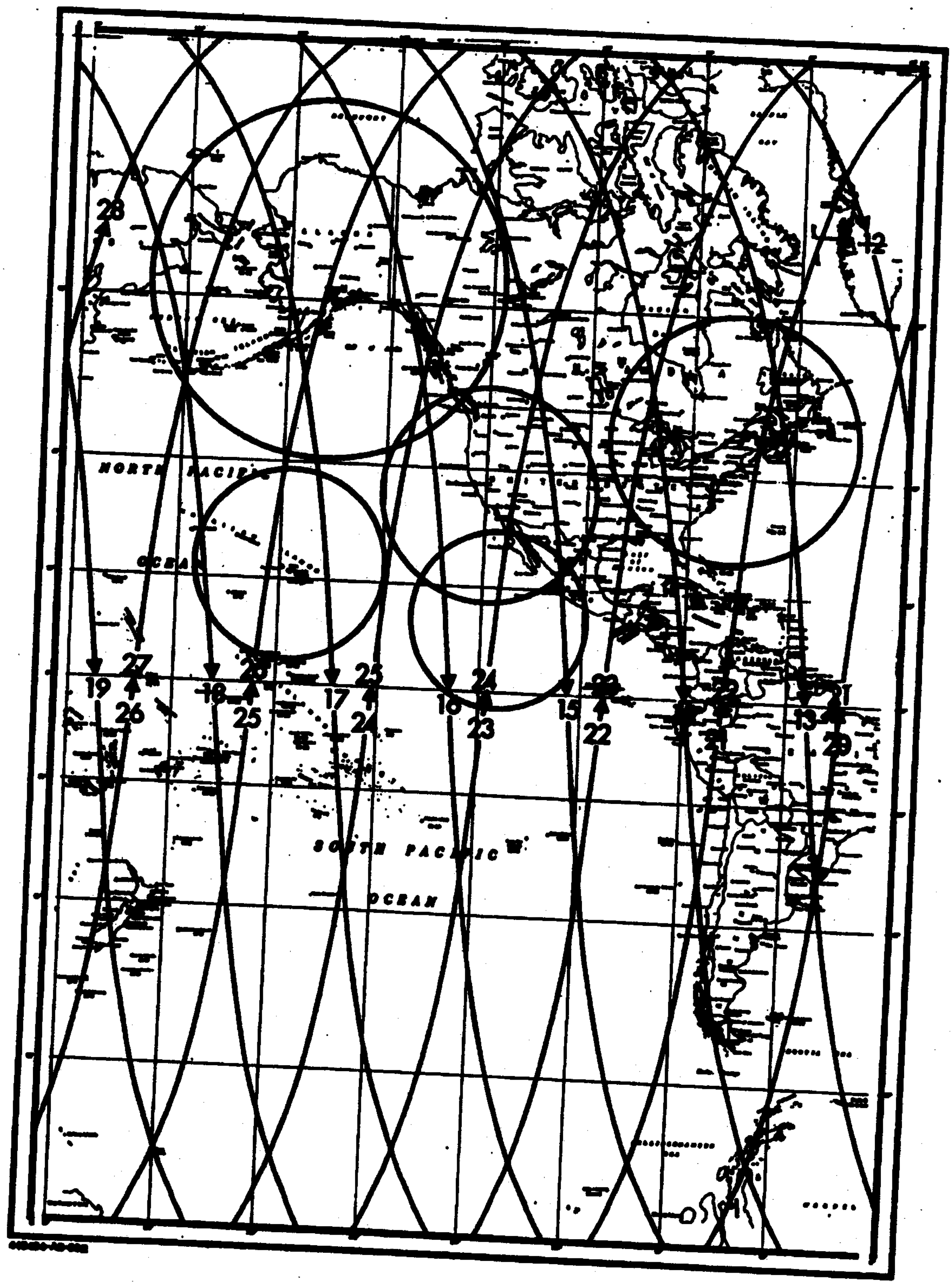

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ORBIT PATTERN
Passes 13 thru 27,
Flight Configuration VI (1107)
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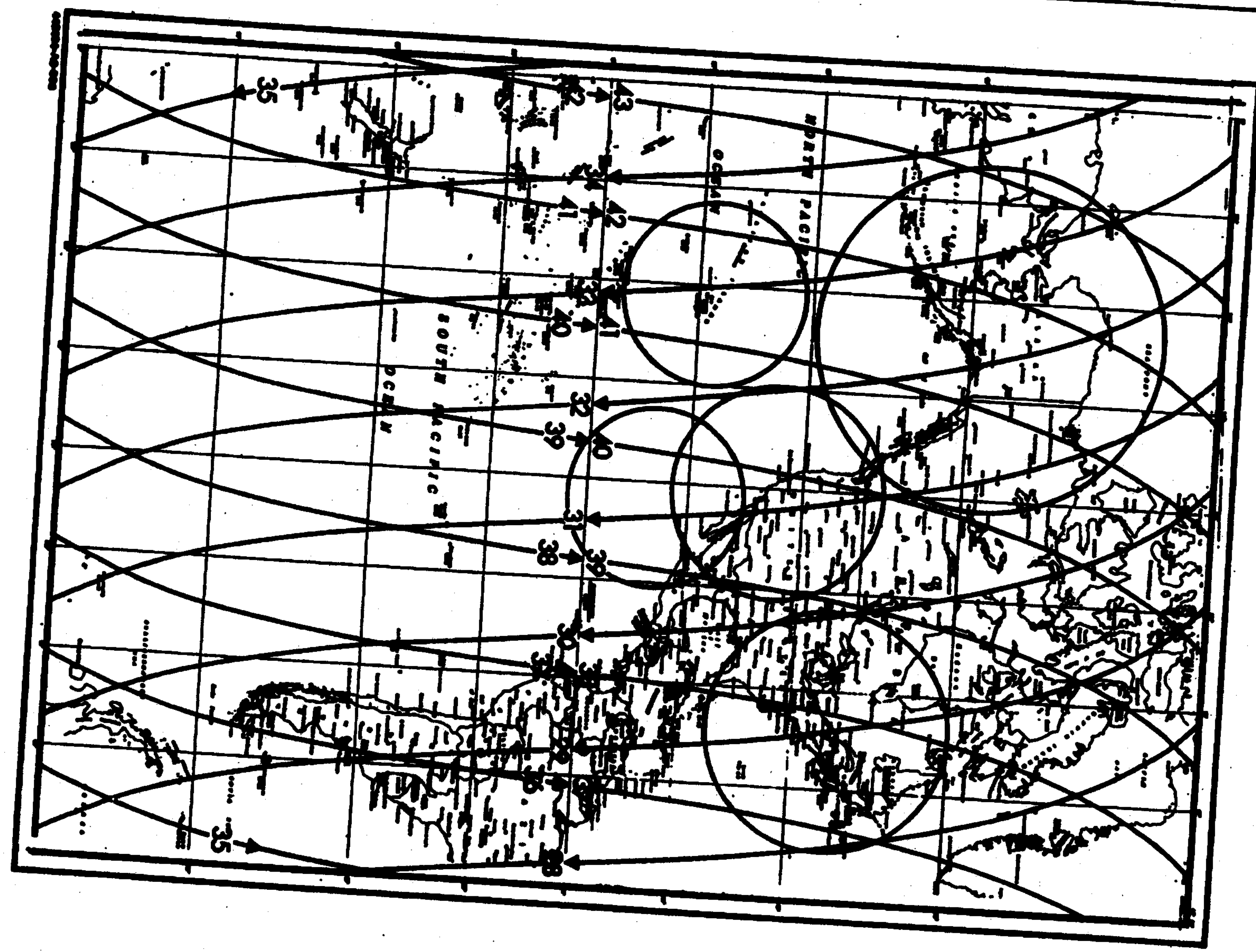

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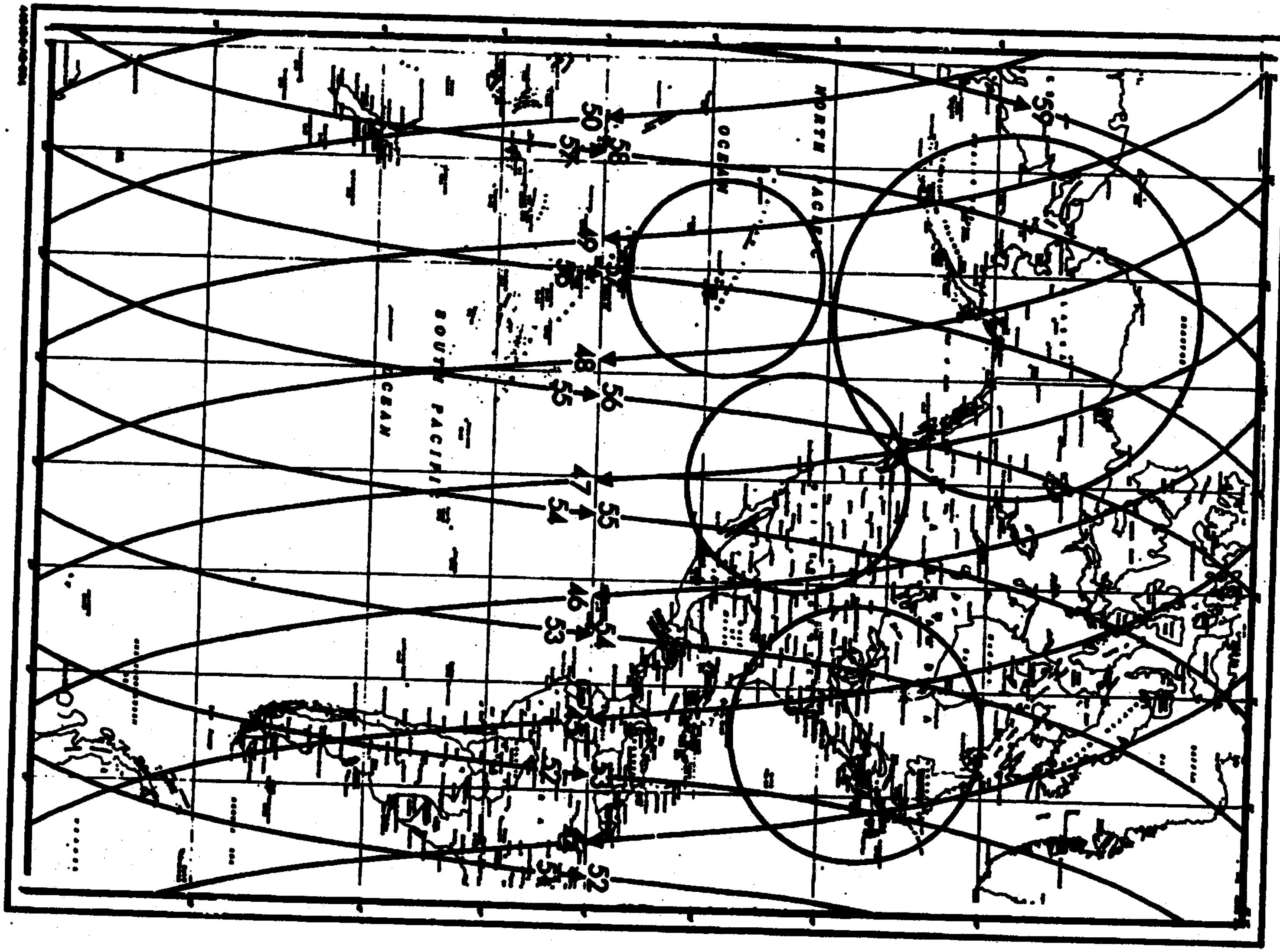

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	ORBIT PATTERN Passes 43 thru 59 Flight Configuration VI (1107) DISCOVERER PROGRAM	
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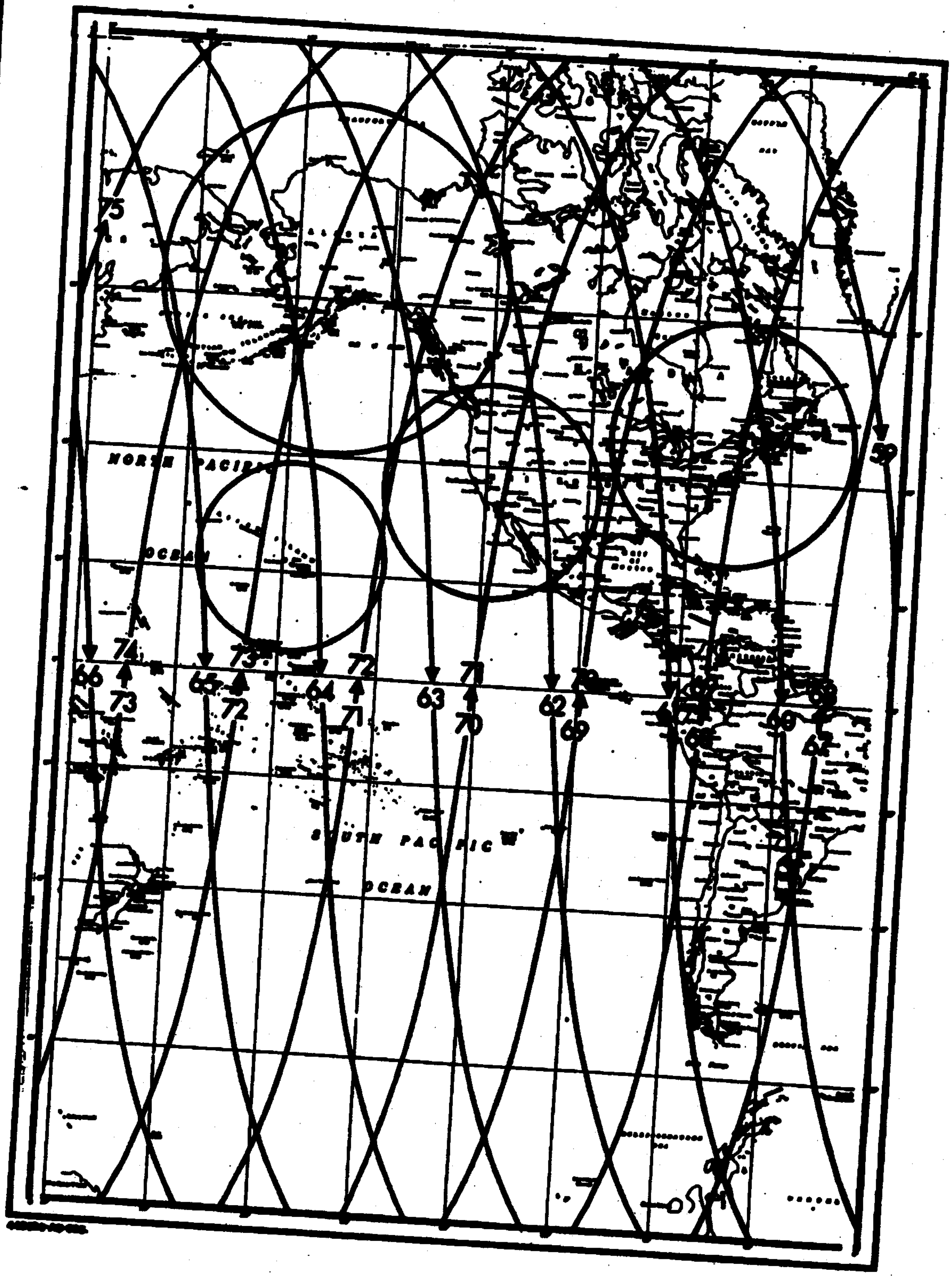

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ORBIT PATTERN
Passes 60 thru 74
Flight Configuration VI (1107)
DISCOVERER PROGRAM

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15 May 1961

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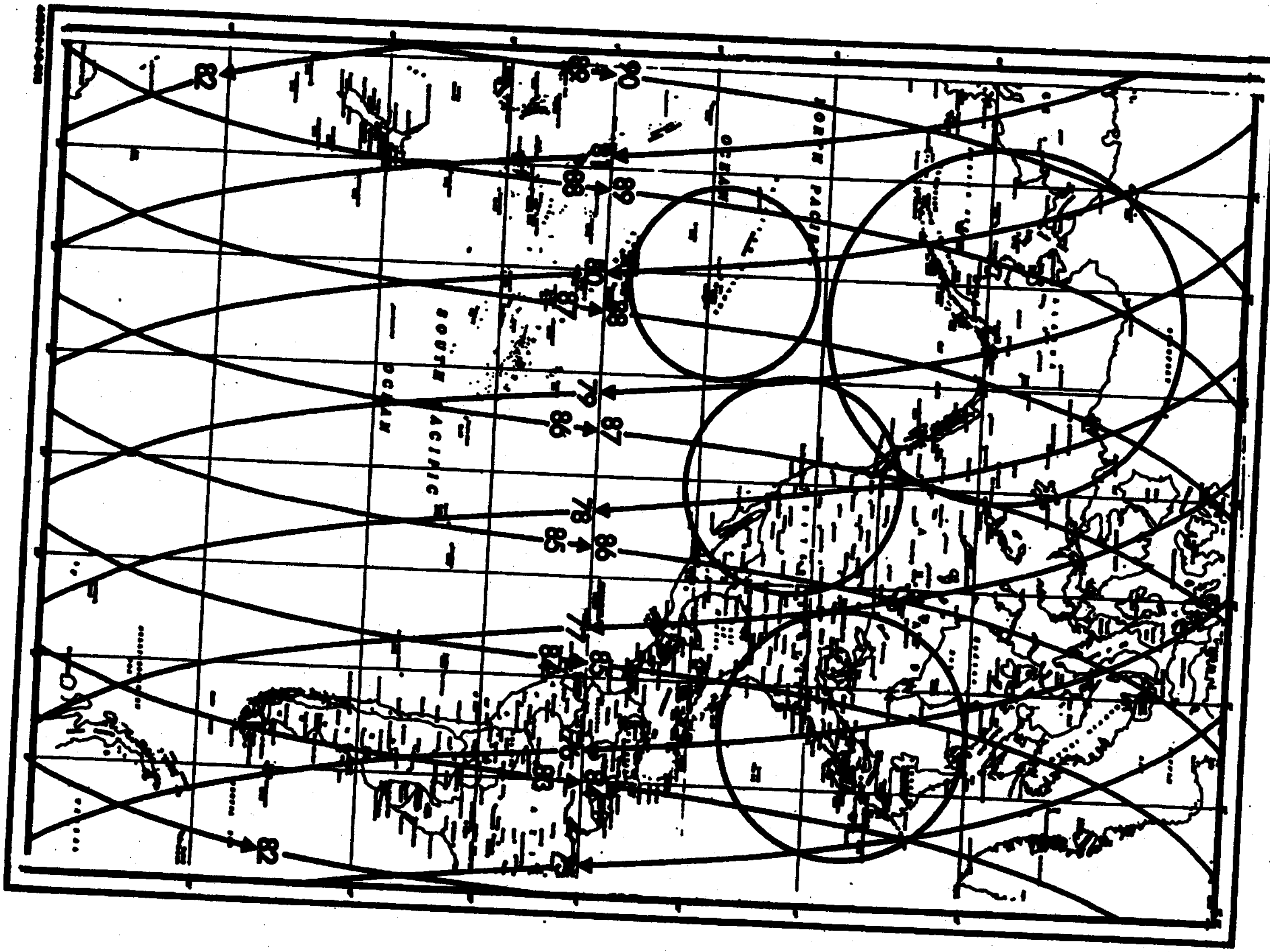

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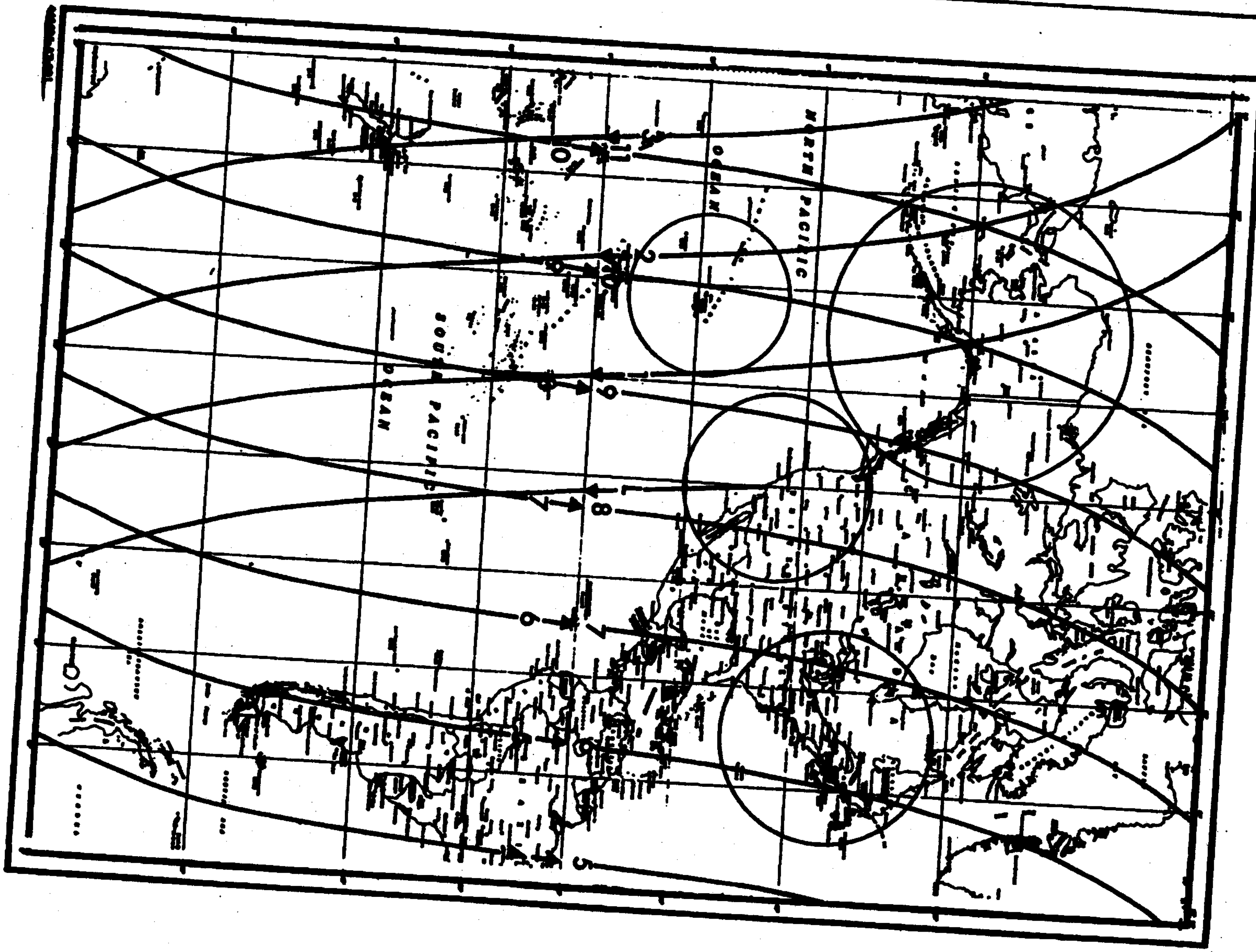

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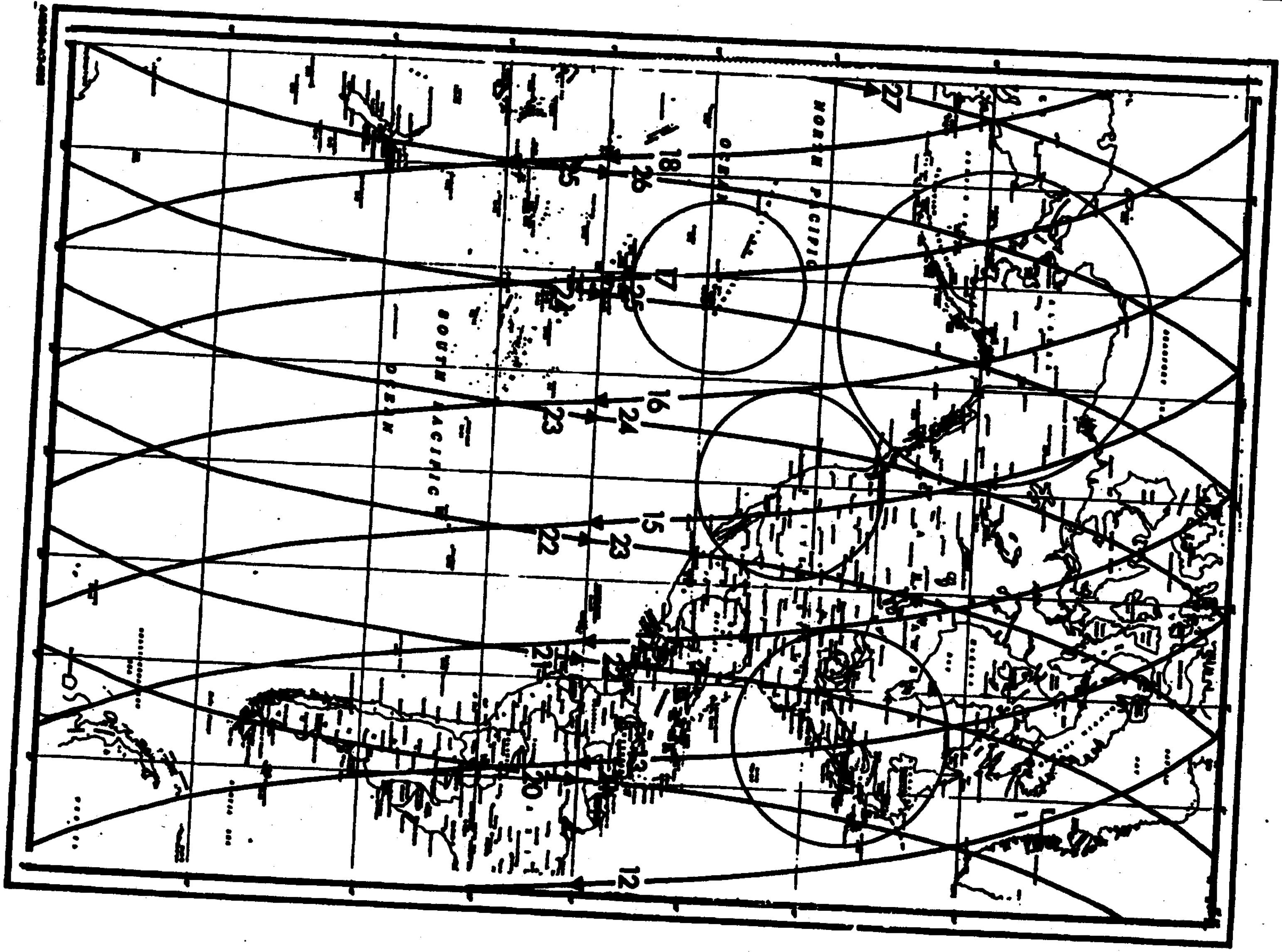

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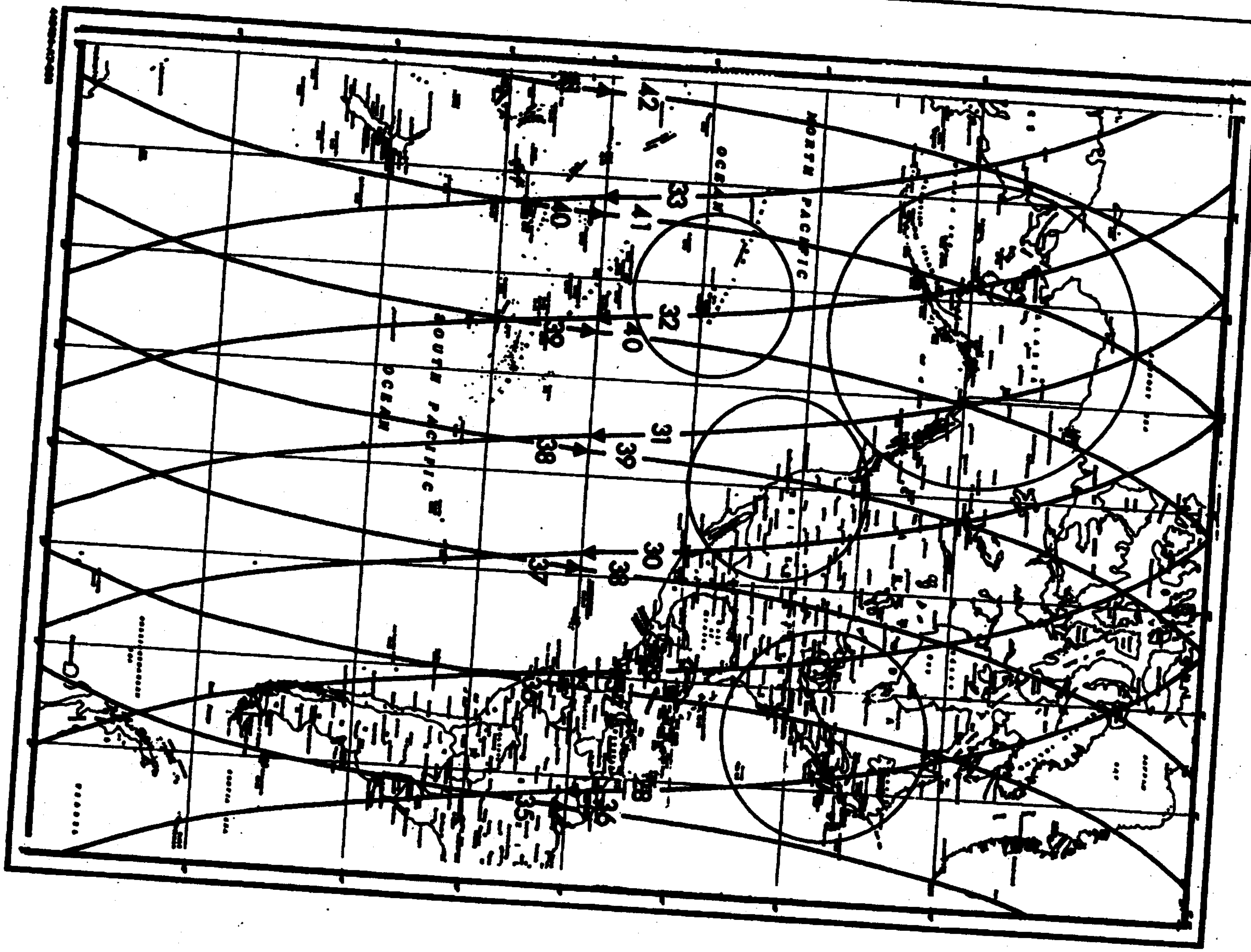

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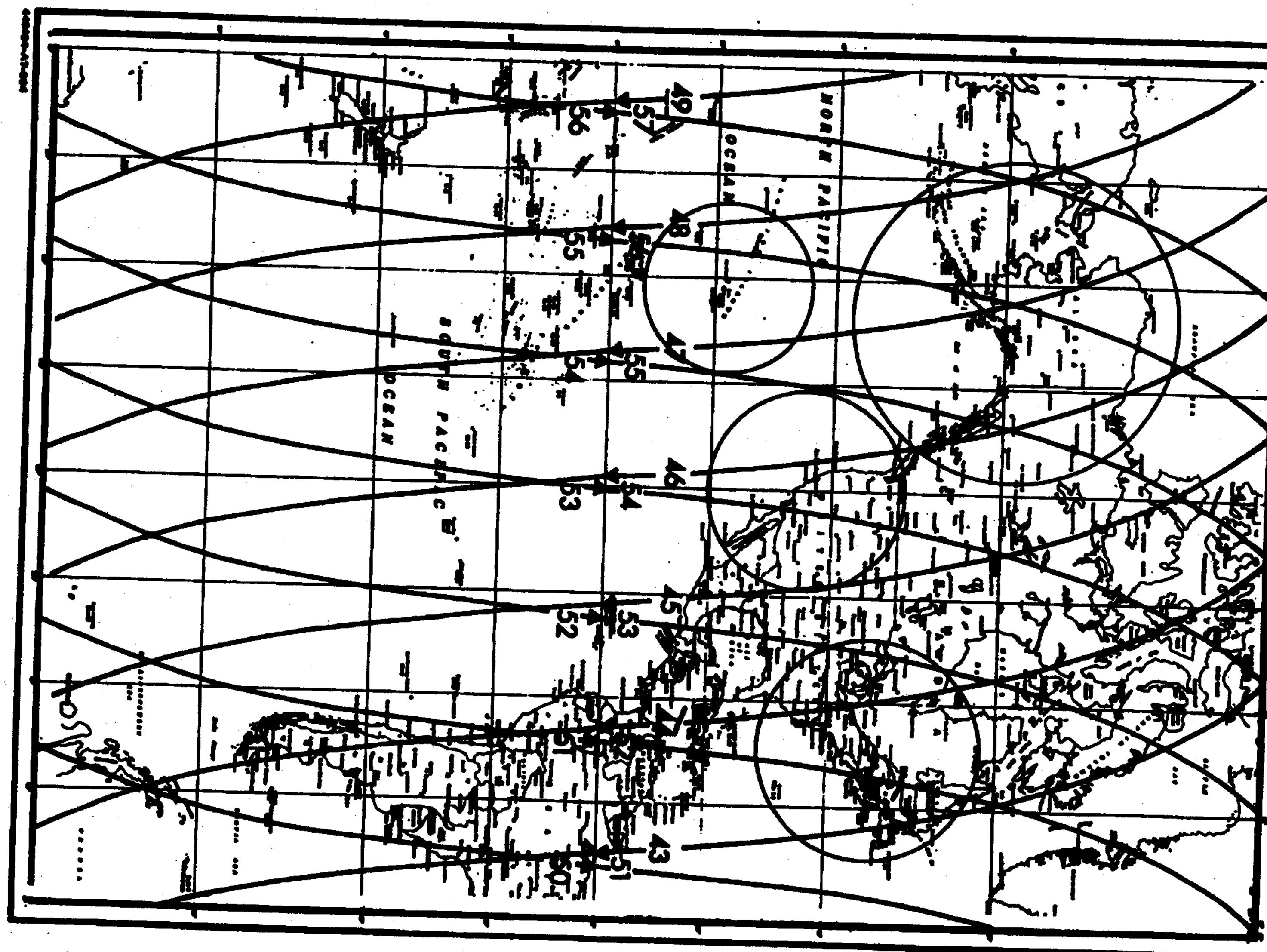

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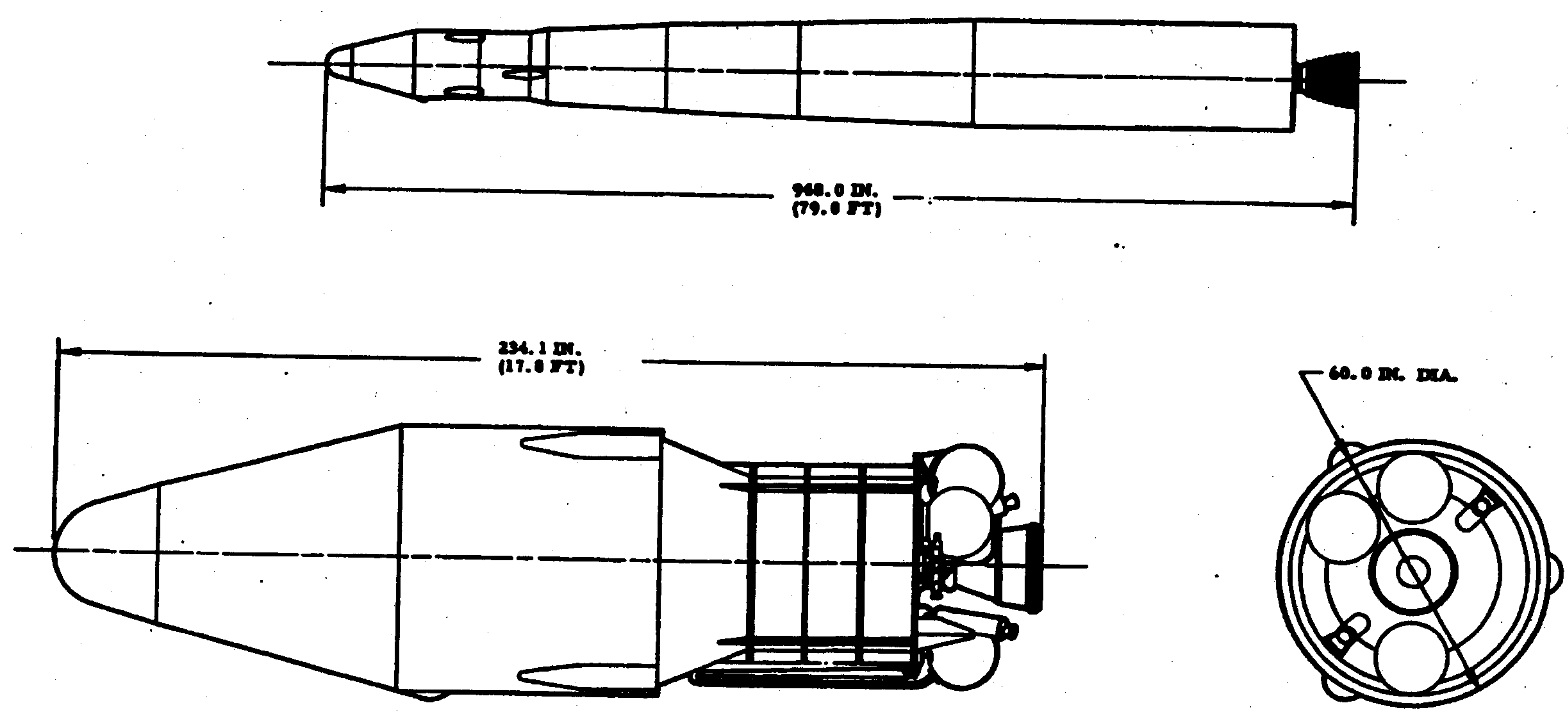

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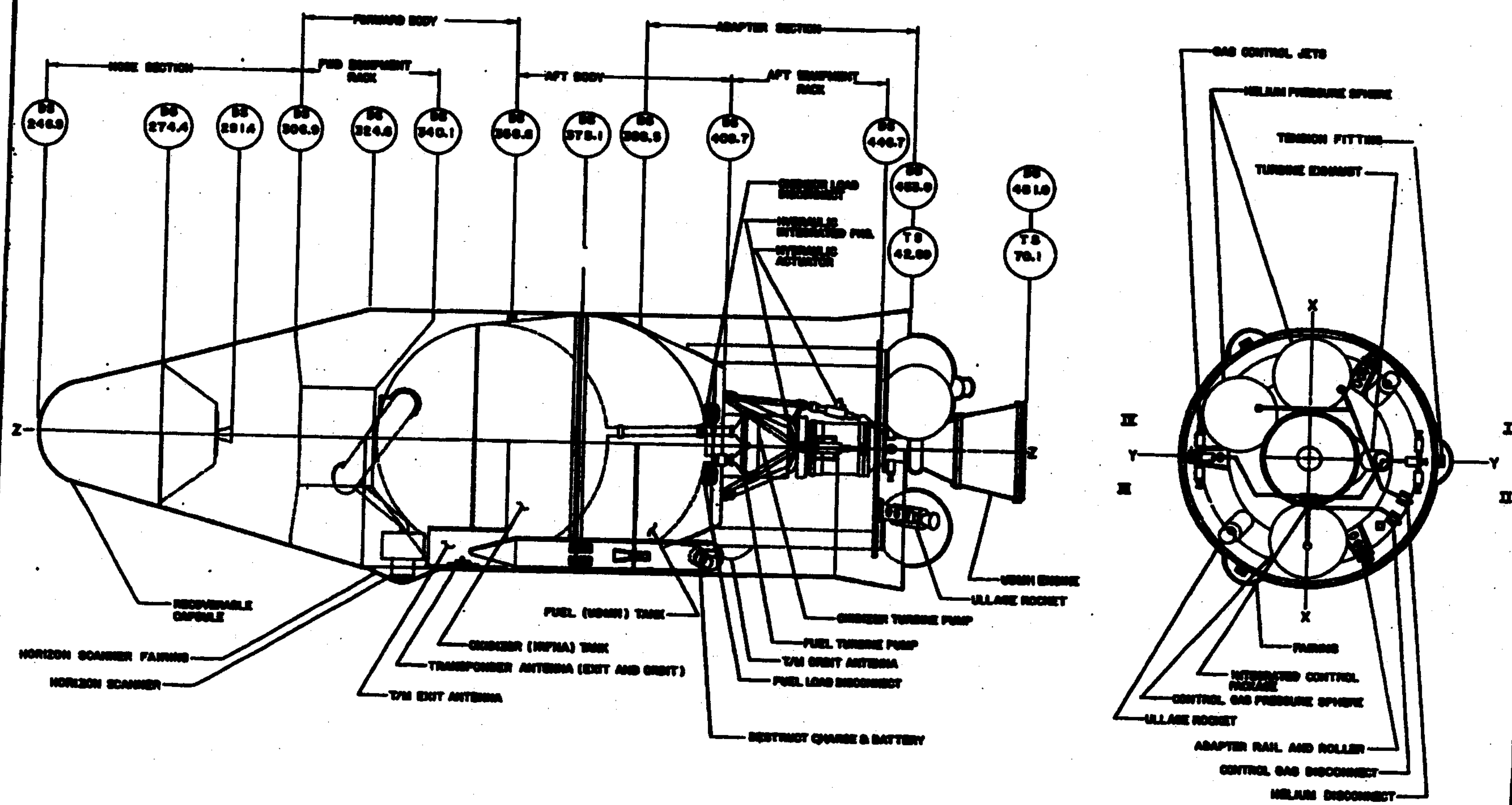

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1.0 GENERAL

The satellite spaceframe for Discoverer Flight Configuration (FC) III consists of the following sections: nose, including the payload and recoverable capsule; forward body, including the forward equipment rack; aft body; aft equipment rack; and the adapter. The spaceframe is a semi-monocoque structure consisting essentially of a magnesium alloy skin and supporting web. The satellite has a cylindrical body which tapers to a blunt nose at the forward end and flares slightly at the rear to accommodate its booster. Functionally, it serves as a carrier for the equipment it houses, supports, and/or jettisons, as applicable, during its ascent to orbit and during its orbital life. During the Coast Phase, the spaceframe separates from its adapter, which has been permanently attached to the Thor booster during preparation for launch. At the end of the Orbit Phase, the payload capsule is ejected for recovery. FC III carries a Mark II Biomedical Recovery Capsule (BRC).

[redacted].0 SIGNAL INPUT REQUIREMENTS

At the times noted in the Timed Sequence of Events (Section IV, pages 2.0 ff, as applicable), the following signal inputs are required for Discoverer spaceframe structure and/or equipment functioning:

[redacted].1 A signal from the SS/D timer fires the explosive bolts/pin pullers (refer Par. 3.1.5) installed on the periphery of the aft-body at Discoverer Station (DS) 388.55 to release the satellite from the adapter.

[redacted]2 A signal from the SS/D timer ignites the retrograde rockets to accomplish complete separation of the satellite from the adapter/booster combination.

2.3 A signal from the SS/D timer fires the explosive bolt that releases the horizon scanner fairing.

2.4 A signal from the SS/D timer arms the payload capsule ejection pin pullers.

2.5 A signal from the SS/D timer ignites the pin puller squibs to initiate release of the payload capsule from the satellite.

3.0 PERFORMANCE

3.1 Equipment Description

3.1.1 Nose Section. The nose section, extending from DS 246.95 to DS 306.95, includes the recoverable payload capsule, the aft nose section, and the equipment housed within the aft nose section. The recoverable capsule forms the nose cap between DS 246.95 and DS 274.40 and extends aft within the aft nose section to DS 291.40. It is attached to the aft nose section at DS 274.40 by two-explosively-actuated pull-type pins. When these pins are pulled, four compressed springs accomplish ejection of the capsule. The aft nose section, which expands from an outside diameter (OD) of 33.126 inches at DS 274.40 to an OD of 50.5 inches at DS 306.95, houses the recoverable capsule propulsion equipment, the recoverable capsule attachment and ejection equipment, and a cooling duct that leads to the recoverable capsule from the plenum chamber on the forward equipment rack of the forward body. The aft nose section is bolted to the forward body section.

3.1.2 Forward Body Section. The forward body section, extending from DS 306.95 to DS 358.55, includes the forward equipment rack, the transponder antenna, the horizon scanner, the T/M exit antenna, and the umbilical and break-away receptacles. The forward body expands from an OD of 50.5 inches at DS 306.95 to an OD of 60 inches at DS 324.55, aft of which the body is cylindrical. The nose section is attached to the forward body section at DS 306.95 by six bolts located at 60 degree intervals around the periphery of the satellite. The foremost station of the forward body section, DS 306.95, is a service joint providing access to the forward equipment rack and to the equipment contained therein. The forward equipment rack, a welded structure situated between DS 306.95 and DS 340.05, carries the horizon scanner, the pressurized plenum chamber with ducts leading to the horizon scanner and to the payload capsule in the nose section, and those equipments detailed in Section III, page 1.3.3. A jettisonable fairing over the horizon scanner is attached to the forward body skin. The transponder antenna is located at DS 336.30, just aft of and in the same plane as the horizon scanner. The T/M exit antenna is located aft of the transponder antenna between DS 340.05 and DS 358.55, with a dielectric window at DS 357. The umbilical and breakaway receptacles are located on the side of the satellite just forward of DS 340.55. The forward body also houses the forward portion of the propellant tank assembly. The aft end of the forward body skin is riveted over the forward half of a T-section ring. The rear half of this ring forms a nut plate to which the forward end of the aft body skin is secured by screws to complete the joint between the body assemblies.

3.1.3 Aft Body Section. The aft body section, extending from DS 358.55 to DS 408.76, houses the balance of the propellant tank assembly, a major portion of the propellant pressurization system, and on satellites prior to 2-285-1050, the primary elements of the flight termination system. Between DS 358.55 and 388.55 the aft body is a cylinder with an OD of 60 inches. The rear portion of the assembly, comprising the engine mounting cone assembly, tapers from its intersection with the skin at approximately DS 383 to an OD of 37.75 inches at DS 408.76. The engine cone assembly consists essentially of a frustrum of welded and riveted construction terminating in an engine mounting ring at its aft end. This assembly is riveted to a support ring which is in turn riveted to the aft body skin at DS 388.55. The forward end of the engine mounting cone assembly and the skin are joined by screws at their intersection point forward of DS 388.55. The forward end of the aft body skin fits over and is secured to the rear half of the T-section ring/nutplate to which the aft end of the forward body skin is riveted. At DS 375.11, a ring-type support fitting welded into the propellant tank assembly mates with a doubler riveted to the aft body skin and is held in place through cutouts in the skin by six bathtub-type tension clamps placed in pairs at 120-degree intervals about the periphery of the vehicle. This clamping arrangement supports the tank assembly and secures it to the aft body structure. The adapter assembly, which houses the rear portion of the aft body assembly, is attached to the aft body at DS 388.55 by three separation brackets which incorporate explosive bolts and are located 120 degrees apart around the periphery of the structure. Forward extensions of the aft equipment rack beams and channels are riveted to the engine mounting cone assembly between DS 399.76 and DS 408.76. Three fairings, situated 120 degrees apart around the satellite and extending from DS 347 to DS 388.55, provide aerodynamic surfaces covering propulsion system equipment, auxiliary power equipment, the bathtub-type clamps, and the adapter separation brackets.

3.1.4 Aft Equipment Rack. The aft equipment rack, extending from DS 408.76 to DS 446.70, is riveted to the engine mounting cone portion of the aft body assembly. The rack is essentially a cylinder of approximately 38.4 inches (OD) formed by panels secured to beams and channels with rivets and screws and encircled by exterior rings of about 44.4 inches maximum OD. The equipment rack houses the engine and supports the following: elements of the propellant subsystem, including the helium pressure spheres and ullage control

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	SPACEFRAME Flight Configuration III DISCOVERER PROGRAM	
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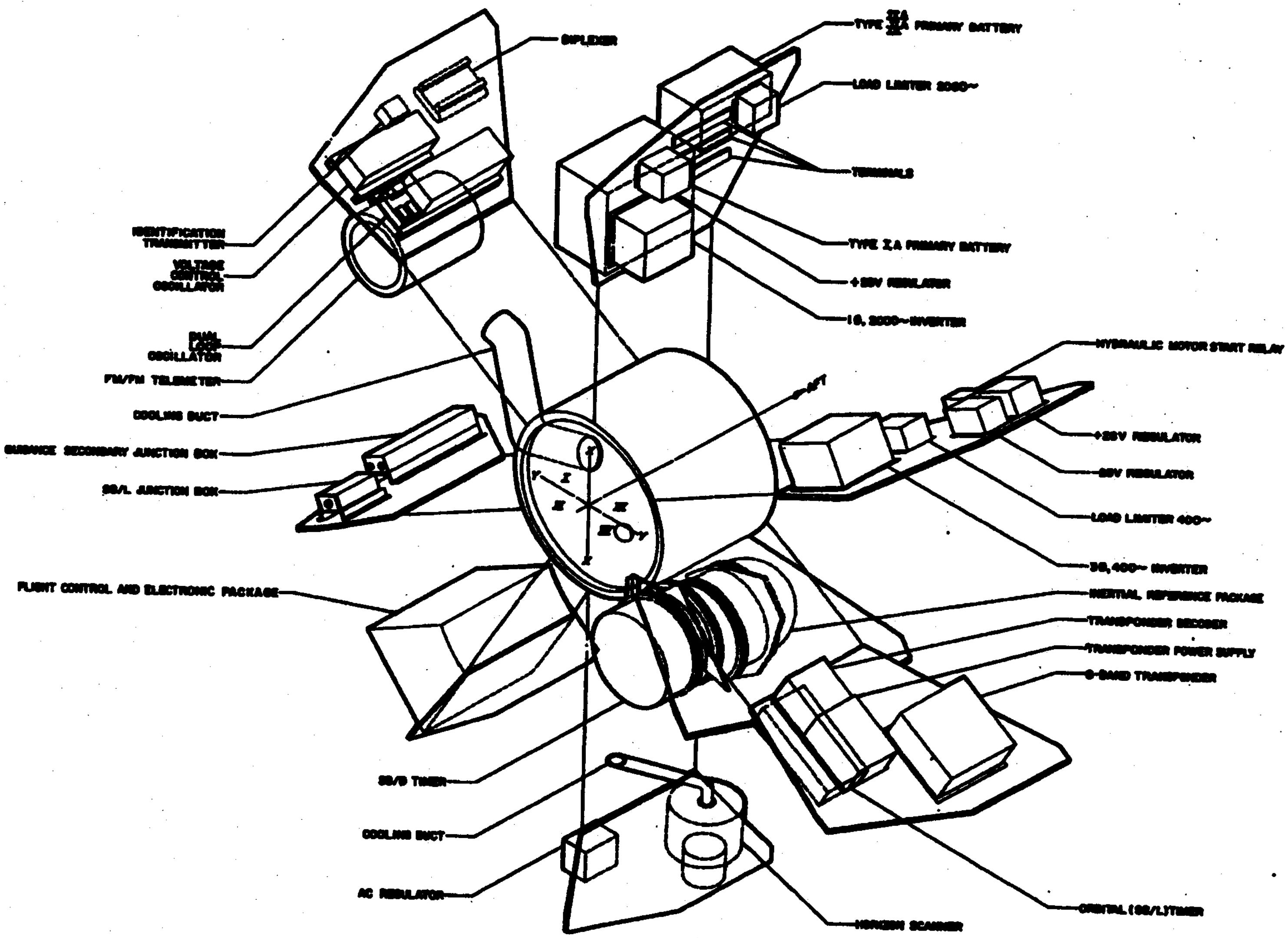

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rockets mounted on the interface at DS 446.70; elements of the guidance and control system, including the control gas pressure sphere mounted on the interface at DS 446.70; the T/M orbit antenna, mounted at DS 407.76 on the forward extension of a channel on the underside of the structure (horizontal attitude) and various items of GRD equipment. The aft equipment rack is equipped with six rollers that travel along matching rails on the adapter assembly in order to facilitate separation. Three of the rollers are mounted 120 degrees apart around the structure at DS 466.70; the other three are mounted correspondingly at DS 456 on brackets extending aft from the interface at DS 446.70. A separation switch is mounted on one of the aft roller extension brackets. A separation plug is installed on the interface at DS 446.70.

3.1.5 Adapter. The adapter section is a cylindrical tube which fits over and encases the aft equipment rack and a portion of the aft body assembly until separation. It provides the means for attaching the satellite to the Thor booster and is retained with the booster after separation. The adapter section extends from DS 388.55 to DS 453, having a constant OD of 60 inches from DS 388.55 to DS 436.25 and then flaring to an OD of 63.31 inches at DS 453. The adapter is joined to the aft body assembly at DS 388.55 by three separation brackets incorporating pin pullers. The aft end (DS 453) of the adapter is permanently attached to the forward end (TS 42.094) of the Thor booster during preparation for launch. The Agena engine and certain equipment items on the aft equipment rack extend beyond DS 453. The engine extends within the Thor booster to DS 481.01. (TS 70.104). The adapter section is equipped with three rails which are mounted at 120-degree intervals around the inside of the adapter shell and which protrude beyond the ends of the adapter section, extending from DS 385.55 to DS 484.15. These rails provide tracks along which the rollers on the aft equipment rack travel during separation. Two retrograde rockets which provide the separating impulse are installed opposite each other on the adapter. Each is secured by two rings at DS 436.90 and DS 444.40 and is covered by a fairing extending from about DS 384 to DS 452. The fairing is equipped with a jettisonable section that is blown off by the rocket upon ignition. The adapter assembly also carries the pin assembly portion of the separation plug. The Flight Termination destruct charge and battery are mounted to the adapter at DS 403.55 and DS 411.76 respectively.

Fairings. A jettisonable fairing is installed over the horizon scanner. Fairings that have jettisonable sections are installed over the two retrograde rockets. Permanent body fairings are installed at 120-degree intervals around the Discoverer body to provide aerodynamic surfaces over propulsion system equipment, auxiliary power equipment, the bathtub-type clamps at DS 375.11, the separation brackets at DS 388.55, and the flight termination destruct charge and battery.

3.2 Separation

3.2.1 Step 1. At the times noted in the Timed Sequence of Events (Section IV, page 2.0 ff, as applicable), SS/D timer signals (a) fire the explosive bolts to free the aft body from its connection to the adapter, and (b) then ignite the two retrograde rockets which are attached to the adapter. The thrust from these rockets decelerates the adapter-Thor combination, causing it to fall behind the satellite. (The retrograde rocket fairings are blown off upon ignition of the rockets.)

3.2.2 Step 2. The first motion of adapter-Thor separation from the satellite pulls the booster away from the spring-loaded separation switch, which is mounted on one of the aft equipment rack roller extension brackets. Prior to separation this switch is held in its depressed position by the forward end of the Thor at about DS 453. In the event of a premature separation (prior to deactivation of the switch by a signal from the SS/D timer), actuation of this switch activates the flight termination automatic subsystem and the vehicle is destroyed.

3.2.3 Step 3. As the adapter-Thor combination begins to separate from the satellite, the separation plug pin assembly mounted on the adapter pulls away from the spring-loaded separation plug mounted on the aft equipment rack interface. When this pin is pulled, the plug immediately disconnects, breaking the umbilical connection between the Thor and the satellite.

3.2.4 Step 4. The roller-rail mechanism of the adapter and aft equipment rack serves to reduce the possibility of drag and/or misalignment during separation.

4.0 POWER INPUT REQUIREMENTS

Power inputs from the SS/D timer are required for the functions noted in Par. 2.0 above.

5.0 ENVIRONMENT

The spaceframe structure and equipments are designed to meet the environmental criteria of LMSD Report 6117A as applicable.

6.0 RESPONSIBILITY.

Spaceframe and Installation Design Department (SS/A).

7.0 NOTES

Not applicable

8.0 SPACEFRAME INDEX

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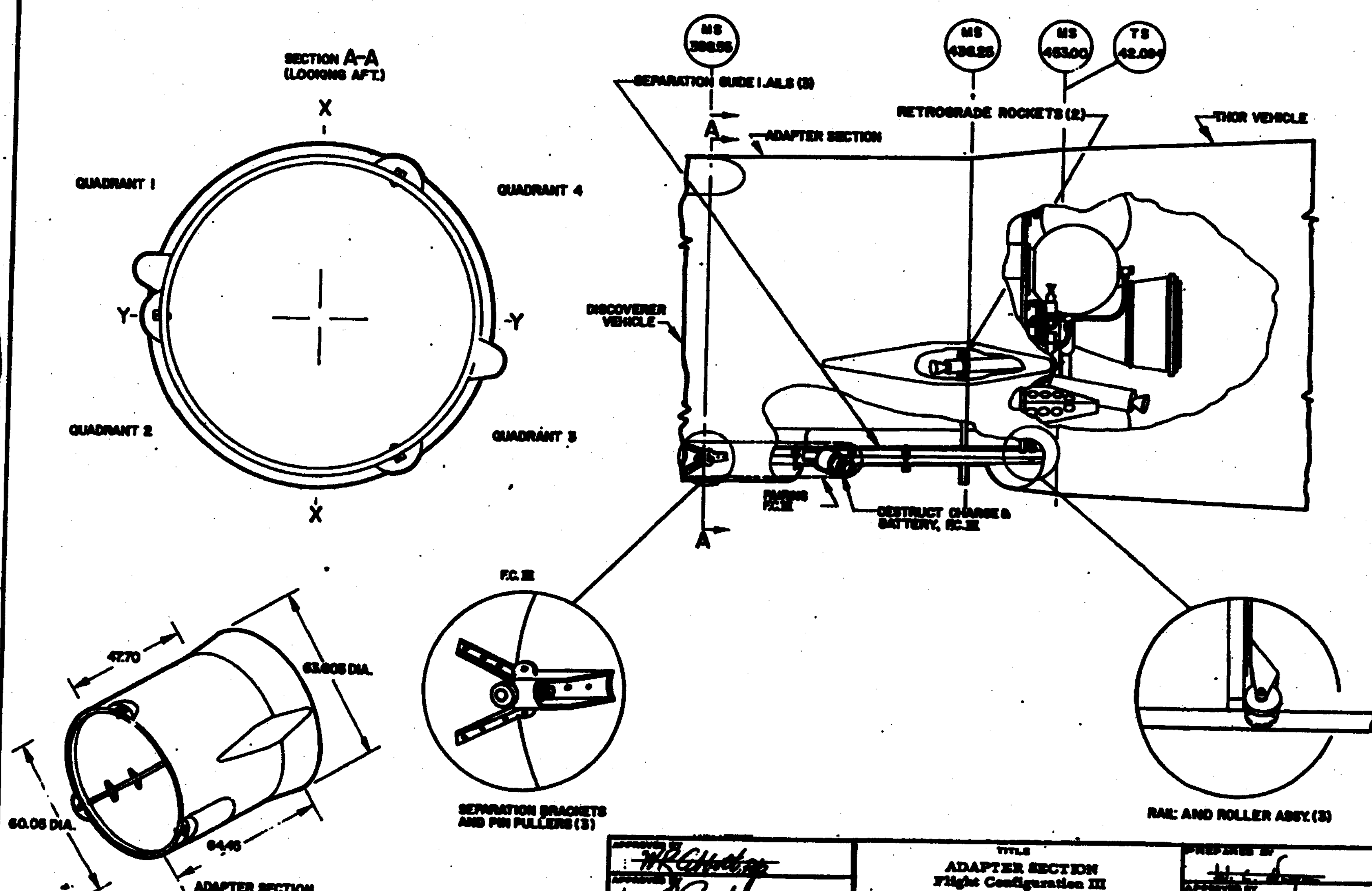

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1.0 GENERAL

The Discoverer satellite spaceframe for Discoverer Program Flight Configuration IX consists of the following sections: nose, including the recoverable payload capsule; forward body, including the forward equipment rack; midbody, including the propellant tanks; aft body, including the aft equipment rack; and the adapter. The spaceframe is a semi-monocoque structure consisting essentially of a magnesium alloy skin and supporting web. The satellite has a cylindrical body, 60 inches in diameter, which tapers to a blunt nose at the forward end and flares slightly at the rear of the adapter to accommodate mating with the Discoverer booster. Functionally, the spaceframe serves as a carrier for the equipment it houses, supports, and/or jettisons, as applicable, during its ascent to orbit and during its orbital life. During the Coast Phase, the spaceframe separates from the adapter, which has been permanently attached to the Discoverer booster during preparations for launch. At the end of the Orbit Phase, which is identified by the initiation of the recovery sequence, the payload capsule is ejected for reentry into the atmosphere for recovery.

In addition to the payload, either Advanced Engineering Test - Heavy or -Light (AET-H or -L), the satellite of this configuration may carry additional equipment for spatial environmental testing (provided by the Geophysical Research Directorate - GRD or School of Aviation Medicine - SAM) or for testing of spaceborne tracking aids, such as Doppler beacons and tracking lights.

Flight Termination equipments are considered a part of the satellite spaceframe. The equipments and the procedures involved, whether for commanded or automatic flight termination, are discussed within this spaceframe description.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in the Timed Sequence of Events (Section IV, pages 2.0 ff, as applicable), the following signal inputs are required for Discoverer Flight Configuration IX spaceframe structure and/or equipment functioning:

2.1 A signal from a barometric switch located in the thrust cone section of the recoverable capsule arms the recoverable capsule ejection system.

2.2 In the event that flight termination is required prior to satellite separation from the booster, a signal is originated at the Range Safety Officer's console in the blockhouse, sent via UHF signal to the Discoverer booster where it is received in one of two UHF range safety radio receivers, and relayed through the satellite separation plug and switch to the Discoverer satellite resulting in the energizing of the initiator which ignites the satellite destruct charge (cf. Par. 3.7.1 and 7.1 below).

2.3 Signals from Douglas Ground Operations and/or from the SS/D timer disable the automatic and the commanded destruct systems.

2.4 A signal from Douglas Ground Operations fires the pinpuller squibs to effect satellite/booster separation.

2.5 A signal from Douglas Ground Operations ignites the retrograde rockets attached to the adapter-booster combination to permit the satellite to coast clear.

2.6 A signal from the SS/D timer fires the pinpusher squibs to eject the horizon scanner fairing.

2.7 A signal from the SS/D timer arms the payload capsule ejection pin pullers.

2.8 A signal from the SS/D recovery sequence timer fires the payload capsule plug disconnect squib.

2.9 A signal from the SS/D timer fires the pinpuller squibs to initiate release of the payload capsule from the Discoverer satellite.

3.0 PERFORMANCE

3.1 Nose Section. The nose section, extending from Discoverer Station (DS) 185.90 to DS 245.90, includes the recoverable payload capsule and the nose cone fairing. The recoverable payload capsule is attached to the nose cone fairing at DS 213.45 with two explosively-actuated pull-type pins. The after portion of the payload capsule, consisting primarily of the capsule retrograde rocket engine assembly, is housed within the nose cone fairing. When the squib-actuated pin pullers are fired, four compressed springs attached to the fairing and acting against the payload capsule effect capsule ejection. The diameter of the payload capsule-nose cone fairing at the separation plane, DS 213.45, is 33.126 inches. The diameter of the nose cone fairing at DS 245.90, at which plane the nose cone fairing is bolted to the forward body, is 50.59 inches.

3.2 Forward Body Section. The forward body section, extending from DS 245.90 to DS 294.08, consists of a magnesium alloy sheet skin supported internally by magnesium alloy rings and by the "wagon-wheel" -shaped racks and shelving of the forward equipment racks which extend from DS 245.90 to DS 279.00. Among the equipments carried in the forward equipment racks are the propulsion system pressurization gas (helium) sphere, inertial reference package (IRP), horizon scanner installation, electrical power components, and various communications equipments, including antennas.

Structurally, the forward body is a truncated cone from DS 245.90 aft to DS 265.50, then cylindrical. A jettisonable fairing is installed along the X-axis, on what is the earth-side of the satellite during orbit, in the cylindrical portion of the forward body to provide an aerodynamic surface over the horizon scanner. The hemispherically-shaped forward end of the fuel tank extends into the forward body area. The three plumbing and wiring fairings (cf. Par. 3.3 below) start at DS 278.00

3.3 Midbody Section. The midbody section, consisting primarily of the integral propellant tanks, extends from DS 294.08 to DS 388.55. The integral tanks actually serve as structural members of the satellite, attaching to the forward body at DS 294.08 and to the aft body at DS 376.69. The hemispherically shaped ends of the fuel and oxidizer tanks extend into forward body and aft body, respectively, beyond the attachment points. Since the propellant tanks constitute structural members, fairings are provided as aerodynamic surfaces to cover and protect the plumbing and wiring which connect components located in the aft body with affected components in the forward body, since this plumbing and wiring must pass outside the tanks and thus over the surface of the satellite. The three plumbing and wiring fairings are located at 120-degree intervals about the periphery of the satellite, one on the Y-axis between quadrants I and II, and the others in quadrants III and IV. The fairings begin at DS 278.00 (in the forward body) and extend into the adapter. At the separation plane between satellite and adapter, the squib-actuated pin pullers are located within the plumbing and wiring fairings (cf. Par. 3.4 below). The fairing in quadrant IV extends aft on the adapter beyond the other two to house the destruct charge (cf. Par. 3.5 below).

Of the propellant tanks, the forward tank contains the fuel and is nearly spherical. The oxidizer tank fits over the fuel tank, the cylindrical portion attaching tangentially to the fuel tank. The aft bulkhead of the fuel tank forms the forward bulkhead of the oxidizer tank. The oxidizer tank is basically cylindrical from its attaching point with the fuel tank aft to attaching point with the aft body, at which point the tank rounds off as a hemisphere. The fuel feed line passes from the fuel tank through the center of oxidizer tank to connections with the fuel pump in the aft body section.

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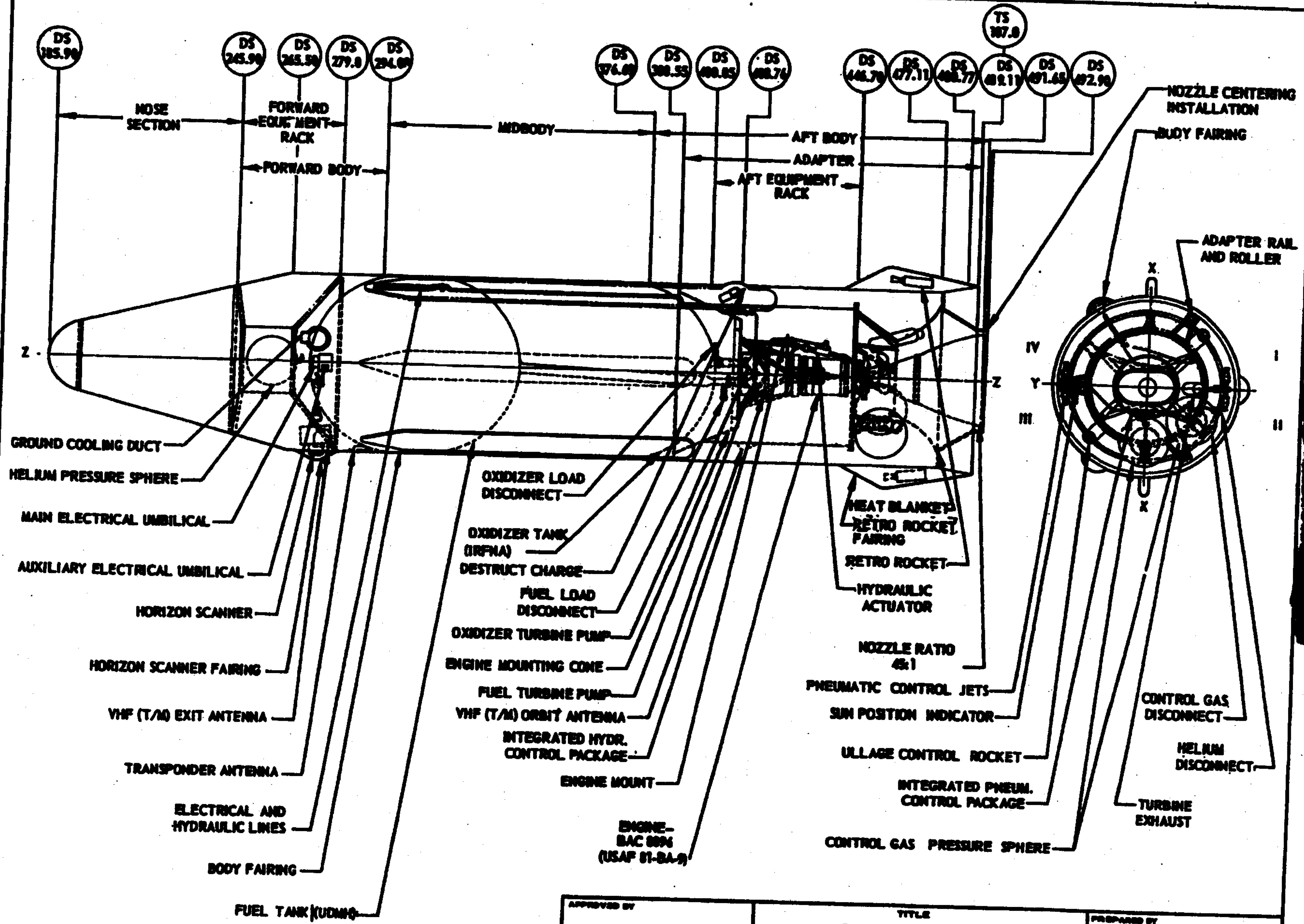

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The adapter abuts against the mid body section at DS 388.55 and is secured to the mid body section by pins. Three separation fittings, incorporating squib-actuated pin pullers, are installed about the periphery of the mid body section at DS 388.55. These fittings are located at 120-degree intervals about this periphery, external to the skin of the satellite, and are protected by the plumbing and wiring fairings. These separation fittings serve to transfer all bending loads to the adapter; shear is transferred by an interlock joint at DS 388.55.

3.4 Aft Body Section. The aft body section, extending from DS 388.55 to DS 491.65, includes the remainder of the satellite, less the adapter. In this section are the engine mounting cone, the aft equipment rack, and the rocket engine. The adapter abuts against the mid body section at DS 388.55 and encloses the aft body section until satellite-booster separation.

The engine mounting cone attaches to the skin of the satellite at DS 382.62. This frustum-shaped engine mounting cone extends aft to DS 408.76, at which point there is an engine mount ring to which are attached the four legs of the BAC-fabricated engine mount. The rocket engine, Bell Aircraft Corporation (BAC), Model 8096, then extends aft to DS 491.65, allowing for the 45:1 expansion ratio nozzle which is housed in the nozzle centering installation on the adapter.

The aft equipment rack, essentially cylindrical in shape with an OD of 38.4 inches, attaches to the engine mounting cone at DS 400.05 and extends aft to DS 446.70. The aft equipment rack encloses portions of the rocket engine subsystem, including the propellant pumps and engine hydraulic actuators, and supports the following: elements of the propulsion system (ullage control rockets mounted on the interface at DS 446.70); elements of the guidance and control system, including the pneumatic control gas jets, the control gas sphere, and the sun position indicator also mounted to the interface at DS 446.70, and the hydraulic power package, installed in quadrant IV at DS 408.76; the telemetry (T/M) orbit antenna, mounted on the X-axis between quadrants II and III at DS 407.76; and extraneous equipments, including certain Geophysical Research Directorate (GRD)-furnished items, an Applied Physics Laboratory (APL) Doppler tracking beacon installation, and a tracking light. The APL beacon installation consists of two whip antennas installed in quadrants II and III, just forward of DS 408.76. The tracking light is installed between DS 411.96 and DS 423.96 near the X-axis in or between quadrants II and III. To facilitate separation of the satellite from the adapter, the aft equipment rack has three pairs of rollers which ride on rails attached to the inside of the adapter (cf. Par. 3.5 below). The forward roller of each pair is mounted 120 degrees apart around the structure at DS 408.76 (on the Y-axis between quadrants III and IV, and in quadrants I and II); the after roller of each pair, aligned correspondingly with its mate, is mounted at DS 484.00 on an aft roller extension bracket attached to the aft equipment rack at the interface at DS 446.70. A separation plug is installed on the interface at DS 446.70. A separation monitor system is provided, which consists of a number of fingers installed longitudinally on the inside of the adapter and a separation monitor switch, mounted adjacent to one of the after rollers. Triggering of the separation monitor switch by the fingers on the adapter as the satellite passes clear of the adapter is telemetered as pulses to the ground as confirmation of separation.

3.5 Adapter. The adapter section of the spaceframe, extending from DS 388.55 to DS 492.90, houses the aft equipment rack until satellite-booster separation. The adapter is secured to the aft body section at separation plane (DS 388.55) by pins. The adapter is a 60-inch cylinder from DS 388.55 to DS 477.11, at which point it flares as a truncated cone to DS 489.11 with a diameter of 63.31 inches. During preparations for launch, the adapter is permanently attached to the Discoverer booster, the mating stations being DS 489.11 for the Discoverer and, in the case of the Douglas-developed booster, Thor Station (TS) 107.0. The adapter is equipped with three rails, installed 120 degrees apart around the inside of the adapter and extending from DS 388.80 to DS 484.15. The rollers which are attached to the aft equipment rack (cf. Par 3.4 above) travel on these rails during separation to reduce perturbations.

The wiring and plumbing fairings which commenced on the forward body section and extended over the midbody terminate on the adapter section. Two of the fairings, the ones on the Y-axis and in quadrant III, terminate at DS 394.50; the other fairing in quadrant IV extends to DS 430.00 and incorporates the destruct charge and hemispherically-shaped destruct charge fairing at DS 408.55. Two retrograde rockets, mounted diametrically opposite on the X-axis on the external surface of the adapter at DS 477.11, provide a regressive impulse to slow the adapter-booster combination during separation and allow the Discoverer satellite to coast clear. Each retro-rocket installation is supported by a magnesium ring at DS 477.11 and is aerodynamically protected by a fairing from DS 448.90 to DS 489.11. A portion of each of these fairings, forward of DS 461.70 is jettisoned by the retro-rocket ignition thrust.

Access through the adapter is provided as follows: two hinged doors, one in quadrant I at DS 401.24 for propellant load disconnect, the other in quadrant II at DS 403.38 for propellant dump disconnect; screw-fastened doors for separation rail-roller inspection and/or adjustment, three at DS 408.51 and three at DS 481.20, each set of three arranged at 120-degree intervals about the structure to correspond with the roller installations on the aft equipment rack (cf. Par. 3.4 above); two hinged doors in quadrant II at DS 434.10 for access to the propellant pressurization gas fill and for access to the attitude control gas fill; and a screw-fastened door in quadrant II at DS 403.95 for access to the engine start plug, engine arming squib, and ullage rocket arming squib.

A nozzle centering installation, consisting of a 33-inch diameter ring supported by six legs equilaterally disposed (X-X axis and 60 degrees therefrom) and attached to the adapter at DS 477.11, is included in the adapter section. This ring extends aft of the booster-adapter interface to DS 492.90 and encloses the aft extremity of the engine thrust chamber nozzle (DS 491.65) to retain the nozzle in a central position during the Ascent Phase until satellite-booster separation.

3.6 Separations. Two separations occur as affects the spaceframe. The first of these is the separation of the Discoverer satellite from the adapter-booster combination at the end of the Ascent Phase, preliminary to the Coast Phase. The second spaceframe separation occurs when the recoverable capsule is ejected.

3.6.1 Satellite-Booster Separation. Subsequent to vernier shutdown, a signal from Douglas Ground Operations fires the squibs to actuate the pinpullers to free the satellite from the adapter which is permanently attached to the Discoverer booster. A simultaneous signal from Douglas Ground Operations fires the retrograde rockets which are attached to the adapter and so positioned to impart a thrust counter to the direction of movement of the adapter-booster combination. This thrust decelerates the adapter-booster combination, permitting the satellite, which has been detached, to coast clear of the combination. The forward portion of the retro-rocket fairings are blown off by the initial blast from the retro-rockets.

In the event that the automatic destruct system has not been deenergized, the first relative motion of separation between the satellite and the adapter (in a practical sense, pre-mature separation) actuates a lanyard-type separation switch to complete the electrical circuit within the satellite to fire the destruct charge.

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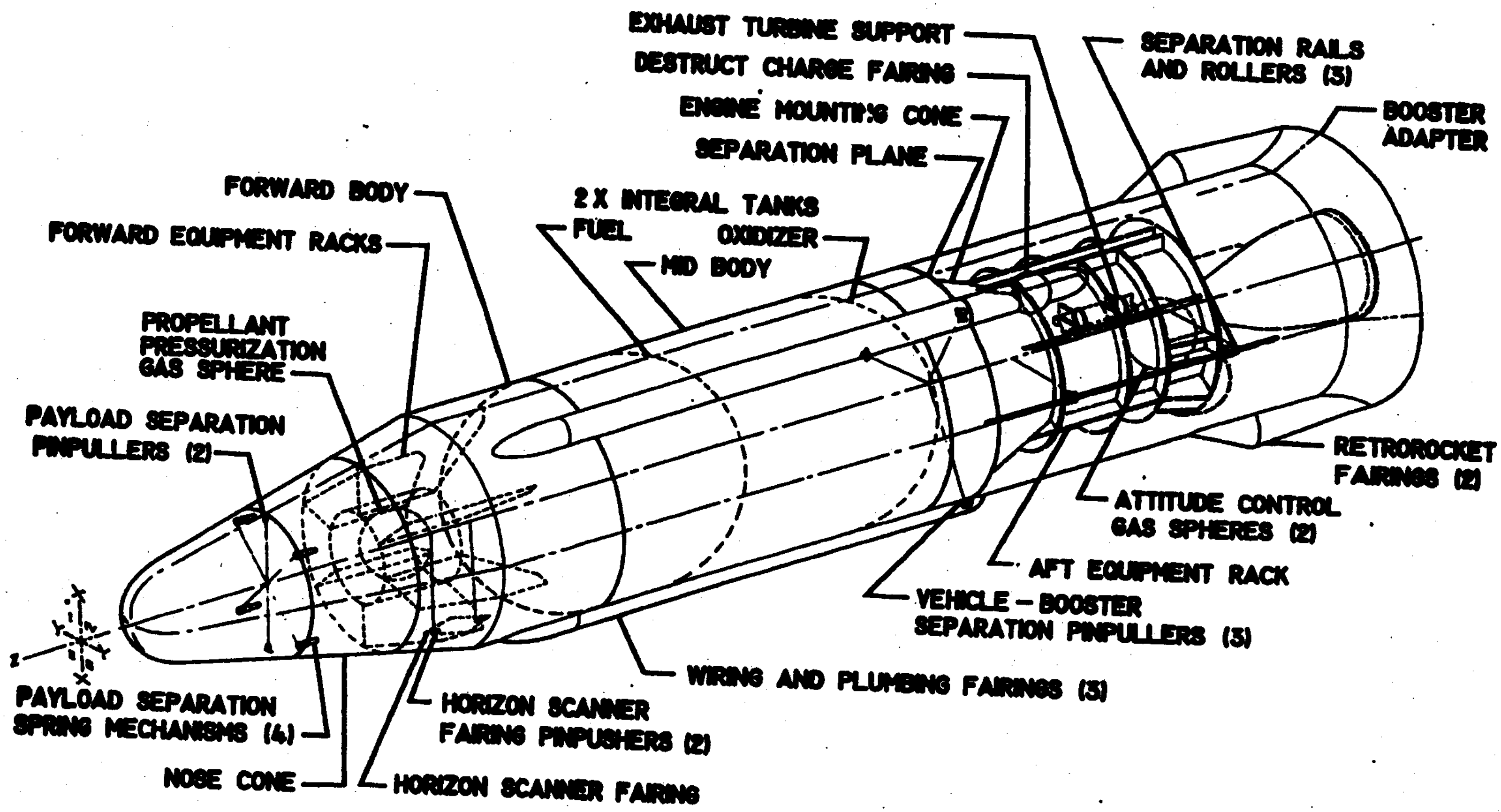

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As the satellite and adapter-booster combination separate from one another, the separation plug pin assembly attached to the adapter disconnects from the separation plug mounted on the after face of the aft equipment rack. This action breaks the umbilical connection between the Discoverer satellite and its booster. The roller-rail mechanism of the adapter and aft equipment rack serves to reduce the possibility of drag and/or misalignment during separation. Booster separation is monitored through the separation monitoring system (cf. Par. 3.4 above). The fingers mounted longitudinally to the interior of the adapter successively trigger a finger on the separation monitor switch as the satellite passes clear of the adapter. These triggerings result in a pulsed signal which is telemetered to the ground to give an indication that separation has occurred.

3.6.2 <u>Payload Capsule Separation.</u> As the Discoverer ascends past approximately 70,000 feet altitude, a barometric switch arms the recoverable capsule ejection system. As capsule ejection time approaches, a signal from the orbital timer restarts the SS/D timer which had been secured after Orbital Boost Phase. Signals in sequence from the SS/D timer arm the capsule ejection squib and then fire the capsule plug disconnect squib and the capsule eject squib. Complete description of the payload capsule ejection sequence is found in Section II, page 7.9 ff.

3.7 <u>Flight Termination System.</u> The Flight Termination System of Discoverer Flight Configuration VI is divided functionally into two subsystems: a Commanded Termination Subsystem and an Automatic Termination Subsystem. Units common to both subsystems include the destruct charge, the initiator, and the arming-disarming circuitry. The 8.31-pound destruct charge assembly, including a 1.3 pound composition A-3 shaped charge, is installed through the skin of the adapter at DS 405.55 and is so positioned that the shaped charge, when detonated, penetrates both propellant tanks. The indiscriminate mixing of the hypergolic propellants, together with the ignition provided by the charge, results in an explosion which destroys the satellite.

3.7.1 <u>Commanded Termination Subsystem.</u> When the flight termination command signal, originating at the Range Safety Officer's console in the blockhouse, is received by one of the two UHF range safety radio receivers in the Discoverer booster, the booster decoder sends a pulse through the satellite separation plug and switch directly to the destruct charge explosive squibs.

3.7.2 <u>Automatic Termination Subsystem.</u> In the event of satellite-booster separation prior to the disarming of the initiator either by signal from Douglas Ground Operations and/or SS/D timer, motion of the satellite away from the booster-adapter combination actuates a pair of "lanyard-type" switches to complete a circuit through the auxiliary battery to the destruct charge explosive squibs. To actuate the explosive squibs, however, both lanyard switches must close. The lanyards, located diametrically to one another, are attached to the after face of the aft equipment rack at DS 446.70; the switches themselves are located on the adapter. The auxiliary battery used is a potted, series connected, 3-cell, nickel cadmium power source having a nominal rating of 3.75v d-c and a capacity of 0.225 ampere hours.

3.7.3 <u>Arming and Disarming.</u> The initiator, a rotary armature device (Ledex) capable of being pulsed to either "armed" or "safe" positions, is armed just prior to launch by a 28-volt d-c pulse sent from the blockhouse through the auxiliary umbilical plug. In the "safe" position, the connections to the destruct charge explosive squibs are grounded and the circuits from the power sources, both booster power and auxiliary battery power, are opened. In the "armed" position, the grounds to the explosive charge squibs are removed and the power sources are again included in the circuitry to the explosive squibs. The flight termination system is disarmed just after booster burnout and prior to satellite separation from the booster-adapter combination by a signal from Douglas Ground Operations and/or SS/D timer that opens the battery circuits once again and grounds the explosive squib connections.

4.0 POWER INPUT REQUIREMENTS

Power inputs for the operations noted above are from the SS/D timer, the booster guidance and control system, and the booster flight termination system, as applicable.

5.0 ENVIRONMENT

The Discoverer spaceframe is designed to meet the environmental criteria of LMSD Report No. 6117A

6.0 RESPONSIBILITY

Spaceframe and Installation Design Department (SS/A).

7.0 NOTES

Not applicable.

8.0 SPACEFRAME INDEX

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QUAD II (upper)
QUAD I (lower)

QUAD IV

The above photographs show two views of the forward equipment rack of a representative satellite (S-208-1104) of Discoverer Flight Configuration VI. The equipments are installed on radial shelving.

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1.0 GENERAL

The propulsion system for the Discoverer Program Flight Configuration III satellite consists of an engine subsystem and a propellant subsystem. The single-burn liquid bi-propellant rocket engine (USAF Model No. YLR-81-BA-5, internally identified as the 8048 engine) employs inhibited red fuming nitric acid (IRFNA) as oxidizer and unsymmetrical dimethyl hydrazine (UDMH) as fuel. It has a nominal burn time capability of 120 seconds. The engine subsystem consists of: (a) the thrust chamber assembly, and associated lines and valves; (b) the turbine pump assembly, including the turbine starter, gas generator, and associated lines and valves; (c) the engine relay box; and (d) the engine mount (BAC fabricated) and the engine mounting cone (LMSD fabricated). The propellant subsystem consists of: (a) the propellant tank assembly and its mount; (b) the IRFNA oxidizer and the UDMH fuel; (c) the pressurization system, including the helium gas and the pressure spheres; (d) the ullage control rockets; and (e) propellant lines and valves, and associated pyrotechnics and wiring. Functionally, the propulsion system provides the second-stage total impulse (The Discoverer Booster provides first-stage total impulse) that enables the Discoverer satellite to achieve the planned orbital injection conditions.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in Section II, pp 2.0 ff, as applicable, the following signal inputs are required for propulsion system operation.

2.1 A SS/D timer signal actuates the ullage control rockets.

2.2 A SS/D timer signal energizes the power relay in the engine relay box to arm the engine ignition system.

2.3 A SS/D timer signal: (a) fires the by-pass valve squib to couple the main helium pressure regulator into the fuel pressurization line; and (b) actuates the gas generator turbine starter to initiate turbine operation.

2.4 A signal from the integrator in the SS/D computer energizes the shutdown relay in the engine relay box to effect engine shutdown when the satellite achieves the predetermined velocity gain.

2.5 A SS/D timer signal actuates the oxidizer, fuel, and helium vent valve squibs to initiate venting of propellant tanks and lines and of pressure spheres and lines after engine shutdown.

3.0 PERFORMANCE

3.1 Description

3.1.1 Engine Subsystem. The gimbal-mounted thrust chamber assembly (with associated lines and valves) consists of: (a) oxidizer valve and fragible retainer disc; (b) injector, with oxidizer manifold; (c) oxidizer manifold pressure switch; (d) solenoid-operated pilot valve; (e) fuel valve and frangible retainer disc; (f) thrust chamber (regeneratively cooled by oxidizer flow through passages in the chamber wall), including contoured nozzle with 20:1 expansion ratio; (g) nozzle closure; and (h) thrust chamber pressure switch. The turbine pump assembly (with associated lines and valves) consists of: (a) turbine starter, including explosive actuators, starter igniter, solid-propellant-grain starter, and rupture disc; (b) gas-driven turbine, including gear box and exhaust duct; (c) oxidizer pump; (d) fuel pump; (e) fuel pump pressure switch; (f) gas generator (fuel/oxidizer) inlet valve and solenoid valve; (g) gas generator fuel and oxidizer flow venturis; and (h) gas generator combustion chamber. The engine relay box, which is mounted on the engine mount, contains a power relay, a thrust chamber relay, a shutdown relay, the circuitry necessary to accomplish starting and shutdown sequence switching functions (Ref. Par. 3.2, below), and a squib monitor assembly. The engine mount (BAC fabricated) consists of four braced legs constructed of steel tubing and an engine gimbal ring supported by the four legs. The forward ends of the legs are attached around a ring at the engine mounting cone at DS 408.76; the gimbal ring supports the forward end of the thrust chamber assembly. Two hydraulic actuating cylinders (hydraulic control system) are attached between the aft ends of two adjacent legs and the thrust chamber just forward of the nozzle throat. In response to differential signals from the guidance and control electronics system (SS/D), these cylinders provide up to ±3.5 degrees of gimballed motion of the thrust chamber in order to effect pitch and/or yaw corrections during propulsion system operation. The engine mounting cone (LMSD fabricated) is essentially a truncated cone of welded and riveted construction terminating in an engine mounting ring at its aft end (DS 408.76). At its forward end (approx. DS 383), the cone intersects the aft body skin and is fastened to it by means of screws. At DS 388.55 the cone is riveted to a support ring which is in turn riveted to the aft body skin, providing the cone with its primary supporting structure.

Performance Characteristics:

Engine System

Rated Thrust	15,490 lbs ±2.5% (at altitude)
Turbine Exhaust Thrust	190 lbs (included in rated thrust)
Effect of Turbine Speed increase at Altitude	300 lbs (included in thrust)
Nozzle Expansion Ratio	20:1
Specific Impulse (minimum)	278 lb-sec/lb
Burn Time (nominal)	120 sec
Thrust Chamber Pressure (nominal)	487 psia

3.1.2 Propellant Subsystem. The propellant tank assembly, extending from DS 326.55 to DS 404.76, consists of two nested aluminum sections welded together and sharing a common bulkhead. The forward section, which contains the oxidizer, nests into the aft section, which contains the fuel. The oxidizer tank is spherical, with a radius of 2.5 feet. The aft end of the fuel tank is approximately hemispherical and tapers slightly inward to intersect the oxidizer tank. The propellant tank assembly is secured to the spaceframe aft body at DS 375.11 by means of a support ring welded to the fuel tank. This ring mates with a doubler riveted to the aft body skin and is held in place through cutouts in the skin by three pairs of bathtub-type tension clamps located at 120-degree intervals around the periphery of the satellite spaceframe. The pressurization system consists of: (a) two helium pressure spheres; (b) main helium pressure regulator and helium filter assembly; (c) slave pressure regulator; (d) squib-operated by-pass valve; (e) quick-disconnect fill valve, pressure relief valves, and squib-operated vent valve; (f) pressure selector valve and lip seal regulator; (g) oxidizer and fuel pressurization lines; and (h) other fill, flow, relief, and vent lines required by the system. The two helium pressurization spheres, which are strapped to the interface at the rear of the aft equipment rack (DS 446.70), store a total of 3988 cu in. of helium (2200+1788) at a maximum pressure of 3000 psi at +120°F.

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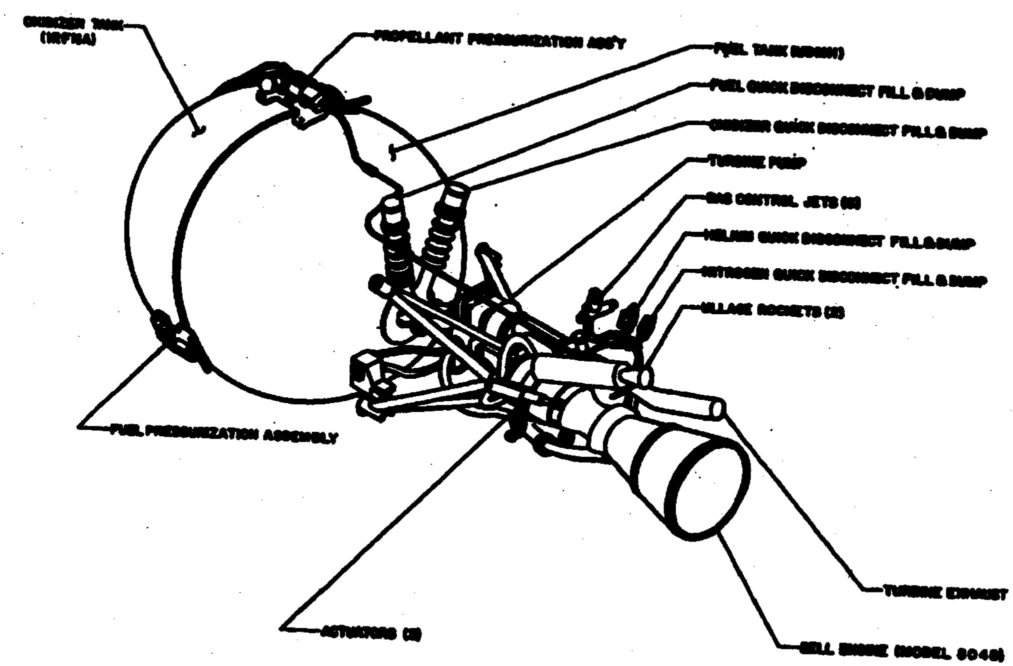
LOCKHEED AIRCRAFT CORPORATION MISSILES AND SPACE DIVISION



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Before engine ignition, the normally-closed by-pass valve is used to prevent contact between any oxidizer and fuel that may have leaked past the oxidizer and fuel check valves. While the by-pass valve is closed, pressure is supplied to the oxidizer tank through the main helium pressure regulator and to the fuel tank through the slave pressure regulator. The slave regulator, which is connected to the main regulator by a pressure-sensing line, maintains a fuel tank pressure approximately equal to that of the oxidizer tank, with a tolerance of +0/-2 psi. When engine operation is initiated, the squib-operated by-pass valve is opened, coupling the fuel tank pressurization line to the main pressure regulator and effectively removing the slave regulator from the system. The pressurization system also supplies, via the lip seal pressure regulator, 5 ±3 psig to the rear side of the oxidizer lip seal in the oxidizer pump. Two ullage control rockets are mounted on the interface at the rear of the aft equipment rack (BS 446.70) in symmetrically opposite locations, with their thrust vectors directed through the satellite c.g. ±0.5 degree. The ullage control rockets burn for a nominal 18.8 seconds, producing sufficient thrust (126 lbs nominal thrust; 4800 ±6% lb-sec nominal total impulse) to displace any gas or vapor which has accumulated in or around the propellant tank outlets during the coast period that follows booster rocket engine (Thor) burnout. The final portion of the ullage control rocket burn period overlaps the beginning of the rocket engine subsystem burn period (ref. Section II, pp 2.0 ff, as applicable, for overlap duration) to assure gas-free propellant flow to the engine subsystem and, hence, a smooth start. The propellant subsystem also includes propellant lines, valves, and associated mechanisms and wiring as follows: (a) quick-disconnect fill and dump valves and lines for oxidizer and fuel; (b) oxidizer and fuel lines from the propellant tanks to the propellant pumps, with filters; (c) oxidizer and fuel check valves between the pressurization lines and the propellant tanks; (d) oxidizer and fuel pressure relief lines and valves; and (e) squib-operated vent valves for both oxidizer and fuel.

Performance Characteristics:

Propellant System

Oxidizer	Inhibited Red Fuming Nitric Acid Type III per MIL-P-7254E (USAF)
Fuel	Unsymmetrical Dimethyl Hydrazine per MIL-D-25604B (USAF)
Oxidizer Tank Volume	49.0 cu ft
Fuel Tank Volume	37.8 cu ft
Total Propellant Weight (nominal)	6640 lbs
Pressurizing Gas	Helium (2 pressure spheres)
Helium Pressure (Maximum)	3000 psi at +120°F
Helium Sphere Volume	2200 + 1726 = 3926 cu in
Total Head, Oxidizer Tank to Pump	65.0 to 73.0 psia
Total Head, Fuel Tank to Pump	58.5 to 59.0 psia
Propellant Temperature	+45 to +60°F
Oxidizer Specific Gravity	1.568 (at +60°F)
Fuel Specific Gravity	0.795 (at +60°F)
Total Propellant Flow Rate (nominal)	55.27 lbs/sec
Total Weight of Impulse Propellants (nominal)	6487 lbs

3.2 Operation

3.2.1 Starting Sequence. Propulsion system operation begins with the actuation of the ullage control rockets by a signal from the SS/D timer (noted in Par. 2.1, above). The burning of these rockets creates a longitudinal force that displaces any gas or vapor which has accumulated in or around the propellant tank outlets, bleeds propellants through the propellant lines to the propellant valves, and prevents the accumulation of vapor or gas bubbles in the propellant lines or at the turbine inlets. While the ullage control rockets are still burning, a SS/D timer signal (Par. 2.2, above) arms the engine ignition system by applying 28v dc to the power relay in the engine relay box. Simultaneously, a SS/D timer signal (Par. 2.1, above): (a) ignites the helium by-pass valve squib to open that valve, coupling both propellant pressurization lines to the main helium regulator and by-passing the slave (fuel tank) pressure regulator; and (b) ignites the explosive actuators of the gas generator turbine starter igniter assembly. Burning of the igniter charge in turn ignites the solid-propellant-grain starter, and the resulting combustion gases initiate turbine operation, accelerating the turbine to about half its rated speed. The turbine, through the turbine gear box, begins to drive the oxidizer and fuel pumps, which start to pump propellants from the propellant tanks. Oxidizer pressure increases (a) in the oxidizer section of the gas generator valve; and (b) in the main oxidizer valve. Fuel pressure increases: (a) in the fuel section of the gas generator valve; and (b) in the main fuel manifold line. Fuel is pumped: (a) through the actuating section of the gas generator valve to the solenoid valve; (b) through the actuating section of the main fuel valve to the solenoid-operated pilot valve. At this point in the starting sequence, the gas generator valve and the main oxidizer and fuel valves are in their normally-closed positions, permitting no propellant flow into the gas generator combustion chamber or the engine thrust chamber; both solenoid valves are in their normally-open positions, permitting fuel flow through return lines to the fuel pump. Propellant tank operating pressures are maintained by the pressurization system, which supplies helium at regulated pressures from the helium pressure spheres. When turbine action raises the fuel pump pressure to 237 psig, the fuel pump pressure switch closes, actuating the gas generator solenoid valve. When this valve closes, the resulting pressure on the gas generator valve's spring-loaded actuating piston forces the oxidizer and fuel sections of the valve open to admit the liquid propellants into the gas generator combustion chamber, where hypergolic ignition occurs. Gas generation by liquid propellant combustion begins about 0.7 seconds after the SS/D timer signal noted above initiates combustion of the solid propellant starter, and for about 0.7 second thereafter both solid and liquid propellants generate gas in the gas generator combustion chamber. Oxidizer and fuel pressures and flow rates to the gas generator are regulated to the required levels by a venturi in each propellant line. The nominal pressure in the gas generator combustion chamber is 495 psia, with temperatures in the range of 1400 to 1550°F at the gas generator outlet. The gas generator at full pressure drives the turbine to its nominal full speed of 24,000 rpm, and the turbine in turn drives the oxidizer and fuel pumps. The gas that drives the turbine is exhausted through the turbine exhaust duct, adding 190 pounds of thrust (thrust vector directed through the satellite c.g. ±1.0 degree) to that of the engine thrust chamber at operating altitudes. Propellant pump action builds up pressure in the main oxidizer and fuel lines. When pressure in the oxidizer line reaches 130 psig, the main oxidizer valve's spring-loaded actuating piston is forced open (about 0.5 second after the SS/D timer signal to the gas generator starting squibs). At 180 psig, the frangible retainer disc in the oxidizer valve ruptures, permitting oxidizer to flow through the cooling passages in the thrust chamber wall and into the oxidizer manifold of the injector. When pressure in the main fuel line reaches 325 psig, the frangible retainer disc in the fuel valve ruptures, permitting fuel flow as far as the fuel valve poppet. As oxidizer begins to flow through the injector and into the thrust chamber, pressure

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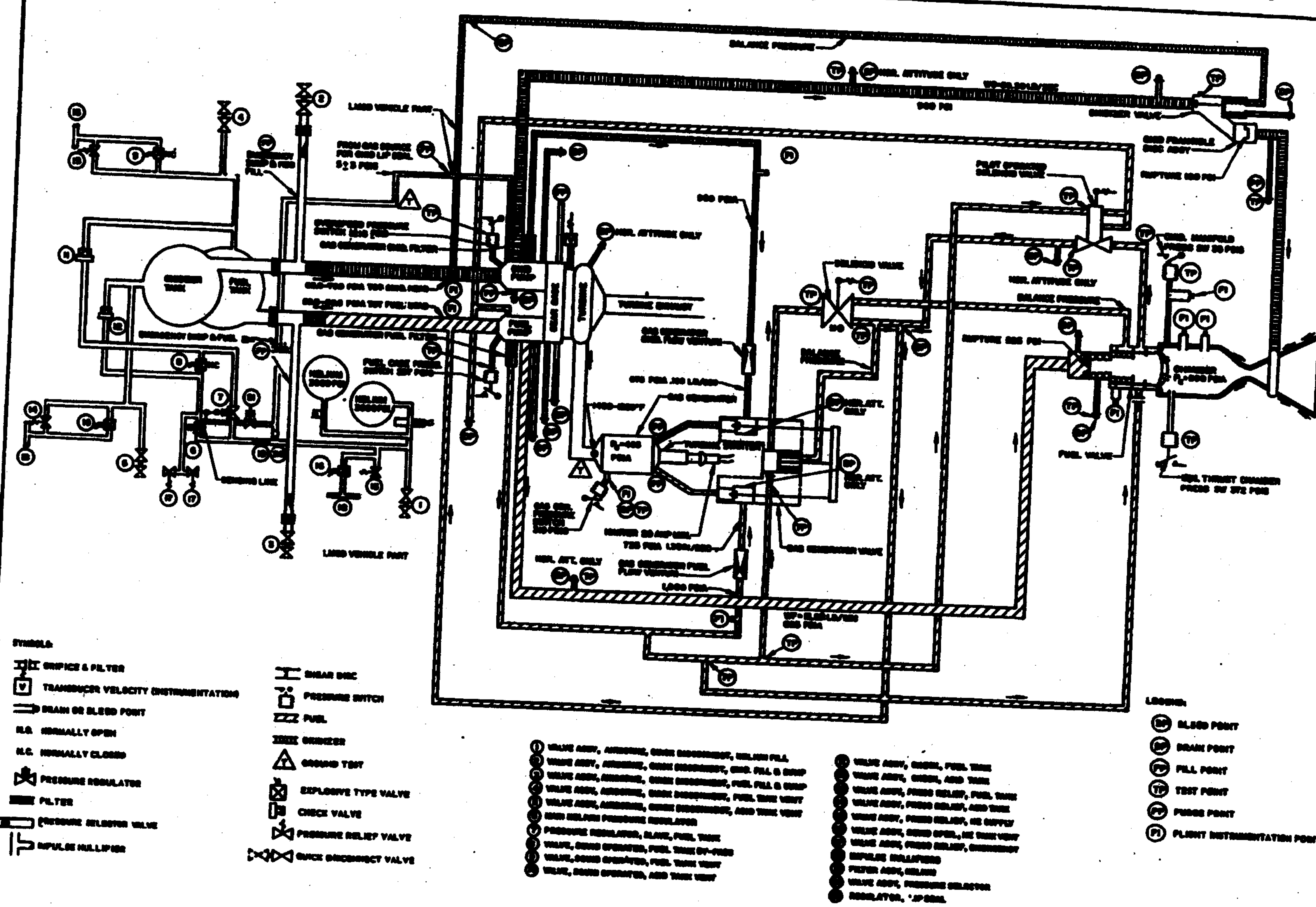

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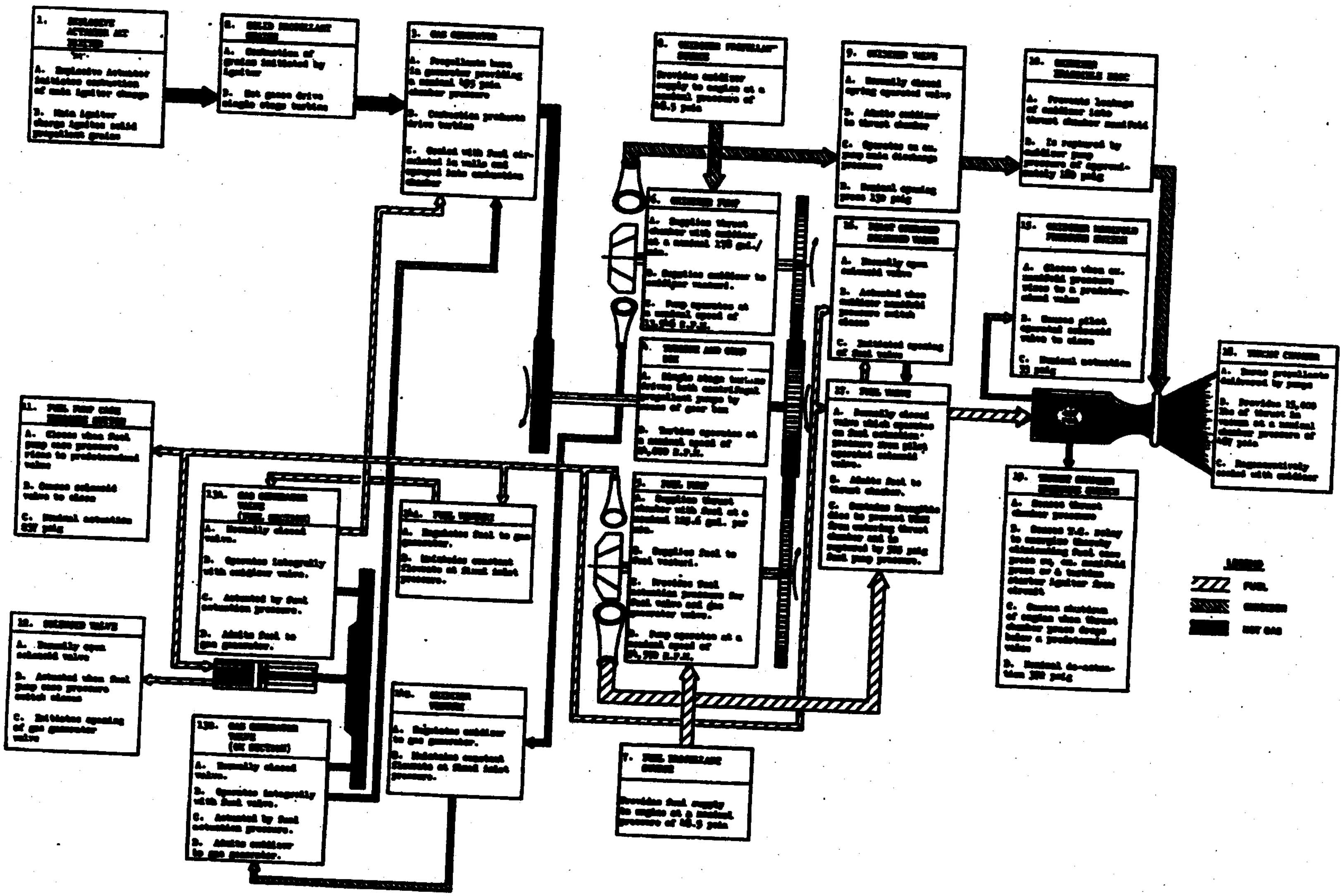

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in the oxidizer manifold quickly reaches 33 psig, and the oxidizer manifold pressure switch closes, actuating the solenoid-operated pilot valve. When this valve closes, the resulting pressure on the main fuel valve's spring-loaded actuating piston forces the valve open (about 1.4 seconds after the SS/D timer signal to the gas generator starter squibs) to permit fuel flow through the injector head and into the thrust chamber. Hypergolic ignition of the propellants occurs, the thrust chamber nozzle closure is ruptured and blown free of the nozzle, and 90 percent of steady-state thrust is achieved about 10 milliseconds after propellant ignition (1.3 to 1.9 seconds after the SS/D timer signal to the gas generator starter squibs). Nominal thrust chamber operating pressure is 407 psia. When pressure in the chamber reaches 372 psig, the thrust chamber pressure switch closes, energizing the thrust chamber relay. This relay: (a) connects 28 v dc directly to the gas generator solenoid valve and the solenoid-operated pilot valve, locking these valves in the closed position so that engine operation is rendered independent of pressures applied to the fuel pump pressure switch and the oxidizer manifold pressure switch; and (b) converts the thrust chamber pressure switch into an engine shutdown switch (Ref. Par. 3.2.2, below).

3.2.2 <u>Shutdown Sequence</u>. Propulsion system shutdown can occur in two ways — by command or as the result of thrust chamber pressure decay. Command shutdown is initiated by the integrator in the SS/D computer, which supplies a signal (Par. 2.4, above) that energizes the shutdown relay when the Discoverer satellite achieves the predetermined velocity gain. Pressure-decay shutdown is initiated by the thrust chamber pressure switch, which closes a circuit to energize the shutdown relay when thrust chamber pressure drops below 372 ± 12 psig as the result of propellant depletion or malfunction. This relay de-energizes the power relay and disconnects the supply of 28 v dc to the gas generator solenoid valve and the solenoid-operated pilot valve permitting these valves to return to their normally-open positions. This in turn allows the actuating pressure to the gas generator valve piston and the main fuel valve piston to decay, and these valves return to their normally-closed positions. Gas generator valve closure shuts off the supply of propellants to the gas generator combustion chamber, the turbine begins to decelerate, and the propellant pump pressures begin to decay. Fuel valve closure shuts off the supply of fuel to the engine thrust chamber. Oxidizer continues to flow for a short time after fuel shutoff, permitting complete combustion of any fuel remaining in the thrust chamber. As oxidizer pump pressure continues to decay, the main oxidizer valve returns to its normally-closed position, shutting off further oxidizer flow to the thrust chamber. When engine shutdown is complete, the oxidizer, fuel, and helium vent valve squibs are actuated by a SS/D timer signal (Par. 2.5, above) to initiate venting of propellant tanks and lines and of pressure spheres and lines. Venting is accomplished through T-type impulse nullifiers to minimize net impulse imparted to the satellite by the venting action.

4.0 POWER INPUT REQUIREMENTS

Voltage	+24.0 to +30.5 v dc (unreg)
Maximum Ripple	0.5 v
Peak Current	20.0 amps
Requirements for 120 Seconds of Operation	500.0 amp-sec at +28.0 v

5.0 ENVIRONMENT

The propulsion system is designed to meet the environmental criteria noted in LMSD Report 6117A.

6.0 RESPONSIBILITY

Propulsion Development Department (SS/B). Spaceframe and Installation Design Department (SS/A).

7.0 NOTES

Not applicable

8.0 PROPULSION SYSTEM INDEX

Item	Para.
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Frangible Disc	3.1.1
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Helium Spheres	3.1.1
	3.1.2

Item	Para.
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1.0 GENERAL

The Propulsion System of the satellite for Discoverer Program Flight Configuration VI develops a nominal 16,000 pounds of thrust in a vacuum, including the thrust derived from the turbine exhaust recovery and turbine speed increase with altitude. In this flight configuration, the engine used, the liquid bi-propellant Bell Aircraft Corporation (BAC), Model 8096, operates in the extended-burn mode for a duration of 240 seconds. Functionally, the thrust provided by this propulsion system is the second-stage thrust which, together with the velocity imparted by the Discoverer booster, enables the Discoverer satellite to achieve the planned altitude and velocity to attain the programmed orbital status.

The Propulsion System consists of the engine subsystem and the propellant subsystem. The engine subsystem, which is basically the engine specified above, includes (a) the thrust chamber assembly, (b) the turbine pump assembly, and (c) the engine mount (BAC-fabricated) and the engine mounting cone (LMSD-fabricated). The propellant subsystem consists of: (a) the propellant tank assembly, (b) the fuel, Unsymmetrical Dimethyl Hydrazine (UDMH), (c) the oxidizer, Inhibited Red Fuming Nitric Acid (IRFNA), (d) the pressurization system, (e) the ullage rockets, and (f) associated regulators, valves, switches, pyrotechnics, plumbing, and wiring.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in the Timed Sequence of Events, Section IV, page 2.6.3 ff, the following signal inputs are required for propulsion system operation.

2.1 A signal from the SS/D timer ignites the ullage control rockets.

2.2 A signal from the SS/D timer ignites the explosive actuators which combust the solid propellant grain starter of the gas generator to initiate turbine operation.

2.3 A signal from the SS/D timer ignites the helium valve squibs to couple the main regulator of the Propellant Tank Pressurization Panel Assembly to the oxidizer tank.

2.4 A signal from the SS/D timer turns off power to both the arm and fire circuits to the gas generator squibs.

2.5 A signal from the SS/D timer turns off power to the helium valve squibs.

2.6 A signal from the SS/D timer deactivates engine cut-off safety switch which has been preventing possible premature engine shutdown by erroneous signal from integrator.

2.7 A signal from the SS/D velocity integrator commands engine shutdown when the satellite has achieved the programmed velocity gain, as corrected by ground control via transponder-decoder channel 6.

2.8 A signal from the SS/D timer ignites the oxidizer and helium vent valve squibs.

2.9 A signal from the SS/D timer ignites the fuel vent valve squib.

3.0 PERFORMANCE

3.1 Component Description - Engine Subsystem. As mentioned above, the engine subsystem consists of the thrust chamber assembly, turbine pump assembly, engine mounts, plus the engine relay box.

3.1.1 Thrust Chamber Assembly. The thrust chamber assembly includes the thrust chamber in which the hypergolic combination of the fuel and oxidizer occurs. Both oxidizer and fuel lines into the thrust chamber are closed off by frangible discs to prevent the premature introduction of oxidizer and/or fuel into the thrust chamber. In a manner to be described below (cf. Par. 3.1.2), the Turbine Pump Assembly delivers oxidizer and fuel to the lines leading into the thrust chamber at pressures and flow rates required to sustain the firing of the rocket engine. Included in the thrust chamber assembly, in addition to the frangible discs, are the injectors through which the fuel is introduced into the chamber, the thrust cone in which there are channels through which the oxidizer passes for regenerative cooling, and the pressure switch through which shutdown of the engine is achieved as the thrust chamber pressure drops off to a predetermined pressure. For this model engine, the thrust chamber is equipped with a contoured nozzle with an area ratio of 45 to 1. Regenerative cooling is achieved only through a portion of the nozzle representing an area ratio of 13:1. The remaining portion of the nozzle, consisting of a titanium extension reinforced with molybdenum, is radiation cooled.

3.1.2 Turbine Pump Assembly. The Turbine Pump Assembly supplies the fuel and oxidizer at flow rates and pressures required to sustain the firing of the rocket engine. In sequence of operation, this assembly includes the two solid propellant starters which, when ignited, provide hot gases which start the gas generator. The gas generator drives a turbine which in turn drives pumps to build up pressure on the fuel and oxidizer lines. The solenoid valve, actuated by the starting signal, closes off a return by-pass line in the fuel line, resulting in this building fuel pressure acting against a spring pressure to open the gas generator valve. Opening of the gas generator valve permits the entrance of both fuel and oxidizer into the gas generator where they mix hypergolically to create the gases which take over driving the turbine and pumps at full operating rpm to furnish the fuel and oxidizer to the thrust chamber. In addition to the above units, this assembly includes a gear box through which the fuel and oxidizer pumps are driven at different rpm from the single gas-driven turbine, fuel and oxidizer venturis which regulate and maintain constant flow rates for their respective propellants to the gas generator, and filters. During operation of the gas generator at altitude, exhausting of the gases accounts for approximately 200 pounds of the total thrust of the engine. A tangential port in the fuel pump case provides a hydraulic power source to drive the hydraulic motor pump package which furnishes the hydraulic pressures to gimbal the engine for satellite directional control during the period of rocket engine firing.

3.1.3 Engine Mounts. The engine mount (BAC-fabricated) includes four legs which are attached at points 90 degrees apart to the engine mounting cone (LMSD-fabricated) at Discoverer Station (DS) 400.76. The legs support the engine gimbal ring at their aft ends. The two hydraulic control system actuating cylinders are attached to the gimbal ring and to the engine housing at the forward end of the thrust chamber so that cylinder action provides gimballing of the engine up to 3.5 degrees in response to differential signals from the electronics system.

3.1.4 Engine Relay Box. The engine relay box, which is mounted on the engine mount, contains a power relay, a thrust chamber relay, a shutdown relay, a squib monitor assembly, and the circuitry necessary to accomplish starting and shutdown sequence functions.

3.2 Component Description - Propellant Subsystem. As mentioned above, the propellant subsystem consists of the propellant tank assembly, the fuel, the oxidizer, the pressurization system, the ullage rockets, and associated hardware.

3.2.1 Propellant Tank Assembly. The propellant tank assembly, which extends from Discoverer Station (DS) 271.0 (approximately) to DS 400.0 (approximately), consists of two integral tanks-

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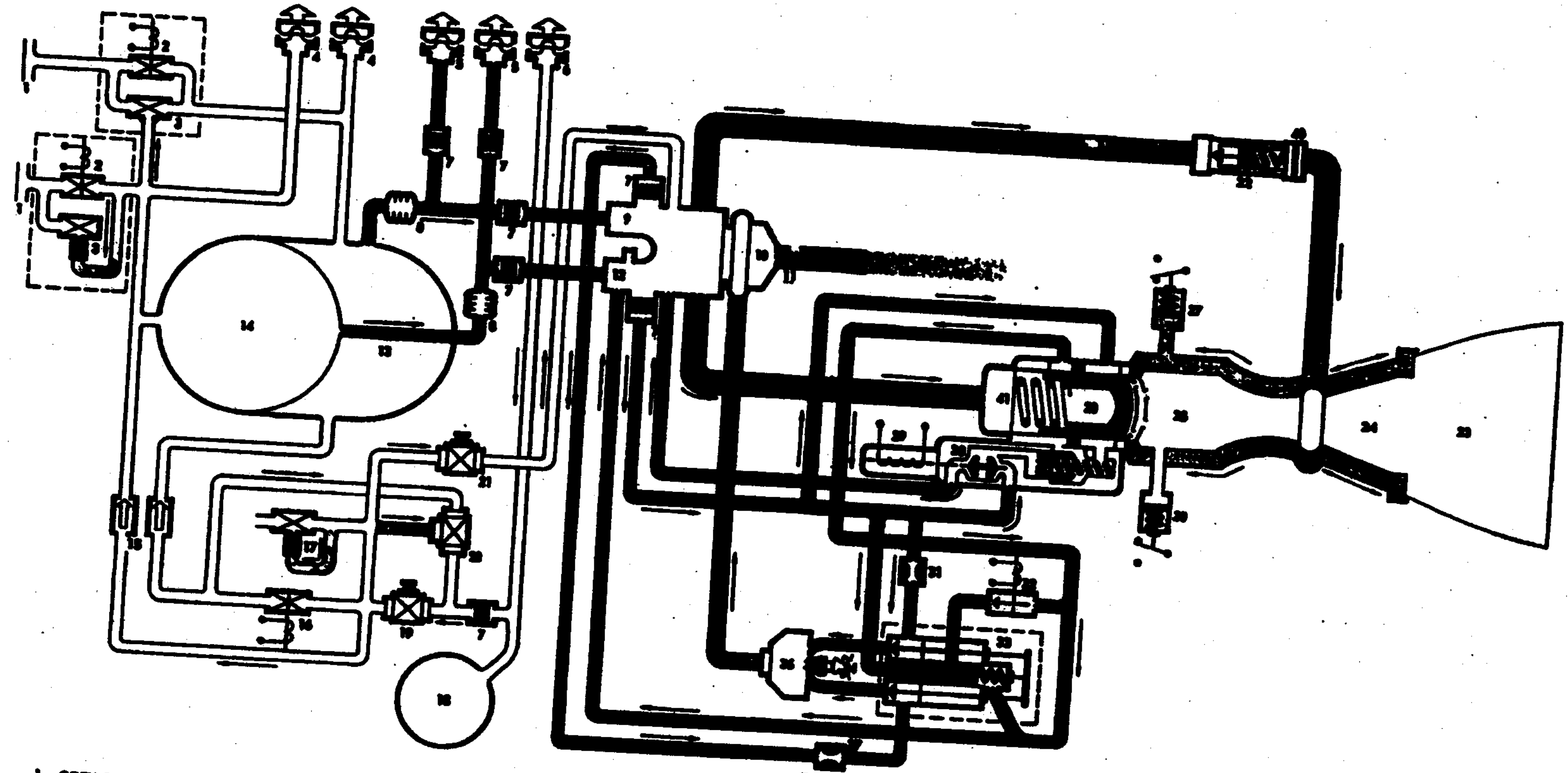

1. OPEN-END T-FITTINGS
2. TANK VENT VALVES
3. TANK PRESSURE-RELIEF VALVES
4. TANK-VENT QUICK-DISCONNECTS
5. TANK LOAD-AND-DUMP QUICK DISCONNECTS
6. HELIUM-FILL QUICK-DISCONNECT
7. FILTERS
8. FLEXIBLE BELLOWS
9. OXIDIZER PUMP
10. TURBINE AND GEAR BOX
11. TURBINE EXHAUST DUCT
12. FUEL PUMP
13. OXIDIZER TANK
14. FUEL TANK
15. HELIUM-LINE CHECK VALVES
16. SQUIB-OPERATED BYPASS VALVE
17. HELIUM-SYSTEM EMERGENCY-RELIEF VALVE
18. HELIUM STORAGE SPHERE
19. HELIUM MAIN-PRESSURE REGULATOR
20. HELIUM SLAVE-PRESSURE REGULATOR
21. OXIDIZER-PUMP-SEAL HELIUM REGULATOR
22. OXIDIZER VALVE
23. TITANIUM NOZZLE EXTENSION
24. THRUST CHAMBER NOZZLE
25. THRUST CHAMBER
26. THRUST CHAMBER COOLING PASSAGES
27. OXIDIZER-MANIFOLD PRESSURE SWITCH
28. FUEL VALVE
29. FUEL CONTROL SOLENOID VALVE
30. THRUST-CHAMBER PRESSURE SWITCH
31. FUEL-FLOW VENTURI
32. GAS-GENERATOR SOLENOID VALVE
33. GAS-GENERATOR BIPROPELLANT VALVE
34. DUAL-SQUIB IGNITER
35. TURBINE STARTER CHARGE
36. GAS-GENERATOR COMBUSTION CHAMBER
37. OXIDIZER-FLOW VENTURI
38. FUEL POPPET
39. ACTUATOR
40. OXIDIZER FRANGIBLE DISC
41. FUEL FRANGIBLE DISC

The above illustration shows the Bell Aircraft Corporation-manufactured rocket engine, Model 8096 (USAF XLR-81-BA-9), as adapted to the Discoverer Flight Configuration VI satellite. This model rocket engine has the dual-start capability, but in this configuration the dual-start feature is eliminated by the removal of the second igniter and start can; the engine is used in the extended burn mode with a nominal burn time of 240 seconds. The turbine-driven propellant pumps, in a "bootstrap" operation, deliver the hypergolic propellants to the thrust chamber.

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the fuel tank and the oxidizer tank. The fuel tank, mounted forward of the oxidizer tank, is practically spherical in shape with a radius of 2.5 feet. The oxidizer tank, basically cylindrical in shape, fits over the fuel tank, tangentially, with the after bulkhead of the fuel tank identical with the forward bulkhead of the oxidizer tank. The cylindrical portion of the oxidizer tank is a stress member of the satellite spaceframe and forms the outer skin of the satellite between DS 294.0 (approximately) and DS 337.0 (approximately). The after end of the oxidizer tank is hemispherical in shape. Both the forward and aft equipment racks attach to the propellant tank assembly. The volumes of the tanks are: fuel - 75.3 cu. ft.; oxidizer - 98.4 cu. ft. The nominal fuel load is approximately 3700 pounds; the nominal oxidizer load approximately 9400 pounds.

3.2.2 **Fuel.** The fuel for this engine is Unsymmetrical Dimethyl Hydrazine (UDMH) which is highly flammable. Its fumes can be ignited by sparks, open flames, or other sources of ignition. In contact with nitric acid UDMH is hypergolic. UDMH should be transferred in closed systems. UDMH will combine with moisture in the air and is subject to oxidization. Air oxidation causes the liquid to change from a clear, colorless condition to a clear yellow color. Extreme caution must be observed by personnel in handling UDMH as it is severely irritating to the skin and respiratory tracts.

3.2.3 **Oxidizer.** The oxidizer is Inhibited Red Fuming Nitric Acid (IRFNA) which is very active and will vigorously attack most metals and organic chemicals. Concentrated nitric acid may cause spontaneous ignition and burn vigorously when in contact with such organic materials as dry grass, sawdust, excelsior, shavings, cotton waste, oil, and greases. To the human being IRFNA is extremely toxic and may be absorbed into the body through skin contact, by inhalation of fumes, and by ingestion.

3.2.4 **Pressurization System.** The Propellant Pressurization System provides helium gas to replace the fuel and oxidizer as the propellants are drawn from their respective tanks by the fuel and oxidizer pumps. The helium gas maintains pressure within the tanks to prevent their collapse which might otherwise occur as the tanks are being evacuated during propellant consumption. The pressurization system consists of a helium sphere, the propellant tank pressurization panel assembly, and associated plumbing. The helium sphere, with a 2200 cu. in. capacity, is located in the forward equipment rack on the satellite centerline at Discoverer Station (DS) 248.80. The propellant tank pressurization panel assembly, also located in the forward equipment rack, regulates and distributes the helium from the sphere into the fuel and oxidizer tanks. The plumbing associated with the pressurization system extends from the helium fill coupling at the aft end of the satellite, through the sphere and pressurization panel assembly, to the propellant tanks.

3.2.5 **Ullage Rockets.** A pair of 17-pound, solid-propellant ullage rockets are mounted, diametrically opposite to one another in relation to the satellite axis, to the aft equipment rack at Discoverer Station (DS) 446.70, in quadrants I and III. Each of these rockets generates a nominal rated thrust of 128 pounds in a vacuum during a nominal burning time of 18.8 seconds. The rockets are so mounted that the axes of the rockets pass through the center of gravity of the satellite. The ullage rockets establish an artificial gravity in the satellite to remove bubbles from the propellants at the inlets to the turbine-driven propellant pumps to preclude a vacuum lock at either or both of the pumps at the beginning of the rocket engine firing cycle.

3.3 **Operation.** In the operation of the Propulsion System there are two distinct areas of concern: starting and shutdown. The starting is further subdivided into engine subsystem and propellant subsystem starting. Shutdown is subdivided into commanded and automatic shutdown.

3.3.1 **Starting Sequence - Engine Subsystem.** Preliminary to actual engine start, the ullage rockets, which are fired by signal from the SS/D timer, create an artificial gravity to eliminate bubbles from the propellant lines and from the inlets to the propellant pumps to preclude vacuum locks in these pumps as the pumping of propellants commences. Actual starting is initiated by the firing of dual explosive igniters which ignite the solid propellant of the turbine starter. Hot gases produced by the combustion of the starter propellant pass through the turbine, bringing turbine speed up to about 50% of rated speed. The oxidizer and fuel pumps, geared to the turbine, build up pressure in the respective feed lines as the turbine speed builds up. When the pressure in the fuel line reaches approximately 250 psig, the resulting fuel pressure buildup in the gas generator valve actuating chamber causes the valve's actuation piston to open the fuel and oxidizer poppets, which permit fuel and oxidizer to flow through the fuel and oxidizer venturis to the gas generator assembly's propellant injection system. Here the propellants combust hypergolically. In a "bootstrap" operation, a portion of the propellants provided to the thrust chamber of the rocket engine by the turbine-driven pumps is diverted to the gas generator to furnish the motive power to drive the turbine and the pumps. The turbine and pumps continue to operate until command shutdown or until propellants are exhausted from the main propellant tanks.

Simultaneously with the rise in fuel pump discharge pressure, the oxidizer pump discharge pressure increases. At approximately 130 psig the oxidizer flows past the oxidizer valve through the oxidizer lines until stopped by the oxidizer frangible disc. This disc ruptures at approximately 180 psig and allows the oxidizer to flow through the regenerative cooling passages in the nozzle of the thrust chamber into the oxidizer injector manifold from whence it is sprayed into the combustion chamber of the thrust chamber. When the pressure within the oxidizer injector manifold reaches 19 ± 2 psig, the oxidizer manifold pressure switch energizes the fuel valve's pilot operated solenoid valve. Closing of this valve, which previously had permitted a circulation of fuel, causes fuel pressure to build up and open the fuel valve to permit fuel to flow through the fuel lines until stopped by the fuel frangible disc. At a pressure of approximately 525 psig, the frangible disc ruptures and permits fuel to enter the injector head to be sprayed into the thrust chamber and on the oxidizer which is already being sprayed into the thrust chamber. Hypergolic combustion occurs and continues until the source of propellants is exhausted or a command shutdown is given.

Pressure in the thrust chamber during operation is 508 psia. As the pressure builds up past 391 psig, the thrust chamber pressure switch actuates to bypass the effect of the oxidizer manifold pressure switch in maintaining fuel flow.

3.3.2 **Starting Sequence - Propellant Subsystem.** The signal from the SS/D timer (as noted in Par. 2.3 above) ignites the helium valve squibs to couple the main regulator of the Propellant Tank Pressurization Panel Assembly into the oxidizer pressurization line. The fuel pressurization line is already connected to the main regulator. As pumping of the propellants from the propellant tanks commences, helium is introduced into each of the propellant tanks to replace the displaced propellants to maintain the operating pressures within the tanks.

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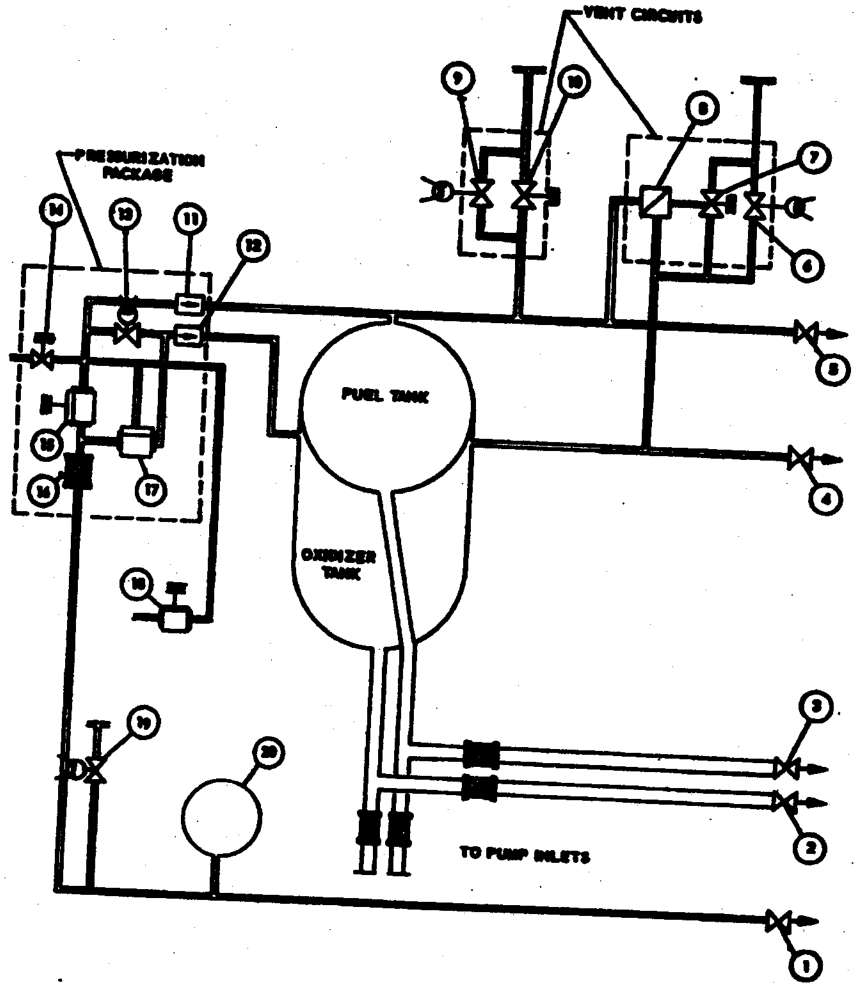

1. HELIUM GAS FILL QUICK DISCONNECT
2. OXIDIZER FILL QUICK DISCONNECT
3. FUEL FILL QUICK DISCONNECT
4. OXIDIZER FILL VENT QUICK DISCONNECT
5. FUEL FILL VENT QUICK DISCONNECT
6. SQUIB-OPERATED OXIDIZER VENT VALVE
7. OXIDIZER RELIEF VALVE
8. TANK PRESSURIZATION EQUALIZATION REGULATOR
9. SQUIB-OPERATED FUEL VENT VALVE
10. FUEL RELIEF VALVE
11. FUEL LINE CHECK VALVE
12. OXIDIZER LINE CHECK VALVE
13. SQUIB-OPERATED BY-PASS VALVE
14. EMERGENCY RELIEF VALVE
15. MAIN HELIUM PRESSURE REGULATOR
16. FILTER
17. BY-PASS SLAVE HELIUM REGULATOR
18. LIP SEAL PRESSURE REGULATOR
19. SQUIB-OPERATED HELIUM VENT VALVE
20. PROPELLANT TANK PRESSURIZATION HELIUM GAS SPHERE

The above illustration shows the Propellant Subsystem of the Propulsion System for the Discoverer Flight Configuration VI satellite. The fuel utilized is UDMH; the oxidizer IRFNA. These two propellants react hypergolicly in contact with one another. The pressurization package incorporates features to prevent these propellants from coming into contact in the propellant pressurization lines due to reverse flow from the propellant tanks.

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3.3.3 **Engine Shutdown - Command.** When the FTV achieves the programmed velocity gain point, the integrator incorporated in the SS/D computer sends a signal to shut off the engine electrical power. This causes the solenoid valve to close, effecting decay of the gas generator valve operating pressure which closes the fuel and oxidizer sections of the valve and shuts down the gas generator. Simultaneously, the loss of power causes the pilot-operated solenoid valve to close with a consequent stoppage in the flow of fuel to the thrust chamber. The oxidizer valve closes under spring load as the oxidizer pump discharge pressure decays, and oxidizer flow to the thrust chamber is stopped.

3.3.4 **Engine Shutdown - Automatic.** Engine shutdown occurs automatically under two conditions, disregarding the ground safety precautions which may be by-passed for the actual launch to minimize the effects of minor malfunctions. The first of these automatic shutdowns occurs at any time during the firing cycle when there is a total loss of electrical power as a casualty rather than as the result of commanded shutdown. The effect is the same, however, as commanded shutdown. The other automatic shutdown occurs when the thrust chamber pressure falls below 372 ± 12 psig, whatever the cause. When this occurs, the thrust chamber pressure switch de-actuates and energizes a relay which removes power from the solenoid valve and the pilot-operated solenoid valve. The sequence of shutdown then becomes the same as for commanded shutdown (cf. Par. 3.3.6 above).

3.4 **Design Criteria**

3.4.1 **Engine Subsystem**

Thrust	
Total (at altitude)	16,000 lb ±1.5%
includes: Turbine exhaust recovery	190 lb
Specific impulse, overall minimum	292.0 lb-sec/lb ±0.70%
Exhaust nozzle expansion ratio	45:1
Nominal burn time	240 seconds (+ 16 or -8 depending on propellant loading)
Gas generator	
Nominal chamber pressure	470 psia
Oxidizer venturi (to gas generator)	
Input pressure	1050 psia
Output pressure	627 psia
Flowrate	.175 lb/sec
Fuel venturi (to gas generator)	
Input pressure	1000 psia
Output pressure	674 psia
Flowrate	1.40 lb/sec
Turbine	
Nominal speed	24,800 rpm
Oxidizer Pump	
Inlet pressure	24.0 psia
Nominal pump speed	14,410 rpm
Nominal output pressure	920 psia
Nominal output flowrate	390 lb/sec
Fuel Pump	
Inlet pressure	24.0 psia
Nominal pump speed	25,389 rpm
Nominal output pressure	750 psia
Nominal output flowrate	13.84 lb/sec
Oxidizer valve	
Nominal opening pressure	130 psig
Oxidizer frangible disc	
Rupture pressure	180 psig
Oxidizer manifold pressure switch (downstream of oxidizer frangible disc)	
Nominal actuation pressure	19 ± 2 psig
Fuel frangible disc (included in fuel valve)	
Rupture pressure	525 psig
Thrust chamber	
Nominal chamber pressure	508 psia
Thrust chamber pressure switch	
Oxidizer manifold pressure switch override pressure	391 psig
Deactuation pressure (propellant exhaustion)	372 ± 12 psig

3.4.2 **Propellant Subsystem**

Propellants	
Oxidizer	Inhibited Red Fuming Nitric Acid (IRFNA) Type III per MIL-N-0072-54D*
Specific gravity	1.570
Fuel	Unsymmetrical Dimethyl Hydrazine (UDMH) per MIL-D-25604B
Specific gravity	0.795
Propellant tank volumes	
Oxidizer	98.4 cu. ft.
Fuel	75.3 cu. ft.
Propellants (nominal load)	
Oxidizer	9400 lbs
Fuel	3700 lbs
Pressurization System	
Gas	Helium U.S. Government Grade A
Volume	2200 cu. in.
Initial pressure	3600 psig
Ullage rockets (2)	
Weight (each)	17 lbs
Nominal rated thrust	128 lbs (in vacuum)
Nominal burning time	18.8 seconds

4.0 POWER INPUT REQUIREMENTS

All components of the Propulsion System using power from the satellite power system operate on d-c power as follows:

Range	19.5 to 30.5 volts dc unregulated, 0.5 volt maximum ripple
Peak demand	20.0 amps
Power requirements, duration of system operation (maximum 246 seconds)	1130 amp-sec at 28 volts

5.0 ENVIRONMENT

The Propulsion System is designed to meet the applicable environmental criteria noted in LMSD Report No. 6117A.

6.0 RESPONSIBILITY

Propulsion Development Department (SS/B)

7.0 Notes:

Not applicable.

* except that solid content shall not exceed 0.05%

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APPROVED BY	PROPULSION SYSTEM Flight Configuration VI DISCOVERER PROGRAM	APPROVED BY

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8.0 **PROPULSION SYSTEM INDEX**

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1.0 GENERAL

The electrical power system for the Discoverer Program Flight Configuration (FC) III satellites consists of the equipments noted in Par. 3.0 below, along with associated connectors, plugs, terminal strips, attachments, and wiring.

Functionally, the system is the main electrical power source for the satellite systems, for special instrumentation, and, until ejection, for the payload (Biomedical Recovery Capsule (BRC), Mark II). The only satellite system item not thus powered is the flight termination system.

2.0 SIGNAL INPUT REQUIREMENTS

Not applicable.

3.0 PERFORMANCE

3.1 Battery Installation. The basic power source for the electrical power system consists of primary batteries mounted on the forward equipment rack. Flight Configuration III employs 1 Type IA primary battery and 1 Type IV primary battery. The Type IA battery consists of 16 series-connected silver peroxide - zinc cells with porous caps for dissipation of gas without loss of electrolyte, packaged in a sealed magnesium container equipped with a relief valve. The Type IV battery consists of 18 series-connected silver peroxide - zinc cells with porous caps, packaged in a sealed stainless steel container with relief valve. Unregulated 28v dc battery power is supplied to the 2000 cps and 400 cps inverters, the +28v dc voltage regulators, and the electronics system throughout the flight; to the computer, the inertial reference package (IRP), the antenna switch, the FM/FM telemeter, the propulsion system, the hydraulic control system, the explosive bolts and pins, and the retrograde rockets during the ascent to orbit; and to the FM/FM telemeter during the operating subcycles. As applicable, unregulated 28v dc battery power is also supplied to the BRC throughout the flight until ejection, and to the Geophysics Research Directorate (GRD) equipment and the tape recorder system throughout the orbit phase.

Design Criteria

Battery	Type IA, Primary Reserve
Cells	Silver peroxide-zinc, 16 series-connected
Voltage (nominal)	24.6 volts dc
Discharge Rate	3.2 amps/100 hours
Voltage Regulation	22.0 to 29.25 volts at 100 hour rate
Capacity	320 ampere-hours
Energy-Weight Factor	87 watt-hours/lb
Battery	Type IV, Primary Reserve
Cells	Silver peroxide-zinc, 18 series-connected
Voltage (nominal)	28.0 volts dc
Discharge Rate	0.1 amp/100 hours
Voltage Regulation	22.0 to 29.25 volts at 100 hour rate
Capacity	10 ampere-hours
Energy Weight Factor	18 watt-hours/lb

3.2 Power Distribution Subsystem. The power distribution subsystem consists of inverters, power amplifiers, voltage regulators, and load limiters which are used to control, boost, and convert the input battery power to necessary output power for satellite system and payload equipment operations.

3.2.1 Inverter, 115 Volt AC 1 ∅ 2000 CPS. The 115 volt ac single phase 2000 cps static electronic Type IV A inverter accepts input battery power and converts it to 115 ±5.0% volts rms, 2000 ±1.0% cps. It supplies this output to the +28 volt dc regulator, the -28 volt dc power supply, and to those equipments noted in the Electrical Power System schematic drawing, Page 3.3.2. The inverter is a transistor-oscillator power amplifier which delivers a maximum of 250 watts (0.8 power factor lagging). A 2000 cps saturable reactor-type load limiter is used to limit the output current of the inverter to its maximum rating. The conversion from d-c power to a-c power is accomplished by push-pull switching circuits, controlled by a square-wave input signal derived from a free-running 2000-cycle oscillator. After the square wave is amplified, it is used to drive the output push-pull power stage. The final power stage output is transformed and filtered by tuned series and parallel filters to produce a sine-wave output with a maximum of 10 percent distortion. The output voltage regulation of 115 (±5 %) volts rms is accomplished by sensing the output voltage, comparing it to a standard voltage, and controlling the voltage to the final driver and power stages by the use of a series boost circuit.

Design Criteria

Inverter, 115V AC 1 ∅ 2000 CPS	Static Electronic, Type IV A
Input	
Voltage (nominal)	25 dc
Voltage Range	22.0 to 29.25 dc
Output	
Voltage and Regulation	115 ±5.0% volts rms
Frequency and Regulation	2000 ±1.0% cps
Power Level	250 watts
Power Load Factor	0.8 lagging (minimum)
Power Conversion Efficiency	70% at 75 watts 80% at 250 watts
Sine Wave Distortion	10% 2nd to 40th Harmonic

3.2.2 Inverter, 115 Volt AC 3 ∅ 400 CPS. The 115 volt ac three-phase, 400 cps static electronic Type I A inverter accepts input battery power varying from 22.0 to 29.25 volts dc, converts it to 115 volts rms, 3 ∅ grounded delta-connected, 400 cps, and supplies a synchronizing voltage to the power amplifier and outputs to the equipments noted on the Electrical Power System schematic drawing, Page 3.3.2. The Type IA inverter consists of a quartz-crystal-controlled oscillator, transistorized amplifiers, and a power converter. Frequency is controlled by the 19.2-kilocycle oscillator. The sine-wave output is transformed through a squaring circuit, differentiating network, and inversion circuit. Countdown and phase shift are accomplished by a magnetic-core ring-counter which, in connection with associated binary flip-flop circuits, performs the required countdown to 400 cycles and establishes an accurate 120-degree phase separation. The ring-counter/phase-shift circuit drives three power amplifiers through transistor push-pull driver circuits. Waveform correction is accomplished by harmonic-series tuned filters in the output transformer secondary. Voltage is controlled by a silicon-Zener rectifier reference bridge, operating in conjunction with a pulse-width-modulated circuit to each of the push-pull driver stages of the power amplifiers.

[redacted]

Design Criteria

Inverter	Static electronic, Type IA
Input	
Voltage (nominal)	25 volts dc
Voltage Range	22.0 to 29.25 volts dc

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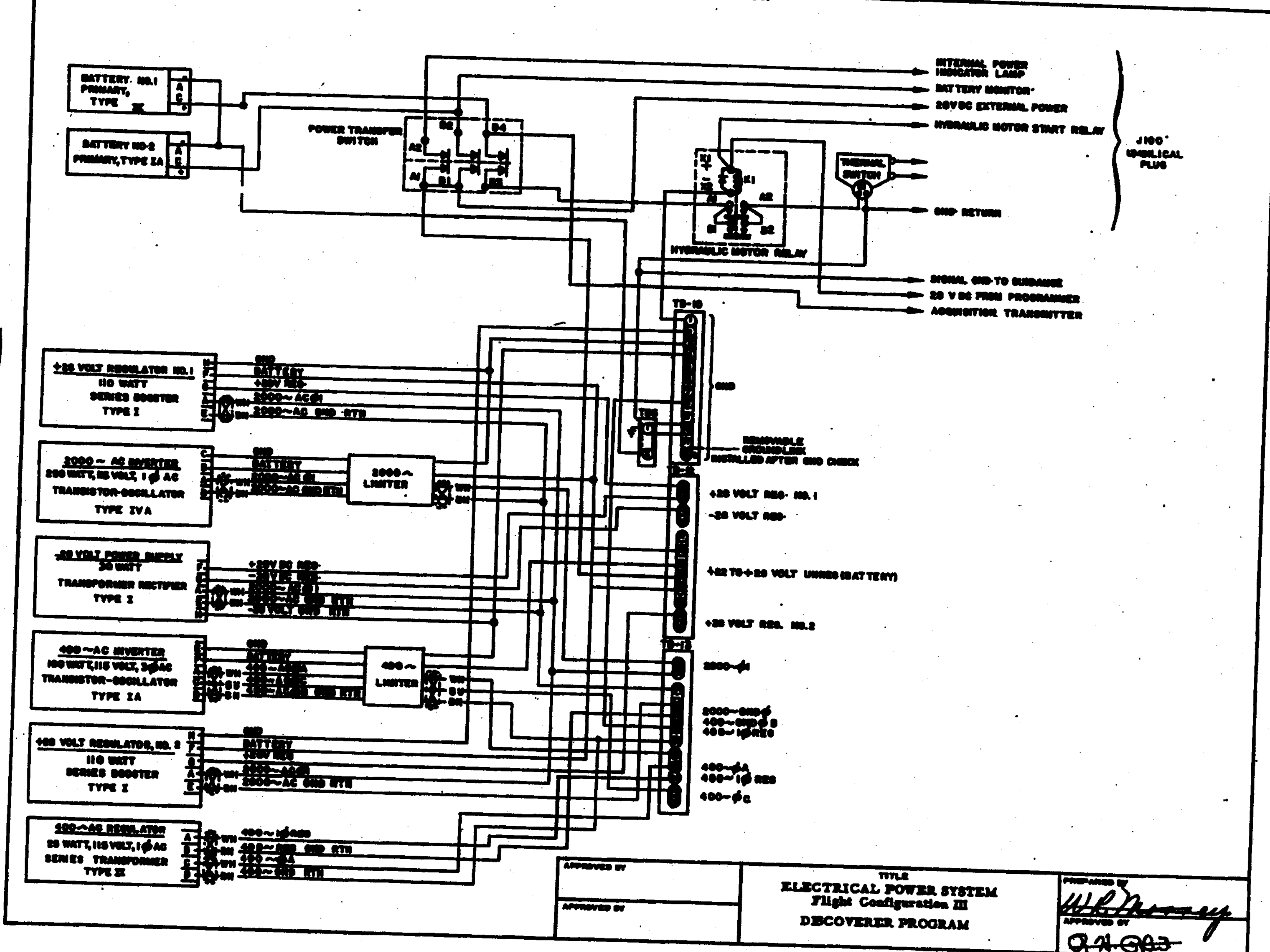
BATTERY NO.1 PRIMARY, TYPE IX
BATTERY NO-2 PRIMARY, TYPE IA
POWER TRANSFER SWITCH
INTERNAL POWER INDICATOR LAMP
BATTERY MONITOR
28V DC EXTERNAL POWER
HYDRAULIC MOTOR START RELAY
J100 UMBILICAL PLUG
THERMAL SWITCH
HYDRAULIC MOTOR RELAY
GND RETURN
SIGNAL GND TO GUIDANCE
28 V DC FROM PROGRAMMER
ACQUISITION TRANSMITTER
+28 VOLT REGULATOR NO.1 110 WATT SERIES BOOSTER TYPE I
2000~ AC INVERTER 250 WATT, 115 VOLT, 1 ϕ AC TRANSISTOR-OSCILLATOR TYPE IVA
2000~ LIMITER
-28 VOLT POWER SUPPLY 30 WATT TRANSFORMER RECTIFIER TYPE I
400~AC INVERTER 100 WATT, 115 VOLT, 3ϕ AC TRANSISTOR-OSCILLATOR TYPE IA
400~ LIMITER
+28 VOLT REGULATOR, NO. 2 110 WATT SERIES BOOSTER TYPE I
400~AC REGULATOR 25 WATT, 115 VOLT, 1ϕ AC SERIES TRANSFORMER TYPE IX
GND
REMOVABLE GROUNDLINK INSTALLED AFTER GND CHECK
+28 VOLT REG. NO. 1
-28 VOLT REG.
+22 TO +28 VOLT UNREG (BATTERY)
+28 VOLT REG. NO.2
2000~ϕA
400~ϕA
400~ϕC
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Output	
Voltage and Regulation	115 ±1.0% at 30 to 100 watts
Frequency and Regulation	400 ±0.02% at 30 to 100 watts
Power	100 watts
Load Power Factor	0.8 lagging
Power Conversion Efficiency	70% at 30 to 100 watts; 65% at less than 30 watts
Sine Wave Distortion	5% 2nd to 40th Harmonic

3.2.3 <u>Voltage Regulators, +28 V DC.</u> The +28 v d-c voltage regulator is a magnetic amplifier controlled, full wave, silicon diode transformer-rectifier operating in series with the battery circuit as a voltage booster. Two such regulators, both installed on the forward equipment rack, add power from the 115 v a-c, 1ø, 2000 cps inverter system to the unregulated battery voltage to assure regulated +28.3 v d-c power input to vehicle equipments. Regulator 1 supplies power to the electronics system, the horizon scanner, the IRP, the identification transmitter, and the orbital timer throughout the flight; to the computer and the BRC during the ascent to orbit; to the GRD equipment during the orbit phase; to the tape recorder system during the operating subcycles; and to the BRC during the operating subcycles until ejection. Regulator 2 supplies power to the transponder during the ascent to orbit and the operating subcycles and also provides reference voltage for the -28 v d-c power supply.

<u>Design Criteria</u>

Voltage Regulator	Transformer-Rectifier, Type I Series Booster
Voltage	+28.3 d.c.
Voltage Regulation	±2.0%
Ripple	110 mv, peak-to-peak
Power (maximum)	160 w
Power Conversion Efficiency (max.)	90%
Dimensions	5.0 x 4.0 x 2.25 in.

3.2.4 <u>Power Supply, -28 V DC.</u> The -28 v d-c power supply installed on the forward equipment rack, is a magnetic amplifier controlled, full wave, silicon diode transformer-rectifier that converts power from the 115 v a-c, 1ø, 2000 cps inverter system to provide a source of -28.3 v d-c power for vehicle equipments. Regulation is provided by a magnetic circuit cross-referenced to one of the +28.3 v d-c voltage regulator circuits. The unit supplies power to the electronics system and the IRP throughout the flight and to the computer during the ascent to orbit. In addition, it supplies power to the GRD equipment.

<u>Design Criteria</u>

-28v Power Supply	Transformer-Rectifier, Type I
Voltage	-28.3 d.c.
Voltage Regulation	±2.0%
Imbalance with +28 v d.c.	±1.0%
Ripple	30 mv, peak-to-peak
Power (maximum)	30 w
Power Conversion Efficiency (max.)	70%
Dimensions	5.0 x 2.0 x 2.0 in.

3.2.5 <u>Voltage Regulator, AC.</u> The 115 v a-c, 1ø, 400 cps voltage regulator is a magnetic amplifier controlled regulator that operates from the AB phase of the 115 v a-c, 3ø, 400 cps inverter system to provide 115 v a-c, 1ø, 400 cps power regulated to ±0.1%. Reference voltage is obtained through a zener diode rectifier bridge. The regulator is installed on the forward equipment rack. It supplies power to the computer and the IRP during the ascent to orbit phase.

<u>Design Criteria</u>

AC Voltage Regulator	Transformer-Rectifier, Type II
Voltage	115 a.c, 1ø
Voltage Input	±2.0% regulation
Voltage Output	±0.1% regulation
Frequency	400 cps
Frequency Regulation	±0.02%
Sine Wave Distortion	1.5%, 2nd to 40th harmonic
Power (maximum)	25 w
Power Conversion Efficiency (maximum)	70%
Dimensions	5.25 x 4.0 x 3.0 in.

4.0 POWER INPUT REQUIREMENTS

Not applicable.

5.0 ENVIRONMENT

The Electrical Power System is designed to meet the applicable requirements of LMSD Report No. 6117A General Environmental Specification for Discoverer, MIDAS, and Samos Programs.

6.0 RESPONSIBILITY

Vehicle Internal Electrical Development (SS/C) Department.

7.0 NOTES

Not applicable

8.0 ELECTRICAL SYSTEM INDEX

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APPROVED BY	ELECTRICAL POWER SYSTEM Flight Configuration III DISCOVERER PROGRAM	APPROVED BY

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1.0 **GENERAL**

The Electrical Power System of the Discoverer Program Flight Configuration VI satellite consists of the battery installation and the power distribution subsystem and of associated terminal strips, connectors, buses, attachments, wiring, and cabling. Functionally, the System is the basic power source for the satellite system and payload equipments from turn-on during the last tasks of countdown through useful orbital life.

2.0 **SIGNAL INPUT REQUIREMENTS**

At the times noted in the Timed Sequence of Events for the affected flight test satellites of this Flight Configuration, the signal inputs noted below are required for System operation.

2.1 A Discoverer blockhouse signal switches power from the ground source to the electrical power system during the last tasks of countdown; from that time the satellite system and payload equipment are powered from the satellite source.

3.0 **PERFORMANCE**

3.1 Battery Installation. The power source of the electrical power system is the connected-in-parallel battery system, consisting of three primary batteries, which supplies certain satellite system equipments directly, the remaining equipments being supplied through the power distribution subsystem. The negative terminals of the battery installation are connected to a single ground point which is common to all electrical power system equipments. Electrical power is supplied to satellite system and payload equipments from a source controlled by the blockhouse until the last minutes of countdown, at which time a blockhouse command actuates the main power transfer switch to transfer electrical power supply to the satellite electrical power system. Each battery housing has an automatically-closing pressure-relief valve for the escape of battery gas without loss of electrolyte. The battery installation has a flexible capability which permits adjustment for lifetime and weight. Adjustment is accomplished by adding or removing batteries.

3.1.1 Battery Installation - AET-L. The battery installation for the Discoverer satellites with the AET Light Payload consists of three Type I-A primary batteries and two Type VI primary batteries. The minimum capability of the Type I-A batteries is 8500 watt-hours, the average 9200 watt-hours; the minimum of the Type VI is 1700, the average 1800; and the total minimum capability is 28,900, the total average 31,200 watt-hours. The calculated requirement for 100 hours of useful orbital life is 24,790.

3.1.2 Battery Installation - AET-H. The battery installation for the Discoverer satellites with the AET Heavy Payload consists of three Type I-A primary batteries. The minimum capability of the Type I-A batteries is 8500 watt-hours, the average 9200 watt-hours. The total power supplied by the three batteries is 25,500 watt-hours minimum, with an average of 27,600. The calculated requirement for 100 hours of useful orbital life is 25,131 watt-hours.

Design Criteria

Battery	Type IA, Primary Reserve
Cells	Silver Peroxide-Zinc, 16 series-connected
Voltage (nominal)	24.6 volts dc
Discharge Rate	3.2 amps/100 hours
Voltage Regulation	22.0 to 29.25 volts at 100 hour rate
Capacity	320 ampere-hours
Energy-Weight Factor	87 watt-hours/lb
Battery	Type VI, Primary Reserve
Cells	Silver peroxide-zinc, 17 series-connected
Voltage (nominal)	26.5 volts dc
Discharge Rate	1.75 amps/70 hours
Voltage Regulation	22.0 to 29.25 volts at 40 hour rate
Capacity	70 ampere hours
Energy-Weight Factor	70 watt-hours/lb

3.2 Power Distribution Subsystem. The power distribution subsystem consists of inverters, power amplifiers, voltage regulators, and load limiters which are used to control, boost, and convert the input battery power to necessary output power for satellite system and payload equipment operations.

3.2.1 Inverter, 115 Volt AC 1 Ø 2000 CPS. The 115 volt ac single phase 2000 cps static electronic Type IV inverter accepts input battery power and converts it to 115 ±5.0% volts rms, 2000 ±1.0% cps. It supplies this output to the +28 volt dc regulator, the -28 volt dc power supply, and to those equipments noted in the Electrical Power System schematic drawing, Page 3.9.1. The inverter is a transistor-oscillator power amplifier which delivers a maximum of 250 watts (0.8 power factor lagging). A 2000 cps saturable reactor-type load limiter is used to limit the output current of the inverter to its maximum rating. The conversion from d-c power to a-c power is accomplished by push-pull switching circuits, controlled by a square-wave input signal derived from a free-running 2000-cycle oscillator. After the square wave is amplified, it is used to drive the output push-pull power stage. The final power stage output is transformed and filtered by tuned series and parallel filters to produce a sin-wave output with a maximum of 10 percent distortion. The output voltage regulation of 115 (±5%) volts rms is accomplished by sensing the output voltage, comparing it to a standard voltage, and controlling the voltage to the final driver and power stages by the use of a series boost circuit.

*AET-L only

Design Criteria

Inverter, 115V AC 1 Ø 2000 CPS	Static Electronic, Type IV A
Input	
Voltage (nominal)	25 dc
Voltage Range	22.0 to 29.25 dc
Output	
Voltage and Regulation	115 ±5.0% volts rms
Frequency and Regulation	2000 ±1.0% cps
Power Level	250 watts
Power Load Factor	0.8 lagging (minimum)
Power Conversion Efficiency	70% at 75 watts 80% at 250 watts
Sine Wave Distortion	10% 2nd to 40th Harmonic

3.2.2 Inverter, 115 Volt AC 3 Ø 400 CPS. The 115 volt ac three-phase, 400 cps static electronic Type I A inverter accepts input battery power varying from 22.0 to 29.25 volts dc, converts it to 115 volts rms, 3 Ø grounded delta-connected, 400 cps, and supplies a synchronizing voltage to the power amplifier and outputs to the equipments noted on the Electrical Power System schematic drawing, Page 3.9.1. The Type 1A inverter consists of a quartz-crystal-controlled oscillator, transistorized amplifiers, and a power converter. Frequency is controlled by the 19.2-kilocycle oscillator. The sine-wave output is transformed through a squaring circuit, differentiating network, and inversion circuit. Countdown and phase shift are accomplished by a magnetic-core ring-counter which, in connection with associated binary flip-flop circuits, performs the required countdown to 400 cycles and establishes an accurate 120-degree phase separation. The ring-counter/phase-shift circuit drives three power amplifiers through transistor push-pull driver circuits. Waveform correction is accomplished by harmonic-series tuned filters in the output transformer secondary.

Voltage is controlled by a silicon-Zener rectifier reference bridge, operating in conjunction with a pulse-width-modulated circuit to each of the push-pull driver stages of the power amplifiers.

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This page is reserved for the Electrical Power System schematic for Flight Configuration VI.

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Design Criteria

Inverter	Static electronic, Type I A
Input	
Voltage (nominal)	25 volts dc
Voltage Range	22.0 to 29.25 volts dc
Output	
Voltage and Regulation	115 ±1.0% at 30 to 100 watts
Frequency and Regulation	400 ±0.02% at 30 to 100 watts
Power	100 watts
Load Power Factor	0.8 lagging
Power Conversion Efficiency	70% at 30 to 100 watts; 65% at less than 30 watts
Sine Wave Distortion	5% 2nd to 40th Harmonic

3.2.3 Power Amplifier, 115 V AC 1 ∅ 400 CPS. The 115 volt ac single phase, 400 cps power amplifier Type I accepts the synchronizing signal from the 3 phase 400 cps inverter and a dc voltage from the battery installation, varying from 22.0 to 29.25 volts, and supplies outputs to the equipments noted on Page 3.9.1. The amount of power drawn from the three-phase supply is one watt. The main power output is supplied by the unregulated dc voltage through a push-pull switching circuit. This square-wave power is transformed and then shaped through an output-filter circuit to an approximate sine wave with no more than 5 percent distortion up to the 40th harmonic. The output voltage is maintained at 115 (±1 %) volts rms by means of a Zener-diode, referenced-bridge, sensing circuit and a series, pulse-width-modulated, voltage-control circuit. The output voltage is transformer-sensed; and if the current exceeds the rating of the power amplifier, a current-limiting action takes place to prevent unit damage.

The type IA also contains a circuit which will continue operation of the amplifier if the ∅A synchronizing signal from the 400 cycle 3∅ inverter is lost.

Design Criteria

Power Amplifier	Static Electronic, Type I
Input	
Voltage (nominal)	25 volts dc
Voltage Range	22.0 to 29.25 volts dc
Output	
Voltage and Regulation	115 ±1.0% volts rms at 30 watt load and 22.0 to 29.25 volts dc input; 115 ±1.0% volts rms at 100 watt load and 24.0 to 29.25 volts dc input; 115 + 1.0/-2.0% volts rms at 100 watt load and 22.0 to 24.0 volts dc input
Frequency and Regulation	400 ±0.02% at 30 to 100 watts load
Power Conversion Efficiency	75% at 100 watts output; 65% at 30 watts output

3.2.4 Voltage Regulator, +28 V DC. The +28 volt dc voltage regulator incorporates a voltage boost circuit which adds a controlled amount of rectified ac voltage to the 22.0 to 29.25 volt dc battery input to provide constant +28.3 volts dc output to affected satellite system and payload equipments. The boost supply is derived from a 2000-cycle, 115-volt source through a transformer rectifier circuit. The full-wave rectifier consists of two silicon-controlled rectifiers which control the amount of boost voltage required to maintain a regulated output. These silicon-controlled rectifiers, in turn, are controlled by magnetic amplifier output. Error sense current, derived from a sense Zener bridge circuit, controls the magnetic amplifier. Whenever the dc supply voltage is below the value necessary to provide the regulated output voltage, the silicon-controlled rectifiers add a rectified average value of ac voltage to the dc supply voltage. This voltage is filtered to smooth out the ripple voltage and then supplied to a transistorized series regulator. Control power for the series regulator is provided by the bias power supply. Series regulator control is accomplished by Zener diode reference as compared to the output voltage. The function of the series regulator is to provide refined control for the regulated output voltage and to achieve an extremely low-output impedance for blocking input transients. In the event that the input voltage magnitude is higher than that of the regulated output voltage, the series regulator reflects additional impedance to the supplied voltage so that the required output voltage level is obtained. Before reaching the output, the regulated voltage is again filtered to reduce the ripple voltage below 30 millivolts, peak-to-peak. A radio interference filter is provided to reduce rf noise.

Design Criteria

Voltage Regulator	Transformer rectifier, series booster, Type III
Input	
Voltage	+22 to +29.25 dc; 115 ±5.0% dc 1 ∅; 2000 ±1.0% cps
Output	
Voltage and Regulation Ripple	28.3 ±1%v dc; 30 millivolts peak-to-peak
Power	350 watts continuous
Power Conversion Efficiency	90% minimum

3.2.5 Power Supply -28V DC. The input voltage from the 115 volt ac single phase 2000 cps inverter subsystem is converted by the full-wave, transformer-rectifier to provide a regulated -28.3v dc output. Regulation is sensed by a transistorized series-regulator circuit with a cross reference carried from the 28-volt regulator. Overall voltage regulation tolerance is 28±0.5 volts. Since only one percent voltage variation is permitted between the negative and positive supplies, a cross-reference voltage-regulation sensing component is required.

Design Criteria

-28v Power Supply	Transformer-Rectifier Type II
Input	
Voltage and Regulation	115 ±5.0%v ac, 1 ∅
Voltage and Regulation	28.3 ± 0.56
Frequency and Regulation	2000 ±1.0% cps
Output	
Voltage and Regulation	-28.3 ±2.0% dc
Output Power	100 watts
Power Conversion Efficiency	80%
Ripple	30 mv peak-to-peak

4.0 POWER INPUT REQUIREMENTS

Not Applicable

5.0 ENVIRONMENT

The Electrical Power System is designed to meet the applicable requirements of LMSD Report No. 6117A General Environment Specification for Discoverer, MIDAS, and Samos Programs.

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6.0 RESPONSIBILITY

Vehicle Internal Electrical Development (SS/C) Department.

7.0 NOTES

Not Applicable

8.0 ELECTRICAL SYSTEM INDEX

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1.0 GENERAL

The guidance and control system for the Discoverer Flight Configuration (FC) III satellite consists of:

a. The guidance system, including the horizon scanner, the inertial reference package (IRP), the computer (including the integrator and the SS/D timer), the guidance (secondary) junction box, and an ac voltage regulator;

b. The flight control system, including the flight control electronics, the pneumatic control system, and the hydraulic control system.

Functionally, the guidance and control system establishes attitude references for the satellite and aligns the satellite to these attitude references during the coast, orbital boost, reorientation, and orbit phases of operation. It also orients the satellite to the attitude required for recoverable capsule ejection. In addition, the system initiates, modifies, and/or terminates programmed satellite equipment operation, including ignition and hard shutdown of the orbital boost rocket engine, during these phases.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in SSDB-DP Section IV, pp 2.0 ff as applicable, the signal inputs noted below are required for operation of the guidance and control system.

2.1 Disconnect of the main and auxiliary umbilicals at launch releases the SS/D timer brake, allowing the timer to start.

2.2 A ground operations hold imposed on the SS/D timer brake by means of a signal received via transponder-decoder channel 5 is used to delay the firing of the ullage control rocket squibs, the first event in the activation of the propulsion system.

2.3 A programmed hold (nominally 20 seconds) from the orbital timer backs up the ground operations hold noted in Paragraph 2.2.

2.4 A ground operations signal via transponder-decoder channel 6 releases the ground operations hold on the SS/D timer (and, if applicable, the orbital timer hold) and transmits minus velocity-gain corrections to the guidance integrator.

2.5 The orbital timer turns off the horizon scanner and restarts the SS/D timer to initiate the ejection of the payload capsule.

3.0 PERFORMANCE

3.1 Guidance System

The guidance system establishes and maintains the various satellite attitudes programmed for the particular flight (subsequent to the Thor main and vernier engine shutdown) and supplies signals to the flight control system as required for satellite orientation from satellite vehicle separation in the ascent-to-orbit phase through capsule ejection in the orbit phase. The guidance system also includes a programming device which triggers satellite equipment operations during the ascent to orbit and the ejection sequence.

3.1.1 Horizon Scanner. The horizon scanner is an optical scanning system which is sensitive to infrared radiation and which operates both day and night. The scanner employs an optical collecting system, a thermistor bolometer, and a phase reference generator. The scanning axis is oriented along the satellite's yaw axis. The scanner system establishes the local vertical by sensing the difference apparent at the horizon between the earth's thermal radiation and that of contiguous space. It then measures the angular difference between this local vertical and the satellite's yaw axis. This measure of difference modulates a 400 cps phase-reversing voltage to provide proportional signals of pitch and roll differences. The phased and proportional signals are fed to the IRP pitch and roll gyros to correct deviations from their respective reference attitudes. After satellite reorientation, the horizon scanner roll signal is also fed to the IRP yaw gyro for correction of possible yaw errors during orbit. This action maintains the longitudinal axis in an attitude parallel to the flight path. The pitch and roll correction signals are used to maintain the satellite's pitch and roll axis in a plane perpendicular to the local vertical.

Performance Characteristics

Horizon Scanner

Infrared Detector Range	2 to 20 microns
Sweeping Signal Scale Factor:	
FC IV	0.772 vrms/°
FC III and V	0.386 vrms/°
Output Signal Voltage (maximum)	2.6 vrms
Time Constant (maximum)	4.0 ± 2.0 sec

3.1.2 Inertial Reference Package (IRP). The inertial reference package consists of three single-degree-of-freedom, floated, integrating gyros which are sensitive to pitch, yaw, and roll attitude errors respectively; one single-degree-of-freedom, servo-restrained, floated pendulous accelerometer; and associated pre-amplifiers, power amplifiers, and temperature regulating circuitry.

3.1.2.1 IRP Operation. The input axes of the IRP pitch, yaw, and roll gyros are oriented along the satellite's pitch, yaw, and roll axes, respectively, so that movement of the satellite about any of these axes causes rotation of the affected gyros about their respective output axes, producing signals proportionate to the displacement of the satellite from the affected axes. These gyro voltage signals are amplified in the preamplifiers and fed to the flight control system as attitude error signals. The applicable control system, pneumatic and/or hydraulic, responds with corrections to maintain the proper satellite attitude and gyro reference position. Programmed changes in satellite attitude are initiated via the SS/D timer and occur: (a) after satellite vehicle separation— -21.8 degrees in pitch to achieve a horizontal satellite attitude in preparation for orbital boost rocket ignition; (b) after orbital boost rocket shutdown— +180 degrees in yaw to place the satellite in a tail-first horizontal attitude for orbit operations; and (c) just before recoverable capsule ejection— -60 degrees in pitch to establish the required satellite attitude for the ejection events. To accomplish a programmed attitude change, the SS/D timer feeds to the affected gyro a torquing signal that causes it to precess about the axial position it occupied before receipt of the signal. Displacement of the gyro from its null signal position causes a proportional error signal to be fed to the flight control system, which responds with corrections that establish a new vehicle attitude and, when the SS/D timer signal is removed, a new gyro reference position. The magnitude of the SS/D timer signal determines the rate of change, and its duration determines the amount of change. In order to maintain a horizontal satellite vehicle attitude with respect to the earth during orbit, a residual pitch rate proportional to the satellite's geocentric angular rate on orbit is initiated by the SS/D timer upon completion of the satellite's reorientation maneuver. This is accomplished by establishing a gyro null signal position that corresponds to a satellite pitch rate of -3.55 degrees per minute. This pitch rate is constant throughout the orbit phase except during the satellite attitude change just before recoverable capsule ejection.

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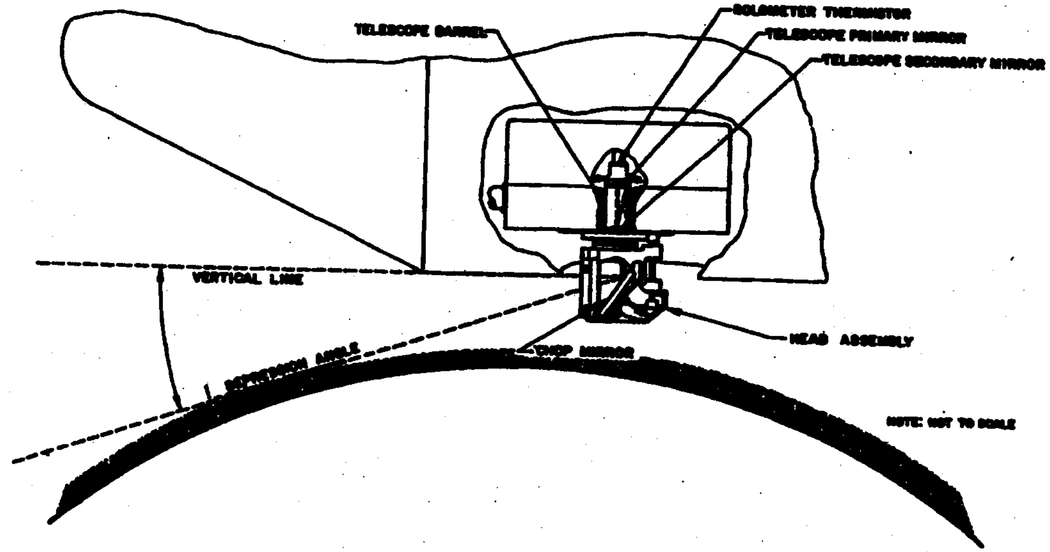

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Performance Characteristics:

IRP Pitch, Yaw, and Roll Gyros

Output Axis Angular Freedom (minimum)	±6°
Input Axis Angular Freedom (minimum)	±18°
Maximum Command Rate Angular Change	50°/min
Torque Generator Linearity at Signal Generator Null	±0.5%
Torque Generator Reproducibility	±0.1%
Maximum Drift Uncertainty (Random Torque)	3°/hr
Operating Temperature	+165°F

3.1.2.2 Accelerometers. The longitudinal (roll axis) accelerometer is oriented with its input axis parallel to the vehicle roll axis during orbital boost, supplying the integrator with a signal proportionate to the satellite's acceleration.

Performance Characteristics:

Accelerometers

Maximum Range	±15 g
Natural Frequency	40 cps
Linearity	0.1%
Operating Threshold (Minimum input)	0.03 ft/sec/sec/ (1 milli-g)
Uncertainty	0.06 ft/sec/sec (2 milli-g)
Operating Temperature	+165°F

3.1.3 Computer. The guidance computer consists of: (a) a programming device, known as the SS/D timer; (b) the guidance integrator; (c) resistive voltage networks; (d) the guidance junction box; and (e) associated circuitry and hardware.

3.1.3.1 SS/D Timer Operation. The SS/D timer controls the event sequence of the principal satellite equipment functions during the coast, orbital boost, and reorientation phases and during recoverable capsule ejection. The SS/D timer operates continuously from launch through reorientation, after which it turns itself off. It is reactivated for the capsule ejection event sequence by the orbital timer, and it turns itself off again when ejection has been accomplished. Just prior to orbital boost, a hold is imposed on the timer brake control by ground operations via transponder-decoder Channel 5. The purpose of this hold is to assure that orbital boost rocket engine ignition occurs at the optimum time. As a back-up to this hold, the orbital timer imposes a programmed hold on the SS/D timer. The duration of the SS/D timer hold is computed on the basis of range radar measurements of the Thor boost trajectory, and the hold is terminated by the transmission of minus velocity gain corrections to the integrator via transponder-decoder Channel 6 (Ref. Par. 3.2.3.2, below). The SS/D timer consists of 24 digital counters (Veeder Root), 24 cams, 72 switches, and a 115v ac, 3ϕ, 400 cps synchronous motor. Performance of the timer programming function is based on timing preset in digital counters attached to a common shaft. Each digital counter can trigger an independent event. Events can occur at 0.5-second intervals when necessary, and an event-choice time range from 0.5 to 2000 seconds is available. When the preset time for a particular digital counter to trigger a particular event is reached, that counter activates an attached cam which trips the affected switch(es) to initiate the required equipment operations.

3.1.3.2 Integrator Operation. The integrator, an ac motor-type tachometer, has two functions. The first of these is to provide a voltage for torquing the IRP pitch gyro at a rate proportional to the satellite's geocentric angular rate during coast and orbital boost in order to maintain the satellite vehicle in a constant attitude with respect to the earth. The basic input signal to the integrator is a voltage derived from the torquing force of the longitudinal (roll axis) accelerometer. This signal drives the tachometer, producing a motor shaft angular displacement proportionate to the satellite vehicle velocity. Through a mechanically-linked potentiometer, the proportional displacement regulates a signal voltage that torques the IRP pitch gyro. The range of the torquing rate thus produced is approximately 2 to 4 degrees per minute. The circuit which supplies the signal voltage to the gyro is turned on shortly after separation and remains on until the reorientation maneuver is initiated after Agena rocket engine shutdown. The second integrator function is to provide the hard shutdown command to the propulsion system when the satellite has reached orbital velocity. A switch closure mechanism and a resolver are coupled to the tachometer motor shaft to form a position servo that makes electrical caging of the shaft possible. The caged position is preset so that after a predetermined number of shaft revolutions, the switch closure mechanism will initiate rocket engine shutdown. This predetermined number of shaft revolutions is proportionate to the velocity-to-be-gained in excess of the Thor-induced velocity. The pre-set velocity-to-be-gained is maintained by the servo caging loop until the SS/D timer hold is applied (after separation and before orbital boost; Ref. Par. 3.1.3.1, above). At this time, the shaft is uncaged, the servo becomes a rate (or velocity) sensing device, and the integrating unit is now ready to integrate the orbital boost acceleration information with the pre-set velocity-to-be-gained increment. The velocity-to-be-gained increment is pre-set slightly high and is adjusted downward as necessary by ground operations transponder-decoder Channel 6 shortly after the integrator is uncaged. The time for transmitting the correction signal and the amount of correction to be applied are calculated on the basis of range radar measurements of the satellite vehicle boost trajectory. The minus velocity gain correction is made by turning the tachometer shaft forward to reduce the number of revolutions required in order to reach the rocket engine cut-off point. The duration of the Channel 6 signal determines the amount of correction.

3.1.3.3 Resistive Voltage Networks. Resistive voltage networks are used to provide the specific voltages required for torquing the pitch and yaw gyros. The SS/D timer switches these signal voltages and the integrator potentiometer signal voltage on the off at the proper times. The circuit that supplies voltage to the pitch gyro for the residual pitch rate on orbit (Ref. Par. 3.1.2.1, above) remains on throughout the orbit phase except during the attitude change just before recoverable capsule ejection.

3.1.3.4 Guidance (Secondary) Junction Box. The guidance (secondary) junction box contains the phase-sensitive converters and the circuitry necessary for converting guidance and control function signals to a form suitable for telemetry purposes.

3.1.4 AC Voltage Regulator. The ac voltage regulator operates from a 115v ac, 400 cps, ϕAB input regulated to 1.0% to produce a voltage output regulated at 0.1%. This output is used in both the IRP and the integrator.

3.2 Flight Control System

The flight control system accepts attitude error signals from the IRP and amplifies them to a level sufficient to operate the pneumatic thrust nozzles (pneumatic control system) and/or to gimbal the rocket engine (hydraulic control system). After satellite vehicle separation, the pneumatic system is used for 3-axis control in all operational phases except orbital boost. During the orbital boost phase, pitch and yaw control are accomplished by the hydraulic control system, while roll control is accomplished by the pneumatic control system. The pitch and yaw pneumatic controls are turned off during this phase.

3.2.1 Flight Control Electronics System. The flight control electronics system consists of: (a) three spring-restrained rate gyros, one each for pitch, yaw, and roll, with their input axes aligned with the satellite pitch, yaw, and roll axes respectively; (b) summing amplifiers, demodulators, limiters, and dc amplifiers; and (c) associated circuits, components, and connectors. IRP pitch, yaw, and roll attitude error signals

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This page is reserved for the Guidance System schematic for Flight Configuration III which will be included in this space in subsequent revision.

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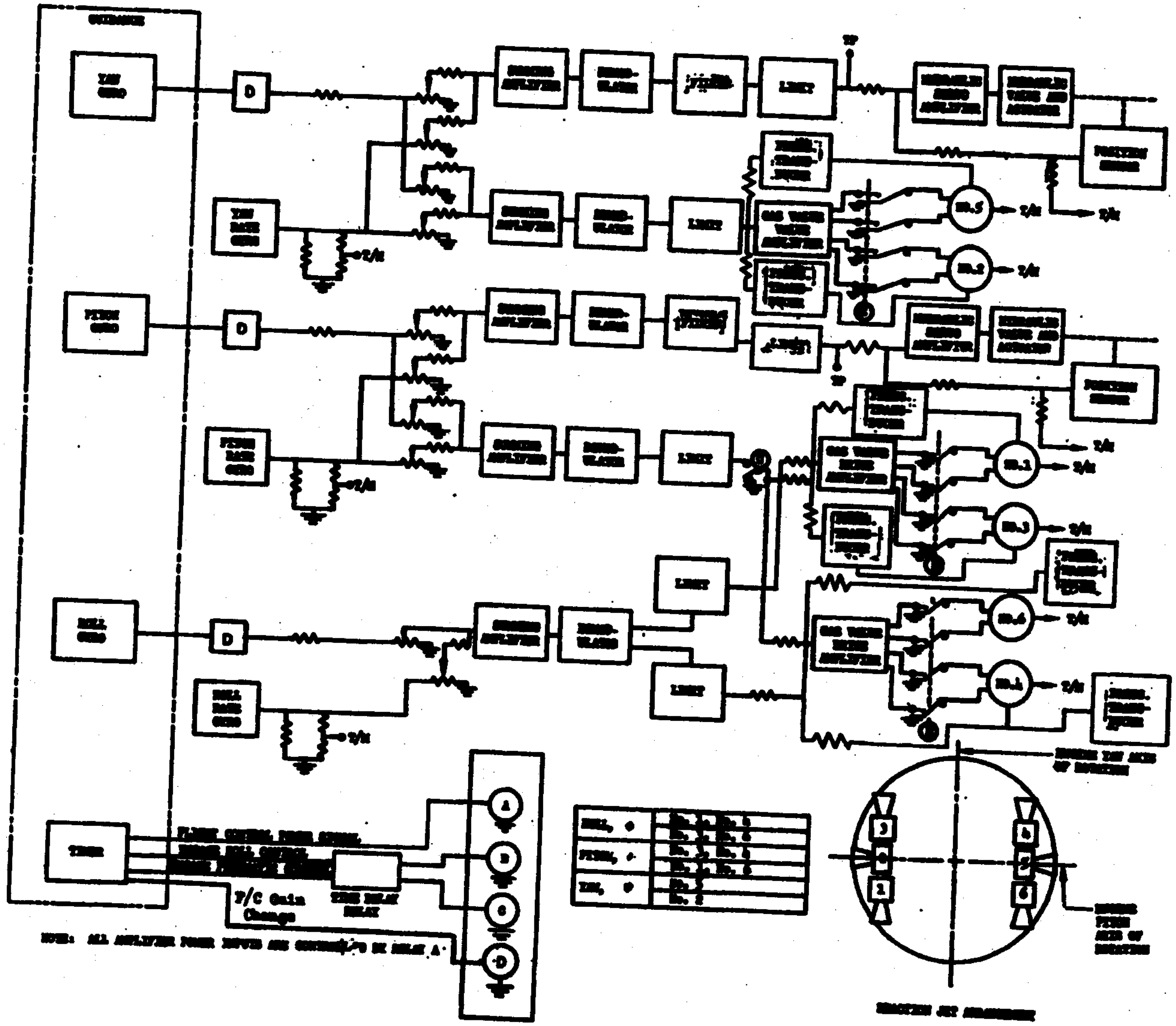

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are added to the rate information from the rate gyros and fed as inputs to the summing amplifiers. The output signals from the summing amplifiers are then fed as the input signal to a full-wave, phase-sensitive demodulator consisting of two diode bridge circuits, a reference transformer, a signal transformer, and filter network. The output from the demodulator is a filtered full-wave dc voltage signal whose polarity is dependent on the phase relationship between the demodulator input signal and the reference voltage. The output signal amplitude is proportionate to the amplitude of the input signals. The demodulator output signals are amplified in dc amplifiers to provide the additional gain required to operate the applicable (pneumatic or hydraulic) control system.

Performance Characteristics:

<u>Pitch, Yaw, and Roll Rate Gyros</u>

Maximum Rate	15°/sec
Sensitivity	0.500°/sec
Threshold under 0 g Environment	0.005°/sec
Maximum In-Phase Null Output	0.015 v at 0 rate input
Natural Frequency	19 to 30 cps
Damping Ratio	0.75 ±0.15

3.2.2 <u>Pneumatic Control System.</u> The pneumatic control system consists of: (a) the integrated power package, including the shutoff valve, pressure regulator, relief valves, and filter; (b) the six gas valve assemblies, including the pressure transducer; and (c) the gas pressure sphere, associated plumbing and circuitry, and the quick-disconnect coupling used when charging the pressure sphere during launch preparation. The pneumatic control system responds to directional control signals from the flight control electronics system. These signals are supplied to the gas valve assemblies, which provide satellite pitch and yaw control during the coast, reorientation, and orbital phases, and roll control during all phases after satellite vehicle separation. The control gas pressure sphere is mounted on the integrated power package installed on the aft interface of the satellite at Discoverer Station (DS) 446.70. The pressure sphere is charged with gas during preparation for launch. During the coast phase, after Thor booster burnout and satellite vehicle separation, a signal from the SS/D timer actuates opening of the shutoff valve, thus starting pressurization of the lines to the gas valve assemblies. A low-pressure relief valve (output side of the regulator) serves to protect the pressure system from surges in pressure caused by thermal expansion of the gas or by rapid cycling through the gas valve assemblies. A wire mesh screen-type filter installed between the pressure sphere and the high-pressure relief valve protects the system from particle contamination. The gas valve assemblies are two-stage pressure-operated servo-mechanisms, each of which is connected to an electro-mechanical transducer. When actuated by a signal from the gas valve amplifier circuit, the gas valve assembly torquing motor deflects the flapper located between the two first-stage control nozzels, thus changing the first-stage control pressure. The changing control pressure lifts the second-stage poppet valve from its seat, throttling the supply pressure in the thrust nozzle chamber. The thrust nozzle chamber pressure fed back to the torquing motor armature applies a force equal and opposite to the force provided by the electro-magnetic input. This action closes the servo loop and reduces the error forces to zero. In addition to the feed-back loop just described, an electrical transducer is used to sense the nozzle upstream pressure for comparison with the dc servo amplifier input signal, as a means of further reducing any possible error. This arrangement also reduces the deadband of the gas valve by a factor of approximately five.

Performance Characteristics:

<u>Pneumatic Control System</u>

System Supply	Nitrogen + Freon 14
Supply Pressure	3500 psi
Nominal Supply Volume	2200 cu in.
Regulated O erating Press.	170 +20/-17 psig
Impulse (approximate)	2000 lb-sec (varies with ratio of Freon to Nitrogen)
Shutoff Valve Operating Pressure (minimum)	30 psig
Filter Operating Pressure	3500 psig
Gas Control Valve:	
Operating Pressure	170 psig
Thrust Output (maximum)	20 lbs
Transducer Operating Press.	0 to 125 psig

3.2.3 <u>Hydraulic Control System.</u> The hydraulic control system consits of: (a) the actuator assembly, which includes the actuating cylinder, the servo valve, and the linear transducer; (b) the power package, including the integrated pump/motor, the accumulator/reservoir, the hydraulic motor starting relay, the filter, valves, and couplings, and (c) associated plumbing and circuitry. The hydraulic control system induces pitch and yaw control moments in the satellite during orbital boost by angular positioning of the thrust vector in accordance with information contained in a command signal from the flight control electronics system. The SS/D timer actuates the hydraulic pump/motor starting relay, and the pump builds up pressure in the lines from the accumulator/reservoir to the actuating cylinders. Each component of the accumulator/reservoir (which uses a common pressurization chamber) is pre-charged during the preparation for launch. A spring-loaded bypass relief connected between the high-pressure and low-pressure sides of the pump passes full pump output flow. The relief valve re-seats and makes the pressure flow available to the servo when the system pressure drops to a preset level. The relief valve also serves to protect the system from excessive pressure surges caused by rapid operation of the servo valve. The relief valve on the low-pressure side of the pump protects the pump inlet seals against rupture due to excessive pressure caused by rapid actuator motion and from rupture due to thermal expansion of the gas in the reservoir. The check valve in the high-pressure side of the pump/motor prevents the pump from "windmilling" or from being driven as a hydraulic motor during ground operations. Quick-disconnect couplings in the high- and low-pressure sides of the system permit both pressure and return connections to be used during ground operations. A stainless steel woven-wire mesh filter, located upstream from the servo valve, protects the valve from oil contaminants. Two electro-hydraulic servo actuators are used to control the engine, one in pitch and the other in yaw. These are attached to the engine mount and to the thrust chamber. Hydraulic flow and pressure to the cylinder is metered by the servo valve in response to signals from the hydraulic servo amplifier in the flight control electronics system. The linear transducer which measures the cylinder piston motion provides a proportional electrical signal to the hydraulic servo amplifier. This signal is fed through the summing amplifiers to provide the output signal which controls the metering action of the servo valve.

Performance Characteristics:

<u>Hydraulic Control System</u>

System Hydraulic Pressure	2500 psi
Electro-Hydraulic Servo:	
Actuating Cylinder:	
Static Hinge Moment	1400 ft-lbs
Effective Piston Area	0.70 sq in.
Total Piston Stroke	1.44 in. = 8.44°
Servo Valve Flow	0.8 gpm with 850 psi pressure drop across valve

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Hydraulic Power Package	
Integrated Pump/Motor:	
Motor	2.5 hp at 12,000 rpm, 28v dc
Pump Flow	0.75 gpm at 3000 psi
Accumulation/Reservoir:	
Pressurization Medium	Nitrogen
Pre-Charging Pressure	950 psi
Accumulator Volume	12 cu in.
Reservoir Volume	85 cu in.
High-Pressure Bypass Relief Valve:	
Full Flow Pressure	2300 psi
Reseat Pressure	2050 psi
Low-Pressure Relief Valve:	
Cracking Pressure	145 psig

4.0 POWER INPUT REQUIREMENTS

The table below presents the basic power input requirements for the FC III Satellite. A number in parentheses after a wattage requirement indicates that the requirement exists for less than the full duration of the noted period. All other requirements shown are for the full duration of the period indicated. Total watt-hour are based on 13 active orbits and 5 passive orbits, with an orbital period of 93.5 minutes, plus an ascent time of 15 minutes and an ejection time of 94.5 seconds.

	Ascent (Watts)	Orbit (Watts)	Eject (Watts)	Total (W-Hrs)
Horizon Scanner				
+28v dc, reg.	2.5	2.5	—	70
115v ac, 400 cps, 3Ø	14	14	—	394
IRP				
+28v dc, unreg.	50(80%)	50(20%)	50(20%)	209
+28v dc, reg.	15	7	7	199
-28v dc, reg.	13	4	4	115
115v ac, 400 cps, 3Ø	12	12	12	338
115v ac, 400 cps, 1Ø, 1% reg.	3	2	2	57
115v ac, 400 cps, 1Ø, 0.1% reg.	5	—	—	1.3
Computer				
+28v dc, unreg.	13(30%)	—	—	1.9
+28v dc, reg.	1(60%)	—	—	0.2
-28v dc, reg.	1(60%)	—	—	0.2
115v ac, 400 cps, 3Ø	15	—	15	4.2
115v ac, 400 cps, 1Ø 1% reg.	4	4	4	113
115v ac, 400 cps, 1Ø 0.1% reg.	6	—	—	1.5

	Ascent (Watts)	Orbit (Watts)	Eject (Watts)	Total (W-Hrs)
F/C Electronics				
+28v dc, unreg.	16(50%)	5	5	143
+28v dc, reg.	4	4	4	113
-28v dc, reg.	4	4	4	113
115v ac, 400 cps, 1Ø, 1% reg.	1	1	1	28
Rate Gyros				
115v ac, 400 cps, 3Ø	15	15	15	405
115v ac, 400 cps, 1Ø, 1% reg.	3	3	3	85
Hydraulic Motor				
+28v dc, unreg.	1960(15%)	—	—	77
Power Input Required (through recoverable capsule ejection)				2566

5.0 ENVIRONMENT

The guidance and control system is designed to meet the environmental criteria noted in LMSD Report 6117A.

6.0 RESPONSIBILITY

Guidance and Control Development Department (88/D).

7.0 NOTES

Not applicable.

8.0 GUIDANCE AND CONTROL SYSTEM INDEX

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AC Voltage Regulator	3.1.4
Computer	3.1.3
Digital Counter	3.1.3.1
Flight Control System	3.2
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1.0 GENERAL

The guidance and control system for Discoverer Flight Configuration (FC) IX Satellites consists of:

a. The guidance system, including the horizon scanner, the inertial reference package (IRP), the computer (including the integrator and the SS/D timer), the guidance (secondary) junction box (including the phase-sensitive converters), and an ac voltage regulator.

b. The flight control system, including the flight control electronics, the pneumatic control system, and the hydraulic control system.

Functionally, the guidance and control system establishes attitude references for the satellite and aligns the satellite to these references during the coast, orbital boost, reorientation, and orbit phases of operation. It also orients the satellite to the attitude required for recoverable capsule ejection. In addition, the system also initiates, modifies, and/or terminates programmed satellite equipment operation, including ignition and hard shutdown of the orbital boost rocket engine, during these phases.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in Section II, pp 2.0 ff as applicable, the signal inputs noted below are required for operation of the guidance and control system.

2.1 Disconnect of the main and auxiliary umbilicals at launch releases the SS/D timer brake, allowing the timer to start.

2.2 A signal from Douglas Ground Operations applies both a hold to the SS/D timer brake and a correction signal to the velocity-to-be-gained setting in the integrator.

2.3 A discrete signal from Douglas Ground Operations to terminate the signal referred to in Par. 2.2.

2.4 A signal from Douglas Ground Operations to re-establish hold on SS/D timer brake.

2.5 A discrete signal from Douglas Ground Operations to terminate the signal referred to in Par. 2.4.

2.6 The orbital timer turns off the horizon scanner and restarts the SS/D timer to initiate the ejection of the payload capsule.

3.0 PERFORMANCE

3.1 Guidance System

The guidance system establishes and maintains the various satellite attitudes programmed for the particular flight (subsequent to the Thor main and vernier engine shutdown) and supplies signals to the flight control system as required for satellite orientation from satellite vehicle separation in the ascent-to-orbit phase through capsule ejection in the orbit phase. The guidance system also includes a satellite equipment programing device which triggers system equipment operations during the ascent to orbit and the ejection sequence.

3.1.1 Horizon Scanner. The horizon scanner is an optical scanning system which is sensitive to infrared radiation and which operates both day and night. The scanner employs an optical collecting system, a thermistor bolometer, and a phase reference generator. The scanning axis is oriented along the satellite's yaw axis. The scanner system establishes the local vertical* by sensing the differences apparent at the horizon between the earth's thermal radiation and that of contiguous space. It then measures the angular difference between this local vertical and the satellite's yaw axis. This measure of difference modulates a 400 cps phase-reversing voltage to provide proportional signals of pitch and roll differences. The phased and proportional signals are fed to the IRP pitch and roll gyros to correct deviations from their respective reference attitudes. After satellite reorientation, the horizon scanner roll signal is also fed to the IRP yaw gyro for correction of possible yaw errors during orbit. This action maintains the longitudinal axis in an attitude parallel to the flight path. The pitch and roll correction signals are used to maintain the satellite's pitch and roll axes in a plane perpendicular to the local vertical.

Performance Characteristics:

Horizon Scanner

Torquing Signal Scale Factor	0.25 v rms/°
Output Signal Voltage (maximum)	2.6 v rms
Time Constant (maximum)	4.0 ±2.0 sec

3.1.2 Inertial Reference Package (IRP). The inertial reference package consists of three single-degree-of-freedom, floated, integrating gyros which are sensitive to pitch, yaw, and roll attitude errors respectively; one single-degree-of-freedom, servo-restrained, floated pendulous accelerometer; and associated pre-amplifiers, power amplifiers, and temperature regulating circuitry.

3.1.2.1 IRP Operation. The input axes of the IRP pitch, yaw, and roll gyros are oriented along the satellite's pitch, yaw, and roll axes, respectively, so that movement of the satellite about any of these axes causes rotation of the affected gyros about their respective output axes, producing signals proportionate to the displacement of the satellite from the affected axes. These gyro voltage signals are amplified in the preamplifiers and fed to the flight control system as attitude error signals. The applicable control system, pneumatic and/or hydraulic, responds with corrections to maintain the proper satellite attitude and gyro reference position. Programmed changes in satellite attitude are initiated via the SS/D timer and occur: (a) after satellite vehicle separation -- -5.0 degrees in yaw, when necessary, in order to align the direction of orbital boost thrust with the direction of thrust velocity prior to Agena engine ignition (the difference is caused by rotation of the earth); (b) after (a) and prior to orbital boost -- -28.6 degrees in pitch to achieve a near-horizontal satellite attitude in preparation for orbital boost rocket ignition; (c) during orbital boost -- +4.5 degrees in pitch to provide the proper direction of orbital boost rocket thrust; (d) after orbital boost rocket shutdown -- -180 degrees in yaw to place the satellite in a tail-first horizontal attitude for orbit operations; and (e) just before recoverable capsule ejection -- -60 degrees in pitch to establish the required satellite attitude for the ejection events. To accomplish a programmed attitude change, the SS/D timer feeds to the affected gyro a torquing signal that causes it to process about the axial position it occupied before receipt of the signal. Displacement of the gyro from its null signal position causes a proportional error signal to be fed to the flight control system, which responds with corrections that establish a new vehicle attitude and when the SS/D timer signal is removed, a new gyro reference position. The magnitude of the SS/D timer signal determines the rate of change, and its duration determines the amount of change. In order to maintain a horizontal satellite attitude with respect to the earth during orbit, a residual pitch rate proportional to the satellite's geocentric angular rate on orbit is initiated by the SS/D timer upon completion of the satellite's reorientation maneuver. This is accomplished by establishing a gyro null signal position that corresponds to a satellite pitch rate of -3.86 degrees per minute. This pitch rate is constant throughout the orbit phase except during the satellite attitude change just before recoverable capsule ejection.

* +1.5° mechanical pitch offset is set into horizon scanner; pitch orientations noted in subsequent paragraphs do not include this offset.

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Performance Characteristics:

IRP Pitch, Yaw, and Roll Gyros

Input Axis Angular Freedom (minimum)	±10°
Torque Generator Linearity at Signal Generator Null	±0.5%
Torque Generator Reproducibility	±0.1%
Maximum Drift Uncertainty (Random Torque)	3°/hr
Operating Temperature	±145°F

3.1.2.2 Accelerometers. The longitudinal (roll axis) accelerometer is oriented with its input axis parallel to the vehicle roll axis during orbital boost, supplying the integrator with a signal proportionate to the satellite's acceleration.

Performance Characteristics:

Accelerometers

Maximum Range	±15 g
Natural Frequency	50 cps
Linearity	0.4%
Operating Threshold (minimum input)	0.03 ft/sec/sec (1 milli-g)
Uncertainty	0.06 ft/sec/sec (2 milli-g)
Operating Temperature	±145°F

3.1.3 Computer. The guidance computer consists of: (a) a programming device, known as the SS/D timer; (b) the guidance integrator; (c) resistive voltage networks; (d) the guidance junction box; and (e) associated circuitry and hardware.

3.1.3.1 SS/D Timer Operation. The SS/D timer controls the event sequence of the principal satellite equipment functions during the coast, orbital boost, and reorientation phases and during recoverable capsule ejection. The SS/D timer operates continuously from launch through reorientation, after which it turns itself off. It is reactivated for the capsule ejection event sequence by the orbital timer, and it turns itself off again when ejection has been accomplished. A few seconds after "MECO command" signal is sent, a signal (D1) is transmitted from Douglas Ground Control via the booster guidance system to apply a correction and a hold on the SS/D timer brake control. A second signal (D2) from Douglas Ground Operations extends the hold, without the correction. The purpose of the correction and the hold is to assure that orbital boost rocket engine ignition occurs at the optimum time. The amount of the correction and the duration of the hold are computed in Douglas Ground Operations equipment on the basis of radar range and velocity measurements of the Booster boost trajectory. The correction is a minus adjustment to the pre-set velocity-to-be-gained setting in the satellite integrator.

3.1.3.2 Integrator Operation. The integrator, an ac motor-type tachometer, has two functions. The first of these is to provide a voltage for torquing the IRP pitch gyro at a rate proportional to the satellite's geocentric angular rate during coast and orbital boost in order to maintain the satellite vehicle in a constant attitude with respect to the earth. The basic input signal to the integrator is a voltage derived from the torquing force of the longitudinal (roll axis) accelerometer. This signal drives the tachometer, producing a motor shaft angular displacement proportionate to the satellite vehicle velocity. Through a mechanically-linked potentiometer, the proportional displacement regulates a signal voltage that torques the IRP pitch gyro. The range of the torquing rate thus produced is approximately 2 to 4 degrees per minute. The circuit which supplies the signal voltage to the gyro is turned on shortly after separation and remains on until the reorientation maneuver is initiated after satellite rocket engine shutdown. The second integrator function is to provide the hard shutdown command to the propulsion system when the satellite has reached orbital velocity. A switch closure mechanism and a resolver are coupled to the tachometer motor shaft to form a position servo that makes electrical caging of the shaft possible. The caged position is preset so that after a predetermined number of shaft revolutions, the switch closure mechanism will initiate rocket engine shutdown. This predetermined number of shaft revolutions is proportionate to the velocity-to-be-gained in excess of the booster-induced velocity. The pre-set velocity-to-be-gained is maintained by the servo caging loop until the SS/D timer hold is applied (after "MECO command" and before separation) (cf. Par. 3.1.3.1 above). At this time, the shaft is uncaged, the servo becomes a rate (or velocity) sensing device, and the integrating unit is now ready to integrate the orbital boost acceleration information with the pre-set velocity-to-be-gained increment. The velocity-to-be-gained increment is pre-set slightly high and is adjusted downward as necessary by Douglas Ground Operations via the D1 signal shortly after the integrator is uncaged. The amount of the correction and the duration of the corrective signal are calculated on the basis of radar range measurements of the vehicle boost trajectory. The minus gain correction is made by turning the tachometer shaft forward to reduce the number of revolutions required in order to reach the rocket engine cutoff point.

3.1.3.3 Resistive Voltage Networks. Resistive voltage networks are used to provide the specific voltages required for torquing the pitch and yaw gyros. The SS/D timer switches these signal voltages and the integrator potentiometer signal voltage on and off at the proper times. The circuit that supplies voltage to the pitch gyro for the residual pitch rate on orbit (Ref. Par. 3.1.2.1, above) remains on throughout the orbit phase except during the attitude change just before recoverable capsule ejection.

3.1.3.4 Guidance (Secondary) Junction Box. The guidance (secondary) junction box contains the phase-sensitive converters and the circuitry necessary for converting guidance and control function signals to a form suitable for telemetry purposes.

3.1.4 AC Voltage Regulator. The ac voltage regulator operates from a 115v ac, 400 cps, ØAB input regulated to 1.0% to produce a voltage output regulated to 0.1%. This output is used in both the IRP and the integrator.

3.2 Flight Control System

The flight control system accepts attitude error signals from the IRP and amplifies them to a level sufficient to operate the pneumatic thrust nozzles (pneumatic control system) and/or to gimbal the rocket engine (hydraulic control system). After satellite vehicle separation, the pneumatic system is used for 3-axis control in all phases except orbital boost. During the orbital boost phase, pitch and yaw control are accomplished by the hydraulic control system, while roll control is accomplished by the pneumatic control system. The pitch and yaw pneumatic controls are turned off during this phase.

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3.2.1 Flight Control Electronics System. The flight control electronics system consists of: (a) three spring-restrained rate gyros, one each for pitch, yaw, and roll, with their input axes lying along the satellite pitch, yaw, and roll axes respectively; (b) summing amplifiers, demodulators, limiters, and dc amplifiers; and (c) associated circuits, components, and connectors. IRP pitch, yaw, and roll attitude error signals are added to the rate information from the rate gyros and fed as inputs to the summing amplifiers. The output signals from the summing amplifiers are then fed as the input signal to a full-wave, phase-sensitive demodulator consisting of two diode bridge circuits, a reference transformer, a signal transformer, and a filter network. The output from the demodulator is a filtered full-wave dc voltage signal whose polarity is dependent on the phase relationship between the demodulator input signal and the reference voltage. The output signal amplitude is proportionate to the amplitude of the input signals. The demodulator output signals are amplified in dc amplifiers to provide the additional gain required to operate the applicable (pneumatic or hydraulic) control system.

Performance Design Characteristics:

Pitch, Yaw, and Roll Rate Gyros

Maximum Rate	15°/sec
Sensitivity	0.500v/°/sec
Threshold under Zero g Environment	0.005°/sec
Maximum In-Phase Null Output (maximum)	0.015 v at zero rate input
Natural frequency	19 to 30 cps
Damping Ratio	0.75 ±0.15

3.2.2 Pneumatic Control System. The pneumatic control system consists of: (a) the integrated power package, including the shutoff valve, pressure regulator, relief valve, and filter; (b) the six gas valve assemblies, including the pressure transducer; and (c) the gas pressure sphere, associated plumbing and circuitry, and the quick-disconnect coupling used when charging the pressure sphere during launch preparation. The pneumatic control system responds to directional control signals from the flight control electronics system. These signals are supplied to the gas valve assemblies, which provide satellite pitch and yaw control during the coast, reorientation, and orbital phases and roll control during all phases after satellite vehicle separation. The control gas pressure sphere is mounted on the integrated power package installed on the aft interface of the satellite at Discoverer Station (DS) 446.70. The pressure sphere is charged with gas during preparation for launch. During the coast phase, after vehicle separation, a signal from the SS/D timer actuates opening of the shut-off valve, thus starting pressurization of the lines to the gas valve assemblies. A low-pressure relief valve (output side of the regulator) serves to protect the pressure system from surges in pressure caused by thermal expansion of the gas or by rapid cycling through the gas valve assemblies. A wire mesh screen-type filter installed between the pressure sphere and the regulator protects the system from particle contamination. The gas valve assemblies are two-stage pressure-operated servo-mechanisms, each of which is connected to an electro-mechanical transducer. When actuated by a signal from the gas valve amplifier circuit, the gas valve assembly torquing motor deflects the flapper located between the two first stage control nozzles, thus changing the first stage control pressure. The changing control pressure lifts the second stage poppet valve from its seat, throttling the supply pressure in the thrust nozzle chamber. The thrust nozzle chamber pressure fed back to the torquing motor armature applies a force equal and opposite to the force provided by the electro-magnetic input. This action closes the servo loop and reduces the error forces to zero. In addition to the feedback loop just described, an electrical transducer is used to sense the nozzle upstream pressure for comparison with the dc servo amplifier input signal, as a means of further reducing any possible error. This arrangement also reduces the deadband of the gas valve by a factor of approximately five.

* Design Characteristics

Pneumatic Control System

System Supply	Nitrogen + Freon 14
Supply Pressure	3500 psi
Nominal Supply Volume	4400 cu. in.
Regulated Operating Pressure	170 +20/-17 psig
Impulse (approx.)	4800 lb-sec
Shutoff Valve Operating Pressure	30 psig (minimum)
Filter Operating Pressure	3500 psi
Gas Control Valve:	
Operating Pressure	170 psig
Thrust Output	20 lbs (maximum)
Transducer Operating Pressure	0 to 125 psig

3.2.3 Hydraulic Control System. The hydraulic control system consists of: (a) the actuator assembly, which includes the actuating cylinder, the servo valve, and the linear transducer; (b) the power package, including the integrated pump, reservoir, filter, valves, and couplings; and (c) associated plumbing and circuitry. The hydraulic control system induces pitch and yaw control moments in the satellite during orbital boost by angular positioning of the thrust vector in accordance with information contained in a command signal from the flight control electronics system. When the SS/D timer actuates the propulsion system, the fuel-driven pump motor begins to operate, and the pump builds up pressure in the lines to the actuating cylinder. A spring-loaded bypass relief connected between the high-pressure and low-pressure sides of the pump passes full pump output flow to the spring-loaded reservoir. The relief valve re-seats and makes the pressure flow available to the servo when the system pressure drops to a pre-set level. The relief valve also serves to protect the system from excessive pressure surges caused by rapid operation of the servo valve. The relief valve on the low-pressure side of the pump protects the pump inlet seals against rupture due to excessive pressure caused by rapid actuator motion. The check valve on the high-pressure side of the pump prevents the pump from "windmilling" or from being driven as a hydraulic motor during ground operations. Quick-disconnect couplings in the high- and low-pressure sides of the system permit both pressure and return connections to be used during ground operations. A stainless steel woven-wire mesh filter, located upstream from the servo valve, protects the valve from oil contaminants. Two electro-hydraulic servo actuators are used to control the engine, one in pitch and the other in yaw. These are attached to the engine mount and to the thrust chamber. Hydraulic flow and pressure to the cylinder is metered by the servo valve in response to signals from the hydraulic servo amplifier in the flight control electronics system. The linear transducer which measures the cylinder piston motion provides a proportional electrical signal to the hydraulic servo amplifier. This signal is fed through the summing amplifiers to provide the output signal which controls the metering action of the servo valve.

* See Notes Par. 6.0

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Performance Characteristics:

Hydraulic Control System

System Hydraulic Pressure	3300 psi
Electro-Hydraulic Servo:	
Actuating Cylinder—	
Static Hinge Moment	1400 ft-lbs
Effective Piston Area	0.70 sq. in
Total Piston Stroke	1.44 in. = 8.44° engine rotation
Servo Valve Flow	0.7 gpm with 3000 psi pressure drop across valve
Integrated Hydraulic Power Package	
Pump Flow	1 gpm at 3000 psi
Reservoir Volume	6 cu in.

4.0 POWER INPUT REQUIREMENTS

The table below presents the basic power input requirements for the satellites discussed herein. A number in parentheses after a wattage requirement indicates a requirement that exists for less than the full duration of the operational period under which it is listed. Total watt-hours shown are based on 42 active orbits and 22 passive orbits, with an orbital period of 95 minutes, plus an ascent time of 15 minutes and an ejection time of 94.5 seconds.

	Ascent (watts)	Orbit (watts)	Eject (watts)	Total (w-hrs)
Horizon Scanner				
+28v dc, reg.	2	2	—	199
115v ac, 400 cps, 3ϕ	12	12	—	1193
IRP				
+28v dc, unreg.	98(30%)	98(20%)	98(20%)	1932
+28v dc, reg.	10(69%)	10(20%)	10(20%)	200
115v ac, 400 cps, 3ϕ	10	10	10	995
115v ac, 400 cps, 1ϕ, 1% reg.	8	6	6	596
115v ac, 400 cps, 1ϕ, 0.1% reg.	7	—	—	1.8
SS/D Timer				
+28v dc, unreg.	13(30%)	—	13	1.4
Integrator				
+28v dc, reg.	1	—	—	0.3
-28v dc, reg.	0.5	—	—	0.1
115v ac, 400 cps, 1ϕ, 1% reg.	3.5	—	—	0.9
115v ac, 400 cps, 1ϕ, 0.1% reg.	5.5	—	—	1.4
Guidance J-Box				
+28v dc, reg.	4.4	4.4	4.4	438
-28v dc, reg.	0.1	0.1	0.1	10
115v ac, 400 cps, 1ϕ, 1% reg.	5	5	5	497
F/C Electronics				
+28v dc, reg.	4	4	4	398
-28v dc, reg.	4	4	4	398
115v ac, 400 cps, 1ϕ, 1% reg.	1	1	1	99
F/C Relays				
+28v dc, unreg.	4	4	4	398
+28v dc, unreg.	4(30%)	—	—	0.5
+28v dc, unreg.	7(27%)	—	—	0.5
+28v dc, unreg.	2(69%)	2	2	199
Rate Gyros				
115v ac, 400 cps, 3ϕ	20	20	20	1990
115v ac, 400 cps, 1ϕ, 1% reg.	9	9	9	895
Power Input Required (through recoverable capsule ejection:				10,464

5.0 ENVIRONMENT

The guidance and control system is designed to meet the environmental criteria noted in the LMSD Report 6117B

6.0 RESPONSIBILITY

Guidance and Control Development Department (SS/D).

7.0 NOTES

Not Applicable

8.0 GUIDANCE AND CONTROL SYSTEM INDEX

ITEM	PARAGRAPH
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AC Voltage Regulator	3.1.4
Computer	3.1.3
Digital Counter	3.1.3.1
Flight Control System	3.2
Flight Control Electronics System	3.2.1.
Gas Valve Assembly	3.2.2
Gyro Caging	3.1.3.2
Gyro Uncaging	3.1.3.2
Guidance (Secondary) Junction Box	3.1.3.4
Guidance System	3.1
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Inertial Reference Package	3.1.2
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Integrated Power Package	3.2.2
IRP Operation	3.1.2.1
Longitudinal Accelerometer	3.1.3.2
Orbital Timer	3.1.3.1
Performance Characteristics	
Accelerometers	3.1.2.2
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Hydraulic Control System	3.2.3
Pitch, Yaw & Roll Rate Gyros	3.2.1, 3.1.2.1
Pneumatic Control System	3.2.2
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Signal Input Requirements	2.0
Summing Amplifier	3.2.3
Three-axis Control	3.2
Timer Operation, SS/D	3.1.3.1
Voltage Networks, Resistive	3.1.3.3

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1.0 GENERAL

The communications system of the Discoverer Flight Configuration (FC) VI satellites consists of the identification transmitter, radar beacon transponder-decoder, orbital timer, unitized telemetry, antennas and associated equipment, and required hardware and wiring. On selected satellites, a dual-frequency Doppler tracking beacon and/or an optical tracking light installation are also carried. The communications system provides the means for: (a) tracking the satellite from launch to orbit and periodically thereafter through the series of useful life orbits; (b) commanding certain satellite equipment operations; (c) changing the time of certain programmed events; and (d) securing data and information about phenomena internal and external to the satellite before, during, and after recoverable capsule ejection. Communications equipment operations are controlled by noncyclic perforated tape timer. The planned sequencing of events during orbit is based on the position of the satellite with respect to ground operations read-in and command (RI/C) stations and to Smithsonian Astronomical Observatory Baker-Nunn optical tracking stations. (Ref. Sect. V, p. 1.1.2 for RI/C, optical tracking, and Transit Doppler readout station locations.) For the purposes of this discussion, the following terms and definitions apply:

(a) Orbit. An orbit is the course the satellite follows in circling the earth, with the "ascending node" as the start/finish point. The first orbit begins with the vehicle's first northward pass over the equator approximately 59 minutes after lift-off at Vandenberg AFB, California. An orbit may include all or part of active, semi-active, and/or passive subcycles.

(b) Subcycle. A subcycle is a programmed time period referenced to the orbital timer's programmed event sequence rather than to the satellite's orbital period. A subcycle begins at the programmed time of an ascending node and ends at the programmed time of the next ascending node. Subcycles are designated as:

(1) Active, in which the transponder-decoder and telemetry are programmed to operate. With a normal orbit, this occurs as the satellite passes within communication range of an RI/C station.

(2) Semi-Active, in which only the tracking light installation is programmed to operate. With a normal orbit, this occurs as the satellite passes within communication range of an optical tracking station.

(3) Passive, in which the transponder-decoder, the telemetry, and the tracking light installation are not programmed to operate since, with a normal orbit, the satellite does not pass within communication range of any RI/C or optical tracking station.

(c) Useful Orbital Life. The useful orbital life of a satellite is the time span, as determined by the last active subcycle, during which transponder and/or telemetry communication with RI/C stations occurs. This excludes tracking via the identification transmitter, the Doppler tracking beacon, and/or the tracking light installation.

(d) Range. The range of an RI/C station is determined by the reception capability of its telemetry equipment and by the command signal transmission capability of its Mod II radar equipment, whichever is less. The range of a Doppler tracking station is determined by the reception capability of its tracking equipment. The range of an optical tracking station is determined by the optical characteristics of its tracking equipment.

2.0 SIGNAL INPUT REQUIREMENTS

2.1 Tracking Equipment. The identification transmitter and Doppler tracking beacon are turned on during countdown and operate continuously until exhaustion of the satellite electrical power supply. The tracking light is turned on and off by the orbital timer during specific active and semi-active subcycles.

2.2 Radar Beacon Transponder and Command Decoder. The radar beacon transponder and command decoder is turned on during countdown and operates during ascent to orbit until turned off by the orbital timer. During this period, the SS/D timer supplies signals for switching the appropriate command channels from exit to orbit functions. Subsequent on-off operation during orbit is controlled by the orbital timer. Pulsed interrogation and command signals from RI/C station Mod II (modified) radars are required for transponder decoder operation. The interrogation signals are transmitted throughout the ascent to orbit tracking period and throughout the tracking periods of each active subcycle. The command signals are transmitted as required during the same periods.

2.3 Orbital Timer. The orbital timer is turned on by the SS/D timer during the ascent to orbit and operates thereafter per its stored program, subject to command modification by ground control.

2.4 Telemetry. The telemetry is turned on during countdown and operates during the ascent to orbit until turned off by the orbital timer. During this period, the SS/D timer supplies signals for telemetry calibration and for switching from exit antenna to orbit antenna operation (Ref. Par. 3.5 below). Subsequent telemetry operation during orbit is controlled by the orbital timer.

3.0 PERFORMANCE

3.1 Tracking Equipment

3.1.1 Identification Transmitter. The VHF identification transmitter transmits an unmodulated CW signal used for satellite tracking by ground control stations. This signal is fed through a multiplexer (which permits simultaneous use of the VHF exit or orbit antenna by the identification and telemetry transmitters) and is radiated to the RI/C stations via the VHF exit or orbit antenna. This lightweight, low-power transmitter is turned on during countdown and operates continuously thereafter until exhaustion of the satellite electrical power supply.

Performance Characteristics:

Acquisition Transmitter

Power Requirements	+12v dc, reg.;0.07 amps
Frequency	232.4 Mc (approximately)
Power Output (maximum)	20 mw

3.1.2 Doppler Tracking Beacon. A dual-frequency Doppler tracking beacon installation furnished by the Applied Physics Laboratory of Johns Hopkins University is carried on selected satellites. (Ref. Sect. V, p. 1.1.2 for Doppler readout station locations.) The beacon consists of a temperature-stabilized crystal-controlled oscillator; frequency multiplier, amplifier, and isolation stages; and two antennas. The beacon operates on two frequencies, 162 Mc and 216 Mc, and is accurate to one cycle in a million. Power for beacon operation is fed through a Type VI dc-dc power supply, which provides outputs of +4v dc and -18v dc. The beacon is turned on during countdown and operates continuously thereafter until exhaustion of the satellite electrical power supply.

Performance Characteristics:

Type VI DC-DC Power Supply

Input Voltage	+28v dc, unreg.
Power Output:	
Output 1	+3.75 to +4.35v dc, 38 to 42 ma
Output 2	-15.5 to -18v dc, 80 to 100 ma
Maximum Transient Voltages	+5v dc & -25v dc

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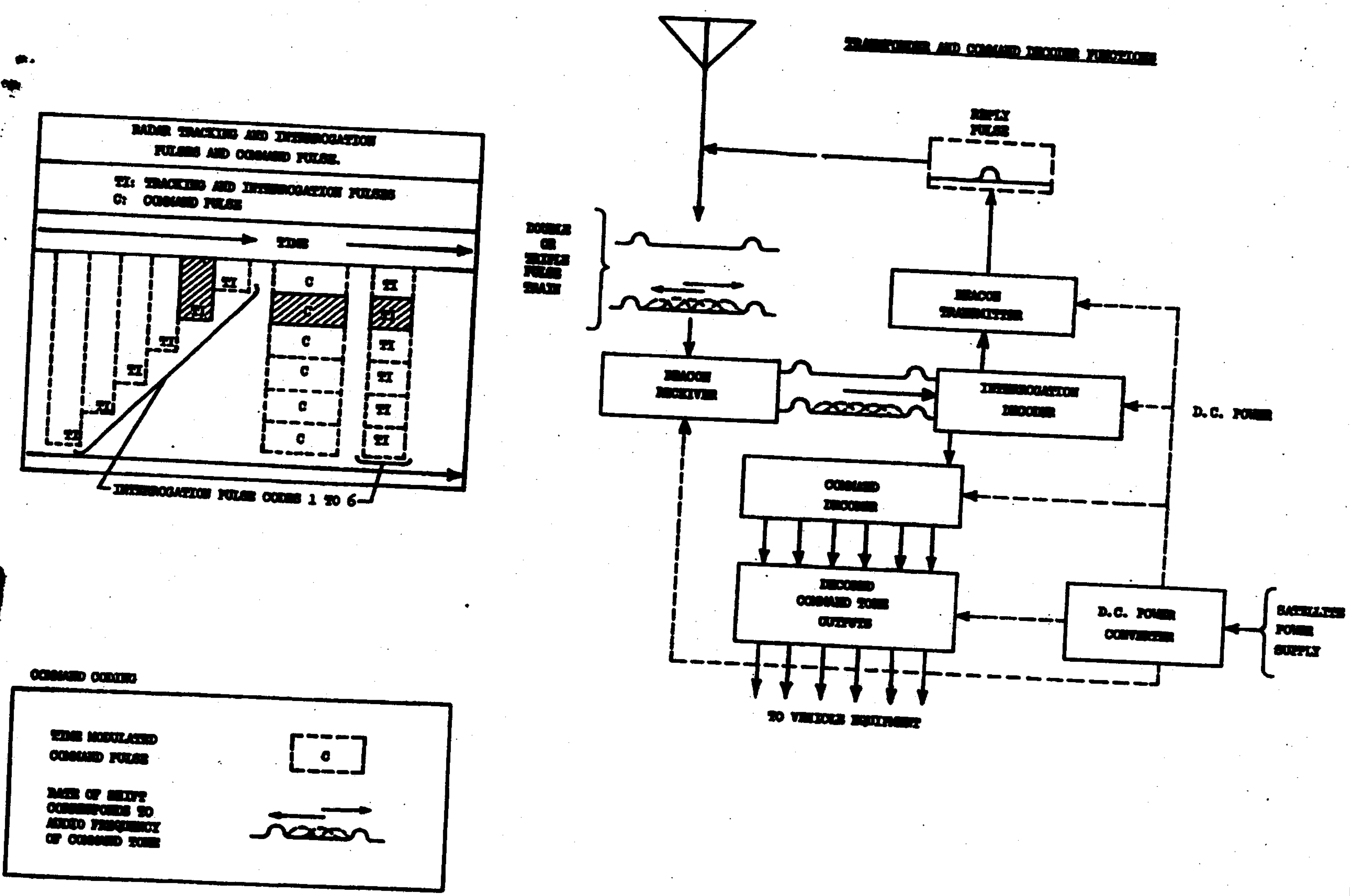

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Maximum Ripple	30 mv, peak-to-peak
Output Impedance:	
Output 1	12 ohms, dc to 1000 cps 10 ohms, 1 kc to 40 kc
Output 2	25 ohms, dc to 1000 cps 10 ohms, 1 kc to 40 kc
Operating Efficiency	70% at minimum input v 47% at maximum input v

Doppler Tracking Beacon

Operating Frequencies	162 Mc & 216 Mc
Power Output	100 mw
Resistive Load	50 ohms

3.1.3 **Tracking Light.** An optical tracking light installation is also carried on selected satellites. The installation, which consists of four aluminized GE #1020 lamps in a series-parallel circuit and a polished flat plate reflector mounted on the underside (horizontal attitude) of the satellite aft equipment rack, provides a minimum total effective illumination of 720 candle power. The tracking light installation is turned on and off by the orbital timer. The timer is programmed to provide a 45-second ON period during nighttime passes over each of 12 Smithsonian Astronomical Observatory Baker-Nunn tracking stations (ref. Sect. I, pp 3.6 ff for programming and Sect. V, p. 1.1.2 for station location). Orbit tracks calculated from telescopic camera azimuth and elevation angles are used to determine the accuracy of ephemerides evolved from the Doppler tracking measurements (ref. Par. 3.1.2)

Performance Characteristics:

Tracking Light

Lamps	4 GE #1020, aluminized
Power Requirements	+27.5v dc, 11.5 amps
Reflector	polished flat plate
Minimum Total Effective Illumination	720 candle power

3.2 **Radar Beacon Transponder-Decoder.** The S-band radar beacon transponder-decoder consists of three functional components: a receiver-transmitter, a command decoder, and a power converter. It has two functions: (a) it responds with single, high-powered pulses when interrogated by double-pulse ground RI/C station radar signals; (b) it receives, decodes, and demodulates additional time modulated pulses carrying RI/C station commands and routes the decoded commands to the affected satellite equipments. The transponder receiver is a superheterodyne type, tunable to any range frequency within the range of 2800 to 3000 Mc. The transmitter is of the plate-modulated triode type, capable of transmitting pulsed signals at any frequency within the range of 2800 to 3000 Mc at pulse repetition rates from 200 to 1600 pps. The transistorized power converter and voltage regulator receive a +28v dc input from the satellite auxiliary power system and supply the power and voltages required for operation of the transponder and command decoder. Reception and transmission are accomplished via a 1/2-wavelength stub exit-orbit antenna mounted in a circular slot flush with the satellite skin. The transponder-decoder is designed to operate in conjunction with a Mod II (modified) radar set. It is turned on during countdown, is turned off by the orbital timer after satellite reorientation, and thereafter is turned on and off by the orbital timer during operating subcycles throughout the useful life of the satellite.

Performance Characteristics:

Radar Beacon Transponder and Command Decoder

Power Requirements	+28v dc, reg.; 3.5 amps

Receiver

Type	Superheterodyne
Frequency	2800 to 3000 Mc
Sensitivity	-65 dbm for 100% interrogation with false alarms per second ≤ 5
Dynamic Range	0 to -75 db
Stability	Local oscillator frequency drift ≤ ±2 Mc per month
Image Rejection	30 db (minimum)

Transmitter

Frequency	2800 to 3000 Mc
Power Output	2 kw peak for 0.8 sec pulse
Interrogation Response Rate	1600 pps (maximum)
Stability	Local oscillator frequency drift ≤ ±2 Mc per month with constant vswr and pulse repetition rate change ≤ 2:1
Pulse Delay Variation	Variation in delay between pulse reception and transmission ≤0.25 sec for 15 db change in received pulse level

3.2.1 **Tracking and Interrogation.** For satellite tracking and interrogation, two spaced radar pulses with a maximum repetition rate of 1600 pps are transmitted from a RI/C station and received by the satellite transponder and interrogation decoder. The transponder responds by transmitting a single, high-powered identification pulse. Six different spacings between the two interrogation pulses are possible:

Code 1	21.35 ±0.5 μ sec
Code 2	24.40 ±0.5 μ sec
Code 3	27.45 ±0.5 μ sec
Code 4	30.50 ±0.5 μ sec
Code 5	33.55 ±0.5 μ sec
Code 6	36.60 ±0.5 μ sec

All pulses spaced more than ±1.5 microseconds from the code spacing are rejected by the interrogation decoder. The threshold level at the receiver output is adjusted to furnish no more than 5 trigger pps to the transponder transmitter in the absence of interrogating signals from RI/C station radar.

3.2.2 **Command.** For Discoverer Ground Operations (DiscOps) commands to satellite equipments, the command decoder decodes and demodulates a time-modulated third (command) pulse introduced between the two interrogation pulses. The command pulse precedes the second interrogation pulse by 9 microseconds. The command decoder transposes the command pulse time modulation to pairs of audio frequency tones which furnish six non-redundant commands that endure for a maximum of 100 milliseconds after receipt of the ground control command pulses. Four tones are used in six combinations of two to form six coded commands.

Tones	Frequencies
1	73.2 ±2% cps
2	91.5 ±2% cps
3	122.0 ±2% cps
4	154.0 ±2% cps

Command	Tones	
1	1 & 2	Selects either the "increase" or "decrease" mode for use with Command 2.
2	2 & 3	Changes the orbital timer subcycle period in 11 second increments, either increasing or decreasing according to the selected mode as determined by Command 1.
3	1 & 4	Resets orbital timer.
4	1 & 3	To payload.
5	2 & 4	Alternately enables and disables payload capsule ejection on one programmed series of ejection orbits.
6	3 & 4	Alternately enables and disables payload capsule ejection on a second programmed series of ejection orbits.

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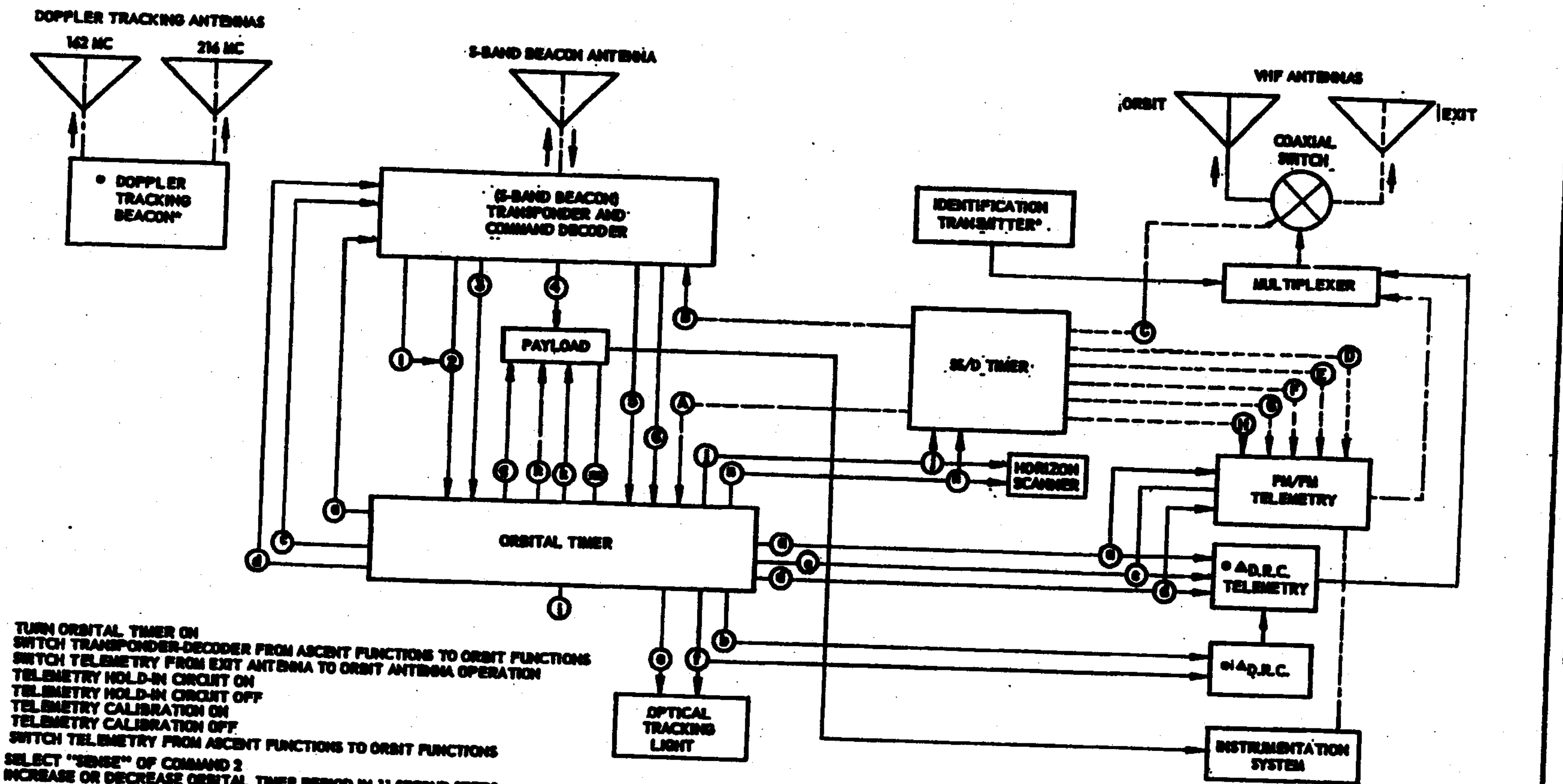

A TURN ORBITAL TIMER ON
B SWITCH TRANSPONDER-DECODER FROM ASCENT FUNCTIONS TO ORBIT FUNCTIONS
C SWITCH TELEMETRY FROM EXIT ANTENNA TO ORBIT ANTENNA OPERATION
D TELEMETRY HOLD-IN CIRCUIT ON
E TELEMETRY HOLD-IN CIRCUIT OFF
F TELEMETRY CALIBRATION ON
G TELEMETRY CALIBRATION OFF
H SWITCH TELEMETRY FROM ASCENT FUNCTIONS TO ORBIT FUNCTIONS
1 SELECT "SENSE" OF COMMAND 2
2 INCREASE OR DECREASE ORBITAL TIMER PERIOD IN 11-SECOND STEPS
3 RESET ORBITAL TIMER
4 TO PAYLOAD
5 ENABLE OR DISABLE RECOVERABLE CAPSULE EJECTION, 1ST PROGRAMMED SUBCYCLE SERIES
6 ENABLE OR DISABLE RECOVERABLE CAPSULE EJECTION, 2ND PROGRAMMED SUBCYCLE SERIES

a TRANSPONDER-DECODER, TELEMETRY, AND DISCOVERER RESEARCH COMPONENTS TELEMETRY WARM-UP ON
b DISCOVERER RESEARCH COMPONENTS ON
c TRANSPONDER-DECODER, TELEMETRY, AND DISCOVERER RESEARCH COMPONENTS TELEMETRY TRANSMITTERS ON; ORBITAL TIMER RESET ENABLE
d TRANSPONDER-DECODER, TELEMETRY, AND DISCOVERER RESEARCH COMPONENTS TELEMETRY TRANSMITTERS AND WARM-UP OFF; ORBITAL TIMER RESET DISABLE
e TRACKING LIGHTS ON
f TRACKING LIGHTS AND DISCOVERER RESEARCH COMPONENTS OFF
g TO PAYLOAD – ON FUNCTION
h TO PAYLOAD – OFF FUNCTION
i INACTIVE
j HORIZON SCANNER OFF AND SS/D TIMER ON FOR COMMAND 6 EJECTION
k TO PAYLOAD – ON FUNCTION
m TO PAYLOAD – OFF FUNCTION
n HORIZON SCANNER OFF AND SS/D TIMER ON FOR COMMAND 5 EJECTION

*CONTINUOUS OPERATION FROM LAUNCH

ΔDISCOVERER RESEARCH COMPONENTS

•ON SELECTED SATELLITES ONLY

SIGNAL FLOW
- - - - - ASCENT OPERATIONS
— - — - ASCENT & ORBIT
———— ORBIT OPERATIONS

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Commands 5 and 6 are effective on the next programmed ejection in the series to which they apply.

3.3 **Orbital Timer.** The orbital timer consists of a tape magazine assembly and a timer chassis assembly. It is used (a) after satellite reorientation, to turn off the transponder-decoder and the telemetry; (b) on orbit, to turn communications equipment on and off in programmed operating subcycles throughout the useful orbital life of the satellite; (c) to initiate recoverable capsule ejection by turning the horizon scanner off and restarting the SS/D timer; and (d) to effect programmed payload operations. The orbital timer is turned on by the SS/D timer during the ascent to orbit and operates continuously thereafter throughout the useful life of the satellite.

Performance Characteristics:

Orbital Timer

Power Requirements +28v dc, reg.: 0.32 amps

3.3.1 **Tape Magazine Assembly.** In the tape magazine, a perforated plastic tape is wound from a supply spool onto a take-up spool by a sprocket drive assembly. The tape travels between an electrically conductive surface on one side and a block of 13 brushes on the other. The circuit for each programmed command is completed by the electrical path made through the tape perforations. Completion of a circuit in this manner moves one of the orbital timer output latching relay contacts to either an ON or an OFF position. Each relay contact remains fixed or "latched" in one of these positions until a subsequent command signal moves it to its alternate position. During timer reset, the block of brushes is lifted from the tape by a solenoid arm.

3.3.2 **Timer Chassis Assembly.** The timer chassis assembly consists of the tape drive and reset, rear plate, and relay mounting plate assemblies. The **Tape Drive and Reset Assembly** consists of a 115v ac 400-cps synchronous motor which operates at 3000 rpm and is geared down to drive the tape at the required speeds. The gear-linkage and high speed clutches provide drive speeds from 1/3 to 1/60 rpm. A reset motor and gear-head, cams, actuators, and a reset clutch are connected to the tape drive through a set of gears. The timer **Rear Plate Assembly** includes stepping switch motors, a phase shift oscillator and power amplifier, and switches and circuitry for providing 11-second incremental timing steps. The phase shift transistor oscillator functions as the frequency generator for regulating the speed of the 400-cps drive motor. The oscillator output is fed through the power amplifier and provides regulating frequencies from 360 to 440 cps for control of the synchronous motor. The **Relay Mounting Plate Assembly** consists of six latching type output relays, additional relays required for operation of the program timer, and level-set potentiometers for regulating the input level of the telemetry signals.

3.3.3 **Operation.** The pre-punched perforations in the orbital timer tape provide a programmed command sequence timed in accordance with specific flight requirements. As noted above, each perforation in the tape is used to complete an electrical path that opens or closes one of the output relays. The circuitry offers thirteen possible commands. The equipment operated and functions performed by each command are follows:

Command	Function
a	Tr-Dec, Tlm, & DRC Tlm* Warm-up ON
b	Discoverer Research Components* ON
c	Tr-Dec, Tlm, & DRC Tlm* Transmitters ON; Orbital Timer Reset ENABLE
d	Tr-Dec, Tlm, & DRC Tlm* Transmitters & Warm-up OFF; Orbital Timer Reset DISABLE
e	Tracking Lights* ON
f	Tracking Lights* & DRC* OFF
g	To Payload -- ON Function
h	To Payload -- OFF Function
i	Inactive
j	Horizon Scanner OFF, SS/D Timer ON for Command 6 Ejection
k	To Payload -- ON Function
m	To Payload -- OFF Function
n	Horizon Scanner OFF, SS/D Timer ON for Command 5 Ejection

The orbital timer programmed sequence of events may be altered in time and/or orbit by the insertion of RI/C station command signals as noted below.

1. Signals sent via Channel 1 (Command 1) will shift the circuitry from the current to the alternate of the two possible modes, increase or decrease.

2. Command signals sent via Channel 2, depending on the "sense" of Command 1, increase or decrease the tape drive speed in steps, adjusting the timer's subcycle period in 11-second increments over a range from 90.0 to 108.15 minutes (99 steps) in order to match the subcycle period with the satellite's orbital period.

3. Command signals sent via Channel 3 move the programmed tape forward or backward to a pre-punched reset perforation, displacing the event sequence with respect to time and to the chosen reset latitude. The magnitude and direction of displacement depend on the position of the tape's reset point at the moment the reset command is given. For subcycle identification, the timer tape is punched at a coded time interval after each reset point. The intervals are coded in multiples of 20 seconds and are allocated so that each subcycle can be distinguished from those preceding and following it, since the interval registers on the reset monitor lamp at ground operations.

4. Command signals sent via Channels 5 and 6 control the activation of recoverable capsule ejection. Ejection is programmed for two series of subcycles, one series controlled via Channel 5 and the other via Channel 6. Ejection on a subcycle in one series is enabled by a command signal via the channel controlling that series and disabled by the next signal via the same channel. The enabling and/or disabling signals are effective on the next programmed ejection in the series to which they apply. (Ref. Sect. 1, pp 3.6 ff for ejection programming.)

3.4 **Telemetry.** The unitized FM/FM telemetry package is an integrated arrangement of modular subassemblies. The unitized FM/FM telemeter includes voltage-controlled subcarrier oscillators, commutators, FM transmitter, and RF amplifier. A power converter, voltage regulator, and required power supply circuits for transducer operation are included as modular units in the unitized telemetry package. Vibration amplifiers are located at or near the instrumentation transducers and are not a physical part of the unitized telemetry package. This equipment is used to convert instrumentation end-instrument signals into VHF signals for transmission to RI/C stations via the exit or orbit antenna. The exit antenna is a cavity slot antenna with a flush-mounted window face installed in the under portion (horizontal attitude) of the satellite's forward body. The orbit antenna is a whip antenna mounted on the aft equipment rack and held in a retracted position until satellite vehicle separation. The telemetry equipment is turned on during countdown and remains on during the ascent to orbit until turned off by the orbital timer. During this period, the SS/D timer switches the telemetry equipment from exit antenna to orbit antenna operation and also switches certain end-instruments from low gain to high gain for orbital operation. Subsequent telemetry operation is controlled by the orbital timer.

* Items carried only on selected satellites.

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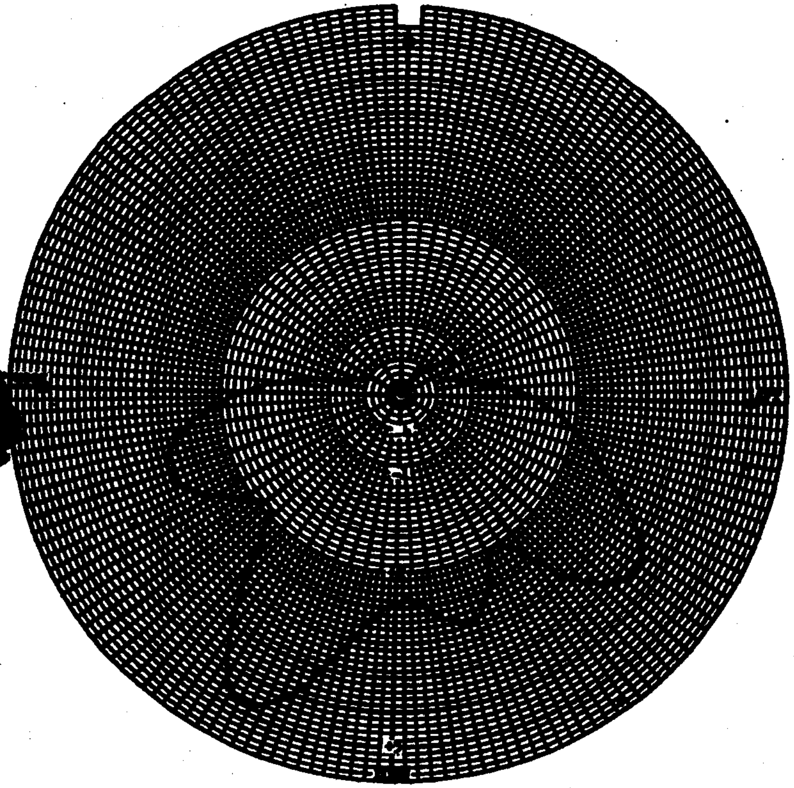
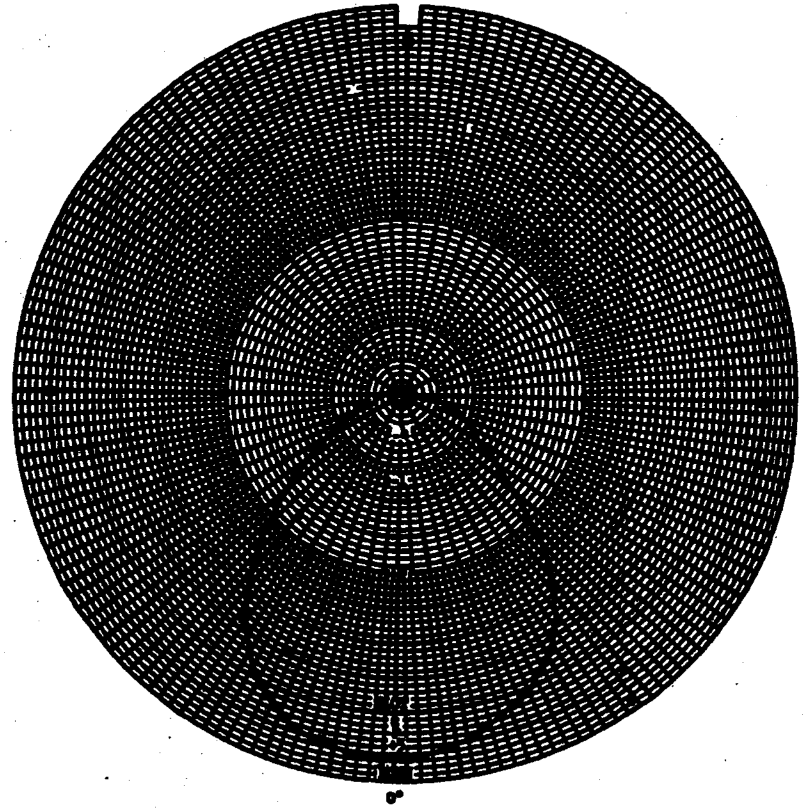


Pitch Pattern



Roll Pattern

NOTES: (1) Curves plotted in power
(2) Gain shown in decibels
(3) Frequency = 920 Mc
(230 Mc for 1/4-scale model)

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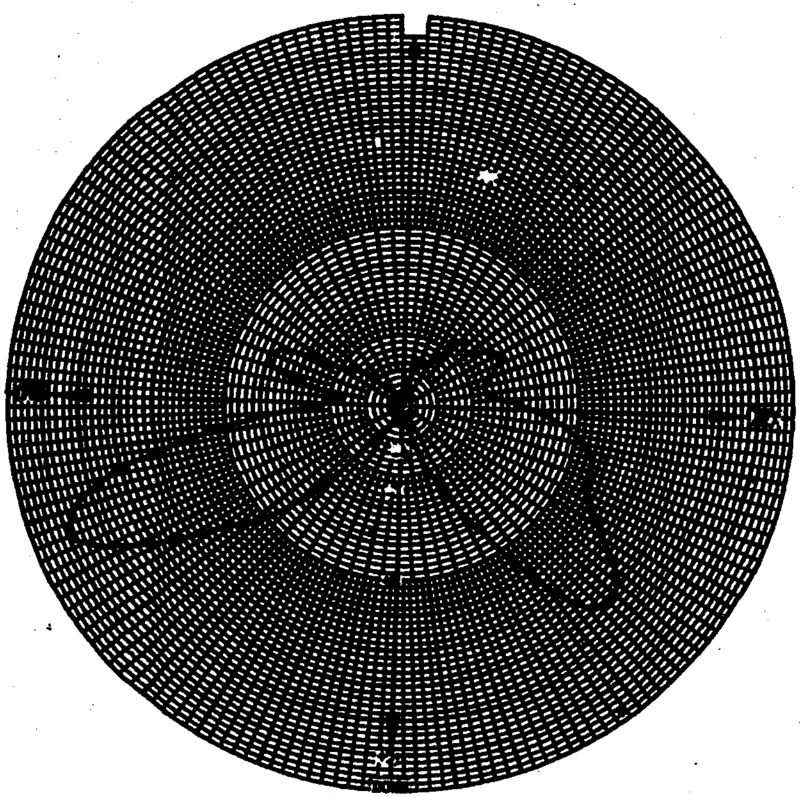
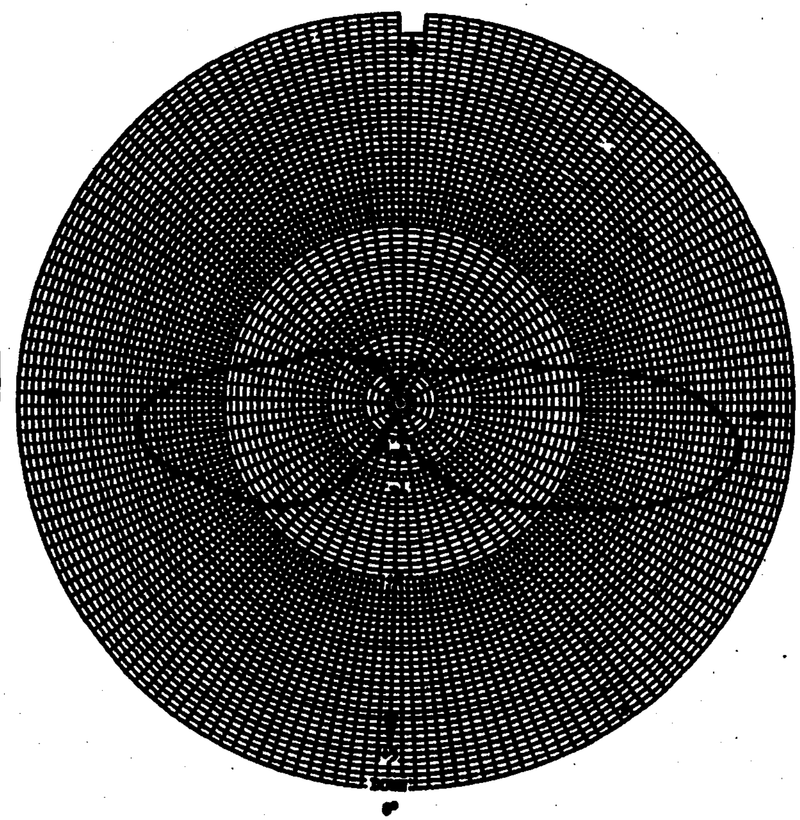


Pitch Plane



Roll Plane

NOTES: (1) Curves plotted in power
(2) Gain shown in decibels
(3) Frequency = 950 Mc
(237.5 Mc for 1/4-scale model)

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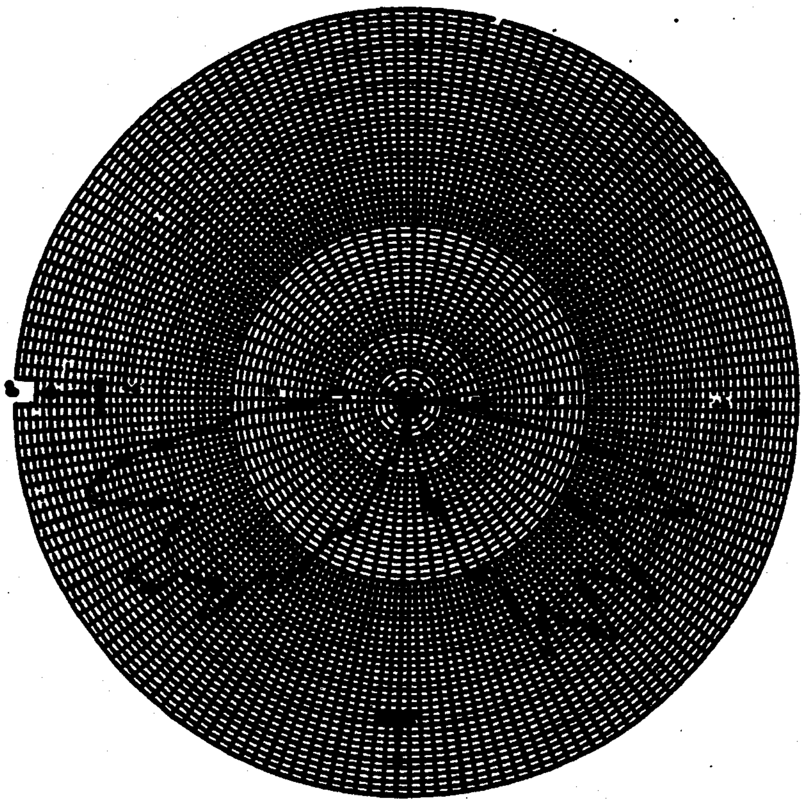
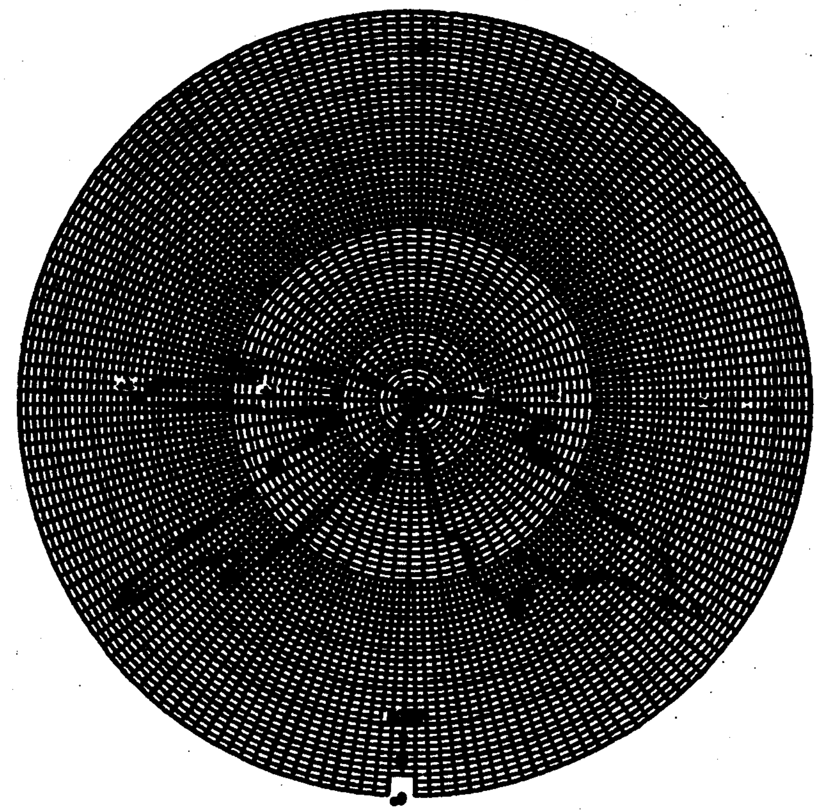
SATELLITE SYSTEMS DATA BOOK



Pitch Pattern



Roll Pattern

NOTES: (1) Curves plotted in power
(2) Gain shown in decibels
(3) Frequency = 2833 Mc

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APPROVED BY	RADIATION PATTERN, PITCH AND ROLL PLANES - S-Band Beacon Antenna Flight Configuration VI DISCOVERER PROGRAM	APPROVED BY

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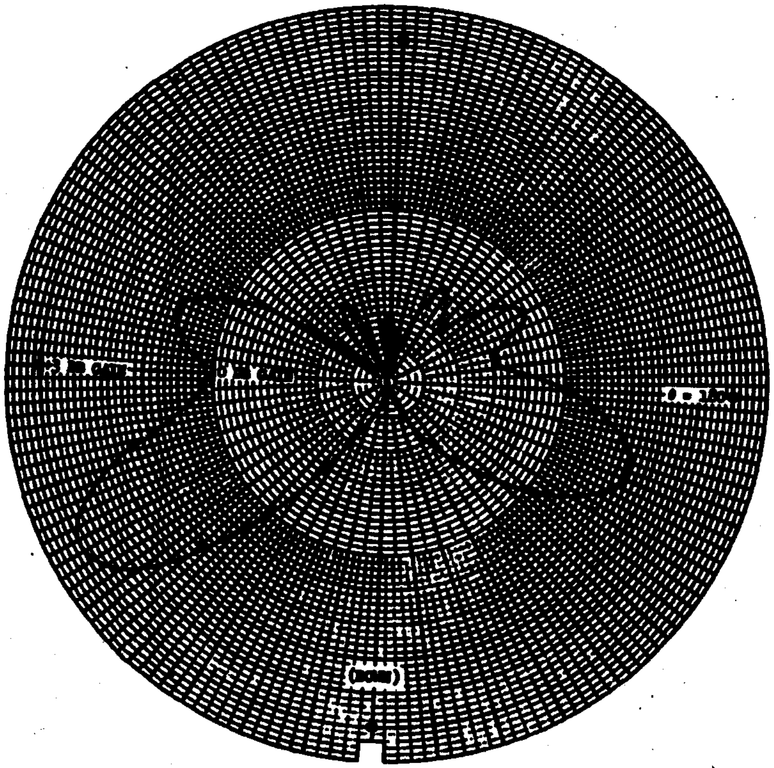
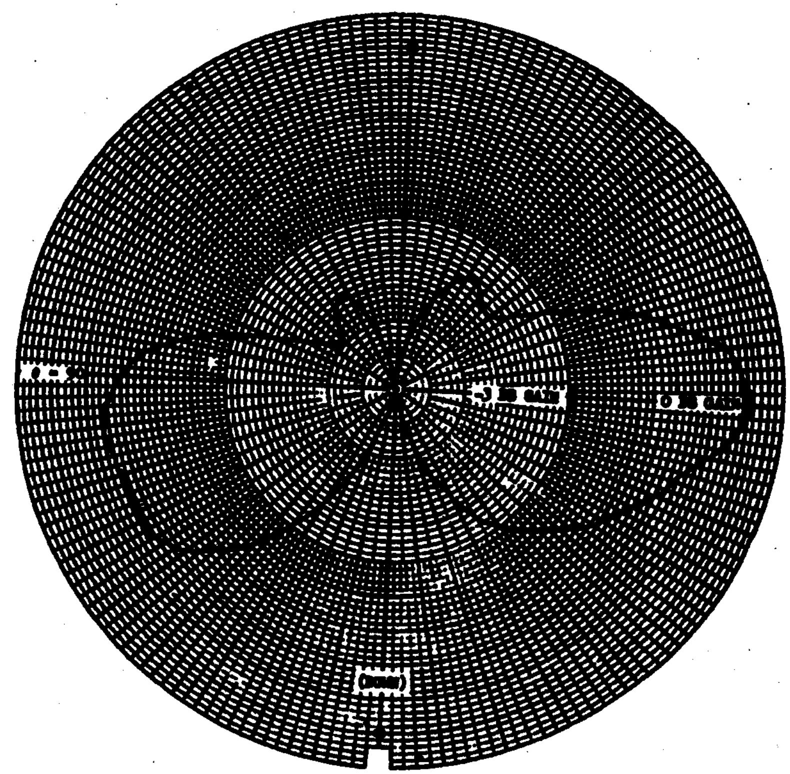


$\theta = 90$

PITCH PLANE



$\theta = 0$

ROLL PLANE

NOTES: (1) CURVES PLOTTED IN POWER
(2) GAIN SHOWN IN DECIBELS
(3) FREQUENCY = 162 Mc

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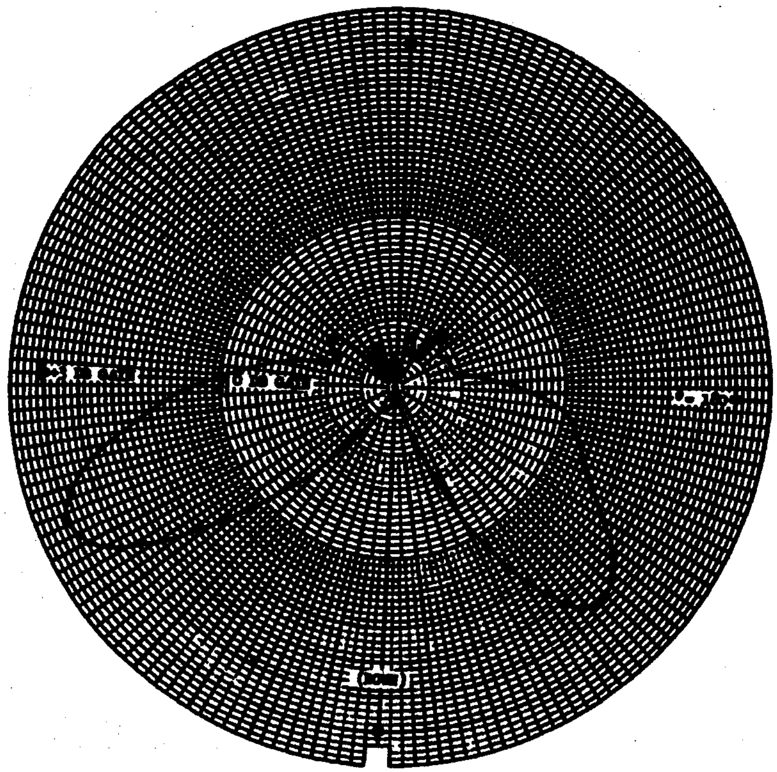
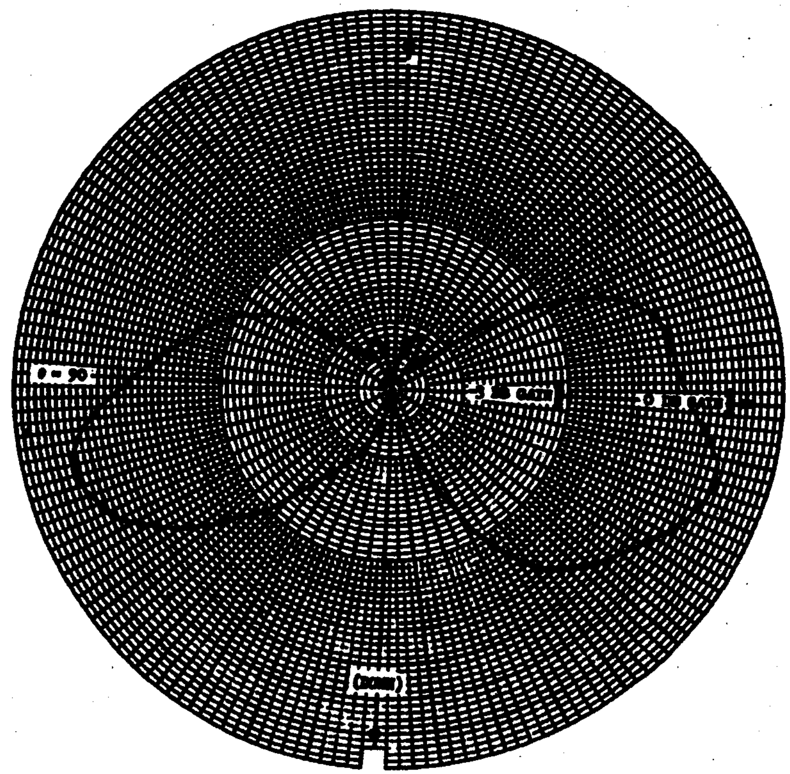


$\theta = 90$

PITCH PLANE



$\theta = 0$

ROLL PLANE

NOTES: (1) CURVES PLOTTED IN POWER
(2) GAIN SHOWN IN DECIBELS
(3) FREQUENCY - 216 Mc

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3.4.1 **Vibration Amplifiers.** The vibration amplifiers amplify the low voltage signal output taken from the piezo-electric vibration transducers, and the amplifiers are installed near these transducers. The amplified signal-voltage output is fed to the phase-sensitive converters which modulate subcarrier oscillator channels.

Performance Characteristics:

Vibration Amplifier

Power Requirements	+28v dc, 10 ma (maximum)
Input Impedance	10 megohms
Output Impedance	5000 ohms
Gain	3 to 50, adjustable
Output Voltage	5.0v, peak-to-peak
Frequency Response	28 to 2100 cps, flat within within 3 db

3.4.2 **Commutators.** The commutators, motor-driven multiple-contact sampling switches, sequentially sample the transducer output and data signals from the instrumentation end-instruments, providing a time-divided multiplexing of these signals.

Performance Characteristics:

Commutators

Power Requirements	+27.5v dc, 100 ma (maximum)
Sampling Rates	60 contact, 0.4 rps, 1 rps, and 5 rps
Minimum Duty Cycle	80%

3.4.3 **Voltage-Controlled Subcarrier Oscillators.** The voltage-controlled subcarrier oscillators change their operating frequency in direct proportion to changes in the signal output received from the phase-sensitive converters or other voltage-shaped signal sources. They are capable of sensing changes in voltage signals in the range from zero to five volts. The output from each subcarrier oscillator is a separate frequency-modulated subcarrier operating at an assigned subordinate channel frequency which forms a part of the total operating frequency of the transmitter.

Performance Characteristics:

Voltage-Controlled Subcarrier Oscillator

Power Requirements	+20v dc, 14.5 ma -20v dc, 6.5 ma
Frequency Deviation	7.5% from center of standard RDB subcarrier channels
Output Voltage	4.0v rms into load of 20,000 ohms at band center

Harmonic Distortion (maximum)	1% at rated output
Linearity	1% straight-line deviation

3.4.4 **FM Transmitter.** The FM transmitter is a low-power frequency-modulated transmitter accepting FM inputs from subcarrier oscillators operating on standard RDB Channels 1 - 18. The amplified subcarrier signals constitute a frequency-modulated time-divided composite signal that modulates the operating frequency of the transmitter carrier wave, forming the multiplexed FM/FM telemetry signal.

Performance Characteristics:

FM Transmitter

Power Requirements	6.5v dc at 0.8 amps 200v dc at 60 ma
Transmitter Power Output (minimum)	2 w into a 50-ohm non-reactive load
Modulation Input Voltage	2.5 to 3v, peak-to-peak
Modulation Input Frequency	100 to 80,000 cps
Transmitter Operating Frequency	Approximately 240 Mc
Frequency Deviation from Center of RF Carrier (maximum)	±125 dc
Frequency Stability	0.01% of center freq.

3.4.5 **RF Amplifier.** The RF power amplifier receives the 2 watt output from the FM transmitter and raises the output power level to 8 watts. The amplified RF carrier signal is then fed through a multiplexer (which permits simultaneous use of the VHF exit or orbit antenna by the telemetry and acquisition transmitters) and radiated to the RI/C stations via the VHF exit or orbit antenna.

Performance Characteristics:

RF Amplifier

Power Requirements	6.5v ac or dc at 1.0 amp +200v dc at 150 ma
Required Driving Power	2.0 w
Power Output (min)	8 w into a 50-ohm load

4.0 POWER INPUT REQUIREMENTS

Not available

5.0 ENVIRONMENT

The communications system is designed to meet the environmental criteria outlined in LMSD Report 6117B.

6.0 RESPONSIBILITY

Communication and Control Equipment Develoment.

7.0 NOTES

Not applicable.

8.0 INDEX

See p. 5.6.12, below.

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8.0 INDEX

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Voltage-Controlled Subcarrier Oscillators (Telemetry)	3.4.3

[redacted]

1.0 GENERAL

The payload for the Discoverer Flight Configuration III satellite is the Mark II Biomedical Recovery Capsule (BRC), designed and fabricated by General Electric Company. Functionally, the BRC is designed to deliver into orbit and return to earth a biological specimen, associated support equipment, and instrumentation required to provide physiological, psychological, and psycho-technical data on the specimen.

2.0 SIGNAL INPUT REQUIREMENTS

At the times noted in Section II, page 2.3.1.1 ff, the following signal inputs are required for operations related to the BRC.

2.1 A signal from the orbital (SS/E) timer arms the psychomotor test circuit. Subsequent signals actuate the psychomotor test (which is on for 15 seconds, off for 45 seconds, and on again for 3 minutes); terminate the psychomotor test; and then disarm the psychomotor test circuit.

2.2 At the beginning of the Capsule Ejection phase a signal from the SS/D timer (identified as the "arming" signal) ignites thermal relays to arm the ejection programmer, and turns on the capsule beacon.

2.3 A SS/D timer signal (identified as the "transfer" signal) actuates a pyrotechnic delay switch to initiate firing of the satellite-capsule electrical disconnect squib; actuates ejection system thermal batteries; and actuates pyrotechnic switches to arm recovery system battery.

2.4 A SS/D timer signal (identified as the "separation" signal) ignites satellite-capsule separation pin-puller squibs (0 to 7 millisecond delay).

3.0 PERFORMANCE

3.1 Description. The Biomedical Recovery Capsule (BRC), which consists of the Life Cell System, the Instrumentation System, the Ejection System, and the Recovery System, contains a specimen for physiological experimentation in space, and provides sensors to monitor the specimen's environmental conditions and psychophysiological reactions. Re-entry and recovery equipments are required for recovery of the capsule in which the specimen is housed.

The recoverable capsule is essentially conical with a bluntly rounded tip. The forward end of the capsule is at Discoverer Station (DS) 246.95. The conical portion, beginning at approximately DS 257, flares to 33.12 inches OD at DS 274.40, where the capsule is attached to the forward end of the satellite's aft nose section by two explosively-actuated pull-type pins. When these pins are pulled, four compressed springs accomplish capsule ejection. A thrust cone, which carries the capsule's propulsion equipment, is attached to the rear of the capsule by two explosive bolts and extends aft within the aft nose section to DS 289.04. The forward end of the capsule is covered with an ablative shell, and the rear of the capsule forward of the thrust cone is protected by an aft thermal cover.

3.1.1 Life Cell System. The life cell houses the biological specimen and provides the food, oxygen, and environment compatible with the subject. It also houses the instrumentation for sensing the internal environmental conditions and the animal's physical condition. This data is transmitted via the satellite telemetry system during orbit and recorded on the capsule magnetic tape recorder during re-entry. A 16 mm camera photographs the subject's face at fixed intervals during the entire mission.

3.1.1.1 Life Cell Housing. The life cell housing, into which the specimen is placed along with sensing and recording instrumentation, consists of a rectangular double-walled box assembly. Space between the outer and inner walls permits cooling air to be circulated around the inner box during pre-launch operations. The box is designed to withstand the pressure differential between the 1/2 atmosphere internal pressure and the ambient (from zero to one atmosphere) pressure outside the box.

Mounted on the outside of the housing assembly are the oxygen supply system and the air regeneration unit (ARU) which removes toxic gases from the housing, and maintains the interior temperature and humidity at acceptable levels.

3.1.1.2 Animal Restraint Seat. The animal restraint seat assembly consists of a foam-rubber-lined plastic contour seat and a webbed restraint suit fixed to a magnesium frame. An excreta collector prevents contamination of the respiratory system. The design of the seat assembly permits the specimen sufficient freedom of movement for feeding and psychomotor task operation.

3.1.1.3 Nutrient Feeder. The nutrient feeder assembly contains nutrient stored in wedged-shaped segments on a 12-segment spool. One segment of the spool is available to the specimen every four hours. The timing is controlled by a viscous fluid in a spring-loaded cylinder. As the fluid escapes through two minute holes in the cylinder cap, the spring withdraws a latch rod which allows the spool to rotate one segment at a time.

3.1.1.4 Oxygen Supply System. The oxygen supply system consists of a spherical oxygen tank and a regulator. The components are mounted on the underside of the housing support assembly. The oxygen tank contains 100%-pure oxygen at an original pressure of 7500 psi, which is supplied upon demand to the inner box of the housing assembly at a reduced pressure. The pressure within the inner pressure box is maintained at 1/2 atmosphere by the oxygen regulator.

To ensure a 100-percent oxygen environment within the housing assembly pressure box, the box is purged of nitrogen and other gases by flushing it with pure oxygen during the pre-launch operations. At the end of the flushing period, the pressure within the box is one atmosphere; this pressure is reduced to 1/2 atmosphere by the breathing of the biological subject. Thereafter, the pressure is maintained at 1/2 atmosphere by the oxygen regulator.

3.1.1.5 Atmosphere Conditioner System. The atmosphere conditioner system consists of a blower fan in the life cell and the air regeneration unit (ARU). The ARU consists of a heat exchanger and purifying chemicals contained in a fiber-glass-epoxy housing.

The ARU removes carbon dioxide, water vapor, and noxious odors from the oxygen exhausted from the pressure box, chemically, and cools the oxygen by passing it over evaporator fins cooled by evaporating water from the cooling system. Carbon dioxide and water vapor are removed from the oxygen by lithium hydroxide and lithium cloride; noxious odors by activated charcoal.

The blower fan within the pressure box draws the contaminated oxygen from the pressure box, and forces it through the conditioner and back to the pressure box. The fan is started when the capsule is assembled. It runs off ground power in the medical complex until it is installed on the Discoverer, and draws power through the satellite from the ground. From launch until separation, it is powered by the Discoverer Electrical Power System. After separation power is supplied by a battery mounted in the life cell.

3.1.1.6 Cooling System. The cooling system consists of a reservoir, a water solenoid valve, a battery, and a thermostat. It supplies water to the ARU for cooling the re-circulated oxygen.

The system is designed to maintain a temperature of 75° ± 10°F within the pressure box. When the temperature in the pressure box rises above 80°F., the thermostat opens the battery circuit to the solenoid valve. This valve opens, allowing a flow of water through the capillary tube to the evaporator. The evaporation of the water cools the fins of the evaporator. Thus the

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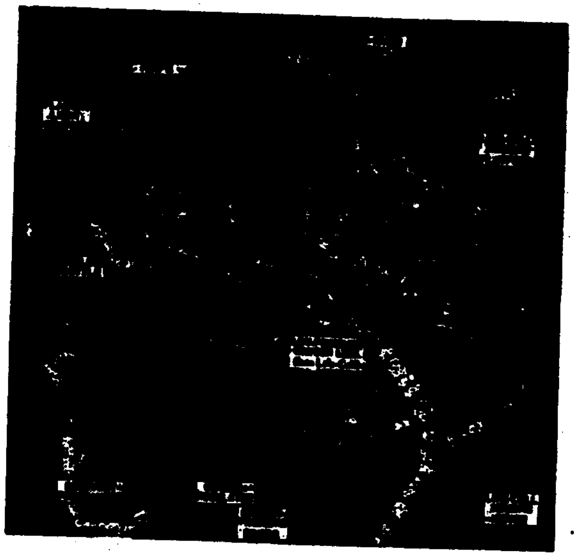

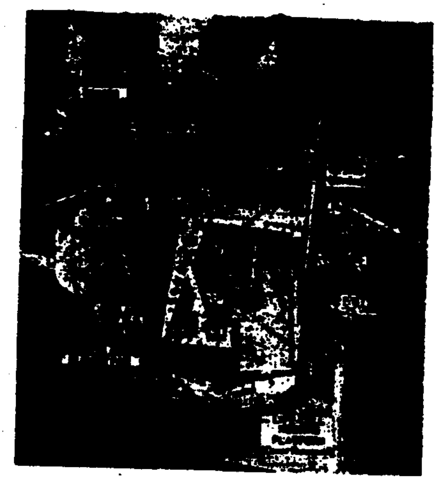

The above photographs show various views of the life cell portion of the Biomedical Recovery Capsule (BRC) in which a primate is launched into space. View 1 shows the life cell as seen externally from below. View 2 shows the interior of the life cell as viewed from above, including the primate's compartment. View 3 shows the assembled life cell unit which attaches to the remainder of the capsule.

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oxygen flowing over and around the fins is cooled before passing on to the inner pressure box. As the water in the reservoir is used, freon gas expands the neoprene bladder in the reservoir, maintaining constant water pressure against the solenoid valve.

3.1.1.7 Heating System. The heating system consists of a thermostat, which turns the heater on when the life cell temperature reaches 70°F, and the heater itself. The heating system is supplied with power from the satellite electrical power system during orbit and is not used during reentry and recovery.

3.1.1.8 Power Supplies. Power for the blower fan and for the sensors after separation is supplied by two batteries: a fan battery and a sensor battery. The sensor battery is a 7.5-volt battery rated at 1 amp-hour; the fan battery generates 18 volts and is rated at 6 amp-hours.

3.1.2 Instrumentation System. The instrumentation system, which consists of a 16 mm camera, the psychomotor test instrumentation, a compartment temperature sensor, a compartment humidity sensor, an oxygen bottle pressure sensor, a fan operation sensor, a compartment pressure sensor, the X-Y-Z axes acceleration sensors, an acoustic noise sensor, an electrocardiograph, and a respiration sensor, functions during the mission as programmed.

3.1.2.1 Camera. The 16 mm camera provides a photographic record of the subject's facial expressions throughout the various operational phases. It is programmed to make a half-frame (8mm by 10mm) exposure every three seconds. The camera, which is mounted on the life cell housing at a 90° angle to the specimen, requires two mirrors to view the animal.

3.1.2.2 Psychomotor Test Instrumentation. The instrumentation unit used in the performance of the psychomotor tests programs the lights and electrical stimuli which initiate the tests and measures the subject's response.

3.1.2.3 Instrumentation System - Launch Phase. During launch the following instrumentation is controlled by the SS/H timer, and the data is read out through the satellite telemetry system.

a) Compartment pressure
b) Compartment temperature
c) Relative humidity
d) Oxygen bottle pressure
e) Fan operation
f) Electrocardiogram (EKG)
g) Respiration

3.1.2.4 Instrumentation System - Orbit Phase. During orbit the following instrumentation is controlled by the SS/H timer, and the data is read out through the satellite telemetry system.

a) Compartment pressure
b) Compartment temperature
c) Relative humidity
d) Oxygen bottle pressure
e) Fan operation
f) Electrocardiogram (EKG)
g) Respiration
h) Psychomotor test performance

3.1.2.5 Instrumentation System - Recovery Phase. At the time of satellite-capsule separation, the magnetic tape recorder (cf. Par. 3.1.2.6) is activated to record instrumentation output continuously on the 1/2 hour supply of tape. The following data is recorded on the eight-channel tape.

a) Pulse-rate and respiration
b) Compartment pressure
c) Compartment temperature
d) Acoustic noise
e) Acceleration - X-axis
f) Acceleration - Y-axis
g) Acceleration - Z-axis
h) Timing index

3.1.2.6 Magnetic Tape Recorder. The recovery magnetic tape recorder, which weighs 3 pounds and runs for 30 minutes at a tape speed of .25 in/sec, is a battery-operated unit capable of recording eight channels of information (cf. Par. 3.1.2.5) on 1/8-in. pre-recorded magnetic tape through the use of the carrier-erase process. Seven of the eight channels are used to record operational data, while the eighth channel records the output of the internal timing oscillator for a timing index. An operating voltage of 18-15v dc is supplied by a capsule battery, triggered by the SS/D timer transfer signal.

3.1.3 Ejection System. The ejection system includes an ejection programmer, a retro rocket, a cold gas stabilization system, a barometric switch, and a thermal battery.

3.1.3.1 Ejection Programmer. The ejection programmer is a timing device that controls the sequence of operations required to place the capsule in a ballistic trajectory toward the earth. It controls the actuation of the retro rocket, the stabilization system, and the thrust cone-capsule separation. The programmer consists of four similar transistorized timers housed in an aluminum-alloy box. The timer components are mounted on printed circuit plug-in cards.

3.1.3.2 Retro Rocket. The solid propellant retro rocket, fired by a pyrotechnic device, burns for 9 seconds, imparts approximately 4-g acceleration, and supplies a total impulse of 10,470 lb-sec.

3.1.3.3 Cold Gas Stabilization System. The stabilization system is a pressurized cold-gas system employing separate but similar installations for capsule spin and de-spin. Each installation consists of a 78-cu-in. pressure sphere containing a mixture of nitrogen and freon at 3000 psi, a manifold into which the gas is fed from the pressure sphere, and two nozzles through which gas from the manifold is expelled at right angles to the capsule radius and the longitudinal axis. The radius of the spin manifold is 14 inches, and that of the de-spin manifold is 12 inches. Each installation develops a maximum thrust of 195 pounds and has an exhaust time of 0.8 second.

3.1.3.4 Barometric Switch. The barometric switch is a single pole, double-throw switch whose contacts are normally open below a threshold altitude of 75,000 (±10,000) feet. This switch prevents activating the pyrotechnic devices in the thermal battery while the SRC is on the ground.

3.1.3.5 Thermal Battery. The parallel thermal battery installation supplies power for the events immediately following ejection.

3.1.4 Recovery System. The SRC Mark II recovery system consists of a protective housing for the capsule during re-entry and recovery, a programmer for the recovery sequence, a parachute to support the capsule during the descent phase, and recovery aids for locating the capsule.

3.1.4.1 Protective Housing. An aft body assembly (recovery bucket) of light-weight aluminum provides protective housing for the life cell during descent and water-impact, and serves as a mounting for the life cell, support structure, beacon transmitter, and shell cover assembly. The shell cover assembly, constructed of high temperature glass-laminated phenolic plastic to withstand the heat during re-entry, serves as a rear cover for the recovery capsule, and as a mounting for the recovery aids, inter-face connector pressure vent, and ground cooling line adapters.

3.1.4.2 Recovery Programmer. The 5-g switch actuates the dimple motor which physically "pops-out" to start the mechanical recovery programmer. This programmer controls the ejection sequence and activation of recovery aids. The main components are a 5-g switch, 180-second timer, thermal batteries, and pyrotechnic switches.

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W. Wright	PAYLOAD Flight Configuration III DISCOVERER PROGRAM	W. L. [illegible]
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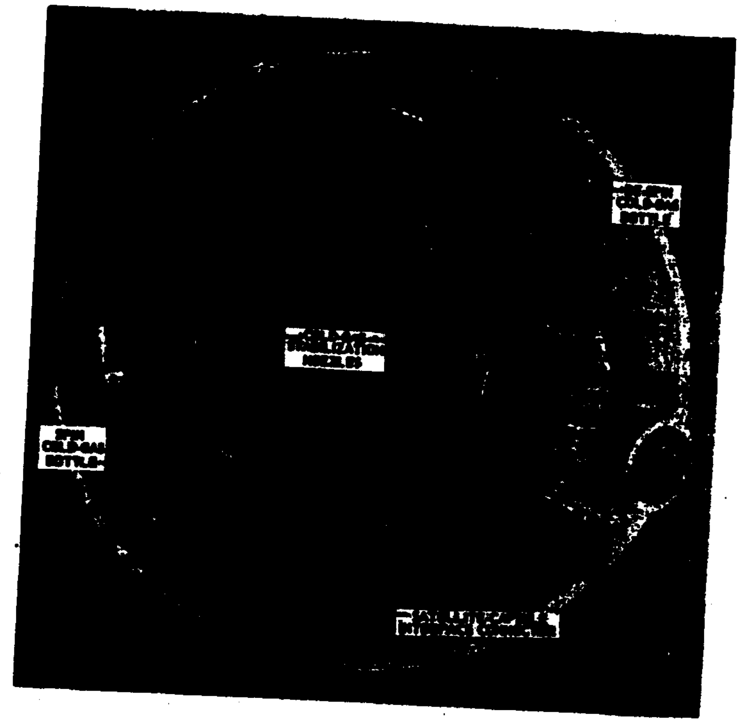

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The photographs above show two views of the thrust cone of the Biomedical Recovery Capsule (BRC) with the various components noted. View 1 shows the interior with the "spin" and "de-spin" gas bottles. View 2 shows the exterior, which is the aft portion of the cone, with the "spin" and "despin" nozzles in prominence. The dial-type unit on the right is a piece of ground test equipment.

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3.1.4.3 Parachute. The parachute assembly consists of a pilot parachute and a main parachute, contained within deployment bags. The panels of each gore of the main parachute are alternately dyed international orange and metallized. A box of chaff is also contained within the main parachute deployment bag. A parachute cover assembly serves as a protective rear housing for the recovery capsule during re-entry, and also protects the beacon light assemblies until cover ejection. The parachute cover is constructed of high-temperature, glass-laminated phenolic plastic.

3.1.4.4 Recovery Aids. A beacon radio transmitter provides a pulse-modulated VHF signal for homing-in by search aircraft and ships. Beacon transmitter life is a minimum of 10 hours; its range is 100 miles at 10,000 feet. The transmitter is mounted inside the recovery capsule, and is activated by the SS/D timer arming signal. The beacon battery provides the required operating voltages for the beacon transmitter and the beacon lights.

The two beacon light assemblies are activated at parachute ejection to provide a visual indication of the position of the recovery capsule. The beacon light assembly has a minimum life of 10 hours, and the signal should be visible at night for 5 miles.

3.2 Operation. Each phase of operation of the BMC is pointed toward acquiring data on the psychophysiological behavior of the specimen and ultimately recovering the specimen and its related instrumentation for analysis. Prior to launch the specimen is prepared for the mission in the Biomedical Support Complex at VAFB (cf. Section IV, Page 1.3.1 ff.)

3.2.1 Ascent to Orbit Phase. At lift-off the separation of the umbilical cabling de-energizes a capsule relay, starting the camera in the life cell, and energizes the thermostat in the life cell, which then controls the water solenoid valve. During this phase data from the life cell instrumentation (cf. Par. 3.1.2.3) is telemetered by the satellite telemetry system. When the satellite reaches an altitude of about 70,000 feet above sea level, the decrease in air pressure closes the barometric switch that arms the capsule ejection system.

3.2.2 Orbit Phase. Readout of the life cell instrumentation is accomplished during a 16 minute period in orbits 1, 2, 6, 7, 8, 9, 10, 13, 14, 15, 16, and 17. At the beginning of the readout period, the orbital (SS/E) timer activates the sensors for readout through the satellite telemetry system. In the middle of the readout period, the psychomotor test is initiated. Two and a half seconds after power is applied, a red light mounted on the housing cover lights, alerting the specimen to pull the psychomotor test lever, mounted on a panel in front of and slightly above him. If the specimen pulls the lever, the light goes out; if he does not, a 20-volt shock is applied to his ankles. If the specimen still does not pull the lever, he is assumed to be either dead or incapacitated. If he does pull the lever during shock, the light goes out and the shock is discontinued. This test procedure is repeated continuously for 15 seconds, then discontinued for 45 seconds, then run again continuously for 3 minutes. The specimen's actions during the test result in a step-voltage readout which is telemetered for decoding and analysis. The sensors (cf. Par. 3.1.2.4) register the specimen's reactions and environment before, during, and after the psychomotor test.

3.2.3 Ejection Phase. On the seventeenth orbit the orbital (SS/E) timer restarts the SS/D timer at T - 94.5 seconds (T - 0 seconds being the time of satellite-capsule separation) to initiate the pitchover and ejection sequence.

At T - 79.5 seconds the SS/D timer arming signal ignites thermal relays to arm the thrust cone (ejection) programmer; turns on the capsule beacon; and torques the inertial reference package (IRP) pitch gyro to rotate the satellite at a -45°/min rate from its tail-first horizontal position in order to achieve a -120° pitch angle for capsule ejection.

At T - 2.5 seconds the SS/D transfer signal changes the IRP pitch rate to a residual of +3.55°/min; actuates the ejection system thermal batteries; actuates recovery system thermal battery 2A8; actuates pyrotechnic switches to arm recovery system battery 2A7BE1; actuates a pyrotechnic delay switch to initiate firing of the satellite-capsule electrical disconnect squib; and transfers capsule equipment from satellite electrical power to capsule-borne electrical power.

At T - 1.5 seconds the electrical disconnect squib fires, disconnecting the satellite-capsule electrical cable. Cable disconnection lifts a ground loop to start the ejection programmer.

At T - 0 the SS/D timer separation signal ignites the satellite-capsule separation pin-puller squibs. When these squibs have fired (0 to 7 milliseconds delay), four compressed springs mounted around the perimeter of the satellite-capsule interface push the capsule away from the satellite, imparting a velocity of about 1.7 ft/sec.

At T + 1.9 seconds the ejection programmer (Event 1) actuates spin valves which spin the capsule to approximately 55 rpm in 0.8 seconds.

At T + 3.15 seconds, the ejection programmer (Event 2) ignites the capsule solid-propellant retro rocket which imparts approximately 4-g acceleration for about 9 seconds. A vector component of this acceleration, opposite to the orbital direction of the capsule, imparts a deceleration to the capsule which modifies the release trajectory (25,000 ft/sec horizontal velocity) to a re-entry trajectory.

At T + 13.9 seconds the ejection programmer (Event 3) actuates the de-spin valves which reduce the capsule spin rate to about 10 rpm.

At T + 15.4 seconds the ejection programmer (Event 4) ignites the thrust cone-capsule electrical disconnect squib and the thrust cone-capsule separation explosive bolts.

With the action of these explosive bolts, the entire thrust cone assembly is jettisoned. After its separation, the thrust cone follows the recovery capsule closely until re-entry. Tracking of the capsule beacon by Kodiak Tracking Station, begun shortly before satellite-capsule separation, continues until the earth's curvature or r-f blackout precludes further signal reception. During the Ejection Phase the magnetic tape recorder (cf. Par. 3.1.2.6) records the output of the sensors noted in Par. 3.1.2.5.

3.2.4 Recovery Phase. When the capsule re-enters the earth's atmosphere, atmospheric friction decelerates the capsule and heats its surface. At an altitude of about 350,000 feet above sea level, the increasing heat causes a layer of ionized particles to form over the surface of the capsule, resulting in r-f blackout.

At about 200,000 feet, capsule deceleration reaches 5 g, closing a switch to permit actuation of recovery system thermal battery 2A7BE1 by battery 2A8. Battery 2A7BE1 actuates dimple motors to start the recovery system mechanical timer and ignites pyrotechnic switches which; (a) connect battery 2A8 to the mechanical timer switch; (b) disconnect the squib of battery 2A7BE1; and (c) arm thermal batteries 2A7BE2 and 2A7BE3.

As capsule deceleration continues, frictional heat reaches a peak and begins to decrease. At about 135,000 feet, the capsule surface cools to a point at which the ionized layer disappears, ending r-f blackout.

At about 55,000 feet, the mechanical timer switch closes, permitting battery 2A8 to: (a) actuate batteries 2A7BE2 and 2A7BE3; and (b) ignite pyrotechnic delay switches. After a 1-second delay, these switches: (a) disconnect their own squibs and those of batteries 2A7BE2 and 2A7BE3; and (b) unshort the ejection pistons. These switches also apply +12v dc from the capsule radio beacon batteries to the recovery system flashing light beacon. The ejection pistons release the ablative shell from the capsule and blow the parachute cover off.

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The ejected parachute cover pulls the pilot chute out of the parachute compartment, and the pilot chute in turn pulls out the main parachute bag with the chute in reefed condition. The radar reflective chaff is dispersed at this time. After a 4-second delay, the mechanically-actuated time-delay pyrotechnic cutter cuts the main parachute reef lines, permitting deployment. The parachute system further decelerates the capsule and positions it for recovery.

4.0 POWER INPUT REQUIREMENTS

This information is not presently available and will be included in a subsequent revision.

5.0 ENVIRONMENT

The Biomedical Recovery Capsule is designed to meet the environmental criteria of LMSD Report 6117A, General Environmental Specification.

6.0 RESPONSIBILITY

Biomedical Systems Development.

7.0 NOTES

Not applicable

8.0 PAYLOAD INDEX

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W. Wright	PAYLOAD Flight Configuration III DISCOVERER PROGRAM	W. L. [signature]
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The security classification of the actual content and function of the payloads for this configuration, popularly known as Advanced Engineering Test - Heavyweight (AET-H) and Advanced Engineering Test - Lightweight (AET-L), is beyond the scope of the Discoverer Data Book.

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The ejection sequence for Flight Configuration III Biomedical Recovery Capsule is included in the Flight Configuration III payload description (Cf. Page 6.3.1.ff).

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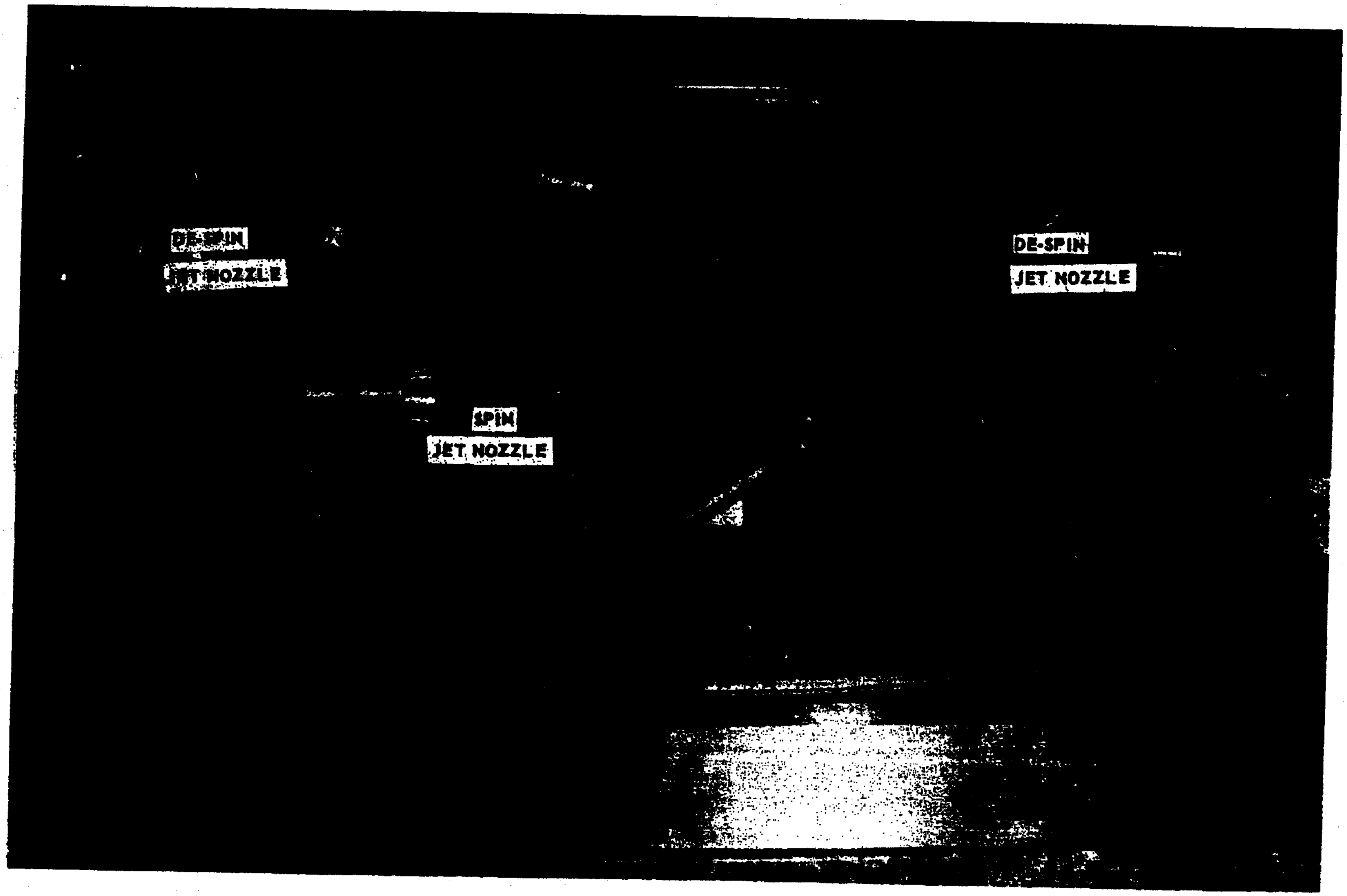

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APPROVED BY	THRUST CONE, EXTERNAL VIEW Flight Configuration VI DISCOVERER PROGRAM	APPROVED BY

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1.0 GENERAL

This section describes the Mark IV Capsule Descent System which is used on Discoverer satellites carrying recoverable capsules. The descent system initiates events following capsule ejection through parachute deployment. Following a discussion of the system is a detailed sequence of events.

2.0 SIGNAL INPUT REQUIREMENTS

Initiation of the Capsule Descent System is accomplished by signals externally supplied by a D-timer located in the Satellite. These signals cause the following actions:

2.1 The telemetry battery and the telemetry transmitter become activated.

2.2 The thermal relays become ignited so a path for power will be available from the thermal battery to the programmer.

2.3 The recovery programmer is armed and the RF beacon is activated.

2.4 Parallel-connected thermal batteries are ignited to power re-entry programmer when ground loop is lifted.

2.5 A 1-second delay squib in the umbilical, between the capsule and the satellite, is ignited so that at the end of the delay the umbilical will separate and lift the programmer ground loop.

3.0 PERFORMANCE

Functions of the Capsule Descent System are performed by two programmer systems: the re-entry programmer, located in the thrust cone, and the recovery programmer, located in the capsule. Both are armed simultaneously at 81.5 seconds before ejection (E-81.5) by a D-timer signal. The letter E is used to indicate time of capsule ejection, and, in this discussion, will precede all events up to re-entry programmer activation, which occurs at E-1.5. However, since the capsule descent system actually starts operating at this time, the letter X will be used to designate this, and will replace E-1.5. Consequently all times following activation of the first programmer will be preceded by plus X. As previously stated, first to start operating is the re-entry programmer shown schematically on page 8.9.4. When the capsule umbilical is disconnected at E-1.5 (see E-1.5 in sequence below), a ground loop is lifted and the programmer initiates the following series of events: capsule spin, retro-acceleration, and de-spin; and separation of the thrust cone from the capsule.

This last event occurs at X+16.9. The recovery programmer system, shown schematically on page 8.9.6, is started approximately 400 seconds following the last re-entry programmer event. This occurs when the free-falling capsule achieves a deceleration of about 6 g's. When this condition is reached, inertia switches in the recovery programmer actuate and start a series of timing circuits which deploy a capsule parachute and turn on a flashing "resculite." An RF beacon and telemetry system are energized at the time of programmer arming, but only the RF beacon continues operating through recovery/retrieval operations. The telemetry system operates for 25 minutes after activation and monitors significant events from the entire performance of the capsule descent system.

3.1 <u>Re-entry Programmer System.</u> Pre-arming of the re-entry programmer system is accomplished during ascent, at 70,000 feet, when a barometric switch, located in the thrust cone, closes. This action provides a path for a D-timer signal, sent at the appropriate time, to fire two 30-volt, 20-second life thermal batteries in the capsule thrust cone. Preceding the battery fire signal, the programmer system receives an arm signal from the D-timer. Arming consists of firing two parallel-connected thermal switches in the thrust cone at E-81.5, causing the switches to close and providing a path between the still unactivated parallel-connected thermal batteries and the re-entry programmer. Also at E-81.5, three additional events occur which prepare the payload capsule and programmers for descent. These are noted under Signal Inputs, paragraph 2.0. At E-2.5, a transfer signal from the D-timer activates the thermal batteries, which then supply 30 volts through the thermal switches to a grounded-out re-entry programmer. At the same time, a signal fires a 1.0 second delay squib in the capsule satellite umbilical disconnect plug, causing the umbilical to separate. Separation of the umbilical removes the ground connection, and permits the programmer to operate. At E-0 or X+1.5, the D-timer fires two squibs, removing the 2 pins holding the capsule to the satellite and allowing four springs to push the capsule away from the satellite. The time relationship between this ejection event and the first re-entry programmer event, capsule spin, is set so the capsule clears the satellite before spin is executed. Spin occurs at X+3.4 when a signal from the programmer causes the neck of the spin gas bottle to break, releasing the compressed gas which rushes through tubing to jets positioned on the outer perimeter of the thrust cone. Orientation of the jets causes the capsule to spin at a velocity of 78 rpm in about 1 second.

The second event occurs at X+4.65, (see sequence below) while the capsule is oriented 60 degrees down from the horizontal. At this angle it is retro-accelerated at about 4 to 5 g's for about 9 seconds by the retro rocket. As the retro rocket loses thrust, the re-entry programmer signals the third event, de-spin, which reduces the capsule spin velocity to about 10 rpm. The reduced rpm is a result of less pressure in the de-spin bottles. The de-spin system is identical to the spin system, but the de-spin jets are oriented so the capsule will tend to revolve in a direction opposite to spin. The fourth and last event separates the thrust cone from the capsule. This is accomplished at X+16.9 when a squib is ignited in the thrust cone/capsule umbilical plug, and electrically separates the thrust cone from the capsule. Simultaneously, a squib is ignited in each of 2 explosive bolts. When the bolts fire, springs push the thrust cone away from the capsule. The capsule free-falls for about 400 hundred seconds and then the recovery programmer begins operation and sequences events which lead to parachute deployment.

<u>Capsule Re-entry Sequence</u>

Time	Signal Source	Event
E-81.5 sec (±.1 sec)	SS/D Timer (arm signal)	1. Capsule T/M Filament on (T/M Batt. activated) 2. Capsule telemetry plates "on" 3. Ignite thermal relays to arm re-entry programmer 4. Capsule beacon "on" 5. Command +45°/min. pitch rate
E-2.5 sec (±.1 sec)	SS/D Timer (Transfer signal)	1. Command +3.55°/min pitch rate 2. Ignite electrical disconnect delay pyro (delay tolerance 500 to 1320 milli-seconds) 3. Ignite re-entry programmer thermal batteries.
E-1.5 sec (±.5 sec +0.32 sec) = X Time 0	Electrical Disconnect Pyro Fires	1. Capsule/satellite cable disconnected 2. Re-entry programmer started (ground loop lifted)

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H. J. Gray	CAPSULE DESCENT SYSTEM Flight Configuration VI DISCOVERER PROGRAM	J. Mitchell
APPROVED BY		APPROVED BY J. Kaplan

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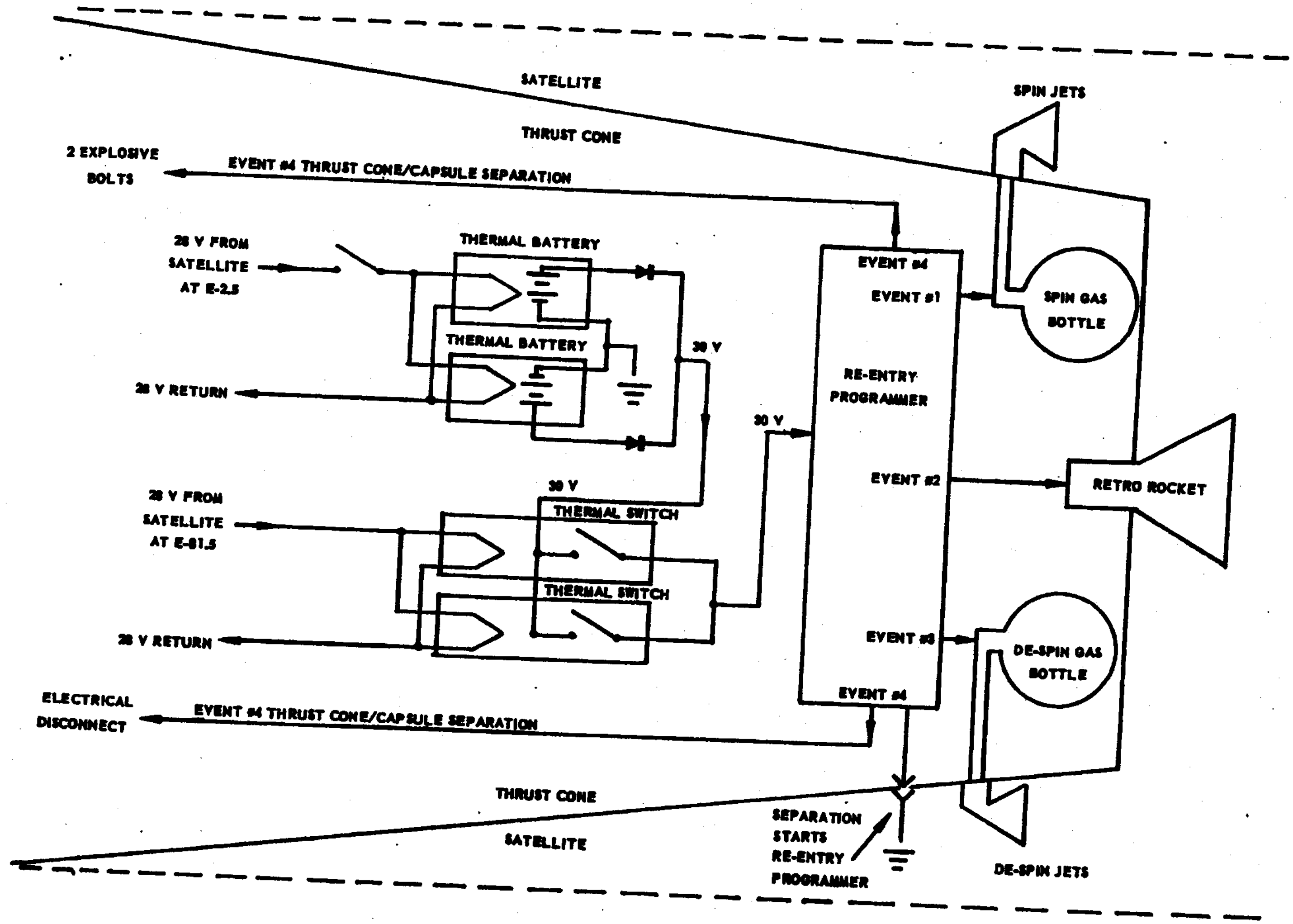

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APPROVED BY	RE-ENTRY PROGRAMMER FUNCTIONAL SCHEMATIC Flight Configuration VI DISCOVERER PROGRAM	APPROVED BY

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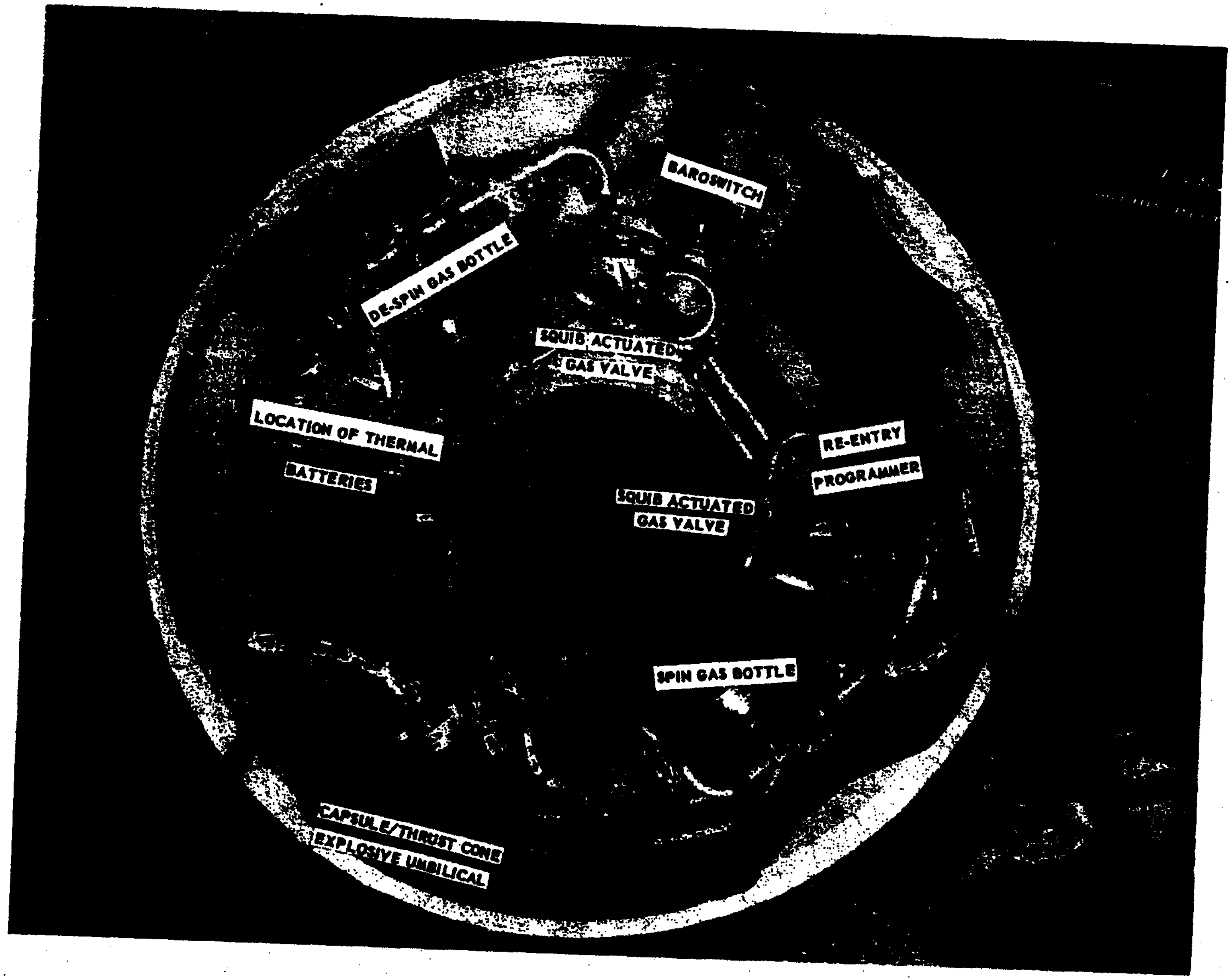

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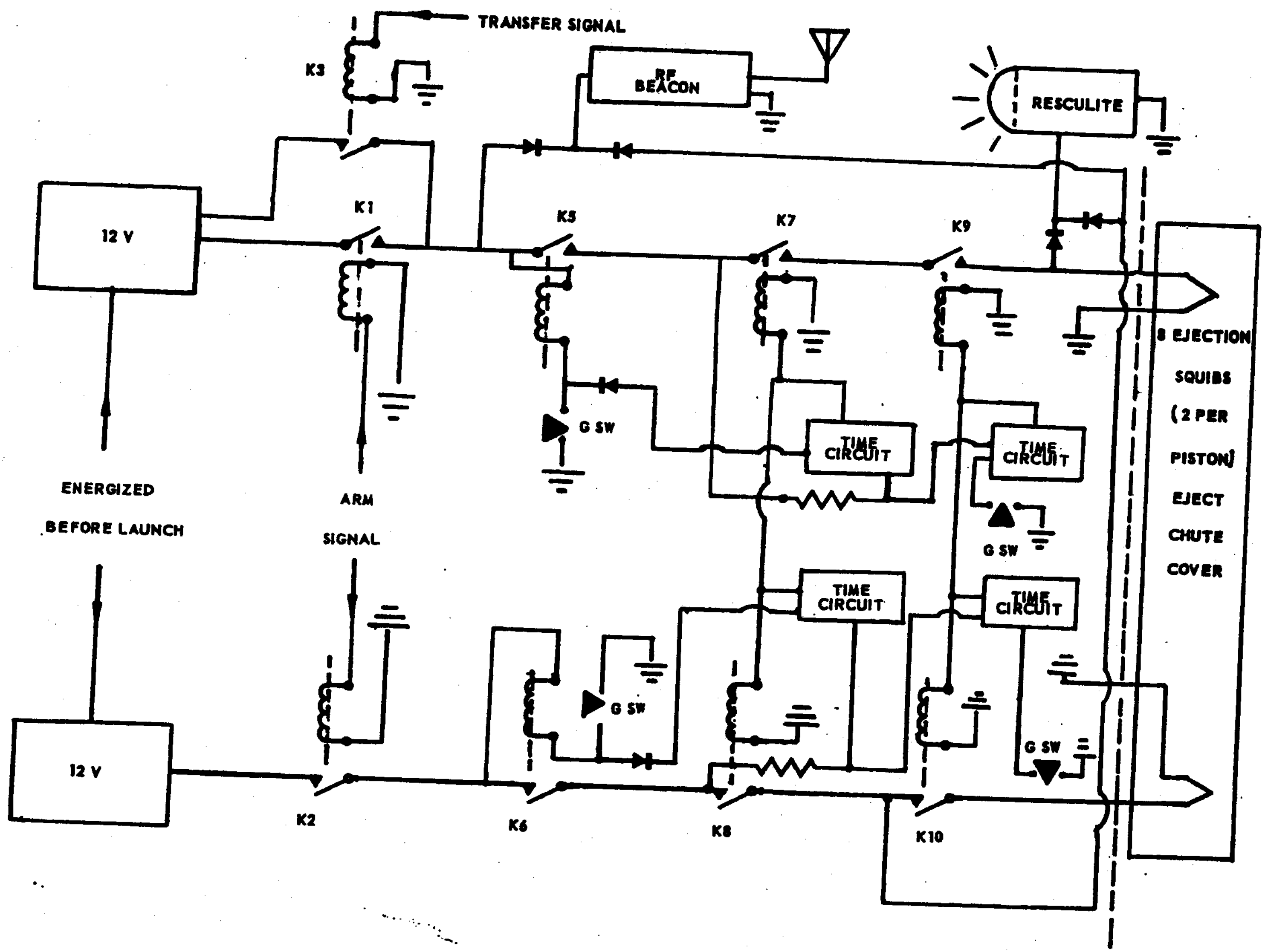
	THRUST CONE INTERNAL VIEW AND LOCATION OF MAJOR COMPONENTS Flight Configuration VI DISCOVERER PROGRAM	



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APPROVED BY	RECOVERY PROGRAMMER FUNCTIONAL SCHEMATIC Flight Configuration VI DISCOVERER PROGRAM	APPROVED BY

Time	Signal Source	Event
E-0 sec X = +1.5	SS/D timer (separation signal)	1. Pin-Puller squibs ignited (0 to 7 milli-seconds delay) 2. Four springs push off capsule to about 1.7 ft/sec
X + 3.4 sec (±.17 sec)	Re-entry Programmer Event 1	1. Spin valve actuated, capsule spins up to about 78 rpm in 0.8 sec
X + 4.65 sec (±.27 sec) (Tolerance from prev. event ± .1 sec.)	Re-entry Programmer Event 2	1. Retro-rocket ignited capsule receives approximately 4 to 5 g's acceleration for approximately 9 sec.
X + 15.4 ±.81 sec (Tolerance from prev. event ±.54	Re-entry Programmer Event 3	1. De-spin valve actuated, capsule de-spins to about 10 rpm.
X + 16.9 sec ± .96 (Tolerance from prev. event ± .15	Re-entry Programmer Event 4	1. Ignite electrical disconnect and explosive separation bolts

3.2 Recovery Programmer System. Preceding turn-on of the recovery programmer, a D-timer signal latches relay K1 and K2 at E-81.5, and, as shown on the recovery programmer functional schematic, page 8.9.6, energizes the RF beacon. At E-2.5 K3 is latched and serves as a back-up for K1. Actual recovery programmer operation begins when the free falling capsule has reached a 6-g deceleration, approximately 400 seconds after ejection. This deceleration causes the g-switches in the programmer to close and latch K5 and K6. Although power is now available to the timer circuits, the timer does not operate because the power is shorted through the inertia switches to ground. At approximately E+500, capsule deceleration has reached 3-g's and the inertia switches open. Now full power is supplied to the timing circuits and the timing sequence begins and continues for 47 seconds. At the end of this time, the timing circuit supplies power to energize and latch relays K7, K8, K9, and K10. Closing of the relays furnishes power to light the "resculite" and fire four ejection pistons, which blow off the chute cover and pull the pilot chute out. The pilot chute brings out the main chute bag in a reefed condition and at the same time activates a 4-second delay pyrotechnic reef cutter. At the end of 4 seconds, the reefing lines are cut and the main chute deploys fully. Chaff, enclosed in a package in the pilot chute bag, is blown out as the pilot chute deploys. As the chute decelerates the payload capsule, the ablative shell, released from the capsule when the ejection pistons fired, falls clear of the capsule. While descending by parachute, the payload capsule signals its position by RF beacon to a searching Recovery/Retrieval force. In the event of water impact, the "resculite", in addition to the RF beacon, will help direct the Retrieval force to the capsule.

3.2.1 Capsule Recovery Sequence

Time	Approximate Altitude	Event
G switches close at between 3 & 6.6 G's increasing and open again at 3 G's ± 5% decreasing.		1. Recovery sequence initiated by G switch opening. (Programmer started)
T + 515 (47.0 ± 1.7 sec after G switch opens) approx.	55,000 ft	1. Recovery programmer transmits Recovery signal. 2. The ejection pistons blow off the chute cover, which pulls out the pilot chute, which in turn pulls out the main chute bag. The main chute bag brings out the chute in a reefed condition. 3. Time delay pyrotechnic cutter disreefs the main chute and permits deployment (4 sec). 4. As the chute system decelerates the capsule, the ablative shell, released from the capsule when the ejection pistons fired, falls clear of the capsule. 5. Relay K9 and K10 in programmer supply + 12v (from batteries contained in the capsule) to the light beacon 6. Radar reflective chaff, packed with the pilot chute, falls free as the chute emerges from its bag.

3.3 Major Components. Major components used in the capsule descent system, in addition to the two programmers discussed, are described in the following paragraphs:

3.3.1 Antenna System, Mark IV. The Mark IV antenna is a beryllium-copper, 1/4 wave whip which tapers from about 1/4 inch diameter at the base down to about 1/8 inch diameter at the tip. This is a semi-rigid antenna and mounts on the ring on the top edge of the bucket. The RF feed passes vertically down through the lip and makes a right angle bend to enter the side of the bucket. Two antennas are used: one for the telemeter, and the other for the beacon. The tip of the antenna extends about 4 inches above the upper edge of the ablative shell. In the recovery mode the antenna serves as a conventional 1/4 wave whip.

3.3.2 Telemetry System. The telemeter contains 3 continuous channels: 7, 9, and 11. It transmits on a frequency of 228.2 mc on an FM bandwidth of 100 KC. Channels 7 and 9 monitor a series of breakwire spin and de-spin events plus the thrust cone and Beacon (Recovery) battery voltage. When the thrust cone is ejected, a relay switches to a series of recovery events related to the chute ejection system. Channel 11 contains a ± 5 g accelerometer to measure retro rocket thrust during cut-down, and to indicate chute deployment during recovery. The telemeter becomes energized when the 28-volt battery becomes remotely activated by the satellite arm signal. Its operating life is about 25 minutes.

3.3.3 Recovery System Batteries. The recovery system batteries are part of a battery pack which consists of two independent 14 volt supplies. Each supply is rated at 5 ampere-hours and has a storage life after activation of 15 days (including flight) at 60°F (higher temperature shortens life). Activation is accomplished prior to launch by the addition of electrolyte. After chute deployment, this battery pack furnishes power to the RF beacon and "resculite" for a minimum of 10 hours.

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3.3.4 Radio (RF) Beacon. The radio beacon uses a dc-dc converter supplied by the capsule 12-volt recovery battery. The beacon is a sealed unit and weighs about 2.7 pounds. It will radiate a peak power of 15 watts. It transmits on a crystal controlled frequency of 235 MC ±.01% and is pulse modulated with a coded signature.

3.3.5 Radio Signature. The radio signature is a small modulator unit which can be used in conjunction with the crystal beacon. Its purpose is to create in the beacon signal a distinctive sound easily recognized by the recovery force. The 1000 cps pulse signal is modulated by a saw-tooth with a 1 second period. The 1000 cps signal is thus varied exponenetially from a nominal 1000 down to a nominal 750 cps. The resultant sound somewhat resembles the meow of a cat.

3.3.6 Resculite. The resculite aids surface vessels and search aircraft in case air recovery is unsuccessful and it becomes necessary to retrieve the capsule from the ocean. The light has a minimum intensity of 0.8 million lumens and a flash rate of 60-90 per minute for a duration of .005 second. Average power consumption is 450 mw, and the weight is 6 ounces, excluding batteries. Power is furnished from the recovery battery when the capsule parachute is deployed. Tests have demonstrated a visual range of 8 nm from 10,000 feet on a dark night.

4.0 POWER INPUT REQUIREMENTS

Not Applicable

5.0 ENVIRONMENT

The Capsule Descent System is designed to meet the applicable requirements of LMSD Report No. 6117B General Environment Specification.

6.0 RESPONSIBILITY

SS/L

7.0 NOTES

Not applicable

8.0 CAPSULE DESCENT SYSTEM INDEX

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LOCKHEED AIRCRAFT CORPORATION

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MISSILES AND SPACE DIVISION

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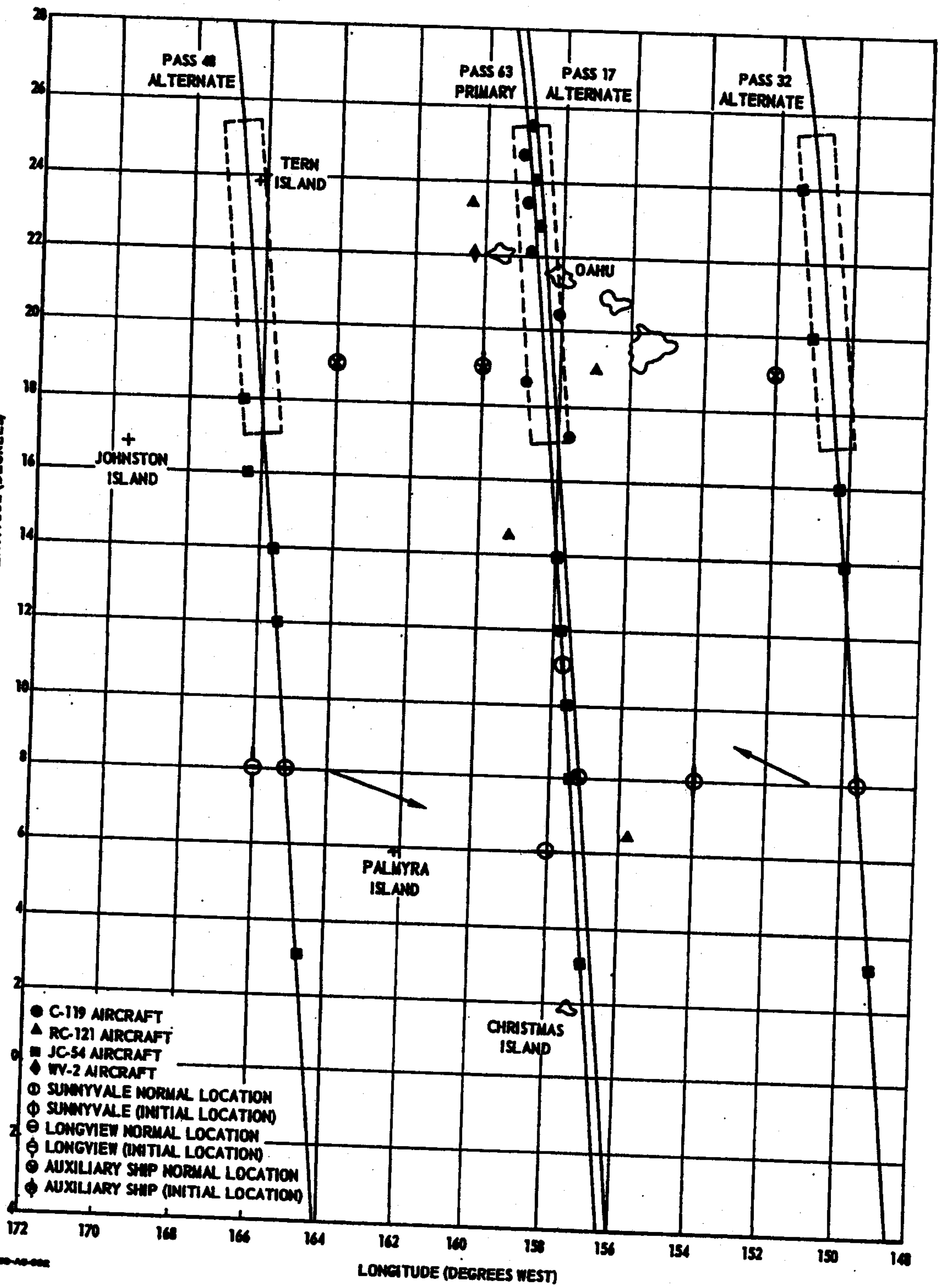

TITLE
RECOVERY/RETRIEVAL FORCE
DEPLOYMENT
DISCOVERER PROGRAM

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1.0 GENERAL

Purpose of the recovery/retrieval program is to recover the capsule in the air or, if the air snatch fails, to retrieve the capsule from the water. To accomplish this purpose requires the use of the Satellite Test Center and the Recovery Control Center, area tracking stations, and the deployment of a recovery/retrieval task force consisting of a number of aircraft and surface vessels. This task force will be deployed in primary (60 x 200 nm), secondary (60 x 330 nm) and extended areas, but the force will attempt the initial air recovery in the secondary area. Capsule ejection and impact is programmed prior to launch for an initial re-entry pass within the Hawaiian area, typically on pass 63, or 64 for back-up. However, conditions during flight, such as poor weather, satellite electrical or control gas depletion, may require re-programming for alternate or emergency re-entry. Alternate re-entry would typically occur on pass 17, 32, and 48 within the Hawaiian area, and emergency re-entry on typical passes 30, 46, and 61 off the coast of Mexico. Satellite progress and capsule re-entry is monitored by tracking stations and communicated to the STC where the specific re-entry pass is determined.

2 RECOVERY/RETRIEVAL OPERATION

The recovery/retrieval operation is conducted from the RCC, which in turn receives direction and information from the STC. When the STC notifies the RCC of the specific programmed re-entry pass, the RCC then deploys the Recovery/Retrieval Force to the proper location. Support of the Recovery/Retrieval Operation is supplied by ground stations which furnish radar coverage, communications, and fuel supply

The Recovery and Control Center (RCC) at Hickam Air Force Base, Hawaii, controls recovery operations in the Hawaiian area. Here status boards, charts, and maps provide an accurate status of all phases of the recovery operation. The RCC is also provided with equipment for continuous communication with the STC and the Recovery Forces.

The Hawaiian Tracking Station (HTS) reports confirmation of capsule ejection and telemetered recovery events to the STC and records all capsule telemetry signals on magnetic tape. The station also receives all antenna bearings from South Point, Barking Sands and Tern Island for correlation with the TLM-18 bearings. This bearing information is relayed immediately to the STC and the RCC.

Tern Island establishes SSB voice and teletype communications with HTS on the assigned recovery operations frequencies for acquisition and tracking instructions and data relay. The Island has an automatic track quad-helix antenna which will track capsule signals and transmit the data to the RCC and STC.

South Point facility uses either a motor-driven quad-helix or canoga quad-helix for search operations and relays data via toll phone to the HTS.

Christmas Island maintains continuous HF communications with the southern telemetry aircraft for exchange of acquisition and tracking information and relays this information to the RCC over SSB radio. A quad-helix antenna is used on the facility for tracking the capsule.

Barking Sands is a PMR facility with a tri-helix antenna installation for tracking the capsule, and a toll phone connection to HTS for communicating tracking data.

Johnson Island maintains a fuel supply for aircraft requiring re-fueling if recovery is made in that area.

2.1 RECOVERY

The Typical Recovery Force, deployed as shown on page 1.0.2, is composed of: 9 C-119J aircraft for catching the capsule in the air; 4 RC-121 aircraft for communications and radar search; JC-54 for capsule detection; SC-54 for para-rescue; 1 WV-2 aircraft for frequency interference control. This force is deployed to provide a maximum air recovery capability in the 60 x 200 nautical mile primary recovery area, but capsule detection will be emphasized within the 60 x 330 nautical mile secondary recovery area. Under normal conditions, the RC-121 aircraft depart for the recovery area so as to be on station approximately three hours before Estimated Time of Parachute Deployment (ETPD). The task force commander aboard one of the RC-121 craft departs at such time as to be on station one hour before ETPD. Communication and time checks are conducted between the command RC-121 aircraft and the surface ships as they rendezvous prior to assuming their assigned positions. Search flight patterns, established by the RCC, provide a straight flight path for the RC-121 aircraft from ETPD -10 minutes to ETPD +20 minutes.

2.1.1 Typical Deployment. The RC-121 telemetry detection aircraft will be deployed along the satellite flight path. Each aircraft is equipped with an APS-95 surveillance radar and APS-45 height-finder radar with a reception range of 120 to 150 nautical miles. The aircraft itself has a range of approximately 3800 nautical miles with an endurance of approximately 17 hours. Three of the RC-121's will be deployed in the northern area to provide overlapping radar coverage of the primary and secondary air recovery areas. A fourth RC-121 aircraft will be deployed in the extended surface recovery area to provide communication control of the forces within that area. At the time of deployment, one of the 4 RC-121 will be designated as command aircraft for the forces in each of the three operational areas. In the event that one of the four RC-121's aborts the mission, the three remaining aircraft will be deployed to assure continuous radar coverage of the primary and secondary recovery areas at the sacrifice of the extended communications control aircraft position. In this event, alternate communications, over HF and Single Side Band (SSB) radio for force control and data talking, will be established between Christmas Island and detection aircraft in the extended recovery area. Six C-119J recovery aircraft will be deployed in the primary recovery area and the remaining three C-119J in the secondary recovery area. Each is equipped with Model 80C air pickup equipment and an FTR-2 Discoverer direction finder for homing on the capsule beacon. By using their auxiliary fuel tanks the C-119J aircraft can travel a distance of 2400 nautical miles at 160 knots true airspeed with a resultant endurance of approximately 15 hours and a remaining reserve of 2 hours of fuel. A WV-2 aircraft will perform a frequency interference (FIC) survey of the predicted impact area and will assume a final position 120 nautical miles south and 100 nautical miles west of the predicted impact point by ETPD -30 minutes. The FIC aircraft will communicate with the primary recovery area command aircraft on the northern primary high frequency, will search for the capsule signals, and will attempt to derive a HF bearing from any of the signals acquired. All telemetry signals received will be recorded. Signal acquisitions and bearings will be reported immediately to the Command RC-121.

2.1.2 Typical Operation. An Electra begins search operations at 10 minutes before capsule separation with receivers tuned for acquisition of the capsule telemetry signal on 280.2 mc and the vehicle telemetry signal on 237.8 mc. If the satellite telemetry signal is received first, reception of this signal is optimized until acquisition of the capsule telemetry signal. Tracking or search is terminated at signal fade or 10 minutes after capsule separation if no signals are received.

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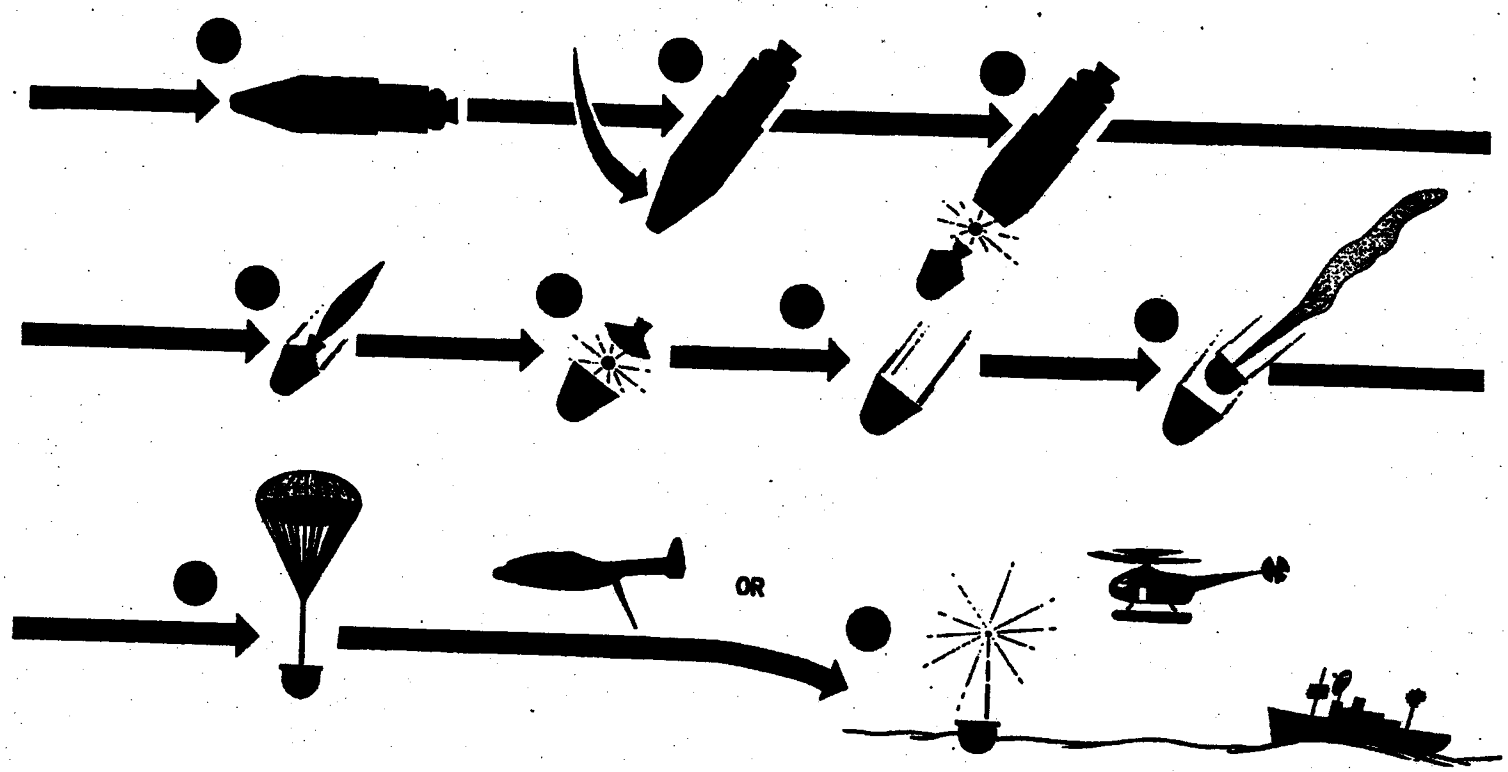

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Immediately following fade of the telemetry signal, the Electra advises the KTS via SSB radio the time of signal acquisition and fade to the nearest second, in GMT and the deviations from nominal frequencies. The Electra will maintain radio silence while receiving telemetry data to avoid r-f interference with data recording.

At approximately 50,000 feet, the recovery capsule parachute will deploy, releasing chaff for radar acquisition; concurrently, a flashing light will be energized. The RC-121 will search and should acquire the chaff as the first radar return. The C-119 recovery aircraft will use the Discoverer direction finder (DDF) to search for and home on the capsule beacon signals. All returns from the recovery force radars and direction finders will be verified as soon as possible to eliminate possible "bogeys". Direction-finder acquisitions by the C-119J aircraft will be plotted to verify that only one intersect point exists. Direction-finder returns will also be checked against radar returns of the RC-121 aircraft and will be used to verify the returns. If "bogey" signals still appear to exist after verification, the Task Force Commander will conduct a systematic visual search for the source of each signal. The director aircraft will not vector the C-119 aircraft on the basis of radar returns if DF bearings are available. When the recovery aircraft visually acquires the recovery capsule parachute, an air recovery will be attempted. Repeat passes will be made, if necessary, until either recovery is effected or the recovery capsule impacts in the water. The C-119J aircraft effecting recovery will return to Hawaii as directed by the RCC and will be escorted by either a C-119J, an RC-121, or an SC-54 (air rescue) aircraft. If the capsule is neither sighted nor acquired by radar, the aircraft will continue to search for chaff or acquisition beacon signals in accordance with previously established search plans until directed by the RCC to terminate the search. Should the air recovery be unsuccessful, the search aircraft will, after sighting the capsule, circle the area of water impact and drop marker aids. The aircraft will be equipped with strobe light bombs, smoke bombs, and dye markers for this purpose. In addition, certain C-119 aircraft will be equipped to provide a beacon marker. The capsule beacon and flashing light minimum operating life is 10 hours and the capsule will float for a minimum of 45 hours. Two parachute teams of the Air Rescue Service will be utilized as a primary capsule water retrieval element of the recovery force and will be deployed by the Recovery Test controller at the RCC. If the capsule has not been located by RTFD +30 minutes in the primary recovery area or RTFD +35 minutes in the secondary and extended recovery areas, the RCC will direct the airborne recovery forces to initiate search, based on the latest impact prediction received from the STC. If this is not available, the RCC will direct a search of the most probable impact areas as determined from tracking triangulation and other available data. If possible, the parachute deployment telemetry sequence will be reported when received and the telemetry aircraft will attempt to determine the capsule bearing at fade or at parachute deployment. If this can be accomplished, the bearing and aircraft position will be reported to the area command RC-121. If one of the telemetry aircraft visually acquires the capsule in the air or in the water, the position will be reported immediately to the area command RC-121. The telemetry aircraft will circle the capsule while maintaining visual contact until arrival of a surface vessel or until fuel supply requires return to base.

2.2 RETRIEVAL

Three surface ships, the USNS Longview, the USNS Sunnyvale, and an auxiliary recovery ship constitute the typical retrieval force and will be deployed within the retrieval area boundaries to assure surface retrieval support for all variations of the orbit period. The ships will depart with sufficient time to arrive at initial deployment stations by T +4 hours. The Pacific Air Command (PAC) will evaluate the tracking data after launch and will provide predicted capsule impact locations and times for each day not later than T +4 hours. On receipt of impact predictions, the STC will provide surface ship redeployment instructions to the RCC.

2.2.1 Typical Deployment and Operation. The USNS Longview and Sunnyvale have speeds in excess of 15 knots and sufficient supplies for 30 days. They will be deployed in the extended recovery area primarily to provide a detection and surface retrieval capability in the event of capsule overshoot, and will be deployed laterally to provide coverage by one ship for each alternate day pass. If the period is such that both ships can cover the primary recovery pass, the easterly ship will be redeployed to 11°N latitude and the westerly ship to 6°N latitude. Initial stations will permit redeployment of either ship to cover its assigned area in one day. For the alternate pass, the Sunnyvale would cover passes 17 and 32. The Longview would move east of its pass 48 on-station position, to a point that would permit it to return to its on-station position and be ready prior to capsule separation. At RTFD -6 hours, if the decision is made to re-enter on the primary pass, the ship will steam south to 6°N latitude as close to the orbit tracks as possible. Either one or both of these ships will be positioned directly under the orbit plane on the recovery pass to enable reception of the signal capsule signals in the extended recovery area. Until acquisition, the quad-helix antenna scans ±90 degrees about 360-degree azimuth, 10 degrees elevation, at the rate of once per 15 seconds, from RTFD -0 until RTFD +2 minutes. From RTFD +2 minutes until RTFD +5 minutes, the quad-helix antenna gives full area coverage by scanning ±90 degrees about 360-degree azimuth, with antenna elevation increasing and decreasing from 10 degrees to 80 degrees in increments of 20 degrees per scan. The scan rate is once per 15 seconds. After RTFD +5 minutes the antenna is positioned at 10 degrees elevation and 180-degree azimuth. In the event that either ship acquires the capsule signals, the telemetry will be recorded on magnetic tape and antenna acquisition and bearing will be immediately reported through PMR to the RCC. Bearings will be relayed to RCC at intervals of one minute. When the parachute deployment telemetry sequence is received or when the antenna azimuth becomes constant, the ship will report verbally over SSB radio through PMR to the RCC and provide ship position, and antenna azimuth and elevation. If no capsule signals are acquired by the ships, a negative verbal report will be submitted over SSB radio through PMR to the RCC at RTFD +30 minutes

A destroyer class auxiliary recovery ship will be used for surface recovery of the capsule or pickup of the pararescue team, should it be necessary. Ship-to-shore communications will be maintained with the RCC/PMR representative for direction. Daily deployment of this ship will be in the primary area.

2.2.2 Ship Equipment. USNS Longview and Sunnyvale each carry HSS-3 helicopters, 360-degree Discoverer direction-finding equipment, and SPS-29 radars. The Longview is also equipped with an SPS-8A height-finder radar. The HSS-3 helicopters have an operating range of 75 nautical miles (limitation when operating in pairs) necessary equipment for effecting water retrieval of the re-entry capsule. Primary mission of the helicopters is to extend the visual acquisition and surface retrieval range of the ships. If operating singly, the helicopters remain within visual range of the retrieval ships for maximum safety since they do not carry DF equipment. Each ship is provided with necessary equipment for handling and storing the re-entry capsule and are both equipped to provide winds aloft weather data. Each ship is provided with the following telemetry receiving equipment: one tape recorder, one NC 1403 and two NC 1300A telemetry receivers, one WWV timing receiver, one quad-helix antenna, and communications HF (SSB/AM/RTTY/CW); VHF; UHF.

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3.0 RECOVERY/RETRIEVAL OPERATION INDEX

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	RECOVERY/RETRIEVAL OPERATION Flight Configuration VI DISCOVERER PROGRAM	J. Muhrl
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The following section describes the Biomedical Support Complex and the units which constitute this complex. Installed at Vandenberg Air Force Base (VAFB), this complex is utilized in aeromedical research for the preparation of specimen and specimen-affiliated equipments for biomedical space launches.

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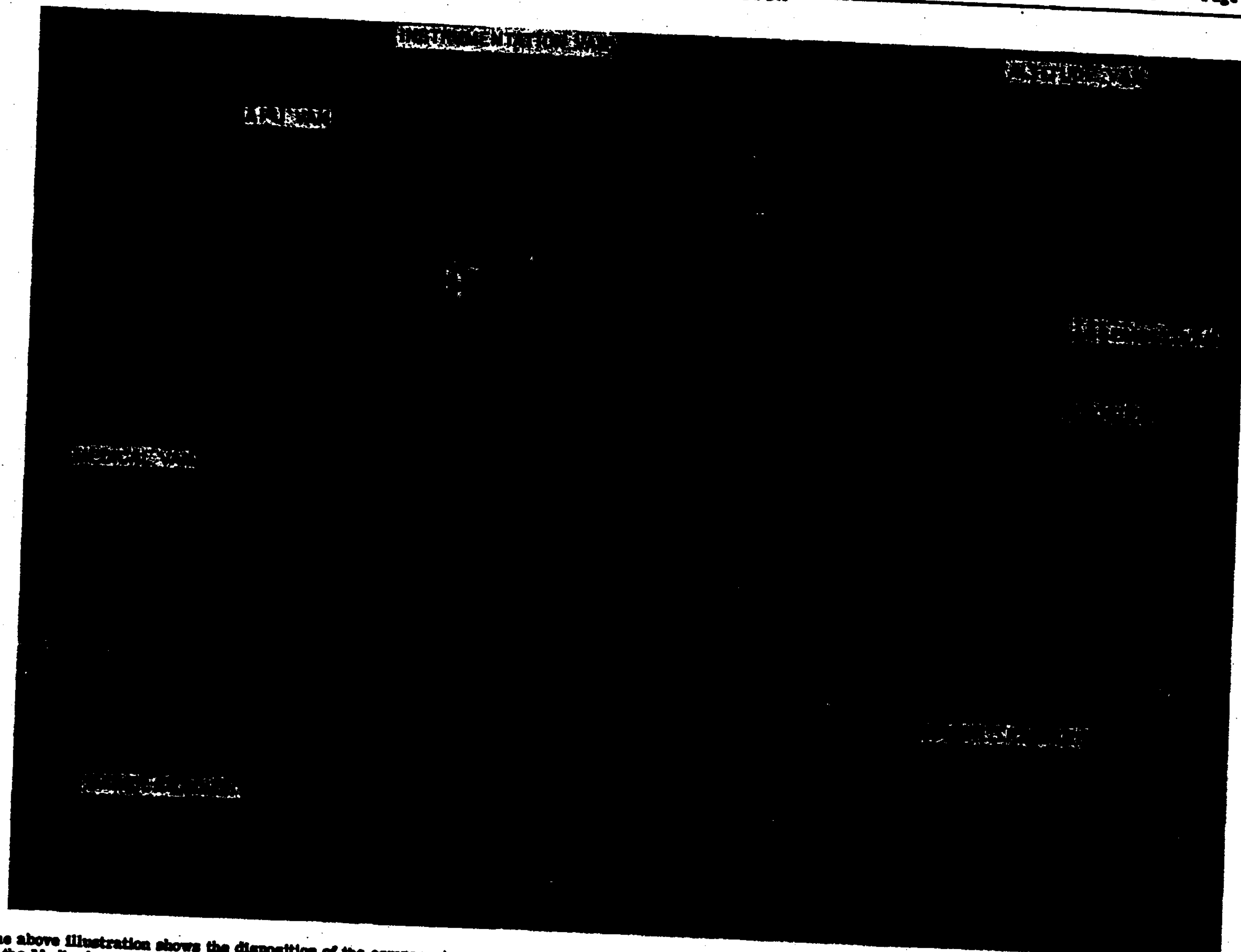
APPROVED BY	MEDICAL SUPPORT COMPLEX Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY



The above illustration shows the disposition of the components of the Medical Support Complex.

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1.0 GENERAL

The Medical Support Complex (LMSD DCD 1062085) consists of two (2) medical vans, an altitude chamber van, an instrumentation van, an assembly van, and an auxiliary power unit van. This complex is used in aeromedical research and in preparing for bio-medical space launches.

2.0 DESCRIPTION

The medical support complex consists of the series of vans, each an air-transportable semi-trailer. These trailers have Type III mobility in accordance with MIL-M-8090A. The vans are designed to provide highly mobile and self-contained facilities for aero-medical support. Living space for research personnel and for medical specimens is provided. The biological laboratory includes X-ray equipment. A high-altitude chamber can simulate the effects of rarified atmosphere. The instrumentation van contains medical electronic instruments and recording equipment. A small machine shop is provided together with the necessary water and power supplies and sanitation facilities. All vans are equipped with communication systems for communication both within the complex and outside. The water supply may be from a base water supply or from government-furnished water supply trucks. The electrical power to the complex may be from an external source or may be generated in the auxiliary power unit. All electrical cabling passes to the complex units through the auxiliary power unit.

3.0 DESIGN CRITERIA

Electrical requirements, complex	480-volt, three phase, 60-cycle 148 KW
Power requirements, components	
Medical vans, each	36 KW
Assembly van	21 KW
Altitude chamber van	30 KW
Instrumentation van	25 KW

4.0 REGULATIONS AND SPECIFICATIONS

MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements, General Specification for
LMSD 6117	General Environment Specification for Satellite Programs
LMSD 6244	Ground Handling and Service Equipment Test Phase Performance Specification for Discoverer Program
LMSD DCD 1062070	Medical Van
LMSD DCD 1062071	Altitude Chamber Van
LMSD DCD 1062073	Assembly Van
LMSD DCD 1062072	Instrumentation Van
LMSD DCD 1062086	Auxiliary Power Unit Van
LMSD 1072274	Acceptance Test Specification, Medical Support Complex

5.0 RESPONSIBILITY

Ground Handling and Service (Medical, altitude Chamber, Assembly Vans)

Checkout Equipment Development (Instrumentation, APU Vans)

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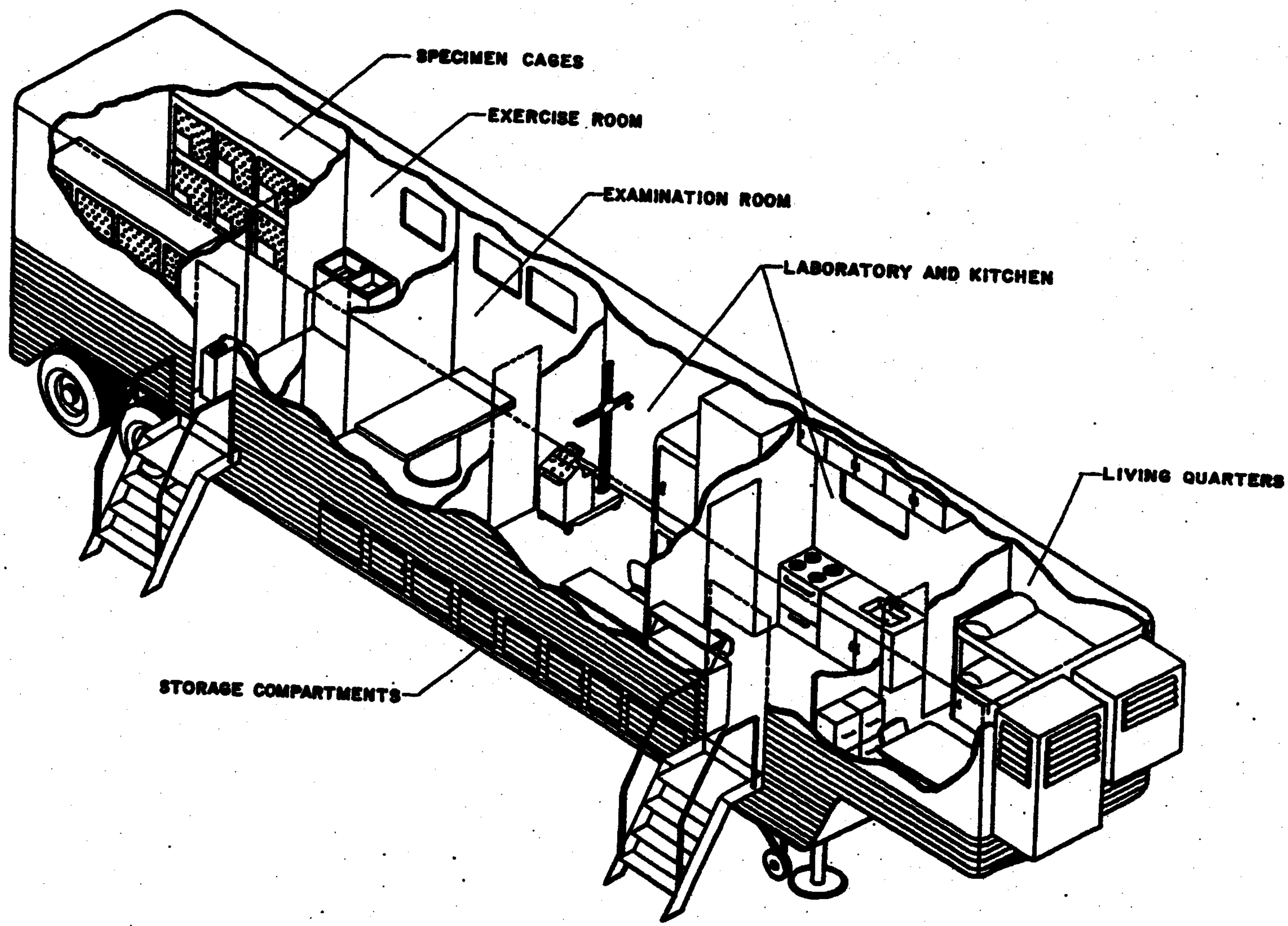

The above illustration shows the arrangements in the Medical Van, which is part of the Biomedical Support Complex. The van is used to conduct whatever pre-flight and post-flight operations on specimens that may be required.

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1.0 GENERAL

The Medical Van (LMSD DCD 1062070) is a mobile facility for the handling, storing and preparing of specimens for space travel. The van is a semi-trailer with accomodations for personnel, cages for specimens, and medical facilities to conduct whatever pre-flight and post-flight operations that may be required. The medical van is a unit in the Medical Support Complex.

2.0 DESCRIPTION

The medical van is a semi-trailer, equipped and outfitted to conduct biological experiments on specimens. Its equipment includes an operating room, complete with X-ray capabilities. The 49-foot long aluminum body is glass fiber insulated and is subdivided into the following compartments (areas): cage, exercise, examination, laboratory, kitchen, and living. The under carriage consists of 20,000-pound spring-suspended bogie-type tandem axle with dual wheels equipped with air brakes and manually-operated brakes. The van is provided with a fifth wheel assembly compatible with a standard military tractor-type truck. A retractable caster assembly is provided to support the van whenever detached from the tow vehicle. Four leveling jacks insure operational ability on uneven terrain. The trailer includes the following systems: air-conditioning and heating, sewage, electrical, water, and communications.

2.1 Compartments (areas).

2.1.1 Cage Area. The after 7 1/2 feet of the trailer is the cage area, separated from the adjacent space with a sliding door. The cage area is lined with stainless steel to facilitate cleaning by means of a pressure hose and hot water. Four tiers of cages, two against either side, are found in the cage area. Access to the cage area from outside the trailer is through a double door in the rear of the trailer. A sliding door provides access to the cage area from the exercise area.

2.1.2 Exercise Area. The area immediately forward of the cage area is the exercise area. The 5-foot long section of the trailer is also lined in stainless steel to facilitate washdown with pressurized hot water. A covered, divided wash basin and hose reel are mounted and plumbed in the exercise area. A single outside access door enters this compartment through the right side of the trailer. Sliding doors separate this compartment from both adjacent spaces.

2.1.3 Examination Area. The 7 1/2 foot compartment forward of the exercise area is the examination area. This area is also lined with stainless steel for ease of cleaning as for the other similarly lined spaces. A foot pedal hydraulic elevated autopsy table is mounted centerline, long table dimension athwart the compartment. Portable floating arm safelight and autoclave are provided. A medical cabinet, wash basin, and hose reel complete the furnishings of this space. The wall and door between the examination and exercise areas are provided with lead shielding compatible with the X-ray equipment used in the trailer. Sliding doors separate this compartment from adjacent spaces.

2.1.4 Laboratory Area. The laboratory area is the next compartment forward. This 9-foot long space contains the X-ray equipment and X-ray developing room, a personnel stall shower, 100-gallon hot water heater, refrigerator, and surgical clothes closet. The developing room is enclosed by an inside black velvet curtain and an outside sliding door. The water heater is enclosed with an external access door opening in the left side of the trailer. Sliding doors separate compartments.

2.1.5 Kitchen Area. The kitchen area, 7 1/2 feet long, lies forward of the laboratory area. Along the right wall of the compartment there is a small, fully equipped chemical laboratory, personnel access door opening outward, and a house trailer type lavatory. Across the compartment from the lavatory is a stowage closet for personal clothing and gear. Both closet and lavatory are enclosed by accordion type folding doors. The remainder of the left wall is taken up with the galley which includes a four burner 30-inch, apartment-type electric stove; sink and cabinets.

2.1.6 Living Area. The living area, the forwardmost compartment, is separated from the kitchen area by a sliding door. Seven feet long, this space contains two bunk beds along the lefthand bulkhead, a small table, two chairs, and a filing cabinet. Additional storage space is provided beneath the bottom bunk and over the table. Since the living area is located over fifth wheel, its deck is higher than the other trailer spaces, and hence a step is required in the kitchen area to enable access to the living area.

2.2 Systems.

Air Conditioning and Heating. The medical van is divided into two zones for air-conditioning and heating. Each zone is supplied by a separate air-conditioning unit, mounted on the forward external bulkhead of the van. Distribution of the conditioned air is via insulated ducts to the various compartments. Zone I includes the specimen areas - cage, exercise, and examination areas. Zone II consists of the laboratory, kitchen, and living areas, those areas intended for human habitation.

Sewage System. The medical van is equipped with two independent sewage systems. A 400-gallon stainless steel tank receives waste from the cage, exercise, and examination areas, and from the stall shower, laboratory area, kitchen sink, and the photographic dark room. A 90-gallon stainless steel chemical tank receives human waste. A sewage transfer pump is provided for emptying the larger tank into a tank truck. Provisions are also made for cleaning and flushing each system.

Electrical System. The trailer receives 480-volt, 3-phase power through cabling from the Auxiliary Power Unit Van. Additional 220-volt, 3-phase and 110-volt power is available within the trailer from the 480-volt source. The X-ray unit requires 220-volt; the 110-volt is for the lighting and appliance outlets. Lighting throughout the van is fluorescent; however, the lighting in the cage, exercise, and examination areas further meets the requirements for medical operating rooms. Flooring throughout the trailer is of the non-static type. The power requirements for the trailer are 36 KW.

Water System. Water is received through a 1 1/2-inch quick-disconnect coupling from an external source. Hot water is generated in the trailer's own 100-gallon water heater. Cold water is routed to the lavatory, to the water heater, and, with hot water from the heater, to the various spaces in the trailer. The spaces in the trailer and the units receiving hot and cold water are as follows: exercise area - hose reel and divided wash basin; examination area - hose reel and medical cabinet wash basin; laboratory - shower and developing lab; kitchen area - chemical lab and kitchen sink. A 90-gallon pressurized water storage tank provides water during periods when water from an external source is unavailable.

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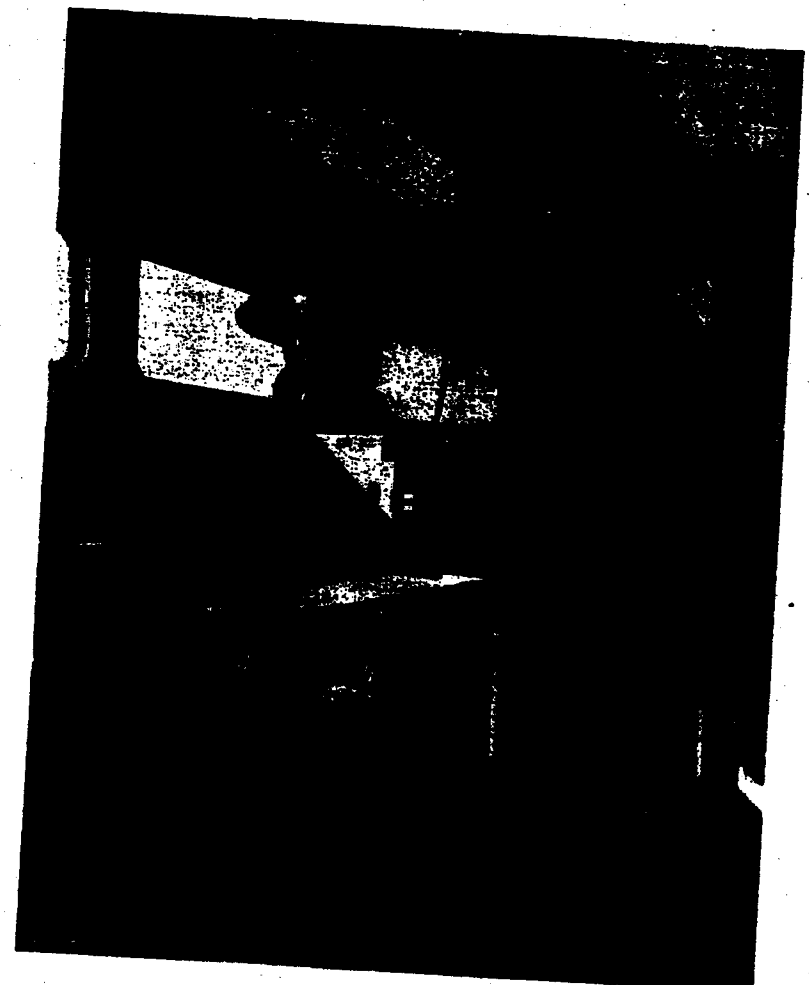
APPROVED BY	MEDICAL VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY



The above illustrations show the cage area and examination area in the Medical Van. Specimen are housed in the cage area during various testing phases; the pre-flight and post-flight operations are conducted in the examination area.

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2.2.5 Communications. The trailer incorporates an inter-communications system to facilitate communication between all areas of the van. The system can deliver and receive sound without the actuation of switches. Through external connections the medical van's communication system is tied into an intra-complex communications system. The van may be equipped with a telephone to connect with the base telephone system.

2.3 Mobility. The trailer is capable of being towed by a standard Air Force tractor equipped with a fifth wheel assembly. The trailer has 14" ground clearance for Type III mobility in accordance with specification MIL-M-8090A. The van is air transportable via C-133 in accordance with Specification MIL-A-8421A.

3.0 DESIGN CRITERIA

Electrical requirements	480-volt, 3-phase a.c. 36 KW
Water Heater	100-gallon, electric
Internal pressurized water tank (standby supply)	50-gallon
Sewage tanks	
General waste	400-gallon
Human waste.	50-gallon, chemical
Air-Conditioning	
Summer design conditions	105°F dry bulb, 80°F wet bulb
Winter design conditions	20°F dry bulb
Zone I requirements (specimen)	±2°F, ±5% RH to preset conditions Limits: 40-100°F; 40-80% RH
Zone II requirements (human)	Normal human comfort conditions
Weight	16,000 pounds
Dimensions	49' x 8' x 11 1/2'

4.0 REGULATIONS AND SPECIFICATIONS

MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements, General Specification for
LMSD 6117	General environment Specification for Satellite Programs
LMSD 6244	Ground Handling and Service Equipment Test Phase Performance Specification for Discoverer Program
LMSD 1067030	Design Control Specification

5.0 RESPONSIBILITY

Ground Handling and Service.

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Biomedical Recovery Capsule Equipment
DISCOVERER PROGRAM

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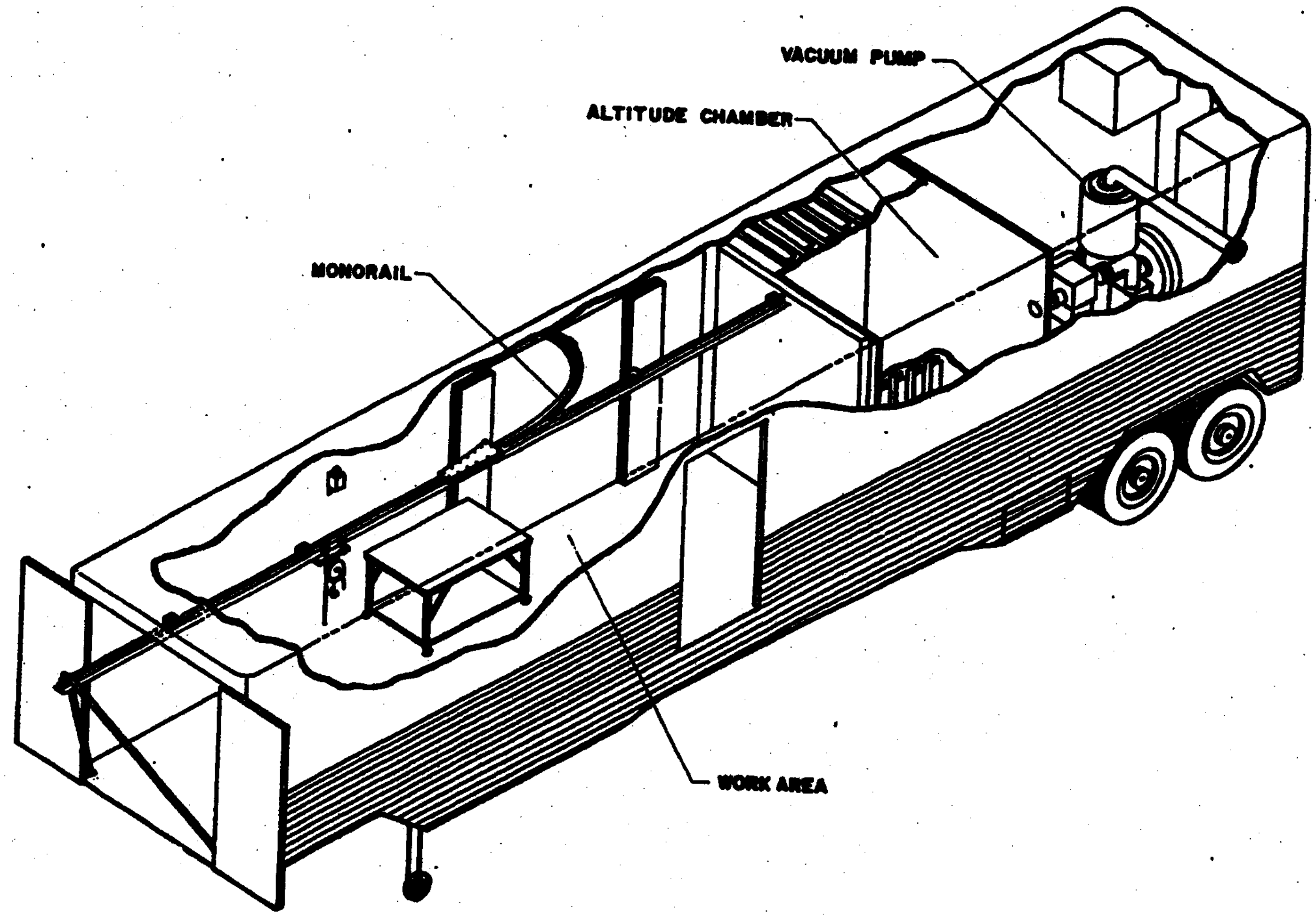

The above illustration shows the arrangement of the Altitude Chamber Van which is used to determine the effects of high altitude (low pressure) on the biological specimen.

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APPROVED BY	ALTITUDE CHAMBER VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY

1.0 GENERAL

The Altitude Chamber Van (LMSD DCD 1062071) is a semi-trailer equipped with a low pressure chamber in which altitudes to approximately 250,000 feet can be simulated. This chamber is used to determine the effects of high altitude on selected biological specimens. The altitude chamber van is a unit of the Medical Support Complex. When used in the complex, the altitude chamber van is coupled with the Instrumentation Van and the Assembly Van.

2.0 DESCRIPTION

The altitude chamber van is a 49-foot long semi-trailer. The under carriage consists of a 20,000-pound spring-suspended bogie-type tandem axle with dual wheels. Both air brakes and manually-operated brakes are installed. The sheet aluminum body, over a steel frame, is fiber glass insulated. The trailer is equipped with a fifth wheel assembly compatible with a standard military tractor type truck. A standard retractable caster is provided to support the van whenever detached from the tractor. Four leveling jacks are provided to insure operational ability on uneven terrain. The altitude assembly van is subdivided into the following compartments (areas): control-work area, altitude chamber area, and vacuum pump area. The following systems are also incorporated: air-conditioning and heating, electrical, and communications.

2.1 Compartments (Areas)

2.1.1 Control-Work Area. The control-work area is contained in the forward 30 feet of the trailer. The aft end of this area abuts against the altitude chamber with the chamber's door opening into the control-work area. Double opening doors in the forward bulkhead open to the outdoors. In each side of the van, approximately midpoint, are two doors. The door in the left side is 6-feet wide, 7-feet high, and leads into the Assembly Van when the assembly van, instrumentation van, and altitude chamber van are inter coupled. The door in the right side of the altitude chamber van gives access into the instrumentation van under the above conditions. Weather-proof coupling doors are provided. A portable work table, caster-mounted, is located in the control-work area, and it may be trundled into the altitude chamber as necessary. A 1000-pound capacity monorail, with a 500-pound capacity chain hoist, is installed. This monorail runs the length of the control-work area and is provided with a 90° switch to the left hand door to splice to a monorail in the assembly van. The control panel console is located against the right bulkhead and is rigidly affixed.

2.1.1.1 Control Panel Console. The control panel console provides a means for controlling the simulated altitude within the altitude chamber. The panel contains the controller unit, switches, and indicators. The desired altitude is set into the controller unit which through pneumatic valves maintains the equivalent pressure (or vacuum). The controller unit also indicates the attained altitude and records this information. The chamber pump motor and the recorder motor are both controlled by switches on the panel with lights to indicate when they are energized.

2.1.2 Altitude Chamber. The altitude chamber, with external dimensions 7 1/2 ft x 7 1/2 ft x 8 ft, is constructed of 1/4-inch stainless steel plate, reinforced with steel "I" beams. The chamber is designed to withstand the forces of one atmosphere of pressure when the pressure within is 1.629×10^{-8} in Hg (4.138×10^{-7} mm Hg) corresponding to an altitude of 6000,000 feet. Entrance to the chamber is through a 6 ft x 7 ft door, gasketed, with "dog" type heavy duty latches to insure a leak-tight closure. An 18 in. x 18 in. viewing window in the door allows the operating personnel to observe the progress of the experiments being conducted within the chamber. The internal lighting is a special vacuum explosion proof type. An instrumentation cable leads into the chamber and connects to a shielded type patch panel. The other end of this instrumentation cable leads to an external connector through which it eventually connects with the instruments in the Instrumentation Van.

2.1.3 Vacuum Pump Area. The remaining eight feet of the van aft of the altitude chamber is the vacuum pump area. The vacuum pump system consists of two units, the Kinny pump and the Connersville-Root booster. These two units, connected in series, evacuate the altitude chamber in less than 10 minutes to a pressure of 1.746×10^{-2} mm Hg. The booster is located in the vacuum line between the chamber and the Kinny pump. The Kinny evacuates the chamber to approximately 1.5 mm Hg, at which point the booster is actuated to further reduce the pressure in the chamber to the desired limit. An "alphatron" is the instrument which measures the degree of vacuum. The alphatron is a device which measures the alpha particle emission from an active source. The rate of emission is inversely proportional to the pressure. The output of the alphatron drives the pressure indicator on the control panel, energizes the booster at a definite pressure, and energizes the controller unit to maintain the preset chamber pressure. The controller unit maintains the preset pressure by bleeding air into the vacuum lines. The Kinny pump is driven by a 15-hp motor while the Connersville-Root blower is driven by a 10-hp motor. Both blower units are water-cooled by a single water circulating system. An air compressor located in this area furnishes air for the pneumatic valves and for cleaning. The compressed air pressure is 90 psig, developed by a 3 standard cu ft/min compressor.

2.2 Systems

2.2.1 Air-Conditioning and Heating. The altitude chamber van is air-conditioned by a unit mounted on the internal bulkhead. Since only the work-control area is cooled, no ducting is required; the air-conditioner discharges directly into the work-control area.

2.2.2 Electrical. The electrical system includes 110-volt and 480-volt. The trailer may receive 480-volts from the Auxiliary Power Unit Van directly, or when the Assembly Van, Instrumentation Van and the Altitude Chamber Van are inter coupled, the chamber van may receive power through the Instrumentation Van which has a patchboard and receives power for combination and distributes it. The trailer converts a portion of the 480-volt to 110-volt for lighting and appliance receptacles. Lighting in the control-work area is fluorescent. The lighting within the chamber is special lighting to withstand the near vacuum. Power requirements for the altitude chamber van are 30 KW.

2.2.3 Communications. The trailer is equipped with a communications system in which transmission and reception are achieved without the mechanical operation of switches. The altitude chamber is tied into a common intra-complex van communications system. The van may be equipped with a telephone to connect with the base telephone exchange.

2.3 Mobility

The trailer is capable of being towed by a standard Air Force tractor equipped with a fifth wheel assembly. The trailer has 14-inch ground clearance for Type III mobility in accordance with Specification MIL-M-8090A. The van is air transportable via C-133 in accordance with Specification MIL-A-8421A.

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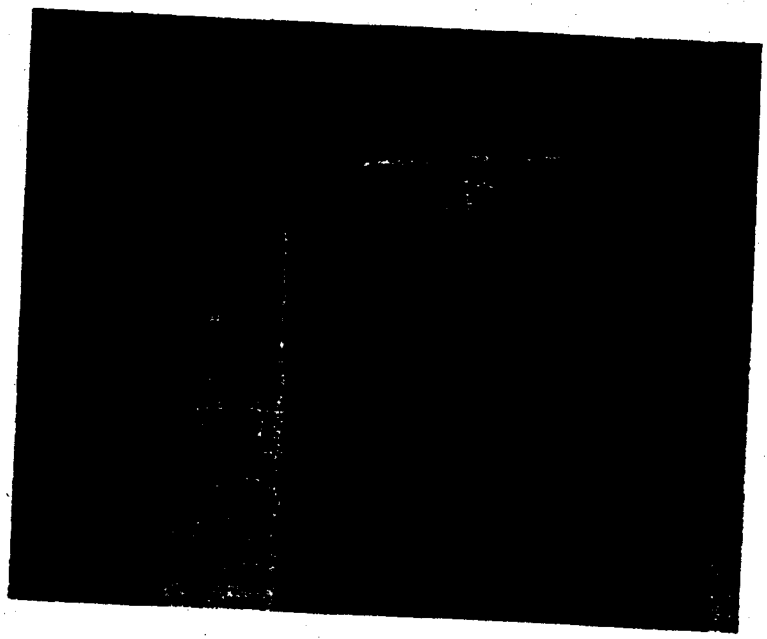
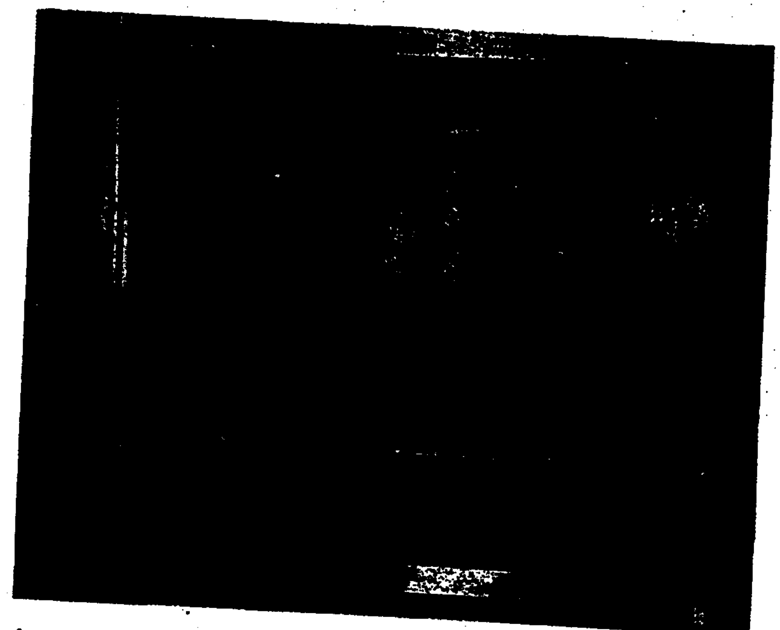
APPROVED BY	ALTITUDE CHAMBER VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY





The above illustrations show the control console for and the interior of the altitude chamber of the Altitude Chamber Van. This altitude chamber is designed to withstand the force of one atmosphere of pressure from without when the pressure within is 1.629 x 10^{-8} in. Hg (4.138 x 10^{-7} mm Hg), corresponding to an altitude of 600,000 feet.

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3.0 DESIGN CRITERIA

Simulated altitude	Greater Than 250,000 feet
Chamber pressure, at 250,000 feet	1.746×10^{-2} mm Hg
Vacuum pump motors	
Main (Kinney)	15-hp
Booster (Connersville-Root)	10-hp
Evacuation time, altitude chamber	Less Than 10 minutes
Air-Conditioning	
Summer design conditions	105°F dry bulb, 80°F wet bulb
Winter design conditions	20°F dry bulb
Requirements	Normal human comfort conditions
Electrical	480-volt, 3-phase a.c.; 30 KW
Weight	25,000 pounds
Dimensions	49' x 8' x 11-1/2'

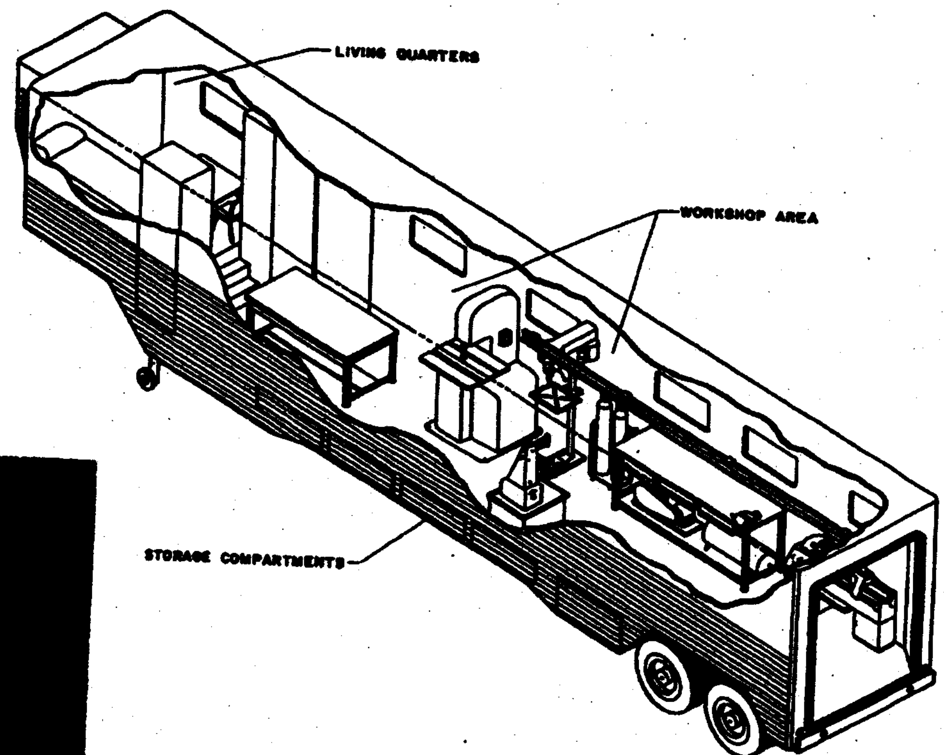
4.0 REGULATIONS AND SPECIFICATIONS

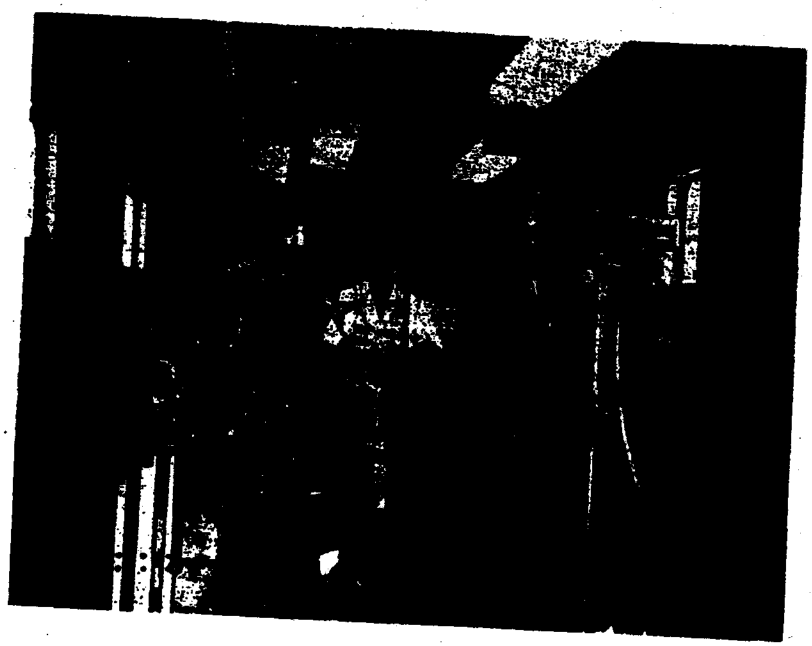
MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements, General Specification for
LMSD-6117	General Environment Specification for Satellite Programs,
LMSD-6244	Ground Handling and Service Equipment Test Phase Performance Specification for Discoverer Program
LMSD-1067031	Design Control Specification

5.0 RESPONSIBILITY

Ground Handling and Service.

APPROVED BY	TITLE	PREPARED BY
APPROVED BY	ALTITUDE CHAMBER VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY





The illustrations above show the Assembly Van of the Biomedical Support Complex. This unit is equipped to maintain, and modify as necessary, the other units of the complex.

APPROVED BY	TITLE	PREPARED BY
APPROVED BY	ASSEMBLY VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY

1.0 GENERAL

The Assembly Van (LMSD ICD 1062073) is a semi-trailer equipped as a portable machine shop to conduct maintenance and modification of Medical Complex units and equipment. In use the assembly van is coupled to the Altitude Chamber Van in a grouping which includes the Instrumentation Van.

2.0 DESCRIPTION

The assembly van is a 49-foot long semi-trailer. The under carriage consists of a 20,000-pound spring-suspended bogie-type tandem axle with dual wheels. Both air brakes and manually-operated brakes are installed. The fiber-glass insulated, sheet aluminum body is mounted over a high-tensile strength steel framework. The trailer is equipped with a fifth wheel assembly compatible with a standard military tractor type truck. A standard retractable caster is provided to support the van whenever detached from the tractor. Four leveling jacks are provided to insure operational ability on uneven terrain. A double door in the rear of the trailer and a personnel access door in the right side just aft of the trailer coupling area provide entry into the trailer. A bumper seal around the outside of the rear doors mates with a similar seal around the outside of the left hand side door of the Altitude Chamber Van for a water proof coupling of the two trailers. The assembly van is divided into two compartments (areas): work shop area and living area. The van incorporates the following systems: air-conditioning and heating, water, electrical, and communications.

2.1 Compartments (areas)

2.1.1 Work Shop Area. The after 37 feet of the van contains the work shop area. The area is open with the various pieces of shop equipment disposed against the bulkheads. Included in this area are the following pieces of equipment: Monarch 10" precision tool room lathe, Bridgeport milling machine, oxygen-acetylene welding hand torch and hand truck (less bottles); 14" drill press - 1/2-inch capacity; Do-All automatic band saw complete with all attachments. In addition to the large pieces of shop equipment there are racks for raw materials, cabinets for the normal complement of machine and hand tools, including electrical hand tools, a welding bench, a work bench, and an assembly table. A 1000-pound capacity monorail, with a 500-pound capacity chain hoist, transits the after half of the work shop area. This rail connects with the monorail in the Altitude Chamber Van when the Altitude Chamber Van is coupled with the assembly van.

2.1.2 Living Area. The 9 feet of trailer space forward of the work shop area is outfitted as the living area. This space is over the fifth wheel assembly and contains two folding bunks, tow chairs, table, and filing cabinet. Since this area is at a higher level than the work shop area, a short centerline stairway is provided for entry into the living area. Two closets, located on each side of the stairway, are provided for personal effects.

2.2 Systems

2.2.1 Air-Conditioning and Heating. The van is air-conditioned and heated by a single unit mounted externally on the front bulkhead. The conditioned air is distributed throughout the trailer via insulated ducts to maintain normal human comfort conditions.

2.2.2 Water System. The integral water system includes a 100-gallon reservoir, transfer pump, 20-gallon hot water heater, and 100-gallon disposal tank. A corner sink is installed against the left bulkhead and the wall of the leftside closet, opposite the personnel access door.

2.2.3 Electrical. 480-volt, 3-phase power is received from the Auxiliary Power Unit Van, either directly or through a patchboard on the Instrumentation Van when the assembly van is inter coupled with the Altitude Chamber Van and the Instrumentation Van. Power (220-volt, 3-phase, and 110-volt) is available within the trailer to operate the machinery and for lighting, respectively. The lighting is fluorescent.

2.2.4 Communications. The assembly van is equipped with a communications system in which transmission and reception are achieved without the mechanical operation of switches. This van is tied into a common intra-complex communications system. There are provisions to receive a telephone to connect the trailer into the base telephone exchange.

2.3 Mobility

The trailer is capable of being towed by a standard Air Force tractor equipped with a fifth wheel assembly. The trailer has 14-inch clearance for Type III mobility in accordance with Specification MIL-M-8090A. The van is air transportable via C-133 in accordance with Specification MIL-A-8421A.

3.0 DESIGN CRITERIA

Air-Conditioning	
Summer design conditions	105°F dry bulb, 80°F wet bulb
Winter design conditions	20°F dry bulb
Requirements	Normal human comfort conditions
Water	
Reservoir	100-gallon
Hot wat r heater	20-gallon
Disposal tank	100-gallon
Electrical	480-volt, 3-phase, a.c. 21 KW
Weight	18,000 pounds
Dimensions	49' x 8' x 11-1/2'

4.0 REGULATIONS AND SPECIFICATIONS

MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements, General Specification for
LMSD 6117	General Environment Specification for Satellite Programs
LMSD 6244	Ground Handling and Service Equipment Test Phase Performance Specification for Discoverer Program
LMSD 1067033	Design Control Specification

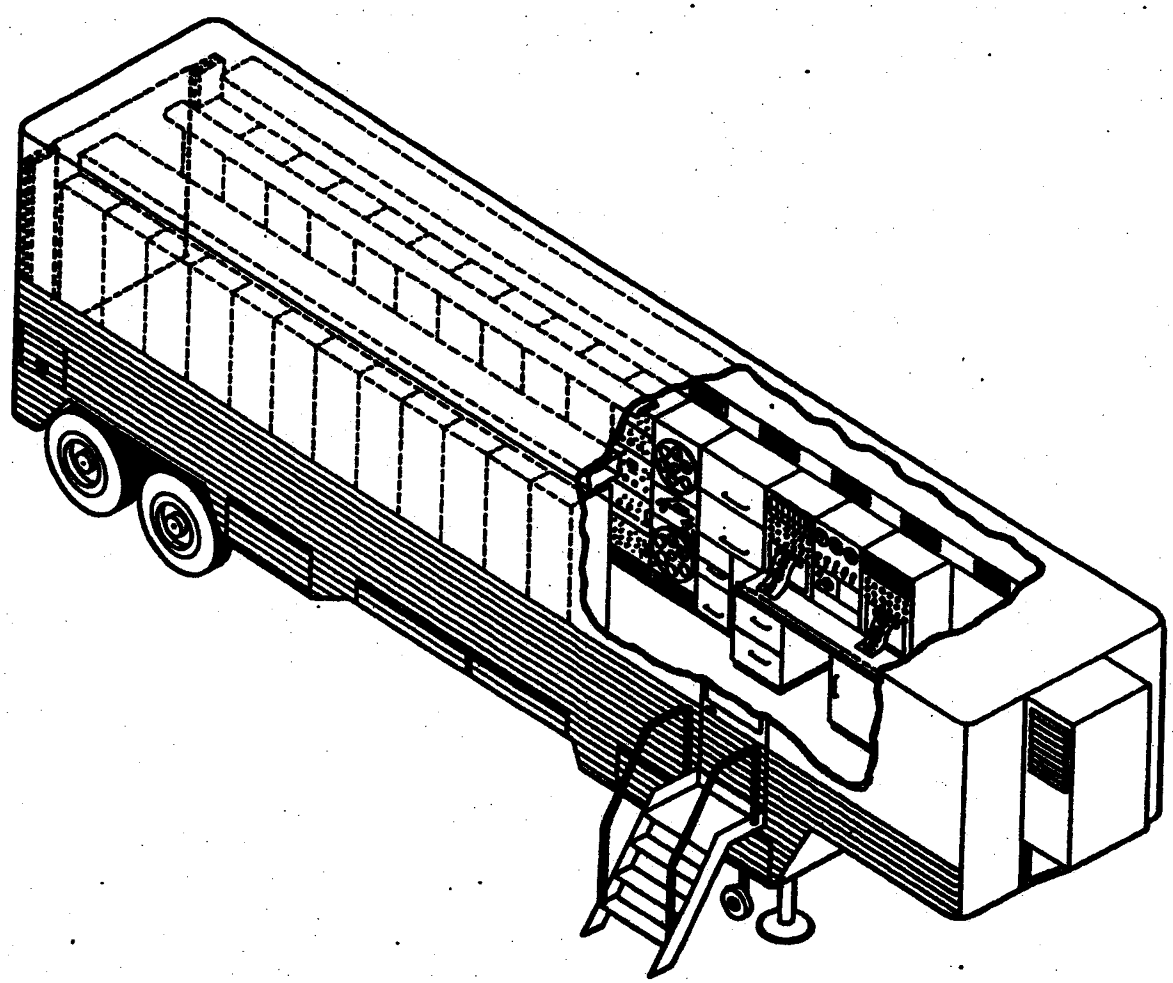
5.0 RESPONSIBILITY

Ground Handling and Service.

APPROVED BY	TITLE	PREPARED BY
APPROVED BY	ASSEMBLY VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY

LOCKHEED AIRCRAFT CORPORATION

MISSILES AND SPACE DIVISION



The illustration above shows the Instrumentation Van of the Biomedical Support Complex. This unit contains the equipment for data display and recording, and for electronic calibration for the other units in the complex.

APPROVED BY	TITLE INSTRUMENTATION VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	PREPARED BY
APPROVED BY		APPROVED BY

GENERAL

1.0 The Instrumentation Van (LMSD DCD 1062072) is a mobile unit within the Medical Complex equipped with data display, recording, and electronic calibration equipment for the complex. When used in the complex, the instrumentation van is coupled with the Altitude Chamber Van in a group which also includes the Assembly Van.

2.0 DESCRIPTION

The instrumentation van is a 45-foot long semi-trailer. The under carriage consists of a 20,000-pound spring-suspended bogie-type tandem axle with dual wheels. Both manually-operated and air brakes are installed. The sheet aluminum body, over a steel frame, is fiber glass insulated. The trailer is equipped with a fifth wheel assembly compatible with a standard military tractor type truck. A standard retractable caster is provided to support the van whenever detached from the tractor. Four leveling jacks are provided to insure operational ability on uneven terrain. A double door in the rear of the trailer and a personnel access door in the right side just aft of the trailer coupling area provide access to the trailer. The equipment in the trailer is arranged in racks along each side between the forward door and the rear of the trailer. An operator's console is located against the left side of the trailer opposite the personnel access door. The area forward of the personnel access door and over the fifth wheel assembly contains storage cabinets and work tables.

The instrumentation van is equipped to receive, process, display, and record data relative to experiments conducted within the altitude chamber of the Altitude Chamber Van, data from the operating areas of the Medical Vans, and inputs from any external source utilizing transducers having electrical outputs. When used in conjunction with the Altitude Chamber Van, the instrumentation van receives signals generated by associated equipment (not furnished with the Medical Support Complex) used to measure specimen reaction within the altitude chamber. These signals are conducted through cabling and connectors to a patch panel in the instrumentation van. Signals are then routed to and distributed by a network of patch panels to the various equipments. Equipments within the instrumentation van fall into the following categories: input, signal conditioning, recording, display, and calibration. Air-conditioning and heating, electrical, and communications systems are also integral parts of the instrumentation van.

2.1 Equipments

2.1.1 Input Equipment. The input equipment includes input plug panel, line balance capacitor switches, and amplifiers. The output of the amplifiers may be used in either the signal conditioning equipment or in the display and recorder equipment. There are 35 differential or single-ended high-gain 0-800 KC amplifiers; 15 high impedance input to high or low impedance output d.c. amplifiers; 3 a.c. instrumentation amplifiers; and 6 galvanometer power amplifiers.

2.1.2 Signal Conditioning Equipment. The signal conditioning equipment includes a 20-channel a.c. modulated carrier and a.c.-d.c. demodulator, 30-channel d.c. bridge, 10-channel thermocouple reference junction, 6-channel pulse rate to d.c. converter, 12-channel potentiometer voltage supply, 2-channel synchro to d.c. phase angle converter, 40 range and balance units, and 90-channel galvanometer scale factoring pads (on console).

2.1.3 Recording Equipment. The recording equipment includes a system to receive 90-channels of analog information, to record this information in digital form, and to play back and reconstruct the 90 analog signals. Additional recording equipments are two 25-channel analog oscillographs (located on the console) and a 14-channel analog tape recorder.

2.1.4 Display Equipment. The display equipment is mounted on the operator's console. Two sets of Simpson meters are located on the console for readout of signal amplitude (from selected circuits via patch panel). The controls for the UHF/VHF communications system and for the intra-complex inter-communications system are located on the console. The two oscillographs (recorder equipment) and associated scale factoring pads (signal conditioning equipment), controls for analog tape recorder, lapse time clock (test duration), and power on-off indicators for equipment are also found on the console.

2.1.5 Calibration Equipment. Calibration equipments consist of a 0-35-volt d.c. supply; a 1-1/2-volt d.c. supply; a 0-115-volt, 60-cycle, a.c. supply; a 0-115-volt, 400-cycle, a.c. supply; a 6-digit, a.c.-d.c. digital ohmmeter and capacity bridge; frequency standard equipment; and time lapse scopes are provided. A standard oscillator is also provided.

2.2 Systems

2.2.1 Air-Conditioning and Heating. An air-conditioning and heating unit mounted to the exterior of the forward bulkhead furnishes conditioned air through an insulated duct system in the van roof. Separate ducting channels filtered cooling air to the instrumentation racks.

Electrical. 480-volt, 3-phase power is received from the Auxiliary Power Unit Van. 220-volt, 3-phase power is provided to operate various instrumentation components. 110-volt is provided for lighting and appliance receptacles throughout the van. The lighting is fluorescent.

When one or more vans are intercoupled with the instrumentation van, power from the APU van is received at and distributed by the instrumentation van to the other vans.

Communications. The trailer is equipped with a communications system in which transmission and reception are achieved without the mechanical operation of switches. The instrumentation van is linked with the other vans in the Medical Complex via a common intra-complex communications system. The van may be equipped with a telephone to connect with the base telephone exchange.

2.3 Mobility

The trailer is capable of being towed by a standard Air Force tractor equipped with a fifth wheel assembly. The trailer has Type III mobility in accordance with Specification MIL-M-8090A. The van is air transportable via C-133 in accordance with Specification MIL-A-8421A.

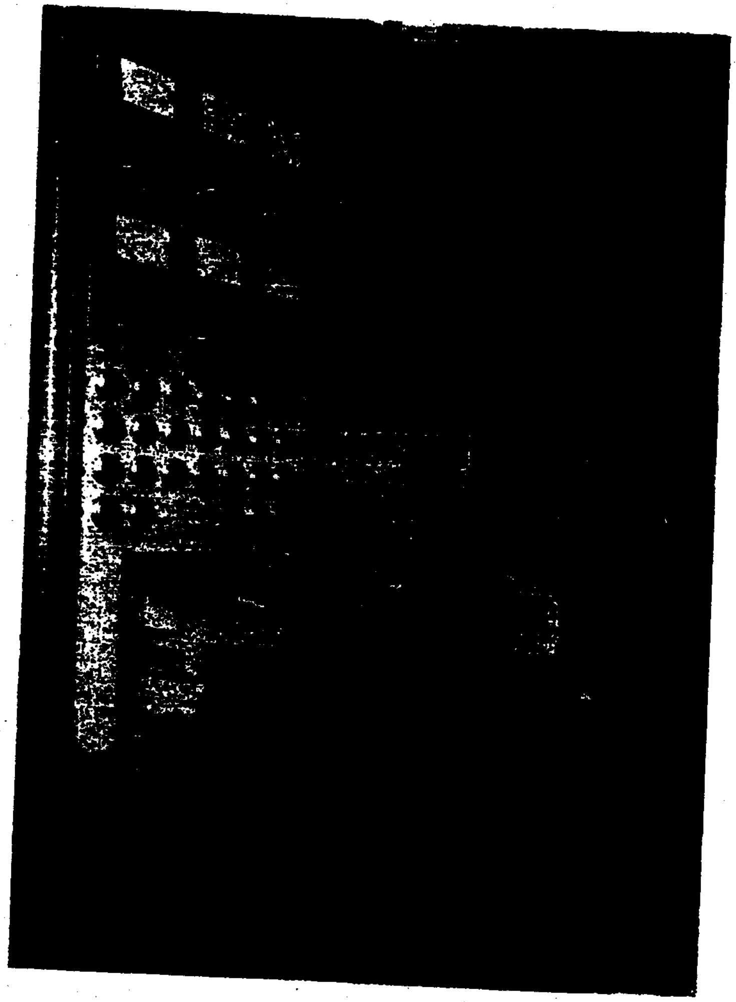
DESIGN CRITERIA

Air-Conditioning	
Summer design conditions	105°F dry bulb, 80°F wet bulb
Winter design conditions	20°F dry bulb
Requirements	Normal human comfort conditions
Electrical	480-volt, 3-phase, a.c. 25 KW
Weight	16,000 pounds
Dimensions	45' x 8' x 11-1/2'

REGULATIONS AND SPECIFICATIONS

MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements General Specification for

APPROVED BY	TITLE	PREPARED BY
APPROVED BY	INSTRUMENTATION VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY



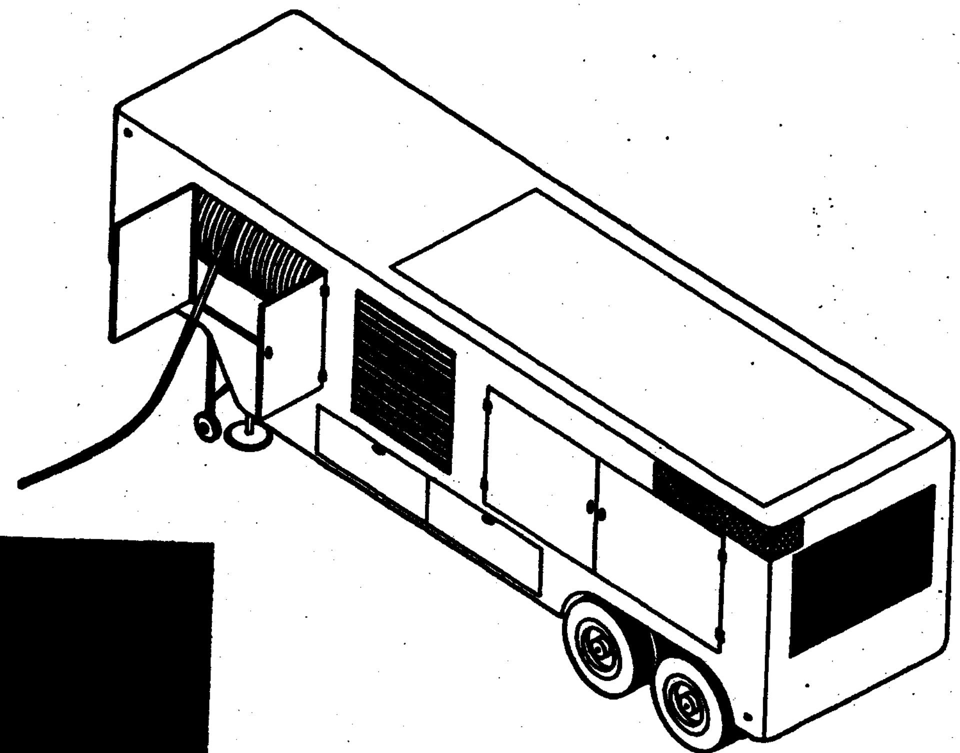
The illustrations above show the interior of the Instrumentation Van of the Biomedical Support Complex. Shown are the Operator's Console, through which the display and recording of data is coordinated, and the banks of electronic equipment for the measuring and recording of the data received from the other units in the complex, such as the Altitude Chamber Van.

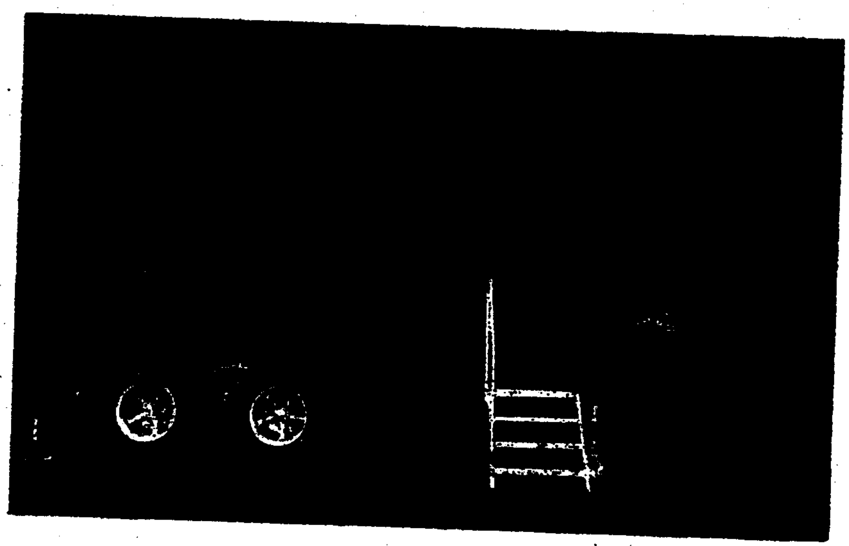
APPROVED BY	TITLE INSTRUMENTATION VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	PREPARED BY
APPROVED BY		APPROVED BY

LMSD 6117	General Environment Specification for Satellite Programs
LMSD 6244	Ground Handling and Service Equipment Test Phase Performance Specification for Discoverer Program
LMSD 1067032	Design Control Specification

5.0 RESPONSIBILITY

Checkout Equipment Development





The unit depicted above, the Auxiliary Power Unit Van, distributes base power to the other units in the Biomedical Support Complex, and in the event of a failure of base power, generates sufficient power to supply the full power requirements for the complex.

APPROVED BY	TITLE AUXILIARY POWER UNIT (APU) VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	PREPARED BY	
APPROVED BY		APPROVED BY	

1.0 GENERAL

The Auxiliary Power Unit (APU) Van (LMSD ICD 1068086) is a semi-trailer equipped with a diesel-generator. As part of the Medical Support Complex the APU van provides power to the remaining vans within the complex. The APU van receives base power and distributes it, or in event of base power failure, generates sufficient power to supply the full power requirements of the complex.

2.0 DESCRIPTION

The auxiliary power unit (APU) van is a 30-foot long semi-trailer. The under carriage consists of a 20,000-pound spring-suspended bogie-type tandem axle with dual wheels. The van is equipped with both air brakes and manually-operated brakes. A fifth wheel assembly compatible with a standard military tractor type truck provides the means of coupling van to tractor. A standard retractable caster is provided to support the van whenever detached from the tractor. The body of the van is sheet aluminum over a steel frame. A personnel access door is provided in the right side of the van about midway along its length. A removable panel in the top of the trailer permits the removal of the APU generator by means of an overhead crane. A storage space for cable reels and other equipment affiliated with the APU van is provided in the forward end of the van. This space is enclosed with access via a door which opens into the main part of the van.

The APU normally receives base power and distributes it throughout the Medical Support Complex. In the event of power failure on the base, the APU takes over the load and furnishes the required power to the vans in the complex. This takeover is automatic. The APU regulates the output to the other vans at 480-volts. Design of the van is in accordance with Specification MIL-P-6457A. The trailer has 14-inch clearance for Type III mobility in accordance with Specification in MIL-M-8090A, and is air transportable per Specification MIL-A-8421A. The APU van has the following major components: generator set, distribution system, automatic control system, and safety equipment.

2.1 Generator Set. The generator set consists of a 225-hp air-cooled, continuous duty diesel engine, flexibly coupled to a 187 KVA, 480-volt, 3-phase, 4-wire alternator. A 180-gallon fuel tank with filters and transfer pumps supplies a full-load fuel supply to the diesel engine for a twelve-hour period.

2.2 Distribution System. The distribution system consists of three 150-foot capacity cable reels. Power is distributed to the two Medical Vans and the Instrumentation Van through these three cables, each of which is equipped with weather-proof quick disconnect plugs. If required, power may be furnished directly to either the Altitude Chamber Van or the Assembly Van in lieu of one of the other vans.

2.3 Automatic Control System. The automatic control system transfers the load from base power to the engine-operated generator in the event there should be a base power failure. Unless otherwise manually overcontrolled this system will shift the load back to base power 30 minutes after restoration of base power. When base power fails or fluctuates, the automatic control system starts the diesel engine and places the generator on the line. Safety devices are incorporated in the system to prevent engine-generator damage.

2.4 Safety Equipment. A 15-pound capacity fire extinguisher is provided.

3.0 DESIGN CRITERIA

Diesel engine	225-hp, air-cooled
Generator	187 KVA, 480-volt, 3-phase, 4-wire
Diesel fuel tank	180-gallon
Power distribution reels, 150-foot capacity	

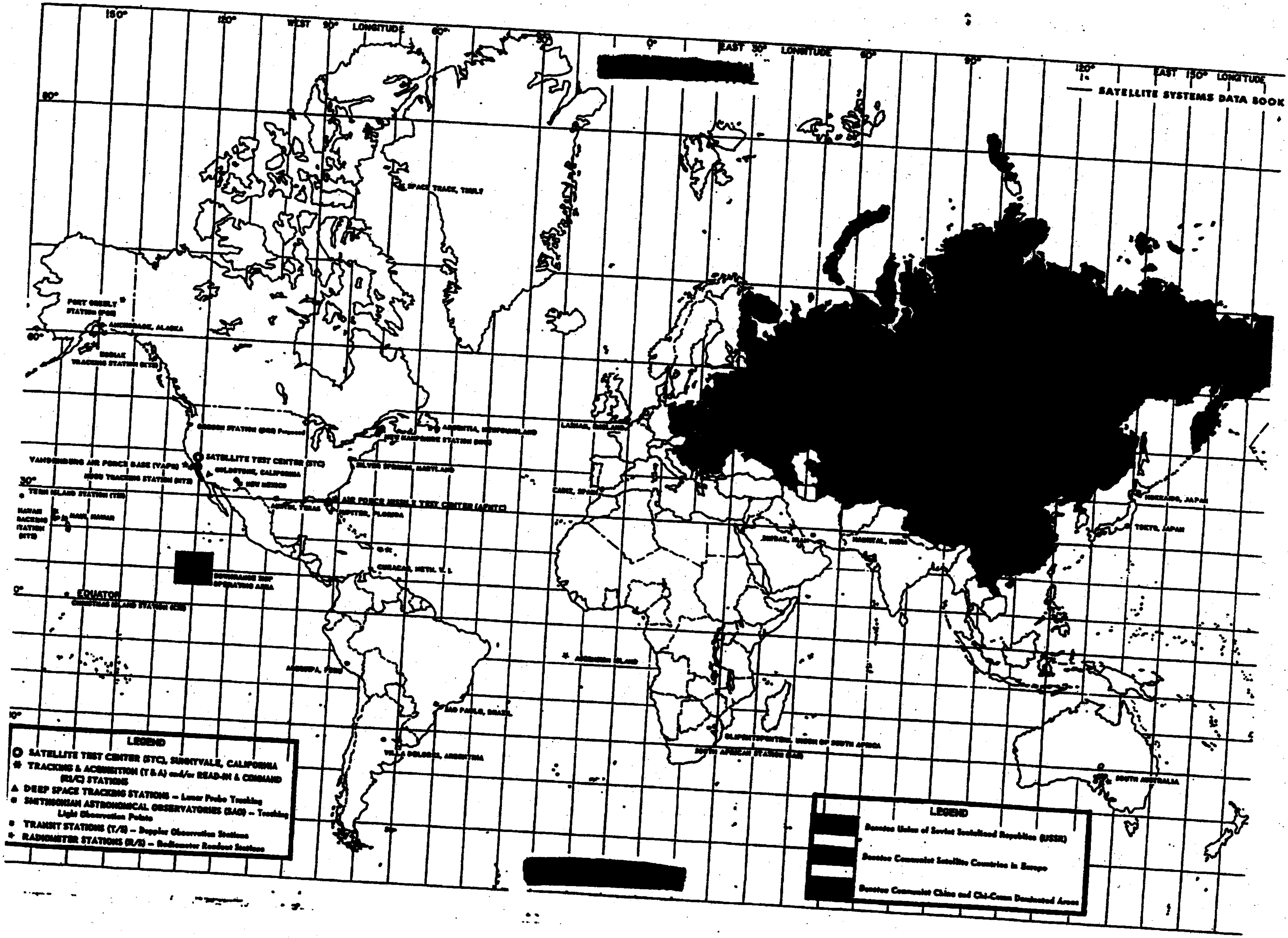
4.0 REGULATIONS AND SPECIFICATIONS

MIL-M-8090A	Mobility Requirements, Ground Support Equipment, General Specification for
MIL-A-8421A	Air Transportability Requirements, General Specification for
MIL-P-6457A	Power Supply, Metallic Rectifier Type, General Specification for
LMSD 6117	General Environment Specification for Satellite Programs
LMSD 6244	Ground Handling and Service Equipment. Test Phase Performance Specification for Discoverer Program
LMSD-1067042	Design Control Specification

5.0 RESPONSIBILITY

Subsystem Checkout Equipment

APPROVED BY	TITLE	PREPARED BY
APPROVED BY	AUXILIARY POWER UNIT (APU) VAN Biomedical Recovery Capsule Equipment DISCOVERER PROGRAM	APPROVED BY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SATELLITE SYSTEMS DATA BOOK
WEST 90° LONGITUDE
EAST 30° LONGITUDE
EAST 150° LONGITUDE
SPACE TRACK, THULE
ANCHORAGE, ALASKA
KODIAK TRACKING STATION (KTS)
SATELLITE TEST CENTER (STC)
VANDENBURG AIR FORCE BASE (VAFB)
GOLDSTONE, CALIFORNIA
NEW MEXICO
SILVER SPRINGS, MARYLAND
NEW HAMPSHIRE STATION (NHS)
ARGENTIA, NEWFOUNDLAND
JUPITER, FLORIDA
AUSTIN, TEXAS
CURACAO, NETH. W. I.
EQUATOR
CHRISTMAS ISLAND STATION (CIS)
AREQUIPA, PERU
SAO PAULO, BRAZIL
VILLA DOLORES, ARGENTINA
ASCENSION ISLAND
CADIZ, SPAIN
SHIRAZ, IRAN
NAINITAL, INDIA
HOKKAIDO, JAPAN
TOKYO, JAPAN
SOUTH AUSTRALIA
LEGEND
SATELLITE TEST CENTER (STC), SUNNYVALE, CALIFORNIA
TRACKING & ACQUISITION (T & A) and/or READ-IN & COMMAND (RI/C) STATIONS
DEEP SPACE TRACKING STATIONS – Lunar Probe Tracking
SMITHSONIAN ASTRONOMICAL OBSERVATORIES (SAO) – Tracking Light Observation Points
TRANSIT STATIONS (T/S) – Doppler Observation Stations
RADIOMETER STATIONS (R/S) – Radiometer Readout Stations
LEGEND
Denotes Union of Soviet Socialist Republics (USSR)
Denotes Communist Satellite Countries in Europe
Denotes Communist China and Chi-Comm Dominated Areas
[redacted]

[redacted]

O FLIGHT TEST CONTROL

Sunnyvale, California
Satellite Test Center (STC)
Palo Alto Computer (PAC) (until 1/61)
Building 104, LMSD (SUN)

* TRACKING & ACQUISITION (T&A)/READ-IN & COMMAND (RI/C) STATIONS

	T & A								RI/C			
	Tracking				Data Acquisition*				Payload			
	D	M	S	N	D	M	S	N	D	M	S	N
Vandenberg Air Force Base (VAFB), California; Vandenberg Tracking Station (VTS); Vandenberg Communications Center (VCC)	X	X	X	X●	X	X	X	RI	C*	X	X	
Oahu, Hawaii; Hawaii Tracking Station (HTS); Hawaii Control Center (HCC) (Air-Recovery/Sea-Retrieval Force Coordination)	X	V	V		X	RI	V		C*	RI	V	
Kodiak, Alaska; Kodiak Tracking Station (KTS)	X	A◇	V◇		X		V		C*		V	
Pt. Mugu Naval Air Station, Oxnard, California; Mugu Tracking Station (MTS)	X	B^	B^	X^	X			RI	C*			
New Boston, New Hampshire; New Hampshire Station (NHS)	X	V	X		X	RI●	X		C*	RI●	X	
Fort Greely, Alaska; Fort Greely Station (FGS)										RI◇		
Johannesburg, Union of South Africa; South African Station (SAS)			B			RI		RI◇				
Central Pacific; Christmas Island Station (CIS); Tern Island Station (TIS)	R R		H				RI					
Fort Stevens, Oregon; Oregon Station (ORS)			X◇				X◇				X◇	
▲ Deep Space Tracking Stations; Johannesburg, Union of South Africa; Woomera, South Australia, Australia; Goldstone, California				X X X				RI RI X				

D = Discoverer Program
M = MIDAS Program
S = Samos Program
N = NASA Program

X = General capability
V = VHF & S-band only
A = S-band only
R = Recovery only

◇ = Potential capability
B = Back-up facility
^ = Ascent only
H = VHF only

● = Implies a satellite command capability within the general capability
RI = A read-in capability only
C = Command capability only
◇ = Early passes only
● = UHF command capability only
● = If S-band radar/beacon is used
* = Single command capability
+ = Single UHF command capability (originates STC; uses FGS beacon)
● = Angle track also

1 or 2 downrange ships are positioned as indicated for telemetry receiving and for exit tracking - All Programs less NASA.

* TRANSIT STATIONS
(APL Doppler Readout)

Utilization: Discoverer Program only

Anchorage, Alaska	(PMR)
Argentia, Newfoundland	(APL)
Austin, Texas	(APL)
Hokkaido, Japan	(PMR)
Las Cruces, New Mexico	(APL)
Lasham, England	(Brit.)
Philippines	(PMR)
Sao Paulo, Brazil	(PMR)
Silver Spring, Maryland	(APL)
South Australia, Australia	(PMR)
South Point, Hawaii	(PMR)

APL = Stations manned by personnel from Applied Physics Laboratory
PMR = Stations manned by personnel from Pacific Missile Range

* SMITHSONIAN ASTRONOMICAL OBSERVATORIES
(Tracking Light Observations)

Utilization: Discoverer Program only

Arequipa, Peru
Cadiz, Spain
Curacao, Netherlands West Indies
Jupiter, Florida
Maui, Hawaii
Nainital, India
Olifantsfontein, Union of South Africa
Organ Pass, New Mexico
Shiraz, Iran
Tokyo, Japan
Villa Dolores, Argentina
Woomera, South Australia, Australia

★ RADIOMETER STATIONS
(Radiometer Readout)

Utilization: Discoverer Program only

Ascension Island (AMR)
Cape Canaveral (AFMTC) (AMR)
Hawaii Tracking Station (HTS)
Kodiak Tracking Station (KTS)
New Hampshire Station (NHS)
Space Track, THULE Air Force Base
Vandenberg Tracking Station (VTS)
Woomera, South Australia, Australia

AMR = Stations within the Atlantic Missile Range network

LMSD-6164B
-6165A
-445641A

15 May 1961

TITLE	PREPARED BY
TRACKING & READ-IN AND COMMAND STATIONS Satellite and Payload ALL PROGRAMS	APPROVED BY

[redacted]