Cline

GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

Memorandum

XII.1

TO : Distribution

DATE: May 12, 1964
R-ASTR-TJ-101-64/I

FROM : Research and Development Operations

SUBJECT : Technical Information Summary Concerning SATURN I Vehicle SA-6

This memorandum outlines, through a series of sketches with accompanying text, the general features of the sixth SATURN I launch vehicle. The sketches are devoted primarily to the launch vehicle but also presents limited information on the spacecraft, the launch facility and launch preparations. The information presented in this summary was compiled through the efforts of R&DO personnel from P&VE, AERO and ASTR.

1. SATURN I Program Background

The SATURN I Program had as its primary objective the development of a two stage launch vehicle which would be used to place manned Apollo Spacecrafts into low earth orbit. Redirection of this program has limited its scope to the research and development phase only. The program now consists of 10 vehicles which are divided into two main groups: Block I (SA-1 through SA-4) and Block II (SA-5 through SA-10). During the past 2 1/2 years the first five SATURN I vehicles were launched successfully from Cape Kennedy.

In the Block I series, only the S-I Stage was propelled and the vehicles were limited to short range ballistic trajectories. Launch vehicles SA-5 through SA-10, which have approximately 20,000 pound payload orbital capability, consist of live S-I and S-IV Stages controlled by a separate instrument Unit located immediately above the S-IV Stage.

Like SA-5, the five remaining Block II vehicles will be used to orbit large unmanned payloads. These vehicles will be launched during the next few years from Launch Complex 37B at Cape Kennedy.

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2. Mission Objectives

In addition to the mission objectives assigned to SA-5, SA-6 will:

a. demonstrate the closed-loop performance of the vehicles ST-124 Guidance System during S-IV burn.

b. demonstrate the compatibility of Launch Vehicle/Apollo Spacecraft (both preflight and inflight) and of Launch Facility/Apollo Spacecraft. SA-6 is the first SATURN/Apollo (boilerplate) flight test.

c. verify the Spacecraft's inflight environmental parameters.

d. demonstrate the Spacecraft's normal LES tower jettison scheme.

3. Mission Profile

SA-6's payload will be launched into a low earth orbit from Launch Complex 37B. After liftoff, the vehicle will roll to the flight azimuth, program downrange during S-I and S-IV burn and guide the 39,300 pound payload (boilerplate Apollo Spacecraft, + IU and S-IV Stage) into orbit.

The orbital parameters for the first pass are expected to be:

Perigee	183 km
Apogee	230 km
Inclination	31.8 deg.
Period	88 1/2 minutes
Lifetime	5 days (nominal).

Verification of orbit and continual updating of orbital parameters will be obtained from the world wide Minitrack system and from additional information provided by the ground and inflight radar systems and long range cameras.

NOTE: In issuing this summary, we would like to acknowledge the assistance given us by several MSFC groups.

Additional copies of this report may be obtained from the following personnel:

AERO, R. Teague, 876-4443
P&VE, O. E. Moon, 876-1277
ASTR, C. B. Jones, 876-3692.

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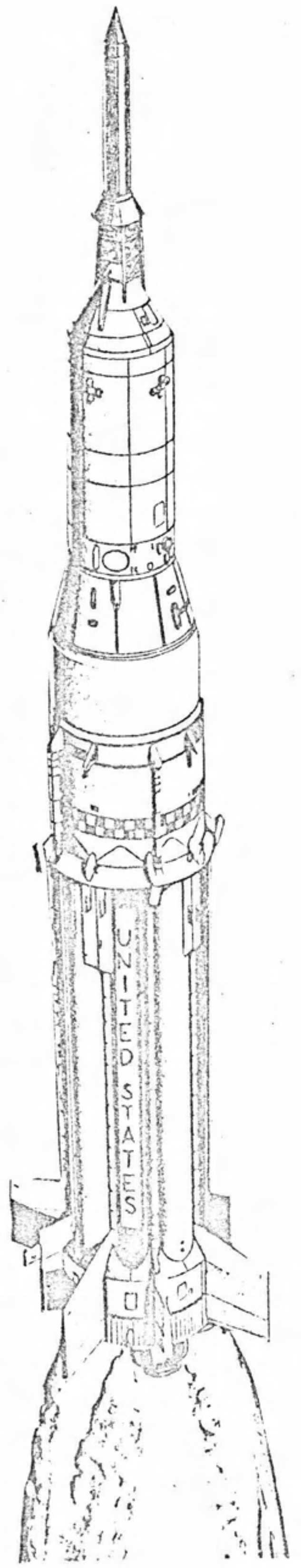


Figure 1

SA-6 Space Vehicle

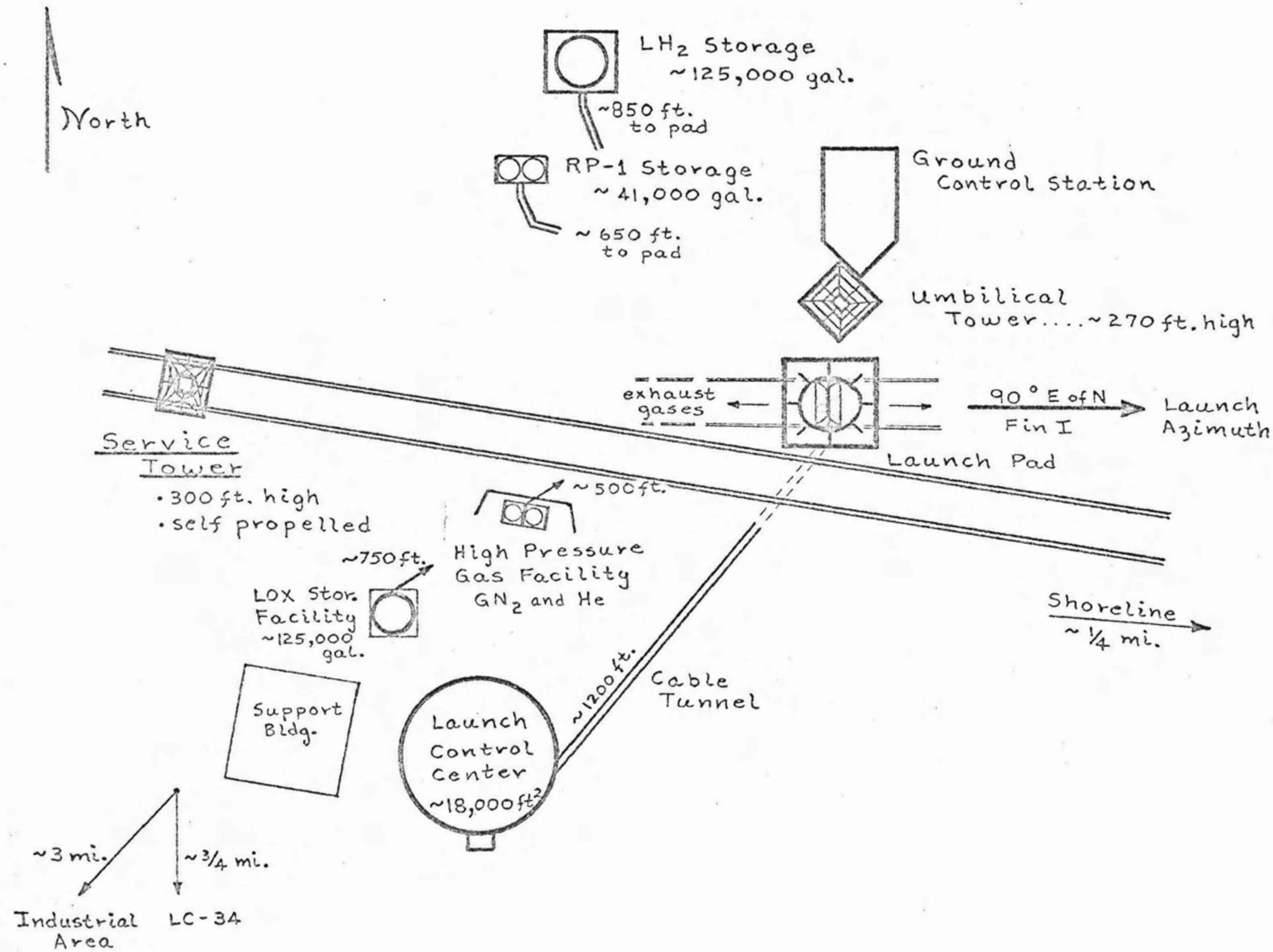


Figure 2

Launch Complex - 37B

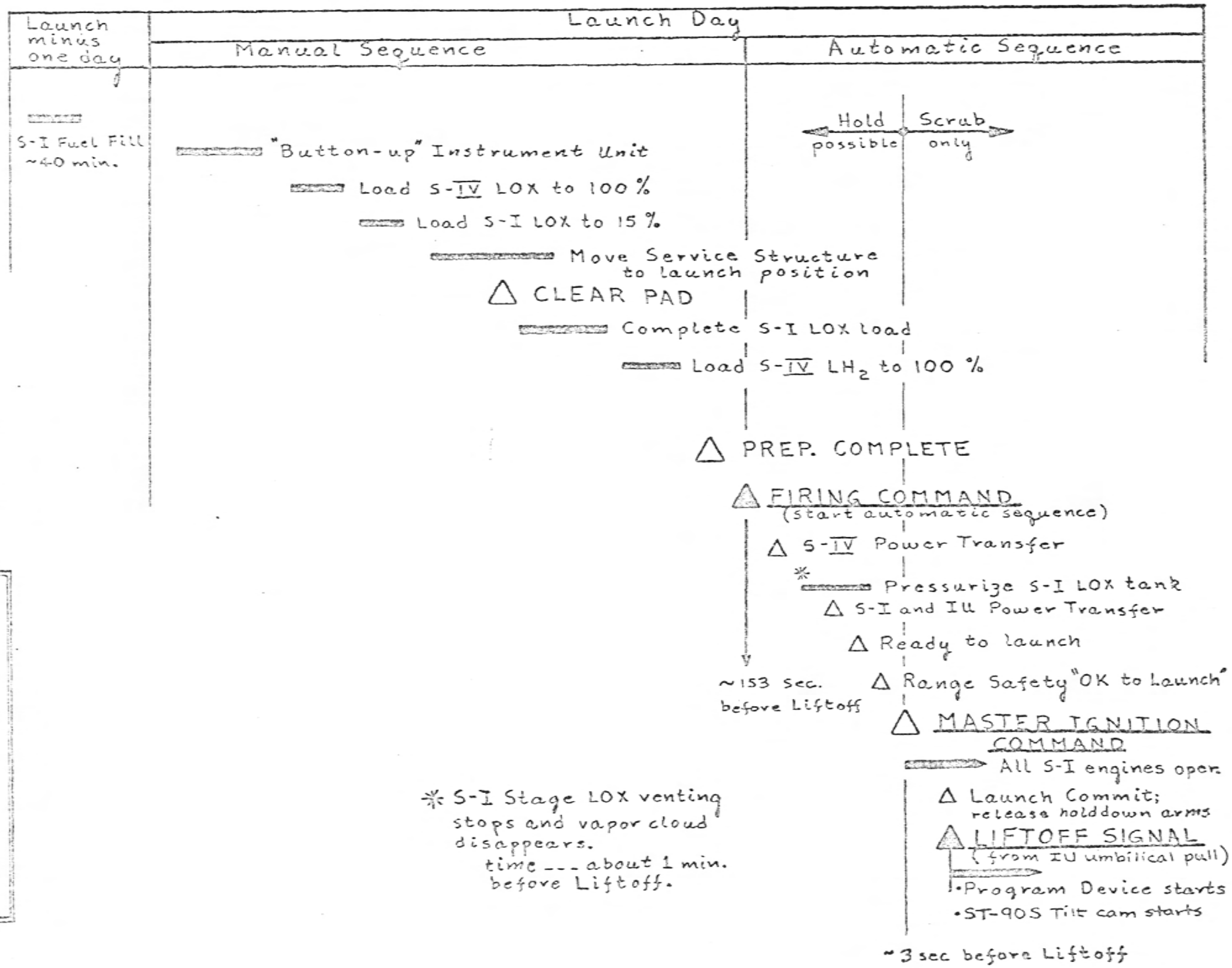
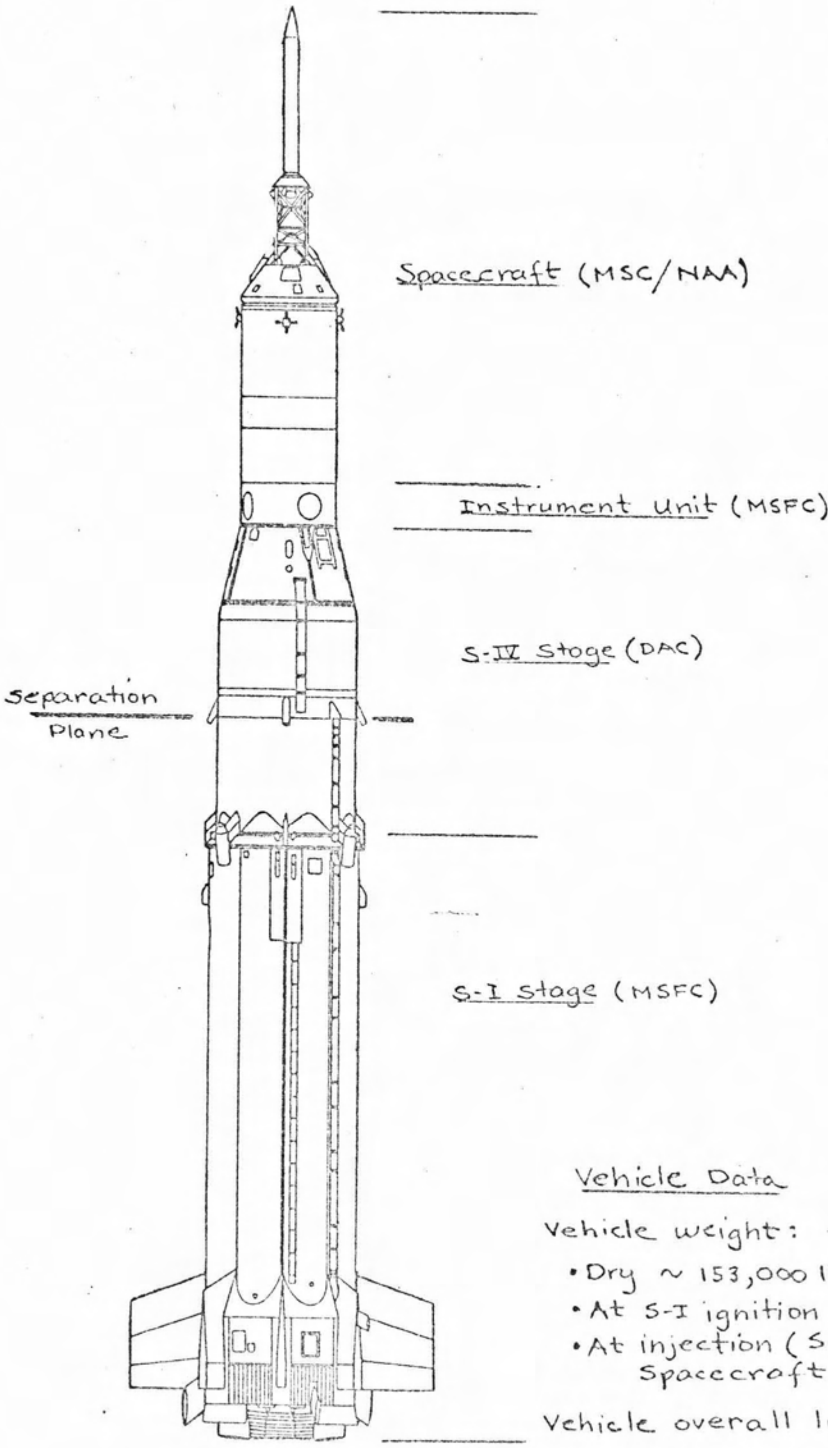


Figure 3

Launch Countdown



Vehicle Data

Vehicle weight:

- Dry ~ 153,000 lbs.
- At S-I ignition ~ 1,145,000 lbs.
- At injection (S-IV, IU, & Spacecraft ~ 37,300 lbs.)

Vehicle overall length ~ 191 ft.

Figure 1 — Space Vehicle Configuration

Trajectory Information

The trajectory for SA-6 was shaped to achieve verification of the environmental parameters (temperature, pressure, etc.) to which the Spacecraft was designed.

In addition, the vehicle will be subjected to a special ST-90S tilt program maneuver to achieve a 4° angle of attack during S-I burn. This is being done to provide aerodynamic data for the continued investigation of the roll moment phenomenon which was present during S-I burn on SA-5.

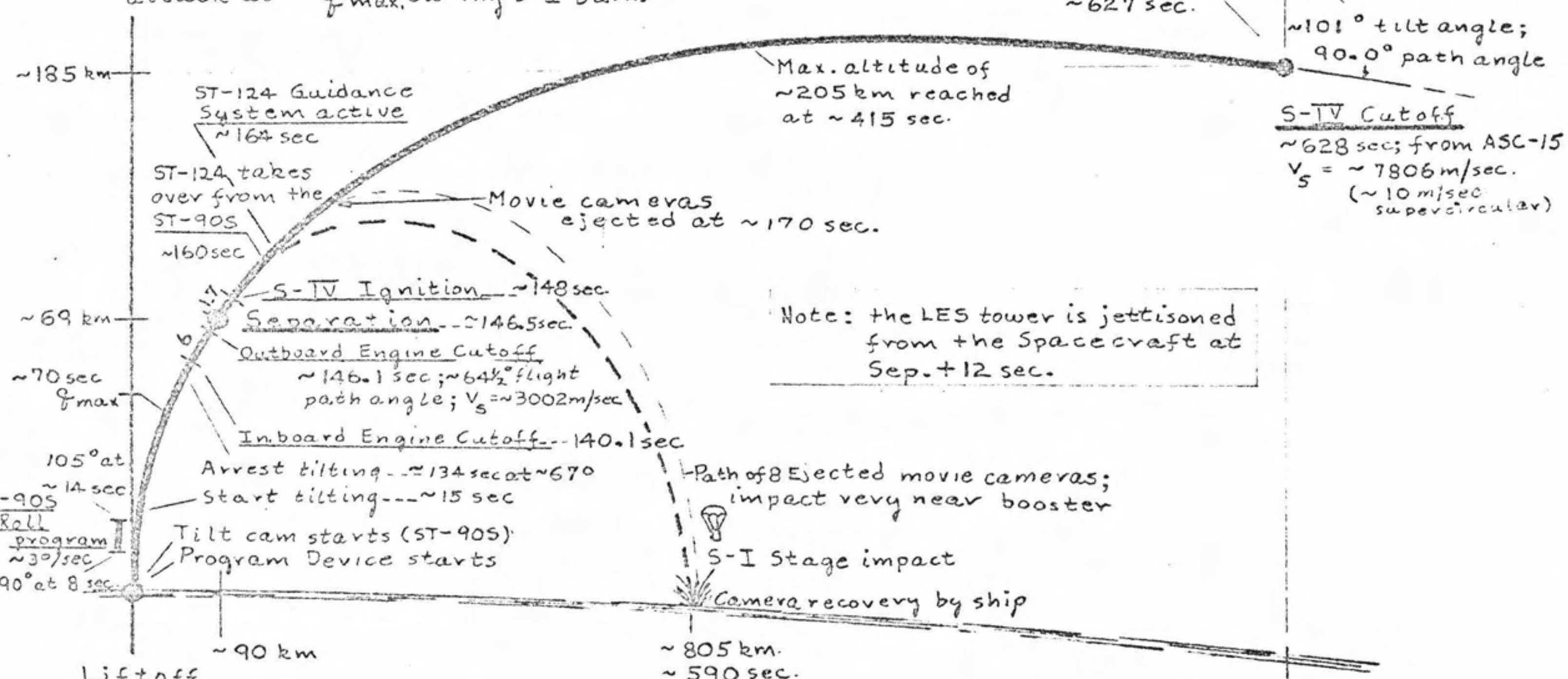
The low orbital injection altitude (183 km), in combination with the near circular orbit results in a rather brief orbital lifetime (nominally 5 days or 80 orbits).

- Cutoff arming --- S-I stage at 134.1 sec.
  S-IV stage at ~593 sec.
- ACTIVE ST-124 Guidance System during S-IV burn; steering signals introduced for pitch plane (path adaptive) and yaw plane (Δ minimum) guidance at Sep. + ~18 sec.
- Tilt program is based upon eight engine operation in the S-I stage and 6 engine operation in the S-IV stage.

  It is biased to create a ~4° angle of attack at ~ q_{max}, during S-I burn.

Orbital Parameters ... 1st pass

- Perigee --- ~183 km.
- Apogee --- ~230 km.
- Inclination --- ~31.8°
- Period --- ~88½ minutes



Liftoff

- Launch Complex 37, pad B
- Launch azimuth 90° E of N
- Flight azimuth 105° E of N

Figure 5

Trajectory Information

Guidance and Control System.

Although the Guidance and Control System hardware for SA-6 is quite similar to that flown on SA-5, its functions in the vehicle system are completely changed. In SA-6:

1. The ST-124 Guidance System provides closed-loop path guidance (steering) signals (path adaptive in the pitch plane and Δ-minimum in the yaw plane) to the Flight Control Computer starting 18 seconds after S-I outboard engine cutoff. When the S-IV attains the prescribed velocity, the ASC-15 Digital Computer goes into the fast computation mode to assure that the velocity cutoff signal is precisely given.

2. The ST-90S provides tilting and attitude error ($\varphi_{P,Y,R}$) signals for vehicle control until the ST-124 takes over control a few seconds before path guidance is introduced.

3. In order to achieve satisfactory guidance and control system hardware performance, the control rate gyros provide the required vehicle attitude rate ($\dot{\varphi}_{P,Y,R}$) signals during S-IV burn. During S-I burn, the φ filters in the Flight Control Computer provide the equivalent $\dot{\varphi}_{P,Y,R}$ signals.

4. The S-IV Stage DAC actuators flown on SA-5 have been superceded by MOOG actuators on S-IV-6.

5. A special Guidance Command System test will be made about 20 minutes after S-IV cutoff from Ascension Island.

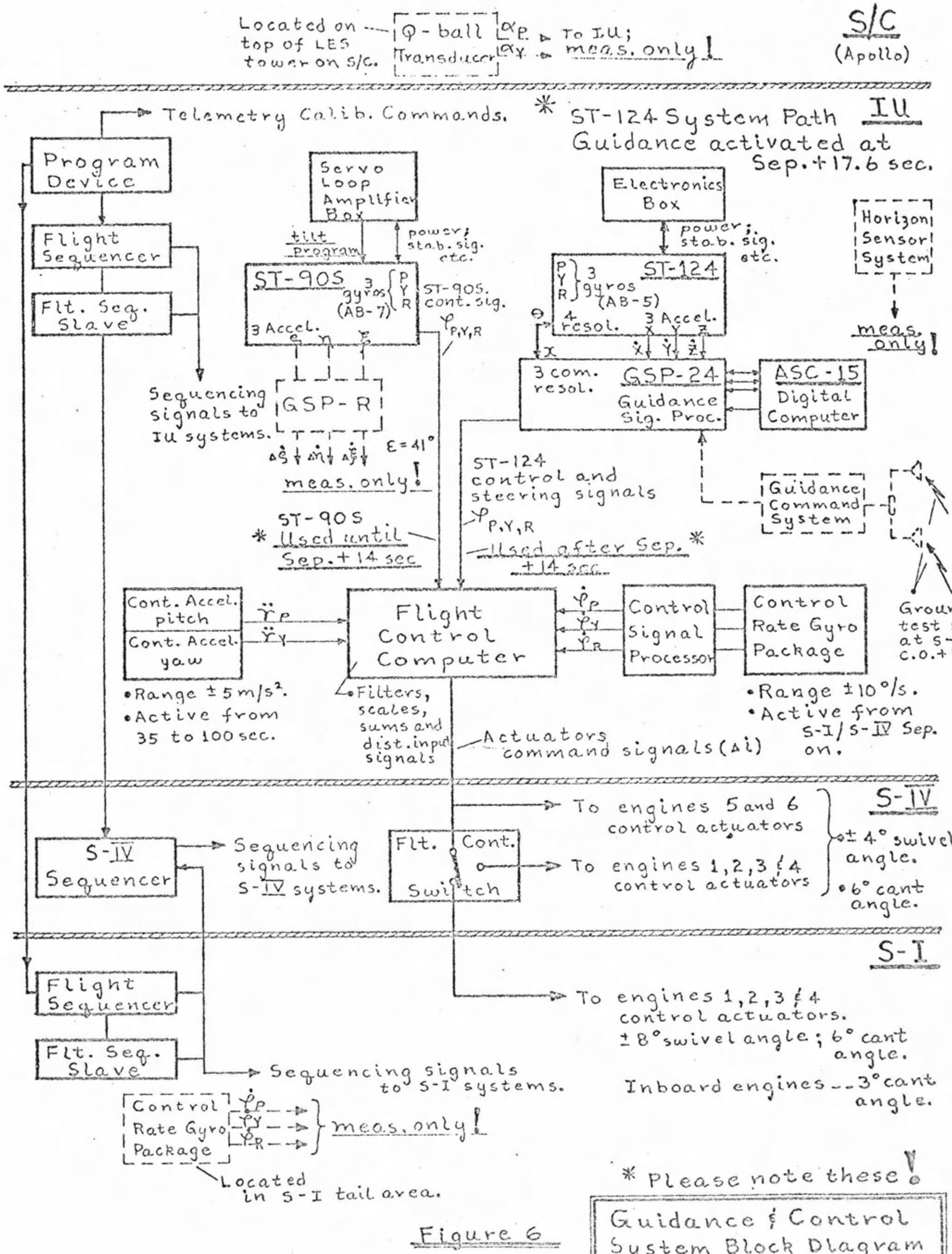


Figure 6

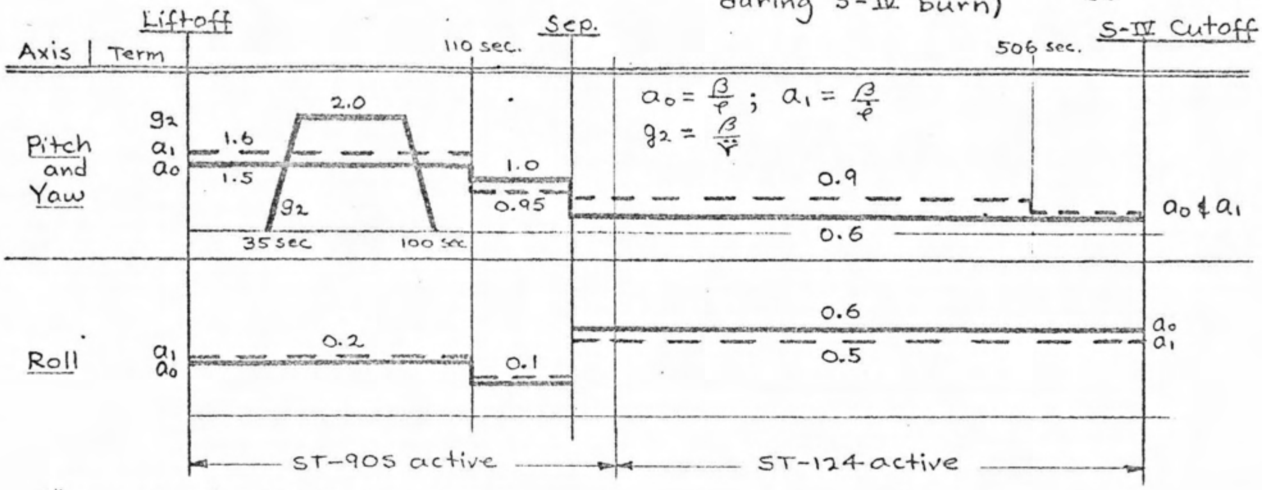
Guidance & Control System Block Diagram

Control and Tilt Program Information

The control system gains (a_0, a_1, g_2) are very similar to SA-5's except for those minor changes required to meet vehicle differences. The introduction of Δ-minimum guidance has added e_0 and e_1 gain terms to the system.

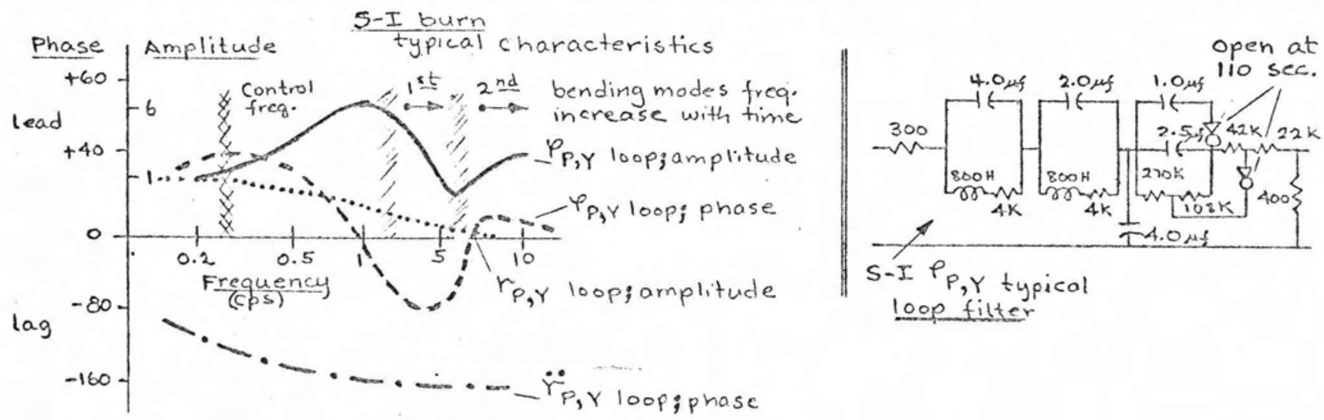
During S-I burn the ST-90S system's pitch program transmission provides the tilting signals to the vehicle control system. Tilt arrest starts at 134 seconds (before S-I inboard engine cutoff) and the vehicle's tilt angle is constant until after the ST-124 system takes over the vehicle attitude control functions some 14 seconds after S-I/S-IV separation. The vehicle's tilt angle is kept constant during this period (134 to 164 sec.) to minimize disturbances, especiall during the switchover of the platform systems.

Control System Gain Factors* (accelerometer control active during S-I burn; Control rate gyros active during S-IV burn)



* For exact values see memo R-ASTR-NG-31 dated April 6, 1964

Control Computer Filter Characteristics (Typical)



Tilt Program

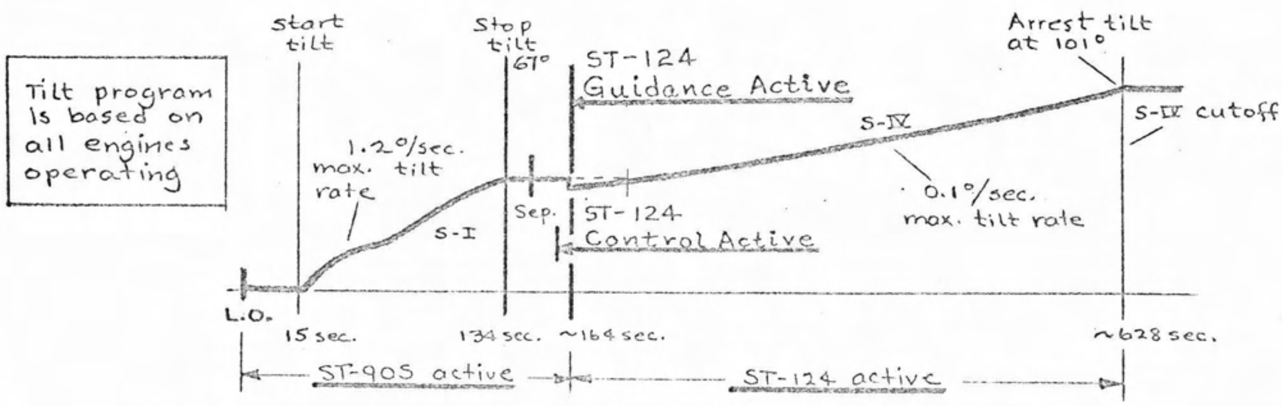
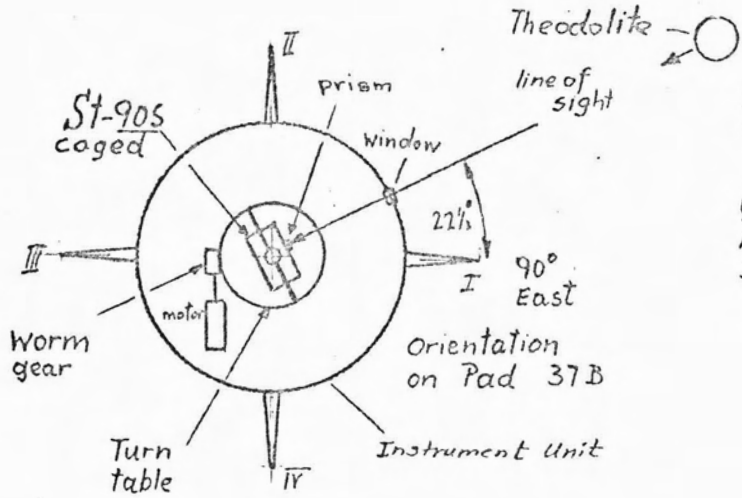


Figure 7 — Control and Tilt Program Information

Roll Maneuver - ST-90S Stabilized Platform

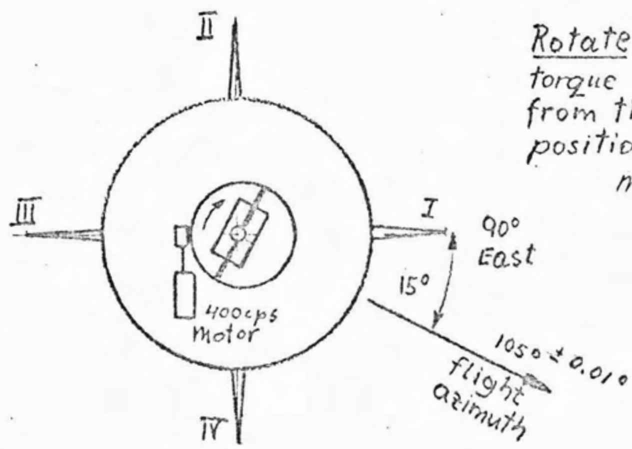
The SATURN vehicle is constrained to be placed on the launch pad at a launch azimuth of 90° E of N; therefore, the ST-90S stabilized platform must be aligned to the flight azimuth (105°) on the pad. To accomplish this, the platform is torqued from the initial geometrically fixed and measured azimuth thru the roll maneuver angle which is then stored in the azimuth drive system. Eight seconds after liftoff the vehicle rolls thru 15° to the flight azimuth driven by the $\Delta\psi_R$ signal from the platform.

The preparation for the ST-124 stabilized platform roll program (passive) is rather similar. After fine erection of the ST-124, it is aligned to a geometrically fixed and measured azimuth and then torqued to the flight azimuth. The roll angle through which the gimbal moves is stored in the ASC-15 computer; 10 seconds after liftoff the command modules in the GSP-24 are unlocked and the ST-124 roll program starts (maximum rate of 1°/sec).

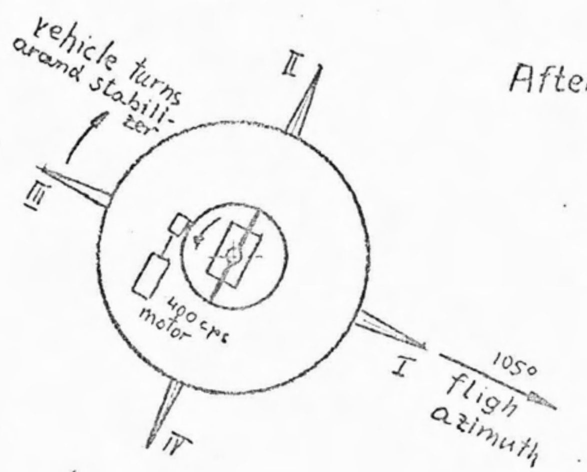


Optical azimuth alignment at this position (unobstructed line of sight)

Uncage St-90S
read out zero-position of stabilizer on resolver and TV-monitor



Rotate azimuth-ring (turn table) and torque uncaged St-90S stabilizer (6°/min) from the original optical alignment-position into 105° firing azimuth.
monitor by TV and resolver



After Lift off turn azimuth ring, and rotate the vehicle around the stabilizer (utilizing the control system) in order to
realign fins (swivel engines) with platform measuring axes.

Roll maneuver starts
8 sec. after L.O.
Turning rate
~3°/sec.

Figure 8

Roll-maneuver St-90S
Stabilized Platform

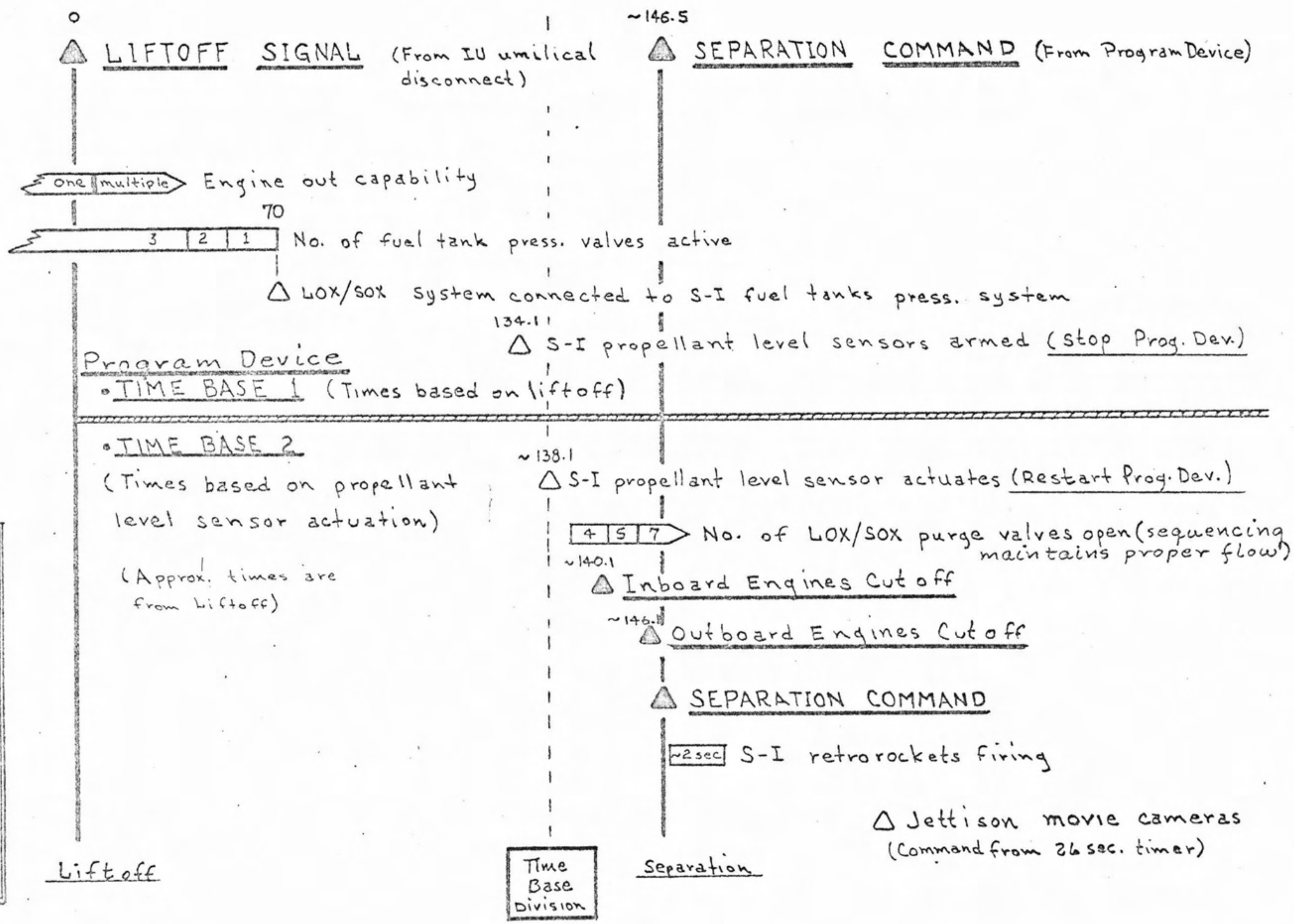


Figure 9

Flight Sequence
S-I Stage Operation

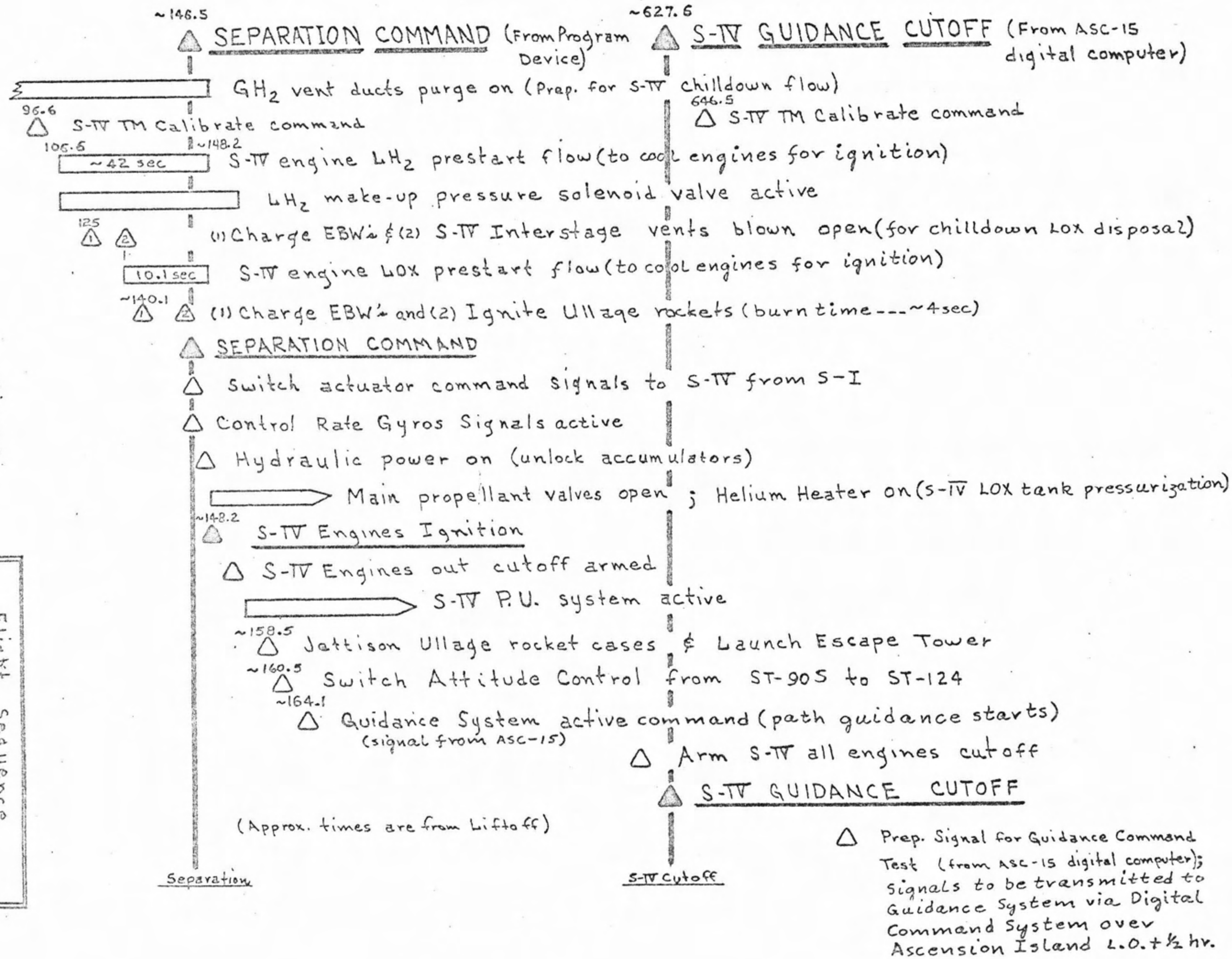


Figure 10

Flight Sequence
S-IV Stage Operations

Space Vehicle Weight vs Flight Time

The total burning time to orbit for SA-6 is 10 1/2 minutes (S-I .... 2 1/2 min.; S-IV .... 8 min.) with a single separation (at the top of the S-IV aft interstage) occuring at about 147 seconds. During this time the total propellant consumption is 952,000 pounds:

1. S-I .... 853,000 pounds including both thrust decays

2. S-IV ... 99,000 pounds, assuming 2000 pounds of propellants remaining after S-IV thrust decay which were allocated for performance reserve and mixture ratio variations.

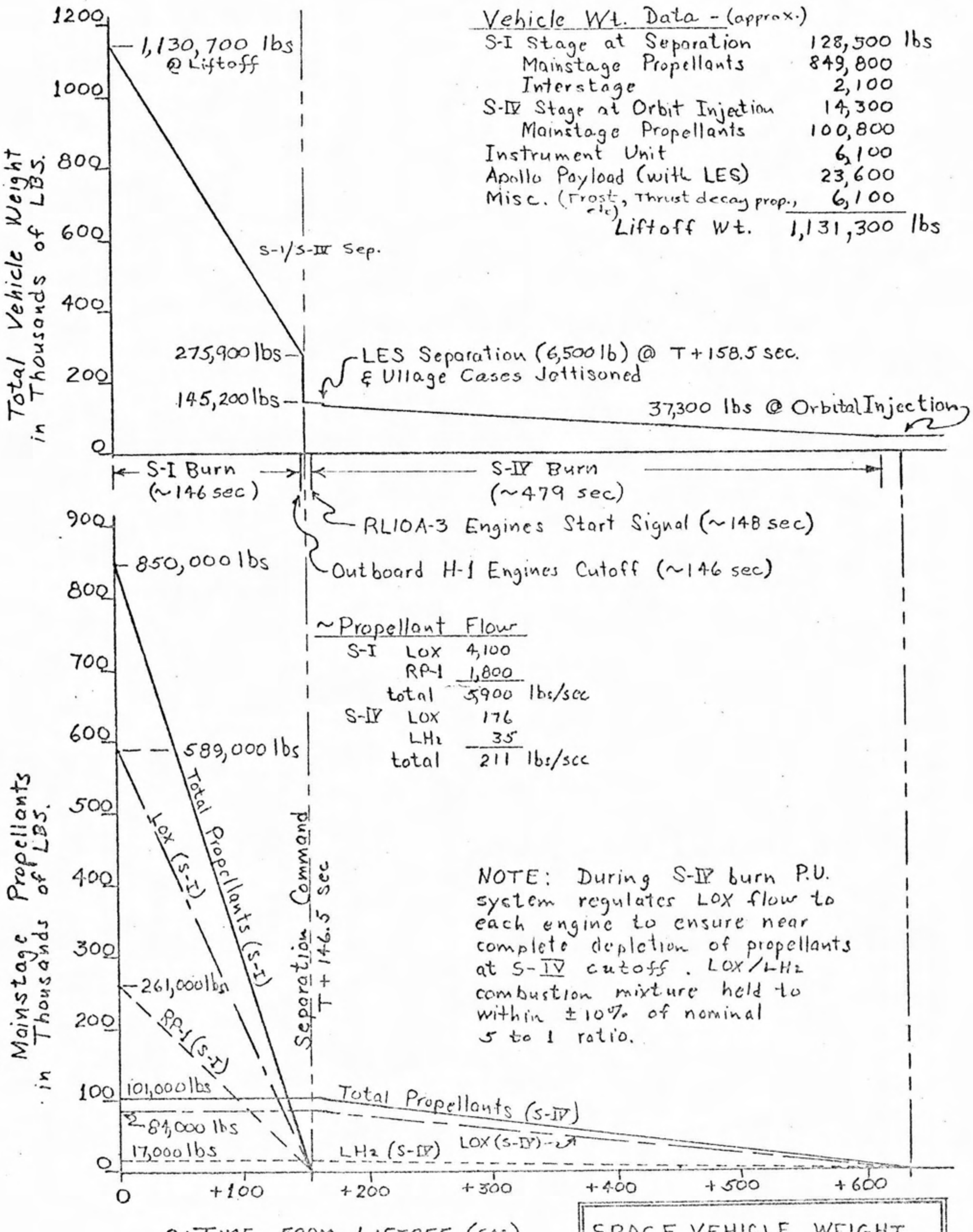
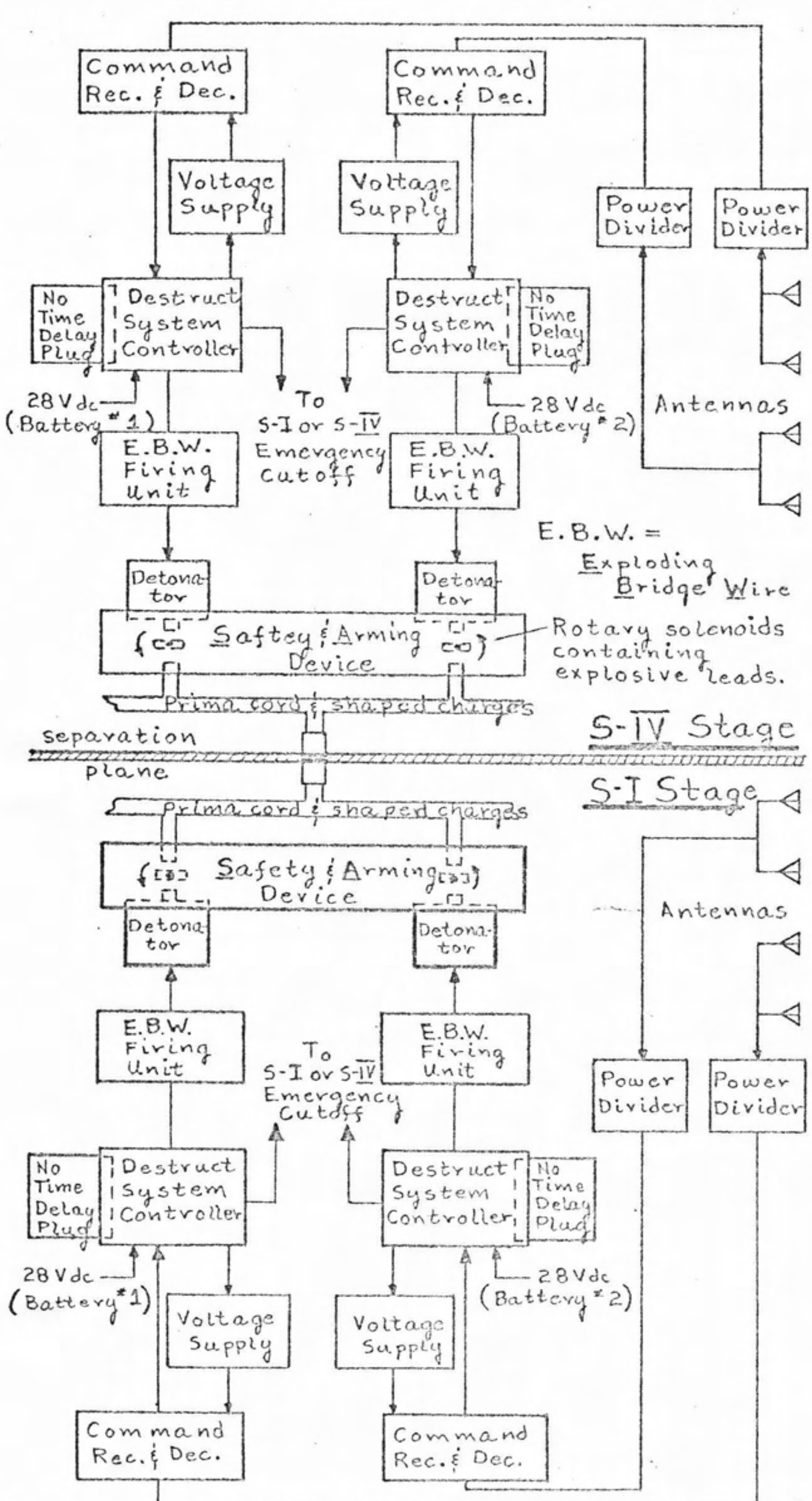


Figure 11

SPACE VEHICLE WEIGHT -VS- FLIGHT TIME

Range Safety System



Notes:

- Both com. receivers on the two stages respond to the same com. signal.
- S&A Device armed by B.H. signal before L.O.
- Range Safety Emer. C.O. command initiates EBW charging, activating the system, permitting subsequent fuel dispersion command.
- The S-IV system is electrically "safed" by a command after normal S-IV cutoff.

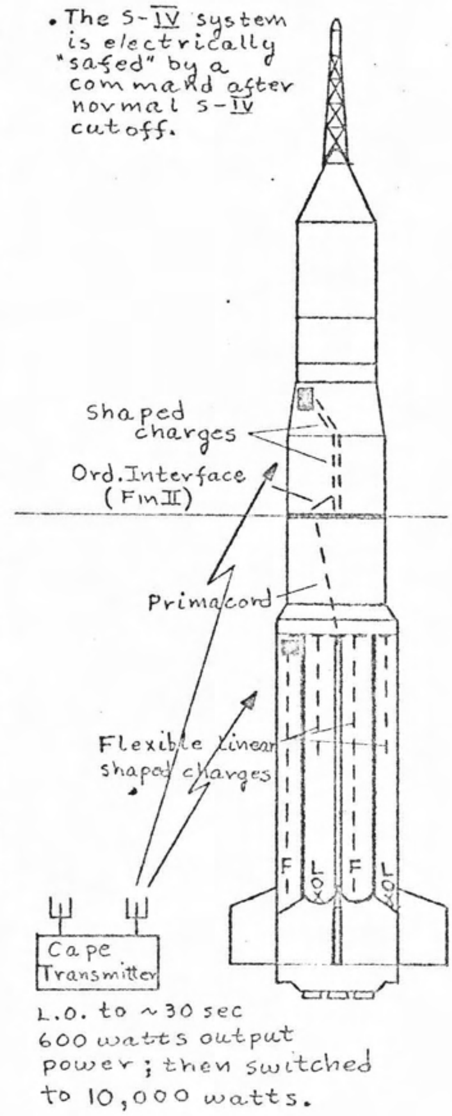


Figure 12

Launch Vehicle Range Safety System

Vehicle Tracking Systems

1. Azusa and C-Band Radar Systems.

The Azusa ground station determines successive trajectory positions of the vehicle by continuous phase comparison between the signals transmitted to and received from the onboard transponder. The C-Band Radar Transponder onboard the vehicle provides (upon interrogation from the ground) a single pulse reply to the FPS-16 ground radar system. These signals are fed to a ground computer complex to obtain position, velocity and acceleration information. These trajectory data are presented on plotting boards for the range safety officer to use in determining "real time" vehicle trajectory parameters. In addition, they are also used for the post flight determination of the vehicle's trajectory.

2. MISTRAM System.

The basic MISTRAM system consists of a vehicle borne transponder, a central ground station and four remote ones. The system determines the position and velocity of the vehicle by the use of long baseline interferometer radar measurements and triangulation techniques. The measured trajectory data is fed into a computer for real time computations and is also stored on magnetic tape for post flight analysis.

3. ODOP System.

The vehicle's ODOP transponder transmits a continuous wave signal (at 70 mc above the received frequency) to the ground receiving stations where the doppler shift (due to the radial velocity of the vehicle) is used to provide trajectory data for the post flight evaluation of the vehicle's performance.

4. Radar Altimeter System.

This system is being flown as a developmental passenger. It is a pulsed radar system which alternately transmits and receives signals through a single antenna installed on the Fin I line of the vehicle to determine vehicle altitude. It transmits the digitally encoded altitude data via the PCM telemeter to ground receiving stations.

5. Minitrack System.

This beacon provides a low power continuous signal to the world wide Minitrack network stations to enable a precise determination of the orbital parameters of the S-IV/IU/Spacecraft over a period of approximately 5 days. (Electrical power is provided to operate the Minitrack beacon for 20 days.) This information will be used to more accurately determine the orbital parameters.

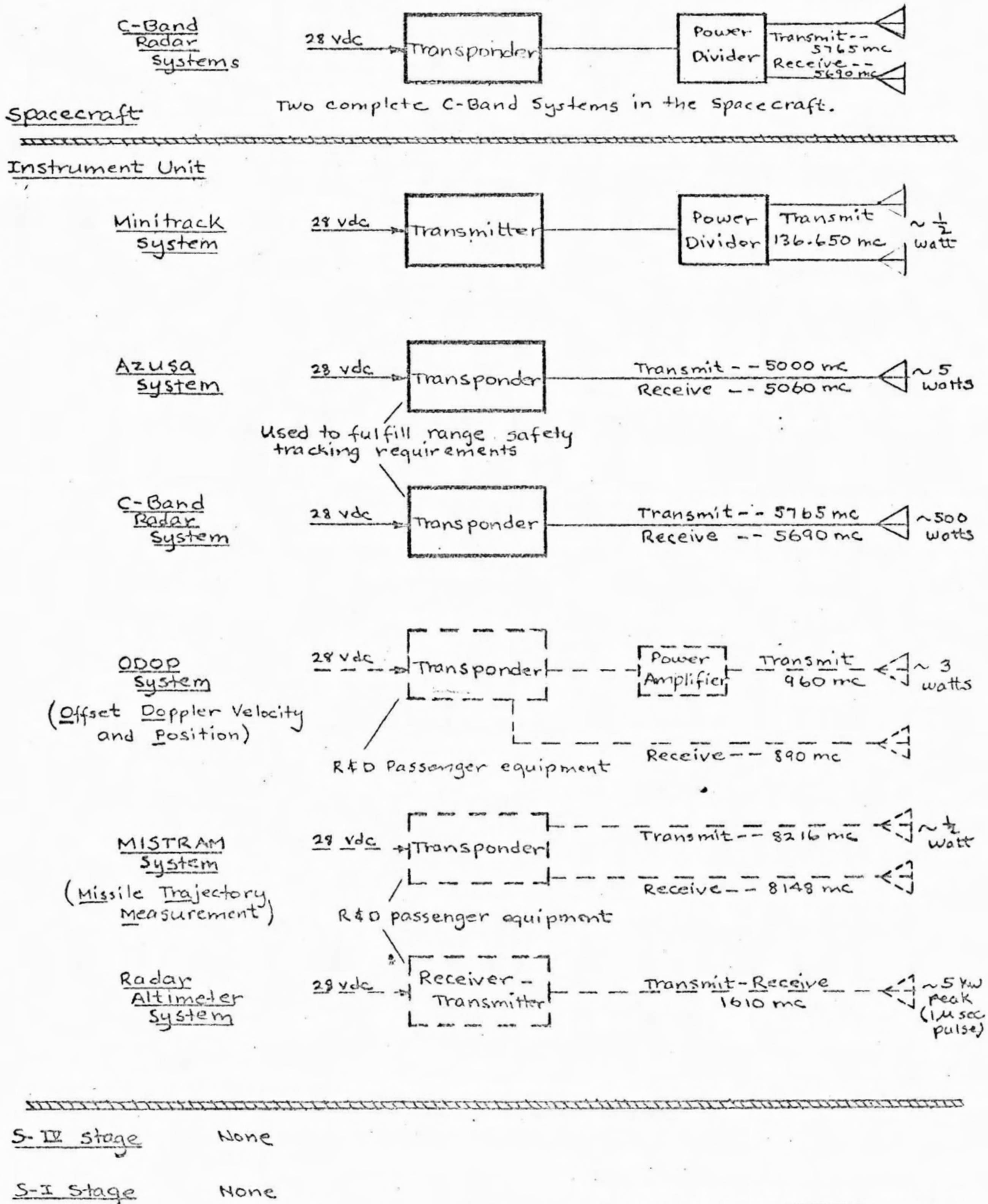
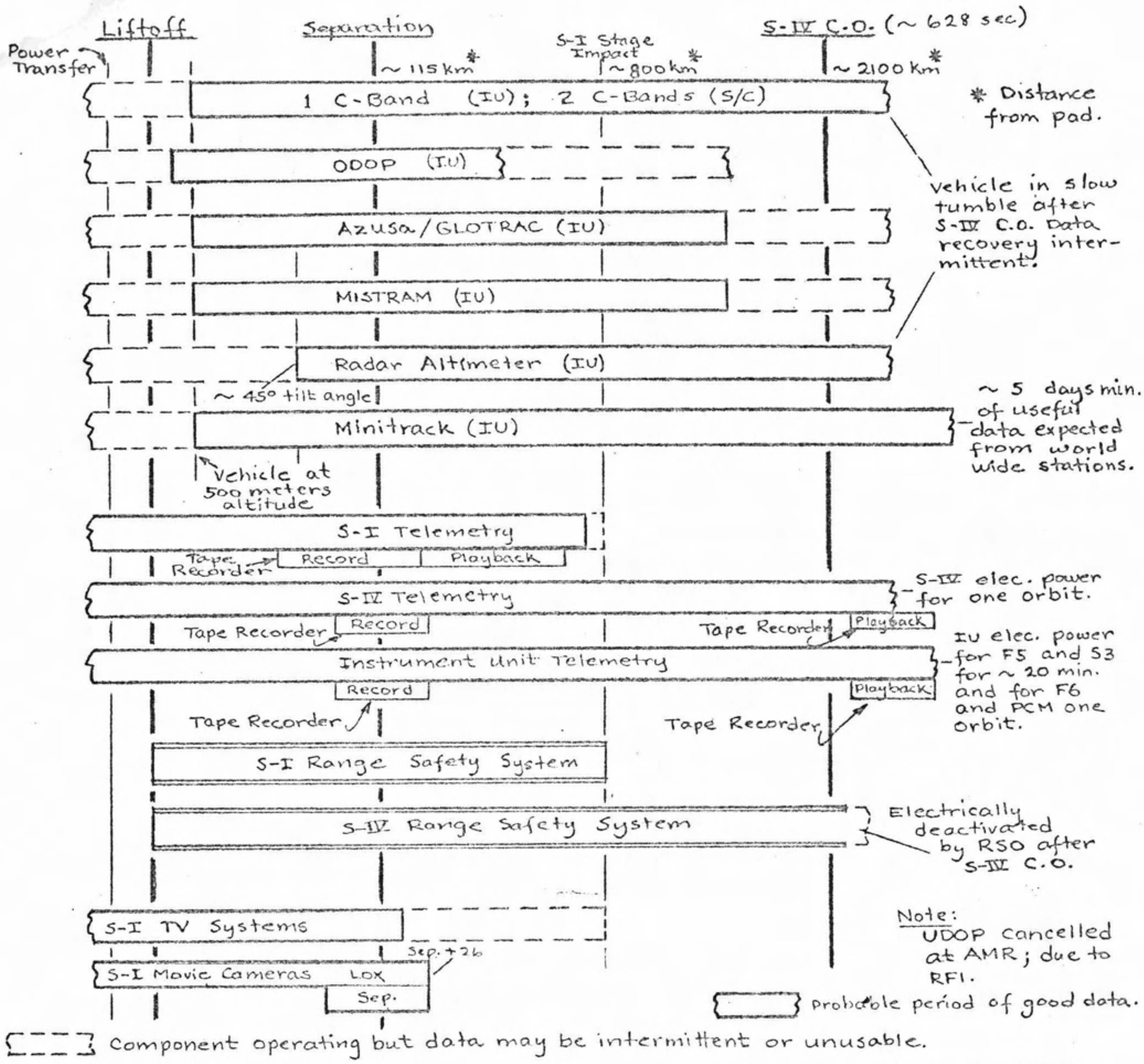


Figure 13

Vehicle Tracking Systems



Ground Tracking

Optical

1. CZR Cameras } Primarily used up to ~ 500 meter altitude. Targets for tracking
2. Theodolites } are vehicle paint patterns.

3. Long Range Cameras -- To track vehicle during flight until loss of view. Some orbital coverage expected.

Other

1. Skin Tracking -- The S-I stage after separation will be skin tracked by radars located at the Cape.

   -- The S-IV/IU/Payload will be skin tracked, while in orbit, by world wide stations.

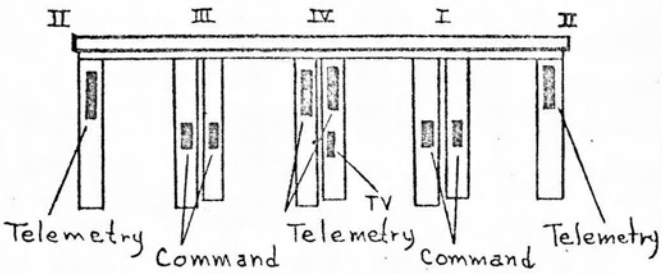
2. Glotrac -- If operational, will use the Azusa transponder after ~ 400 sec.

Figure 14

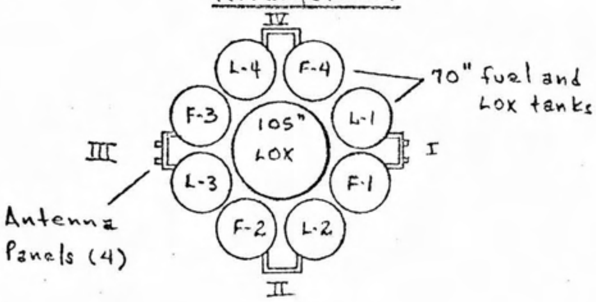
Tracking, Telemetry, Optical and Range Safety Coverage

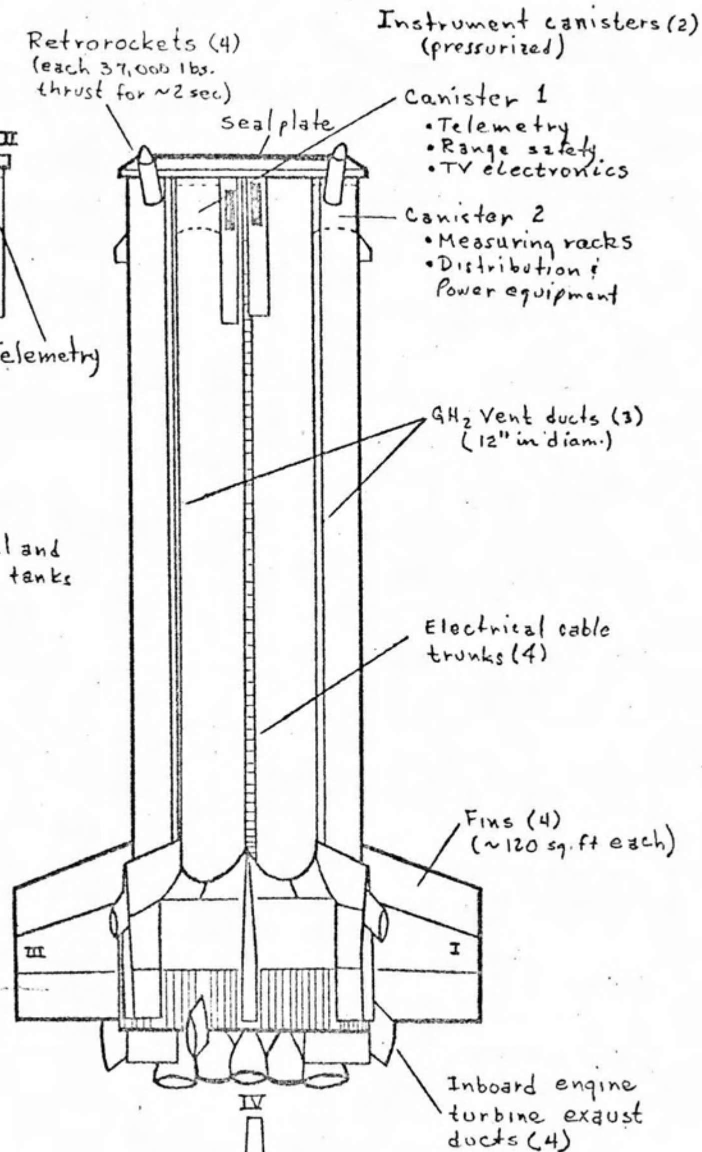
Note:
Tracking may be expected from the Cape until ~ 500 sec.

Antenna Arrangement



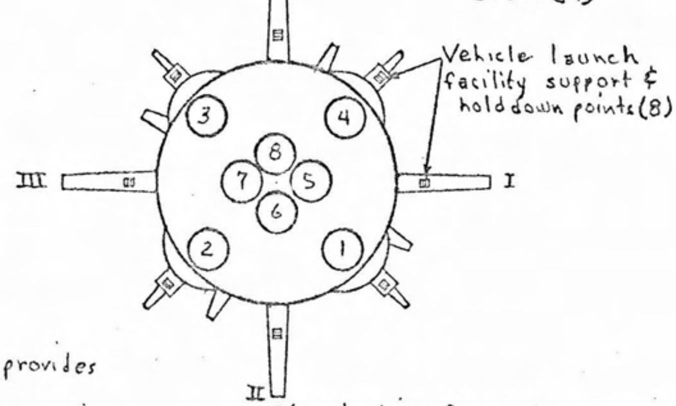
LOX and Fuel Tank Arrangement





S-I Stage Data

- Stage weight (includes S-IV aft interstage)
  - Dry ~ 109,000 lbs
  - At ignition ~ 997,000 lbs.
  - At separation ~ 130,000 lbs.
- Stage dimensions
  - Length 80.2 ft.
  - Diameter 21.4 ft.
  - (~40 ft. including fins)



Engine Data

- Start System: Solid propellant turbine spinner plus hypergolic fuel mixture.
- Four inboard engines fixed mounted; 3° cant angle.
- Four outboard engines gimbal mounted provides ± 8° gimballing; 6° cant angle
- ~188,000 lbs thrust each engine. (Total thrust ~1,500,000 lbs.)

Figure 15

S-I Stage Details

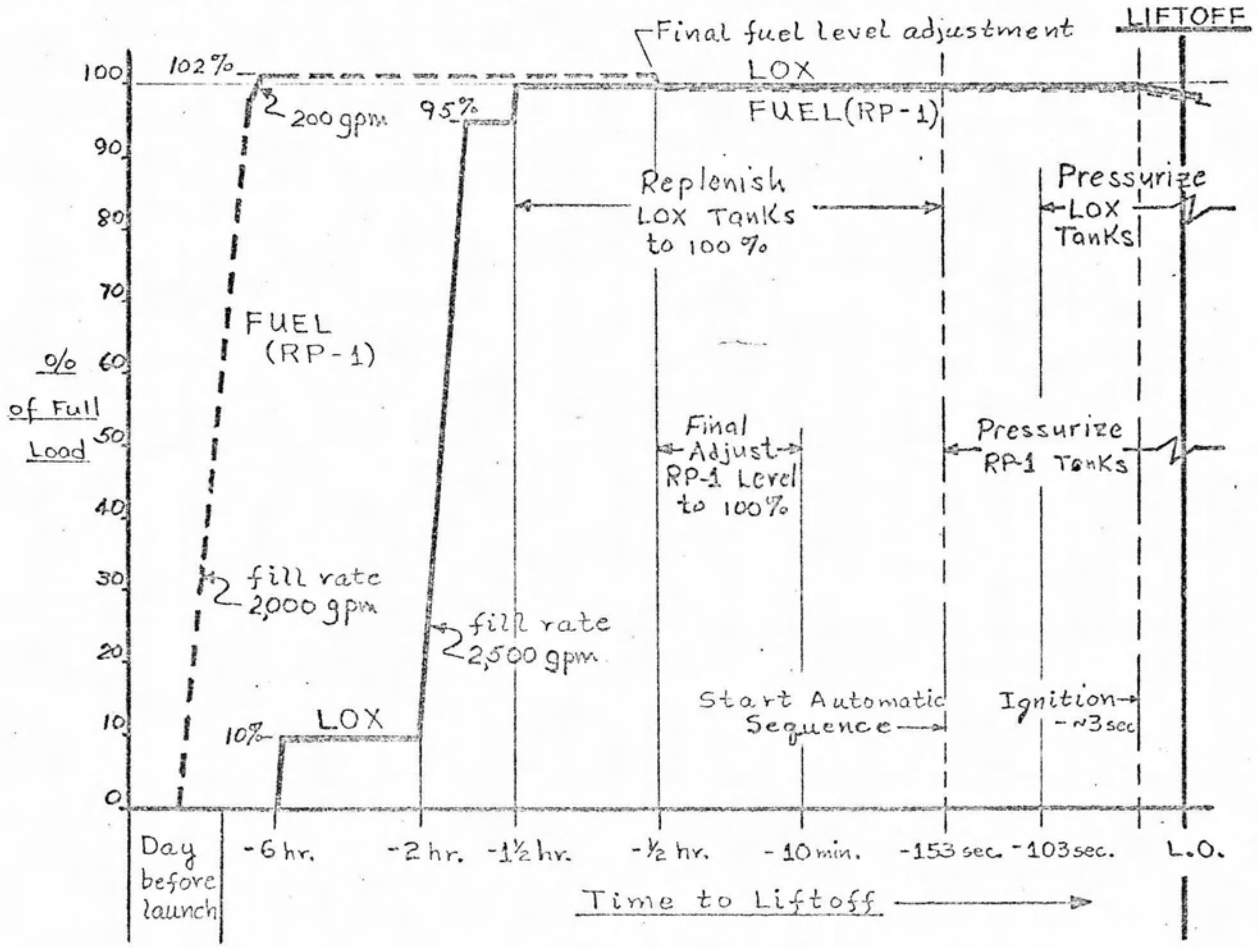
RP-1 System.

The sumps of all RP-1 tanks are interconnected to ensure uniform RP-1 usage. To prevent temperature stratification, in order to improve engine starting characteristics, GN_2 from a ground source is bubbled through the engines RP-1 suction lines from the start of LOX loading until T-150 seconds. The fuel tanks are pressurized with GN_2 from two, 20 cu. ft. spheres in order to meet the net positive suction head (NPSH) requirement of the engines.

LOX System.

The LOX tank sumps are interconnected for uniform LOX usage. To prevent temperature stratification of LOX in the engine suction lines and LOX tanks, cold helium from a ground supply is bubbled through the system and out the forward vent valves. Helium bubbling continues from the start of automatic sequencing until the propellant tanks are pressurized. Initial pressurization of the LOX tanks is provided by ground supplied helium. During flight GOX produced by a heat exchanger on each engine supplies the required LOX tank pressure.

S-I Stage Propellant Loading.



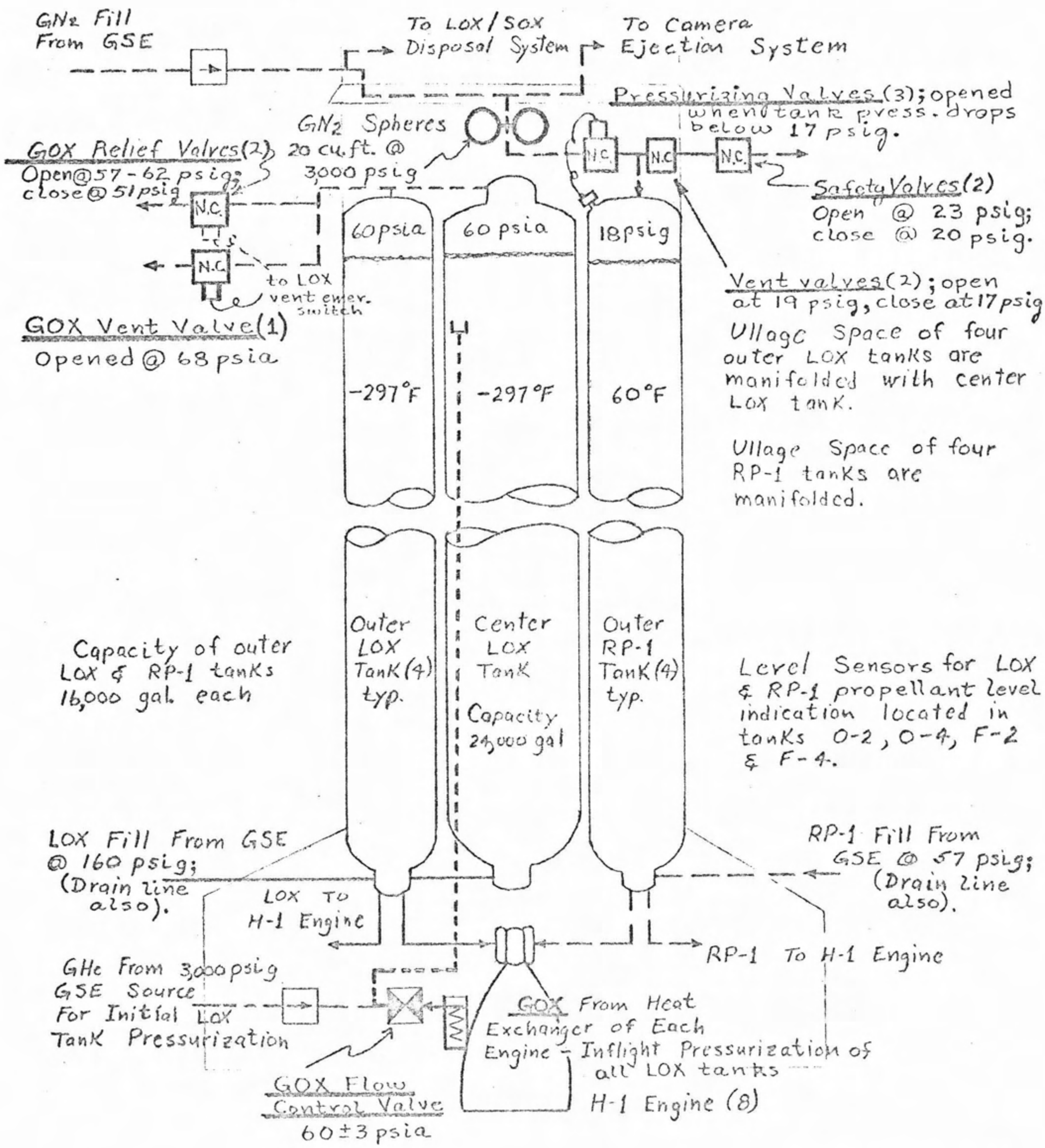


Figure 16

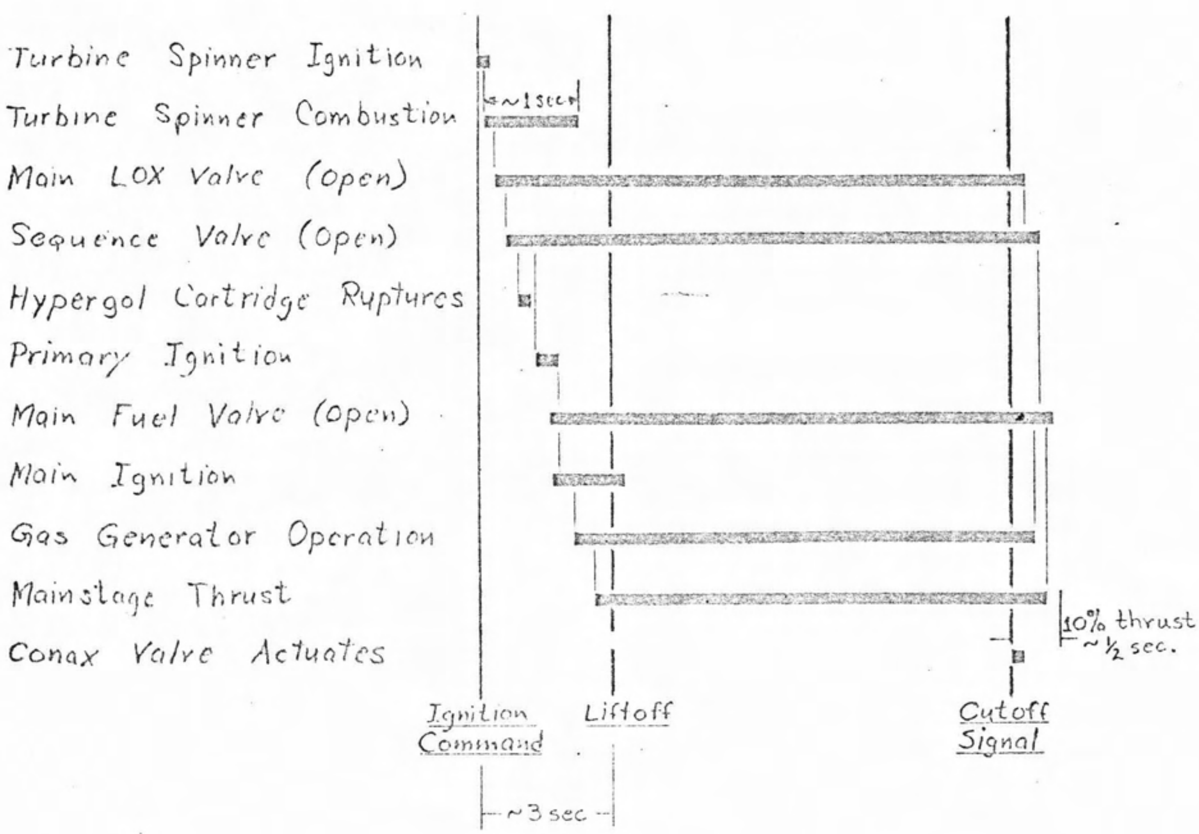
PROPELLANT SYSTEM, S-I

H-1 Engine Operation

The start signal from a ground source ignites the solid propellant gas generator (SPGG) or turbine spinner, which accelerates the LOX and RP-1 pumps. Increasing fuel pressure opens the main LOX valve which in turn opens the sequence valve. Fuel pressure ruptures the hypergolic cartridge. When the RP-1 and hypergolic fluid contact the LOX in the thrust chamber primary ignition occurs. Propellant pressure in the injection results in opening of the main fuel valve and also causes propellants to flow into the liquid propellant gas generator which provides sustained turbine operation.

Engine cutoff signals are initiated by the program device in the instrument unit through the S-I stage flight sequencer. The cutoff signal opens the explosively actuated Conax valve causing equalization of RP-1 pressure at the main LOX valve and the valve closes.

H-1 Operational Sequence



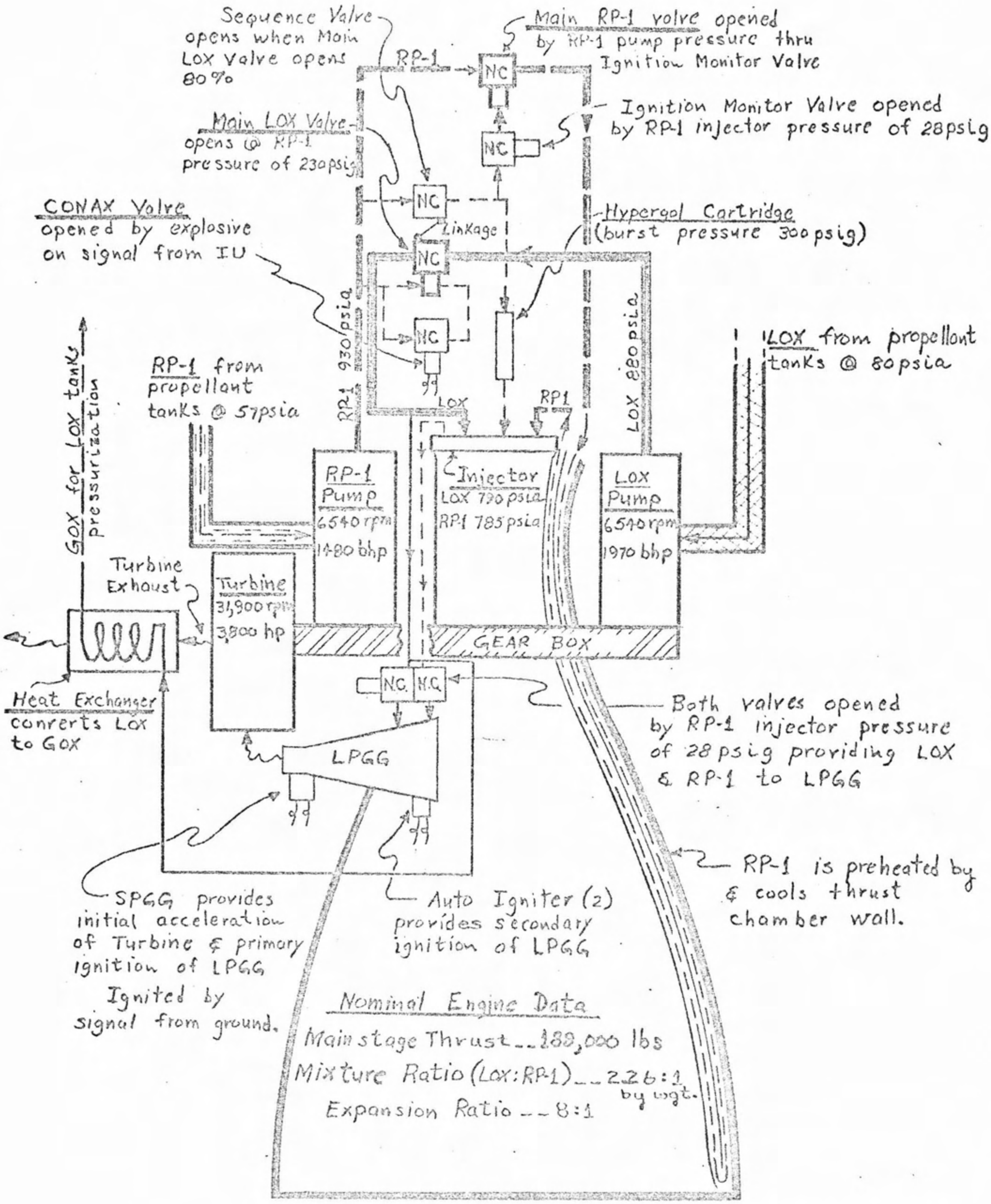


Figure 17

H-1 ENGINE

H-1 Engine Gimballing System

Each of the four outboard H-1 engines is suspended from the base of the stage thrust structure by a gimbal mount so that engine thrust can be directed for vehicle attitude control and steering. Two hydraulic actuators gimbal each engine in response to signals from the flight control computer located in the IU.

The actuators are part of an independent hydraulic system on each gimballed engine. Hydraulic fluid flows to the actuators from the high pressure accumulator and is returned to the low pressure reservoir. The electric motor driven auxiliary pump operates only during prelaunch checkout of the gimballing system.

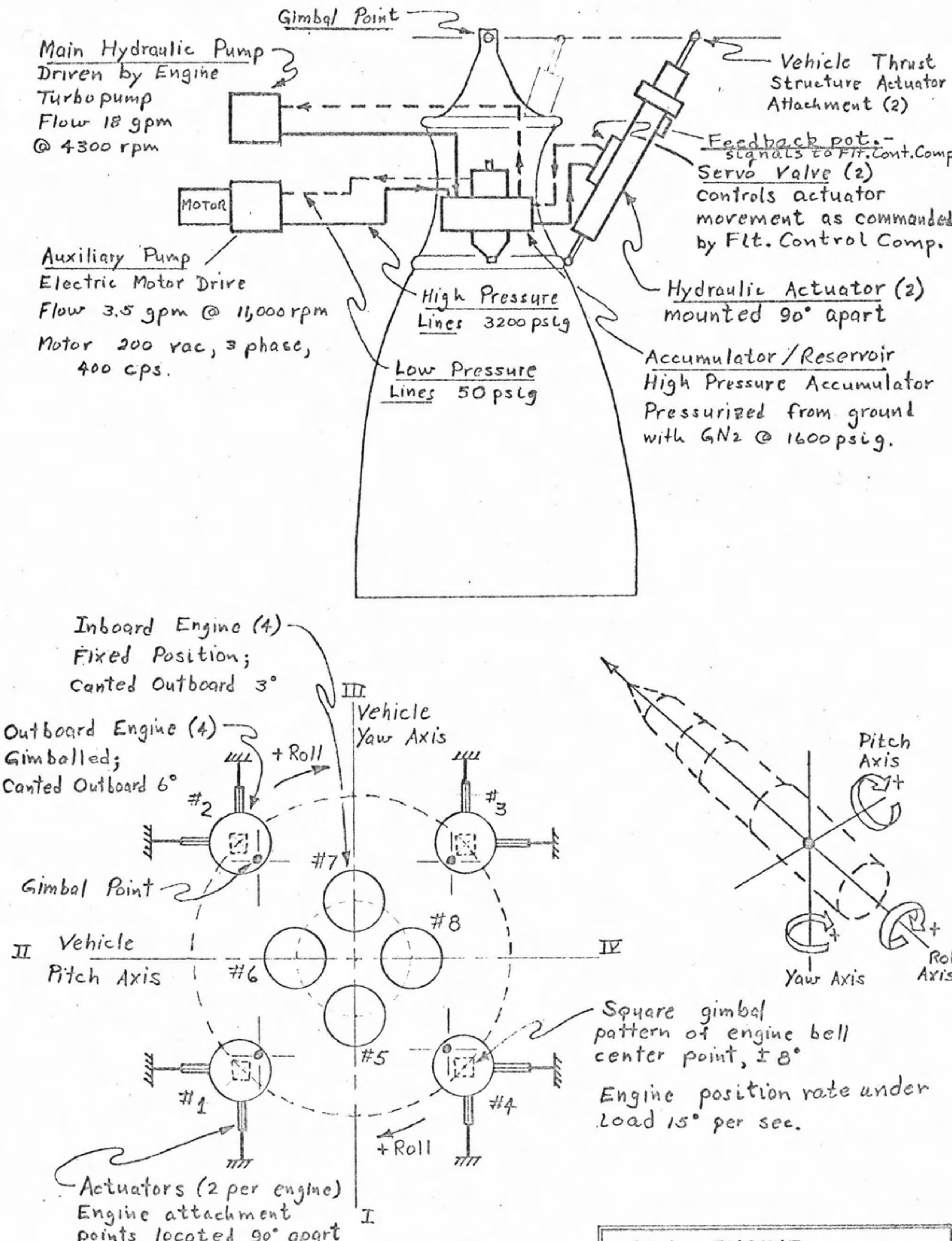


Figure 18

H-1 ENGINE GIMBALING SYSTEM

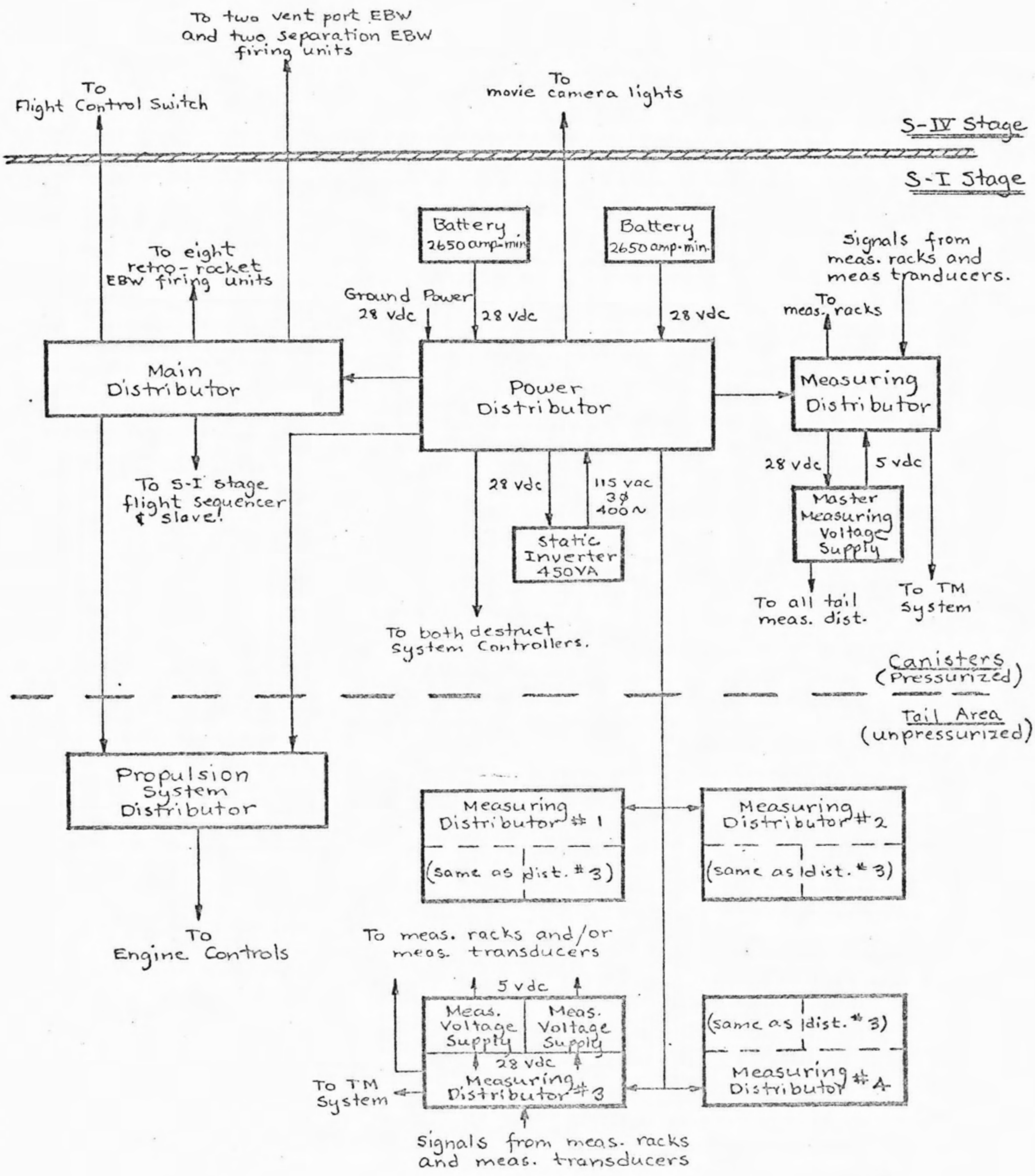


Figure 19 — S-I Stage Electrical Power and Distribution System

Telemetry System Capacity		
System	Data Channels	
	Continuous	Commutated
F1	13	216
F2	13	216
F3	13	54
F4	13	27
S1	11	16
S2	—	36
Totals	63	565
Grand Total = 628 channels		

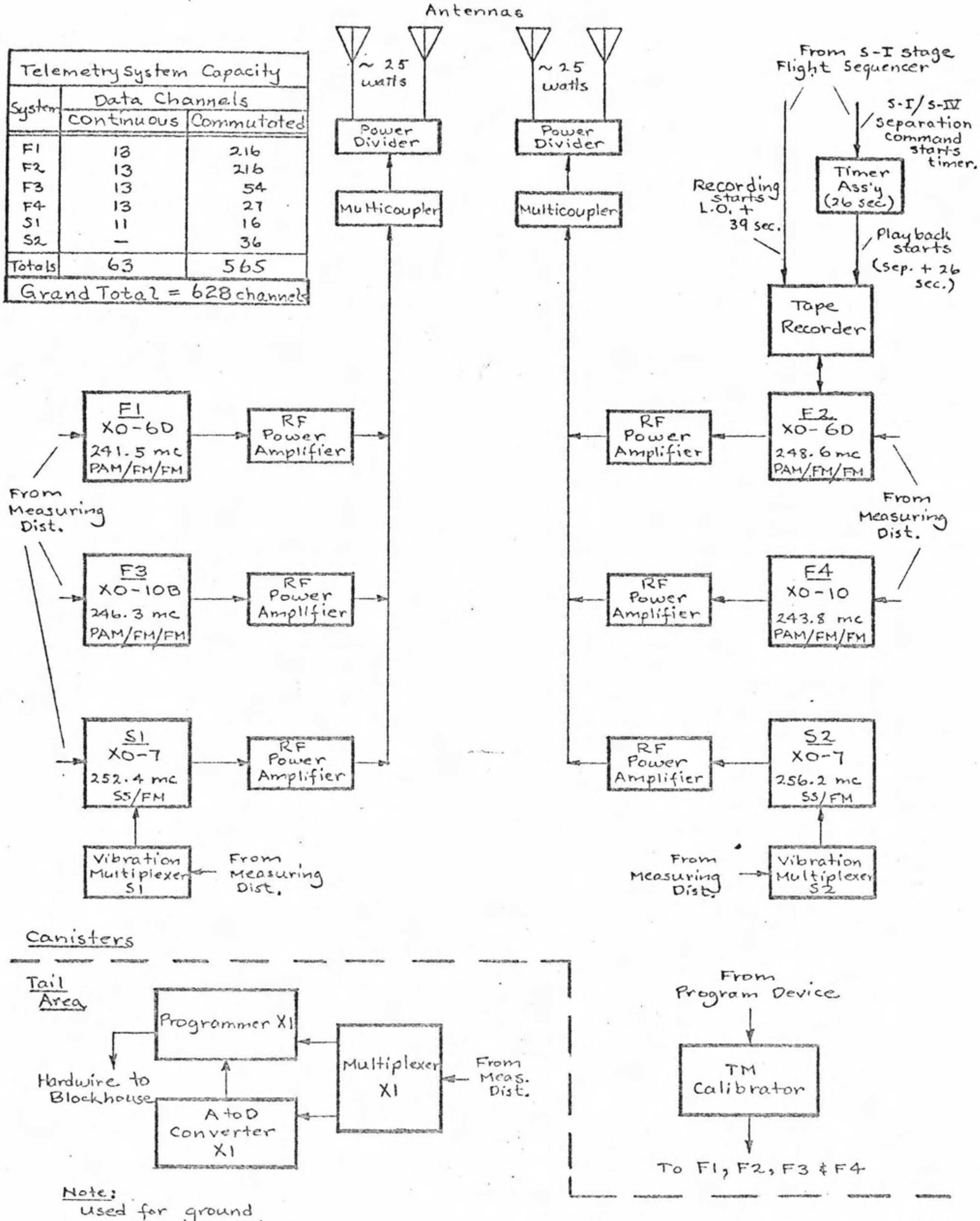


Figure 20

S-I Stage Telemetry System

Summary of Measurements	S-I	IU
Inflight		
1. Propulsion	51	—
2. Temperature . . .	195	15
3. Pressure	128	10
4. Strain & Vibration .	184	39
5. Guidance & Control .	—	88
6. Signals	38	5
7. Miscellaneous . . .	38	42
Total	~634	~199
Blockhouse	~105	~11

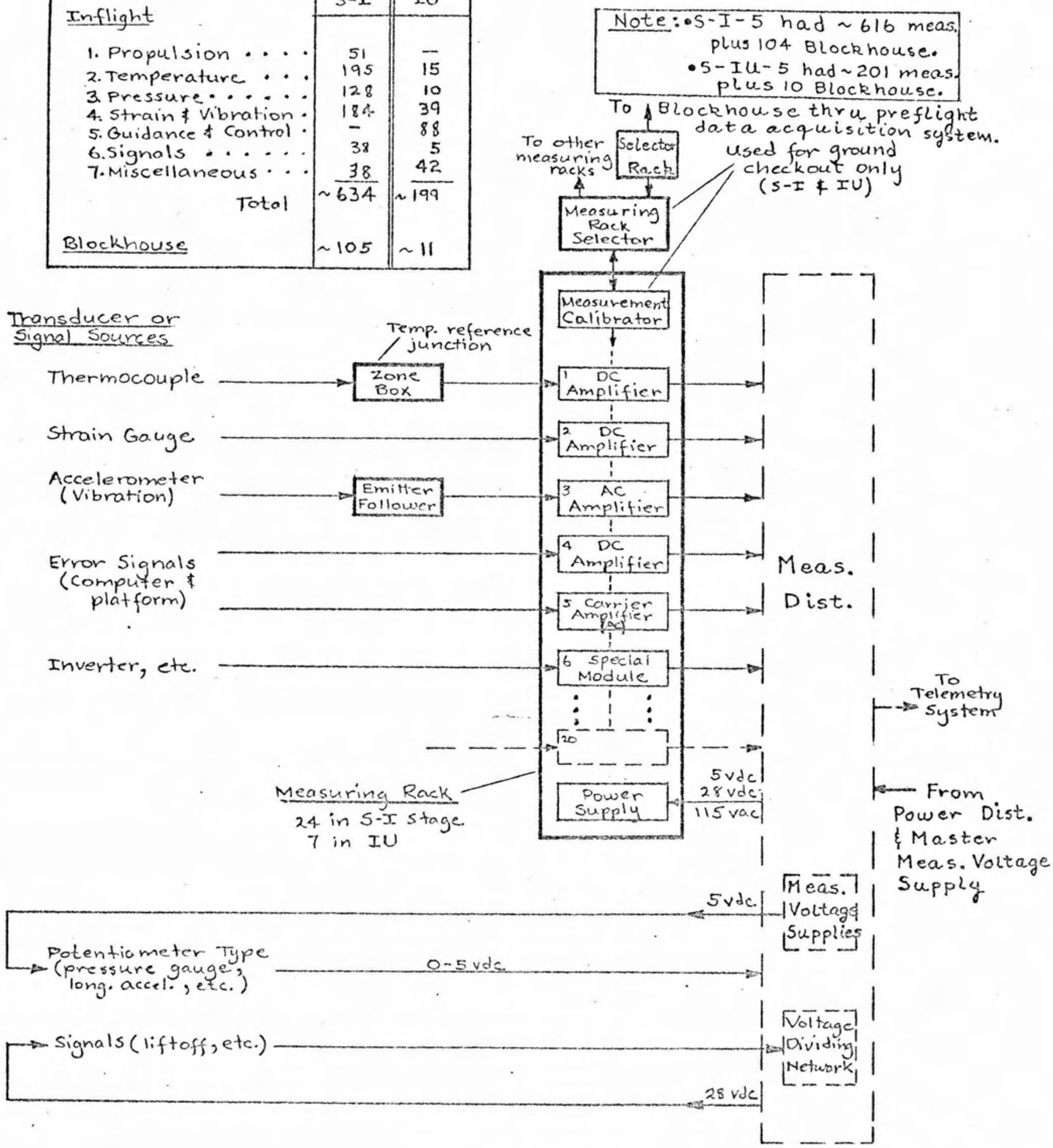
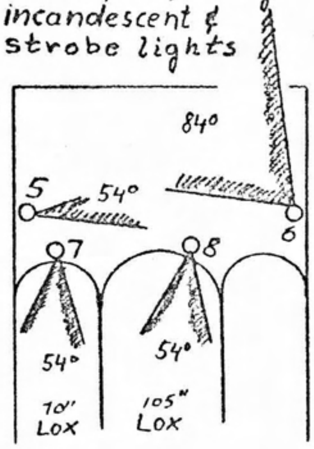
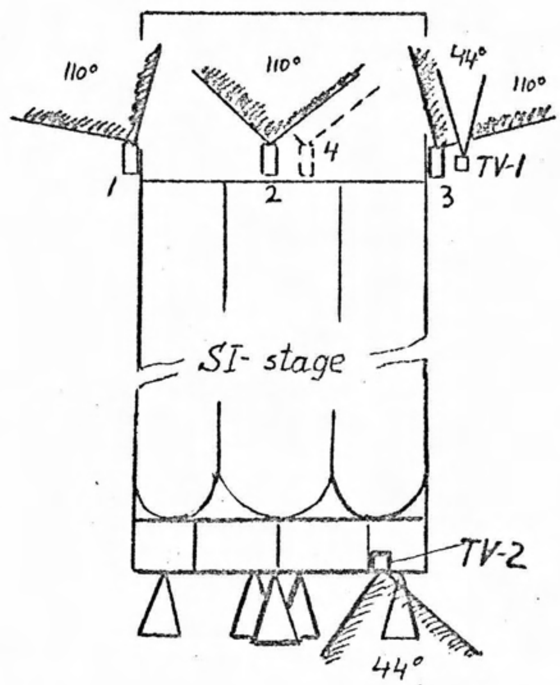


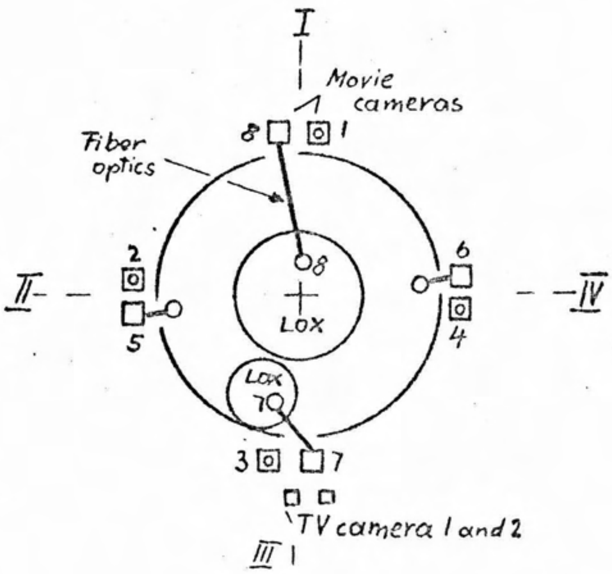
Figure 21 S-I Stage and Instrument Unit Measuring System

4 *Indirect* viewing cameras (Fiber optics) using incandescent & strobe lights



4 *Direct* viewing cameras plus 2 TV cameras





Cameras 5, 6, 7, 8 used for *indirect* viewing (fiber optics applied) of:

- 5- liquid + solid oxygen disposal
- 6- chill down, initial burning of S IV engines
- 7, 8- LOX-motion, emptying

Cameras 1, 2, 3, 4 and TV-1 used for *direct* viewing of S-I/S-IV separation;
- interstage vent ports opening;
- ullage- and retrorockets ignition and initial burning;
- S-IV engines initial burning;
- booster tumbling.

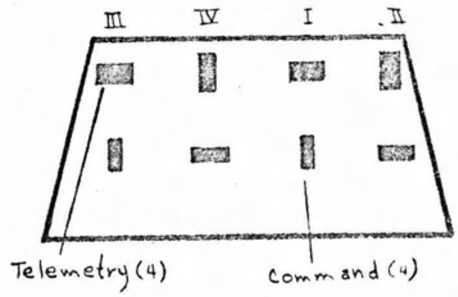
All 8 *movie* cameras will be *ejected* 26 sec. after separation.
Recovery by parachute, floating bag radar + light beacon.

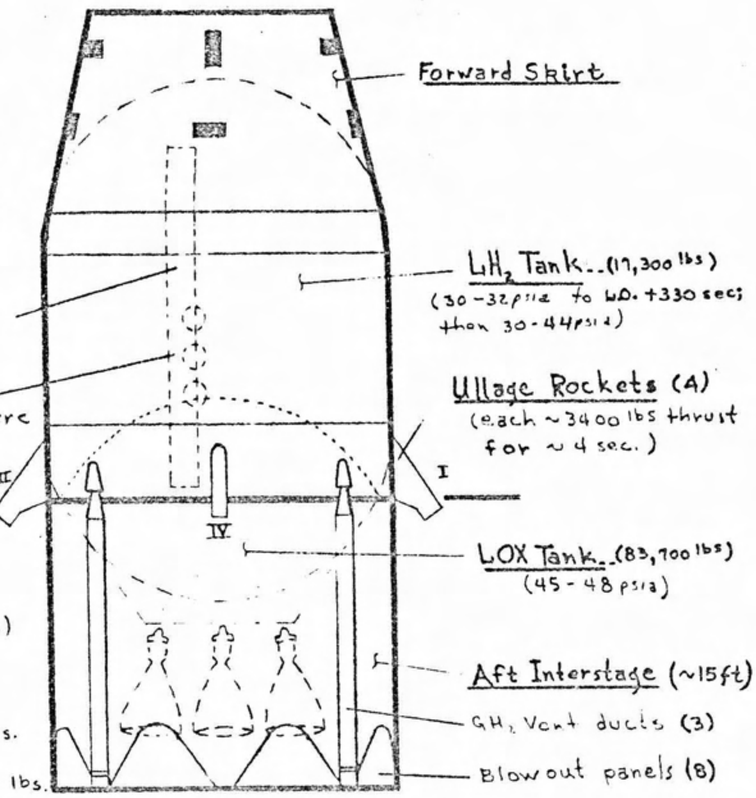
TV-2 is used to look at:
- wraparound lines; the gas generator;
- the heat exchanger;
- engine curtain.

Figure 22

TV and Movie Camera Systems

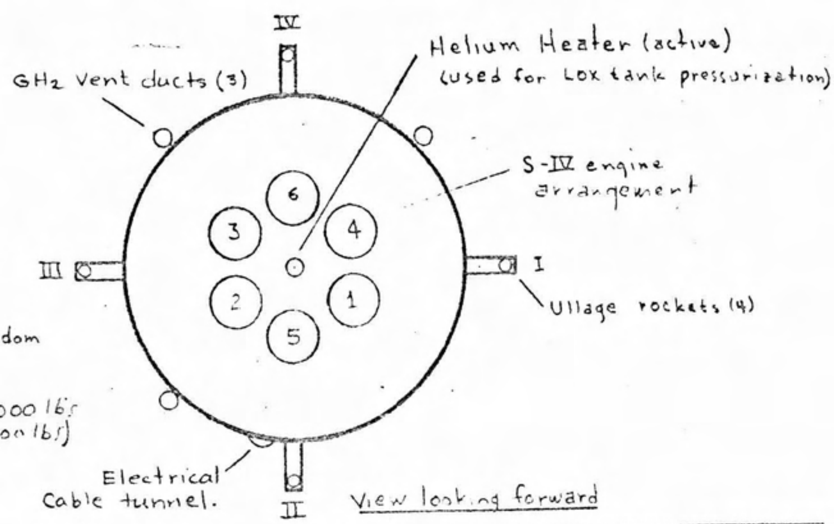
Antenna Arrangement





Stage Data

- S-IV Stage weights (w/o Aft interstage)
  - Dry ~13,800 lbs.
  - At S-I stage ignition ~116,000 lbs.
  - At injection (S-IV wt. only) ~14,300 lbs. (Depletion weight)
- S-IV Stage Dimensions
  - Length ~41 ft.
  - Diameter ~18 ft.



Engine Data

- Six LOX/LH_2 engines gimbal mounted at 6° cant angle.
- Gimballing provides ± 4° freedom in square pattern
- Each engine rated at ~15,000 lbs thrust (Total thrust ~90,000 lbs)

Figure 23

S-IV Stage Details

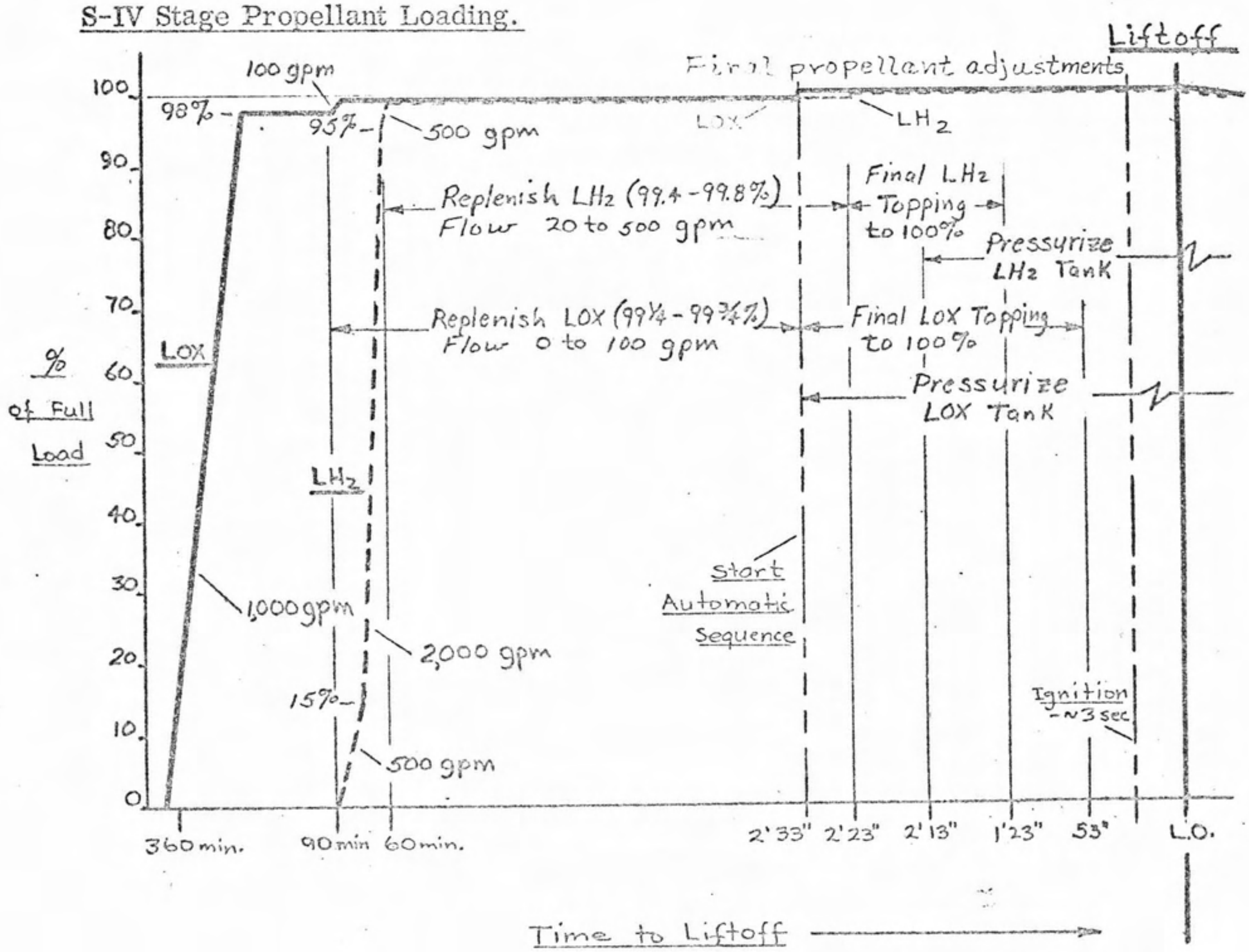
LOX System.

The LOX tank is filled from a 30 psig ground storage system. Loading is controlled by the propellant utilization (P. U.) system which receives data from the LOX mass sensor. Pressurization of the LOX tank is required to meet the net positive suction head (NPSH) requirements of the RL10A-3 engines. At the start of automatic sequencing the tank is pressurized to 47 psia by cold helium from a ground source. Tank pressure is maintained during S-I burn by cold helium from the spheres in the LH_2 tank. During S-IV burn, helium from these spheres is heated and expanded by the helium heater. If tank pressure drops to 40 psia the ten back up pressurization spheres supply necessary tank pressure.

LH_2 System.

The LH_2 tank is filled from a 70 psig ground source. The P. U. system controls LH_2 loading receiving data from the LH_2 mass sensor. Pressurization of the LH_2 tank is required for increased stage structural strength and to meet NPSH requirements of the RL10A-3 engines. Prior to liftoff the tank is pressurized to 37 psia by cold helium from a ground source. During LH_2 cooldown the tank pressure is maintained at 37 psia by the make-up helium sphere. After engine ignition GH_2 from each RL10A-3 engine maintains a tank pressure of 35-38 psia.

S-IV Stage Propellant Loading.



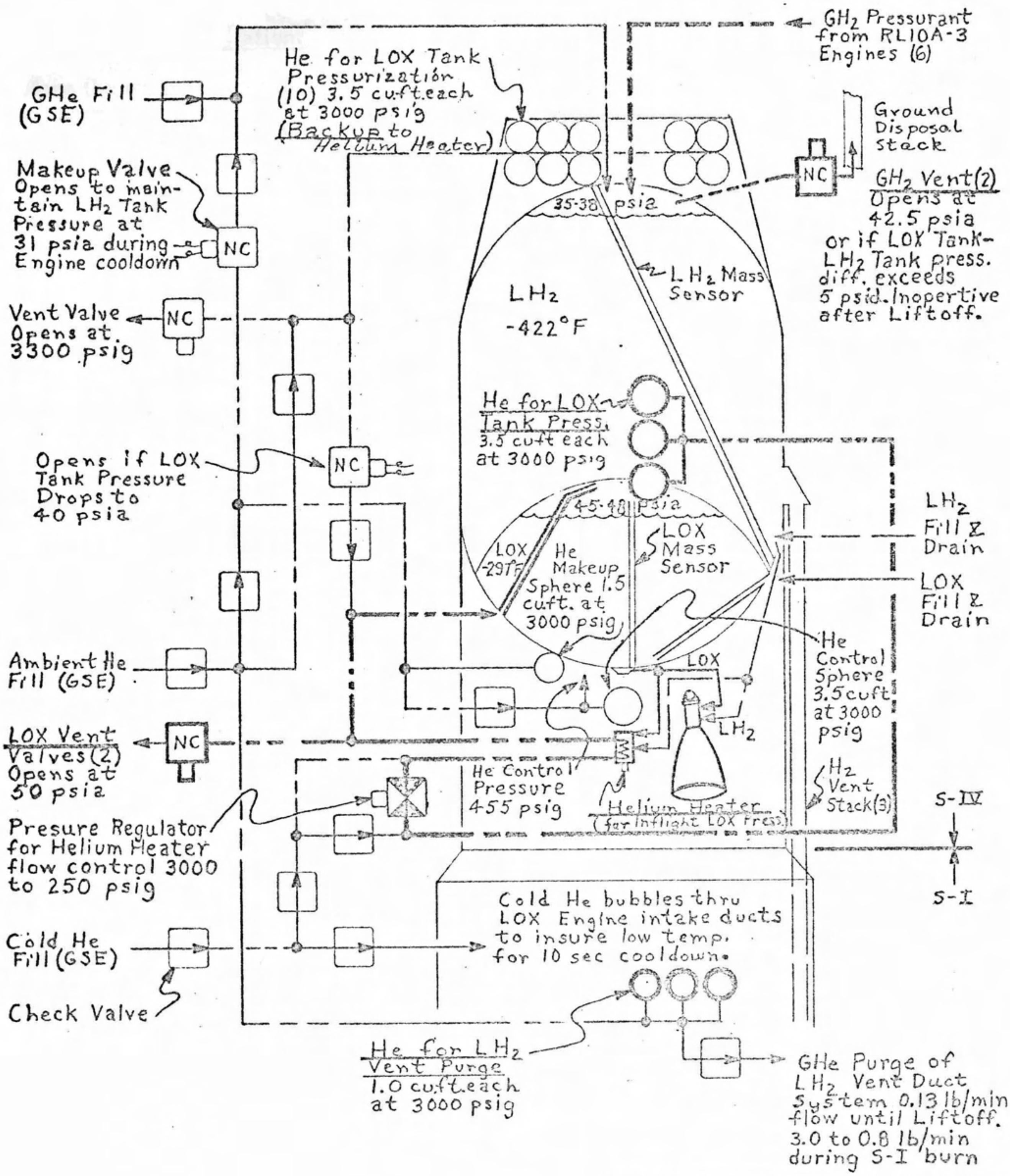


Figure 24 PROPELLANT SYSTEM, S-IV

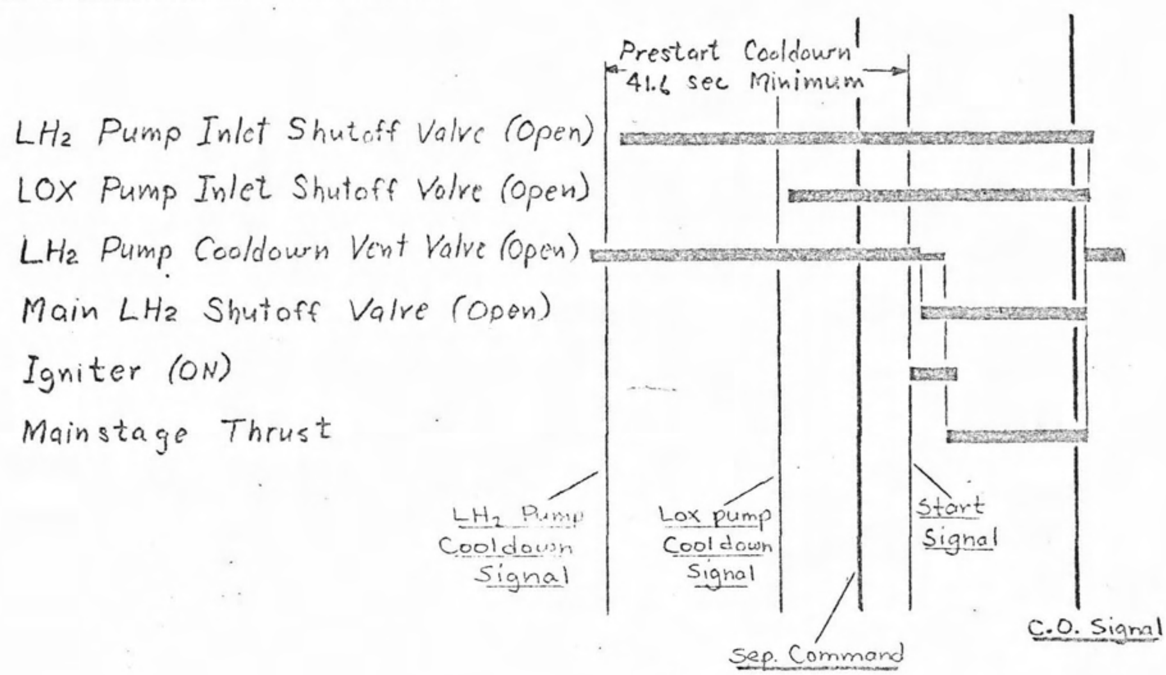
RL10A-3 Engine Operation.

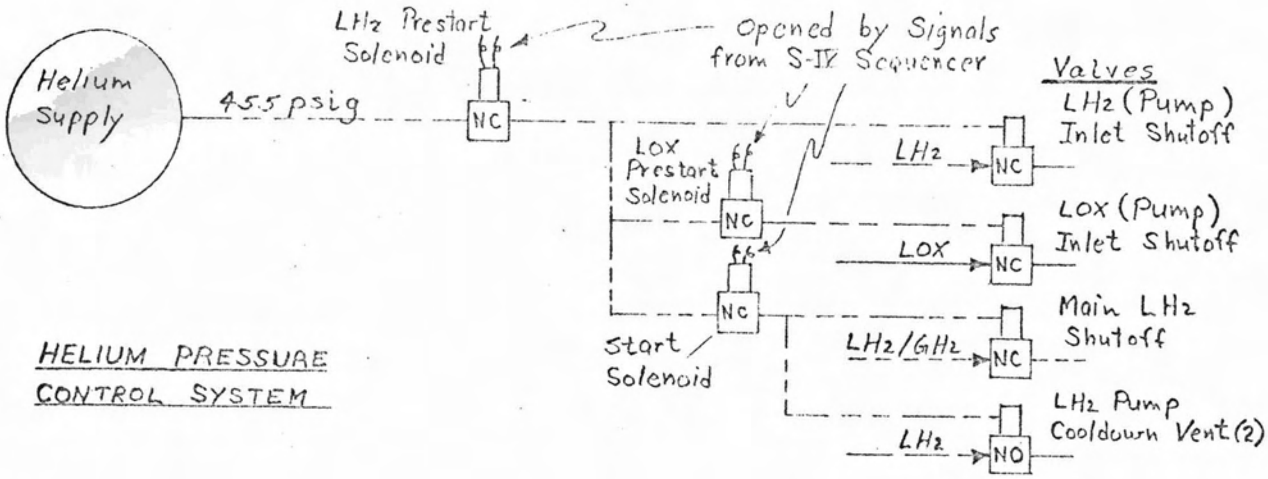
Before the engines are started, LH_2 and LOX flow through the engine lines at a reduced rate in order to cool down engine components to prevent the vaporization of LH_2 or LOX. Vaporized propellants would cause pump cavitation, and GOX in the propellant utilization valve or the propellant injector would restrict LOX flow.

During engine operation LH_2 flows from the fuel pump through the thrust chamber cooling tubes where it cools the thrust chamber, absorbs heat and is converted to GH_2. The GH_2 expanding through the turbine drives the LH_2 and LOX pumps.

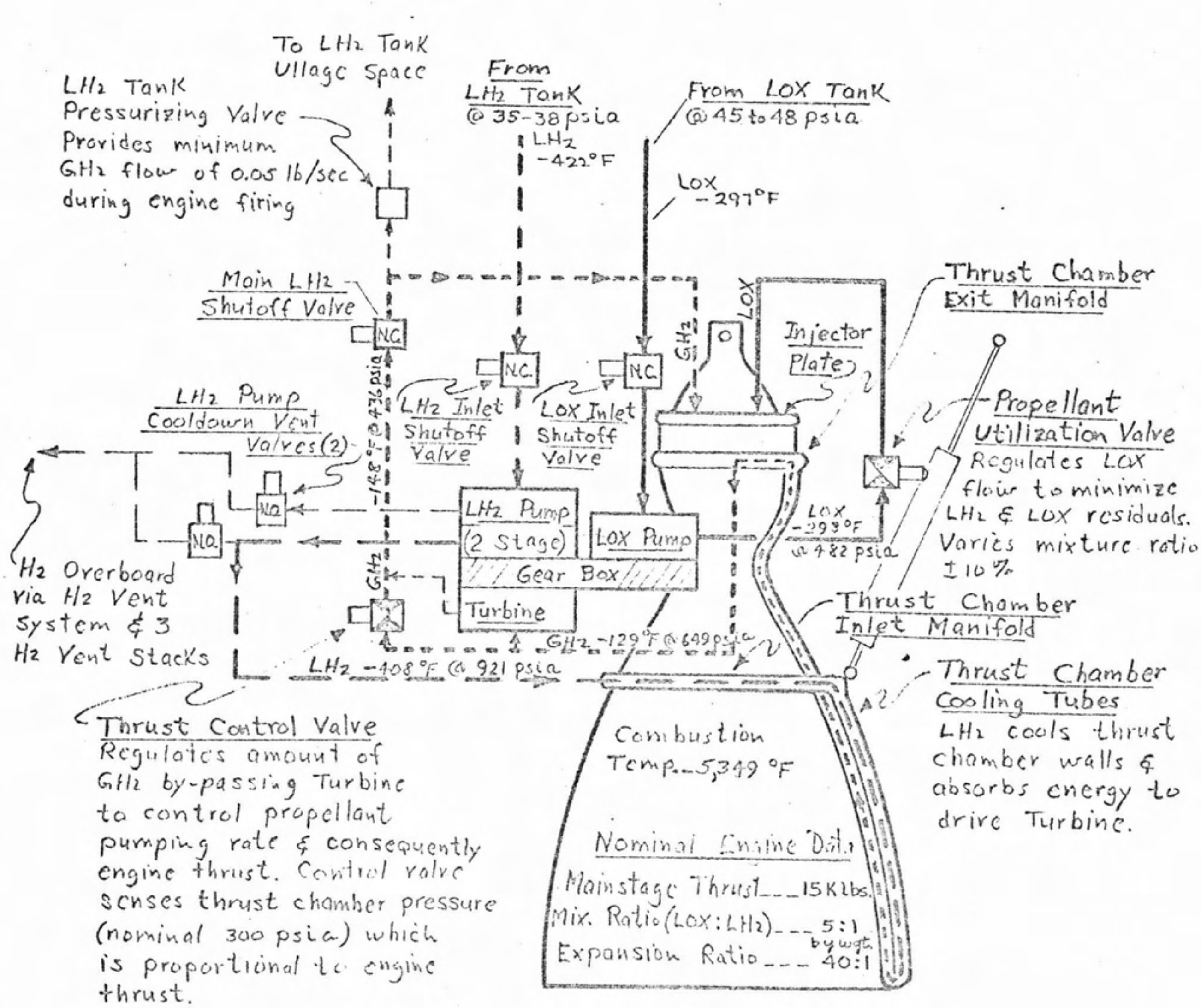
The three engine shutoff valves and the two LH_2 pump cooldown valves are activated by helium at 455 psig which is controlled by the LH_2 prestart, LOX prestart and start solenoids.

RL10A-3 Operational Sequence.





HELIUM PRESSURE CONTROL SYSTEM



Turbopump Data -

- Turbine—592 HP @ 28,400 rpm
- LH2 Pump—28,400 rpm
- LOX Pump—11,350 rpm

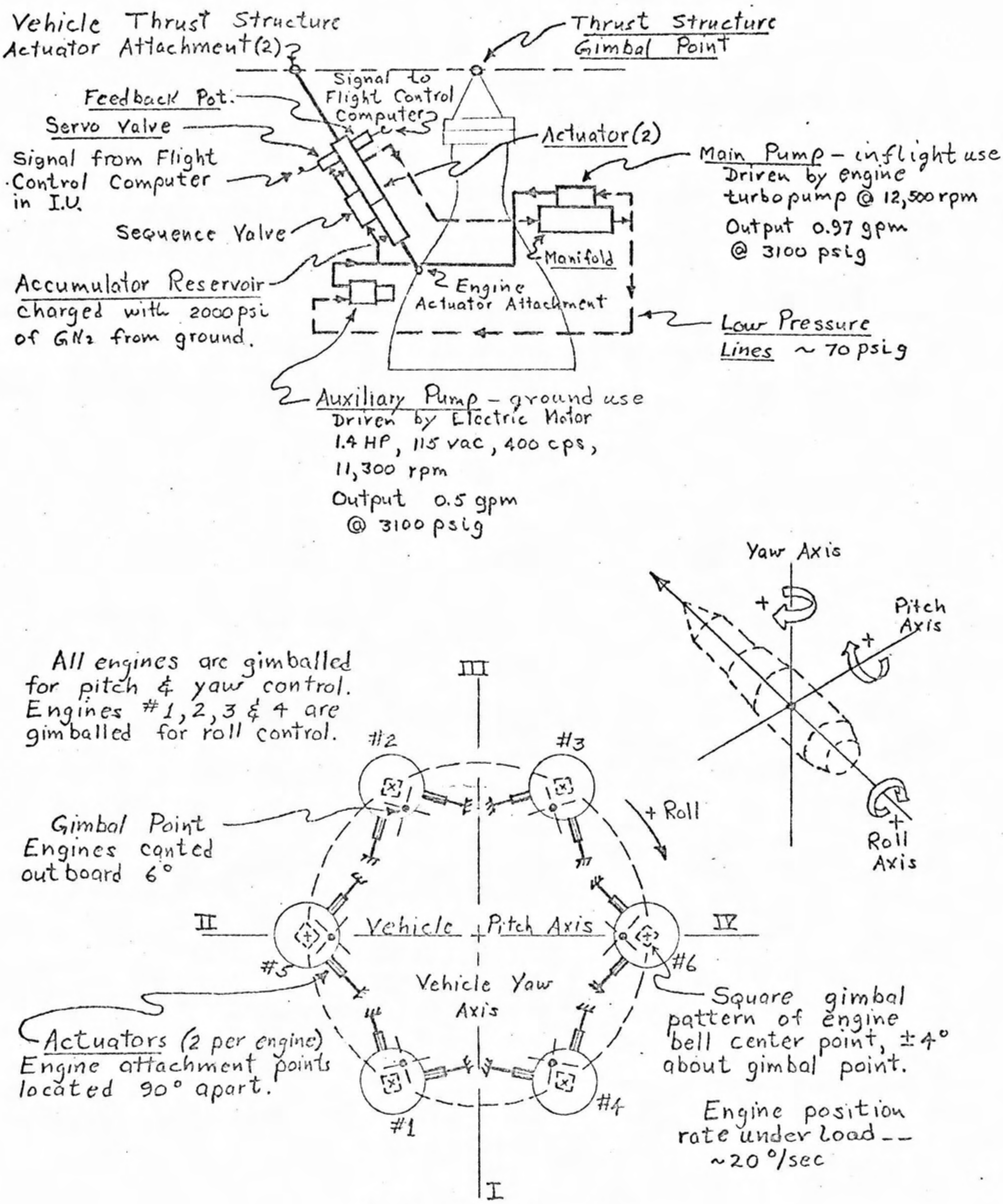
Figure 25

RL10A-3 ENGINE

RL10A-3 Engine Gimbaling System.

Each of the six RL10A-3 engines is suspended by a gimbal mount from the base of the stage thrust structure so that engine thrust can be directed for vehicle attitude control and steering. Two hydraulic actuators gimbal each engine in response to signals from the flight control computer in the instrument unit.

The two hydraulic actuators are part of an independent hydraulic system on each engine which includes two pumps which provide hydraulic pressure to each actuator. The electric motor driven pump operates only during prelaunch check-out of the gimbaling system.



GIMBALLING PATTERN
(looking forward)

Figure 26

RL10A-3 ENGINE
GIMBALING SYSTEM

Propellant Utilization System

The P.U. system controls propellant loading and engine mixture ratios (LOX to LH_2) to assure balanced consumption of LOX and LH_2. The system controls propellant loading, measures propellant mass at liftoff and stage ignition to within ±2% and limits engine mixture ratio errors. The system can vary the nominal 5 to 1 (LOX to LH_2) mixture ratio ±10%.

The S-IV engine cutoff arming signal from the P.U. system is a backup to the program device arming signal. The step pressurization command increases the flow of GH_2 to the LH_2 tank raising the pressure to about 40 psia at S-IV cutoff.

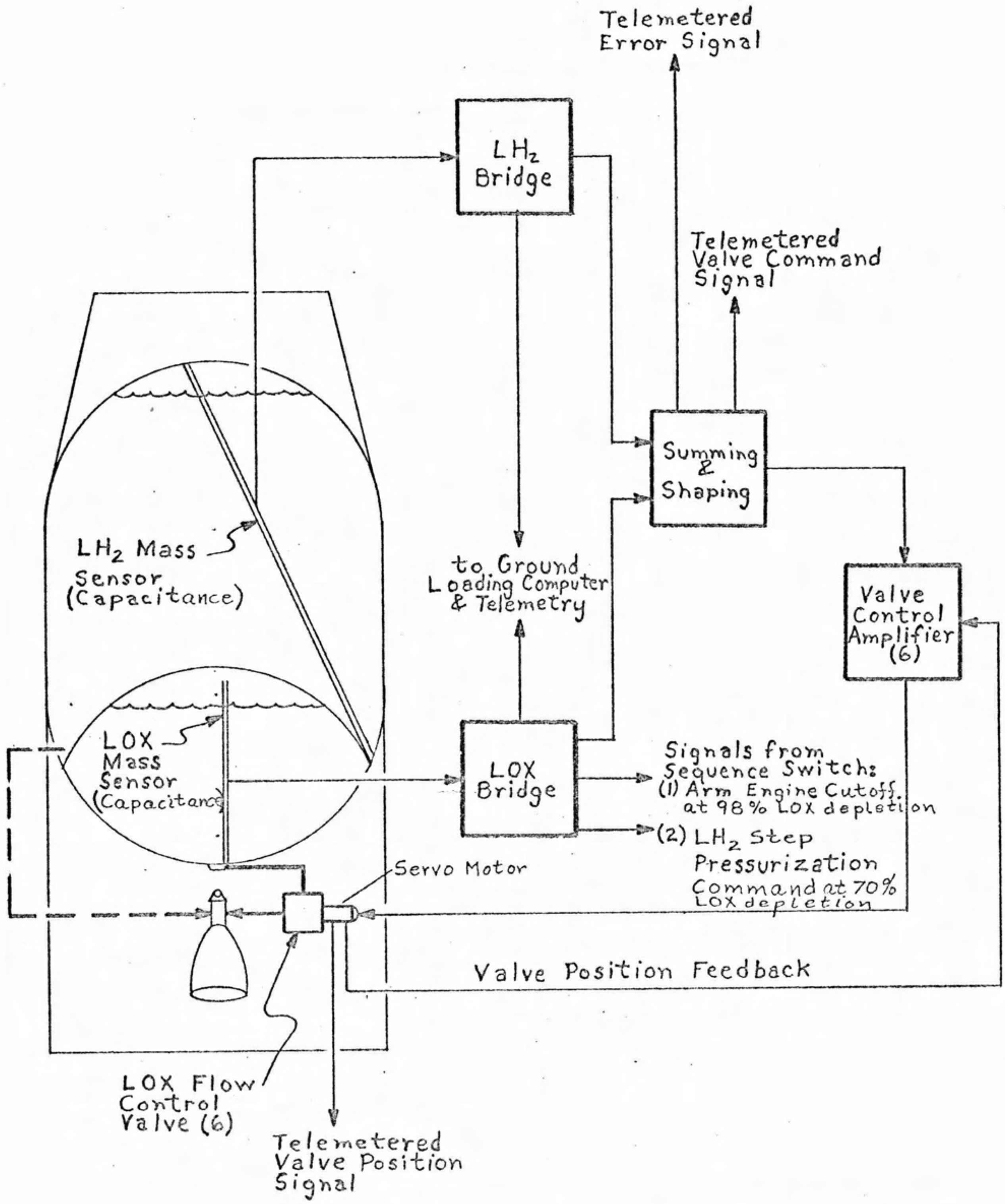


Figure 27

PROPELLANT UTILIZATION SYSTEM, S-IV

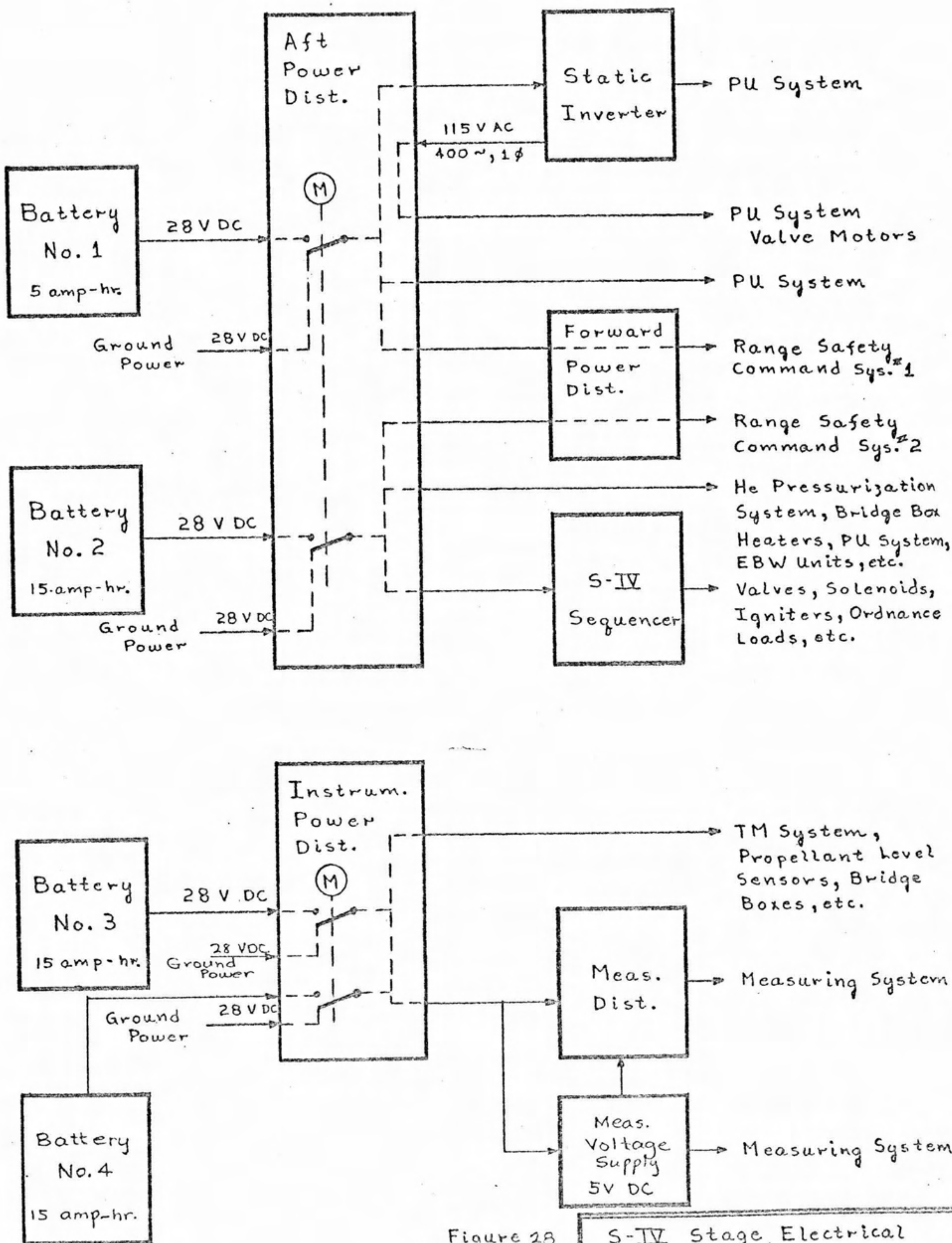
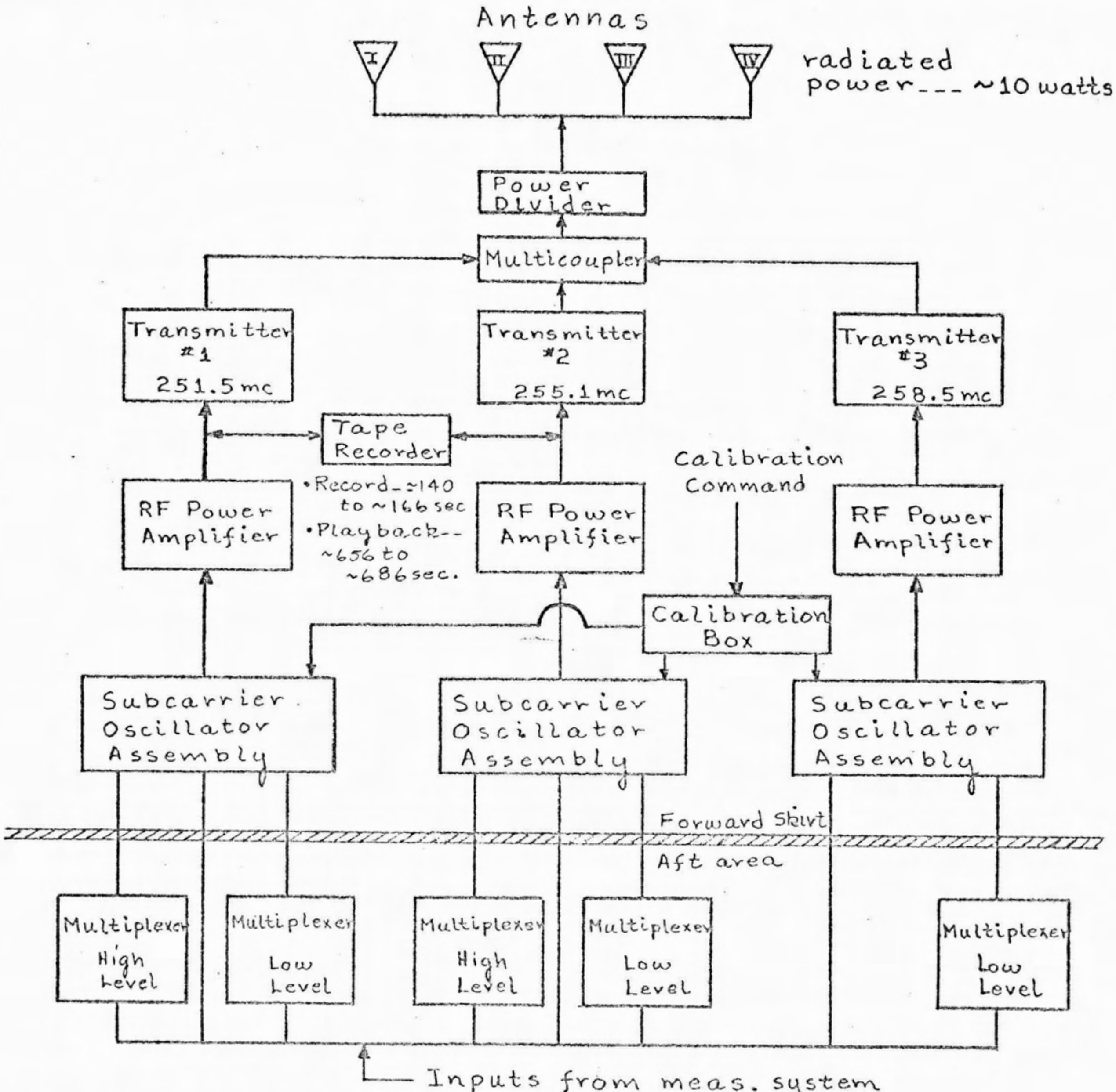


Figure 28 S-IV Stage Electrical Power and Distribution System

Telemetry System Capacity	
* System No.	Data Channels
1	139 (8 FM; 131 PDM)
2	139 " "
3	52 (9 FM; 43 PDM)
Total Capacity	330

* all three telemetry systems are PDM/FM/FM --- Pulse Duration Modulation/ Freq. Mod./ Freq. Mod.



• High Level Multiplexer---
90 channels; 10 pulses/sec; 70 Kc channel.
• Low Level Multiplexer---
45 channels; 2.5 pulses/sec; 40 Kc channel.

Figure 29

S-IV Stage Telemetry System

Summary of Measurements	
Inflight	
1. Propulsion ·······	12
2. Temperature ·····	105
3. Pressure ·······	103
4. Strain & Vibration ··	23
5. Flight Mechanics ··	7
6. Signals ········	52
7. Voltage & Current ·	39
8. Miscellaneous ····	17
Total	~358
Blockhouse	~103*

Note: • S-IV-5 had ~ 342 meas. plus ~ 93 Blockhouse.

* All blockhouse meas. are also used as inflight meas.

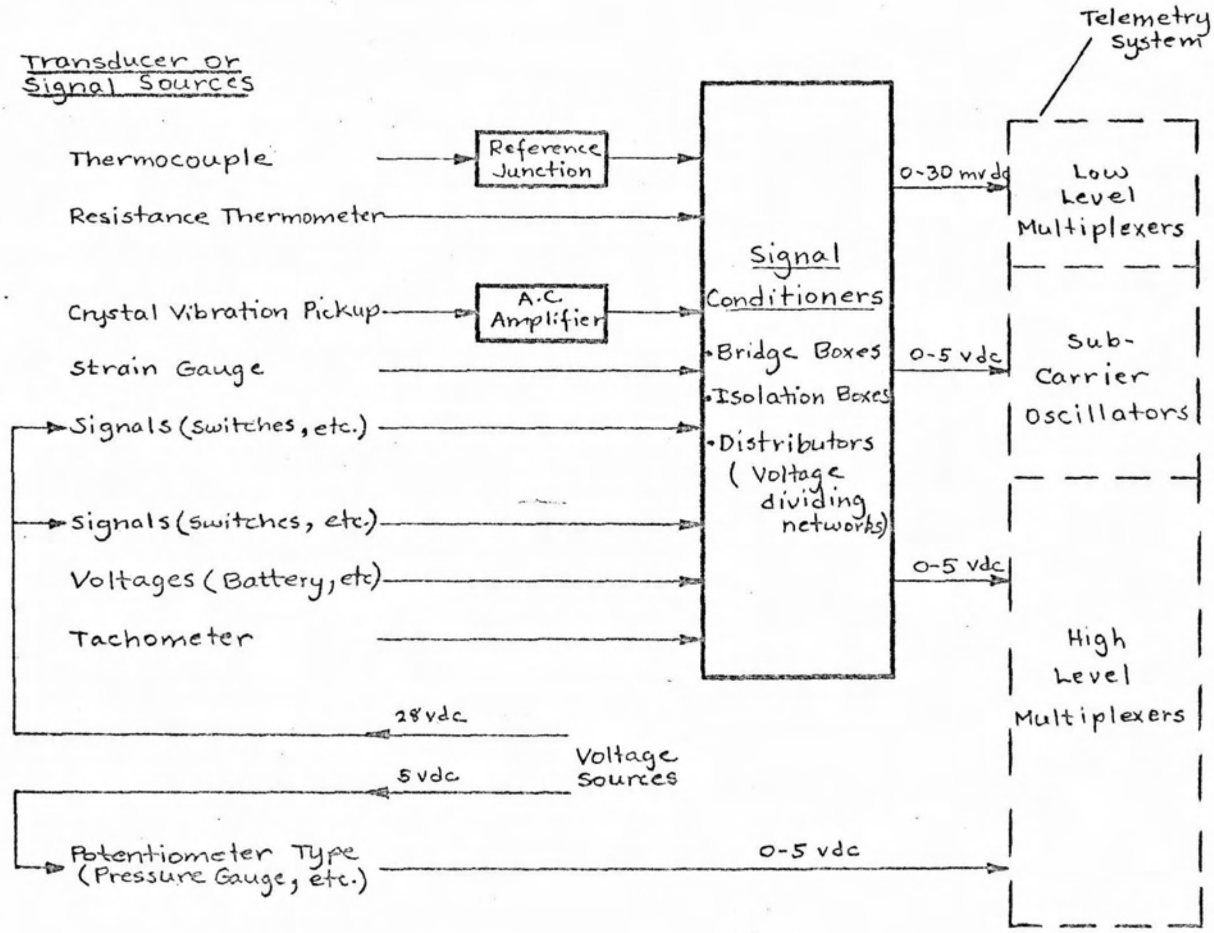
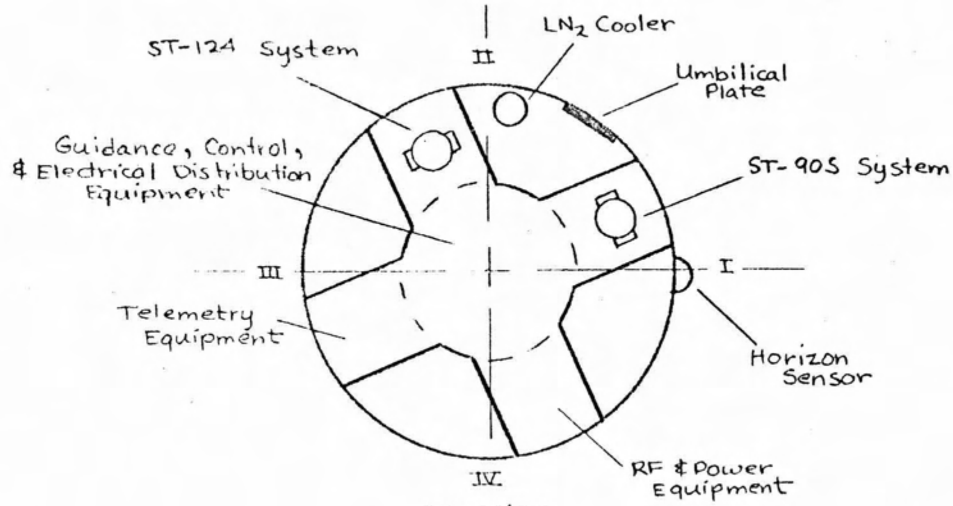
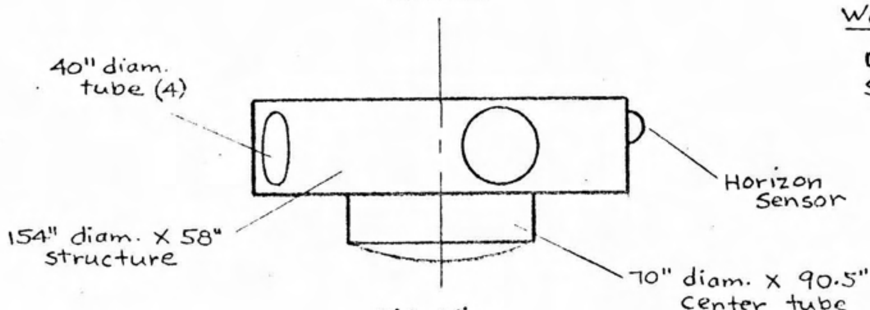


Figure 30 S-IV Stage Measuring System



Top View



Side View

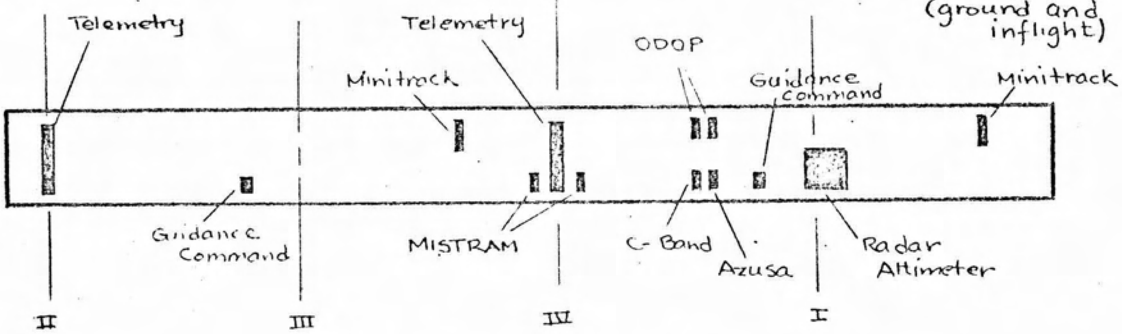
Weight

Dry: ~ 6050 lbs.
Serviced: ~ 6100 lbs.

IU Environmental Parameters

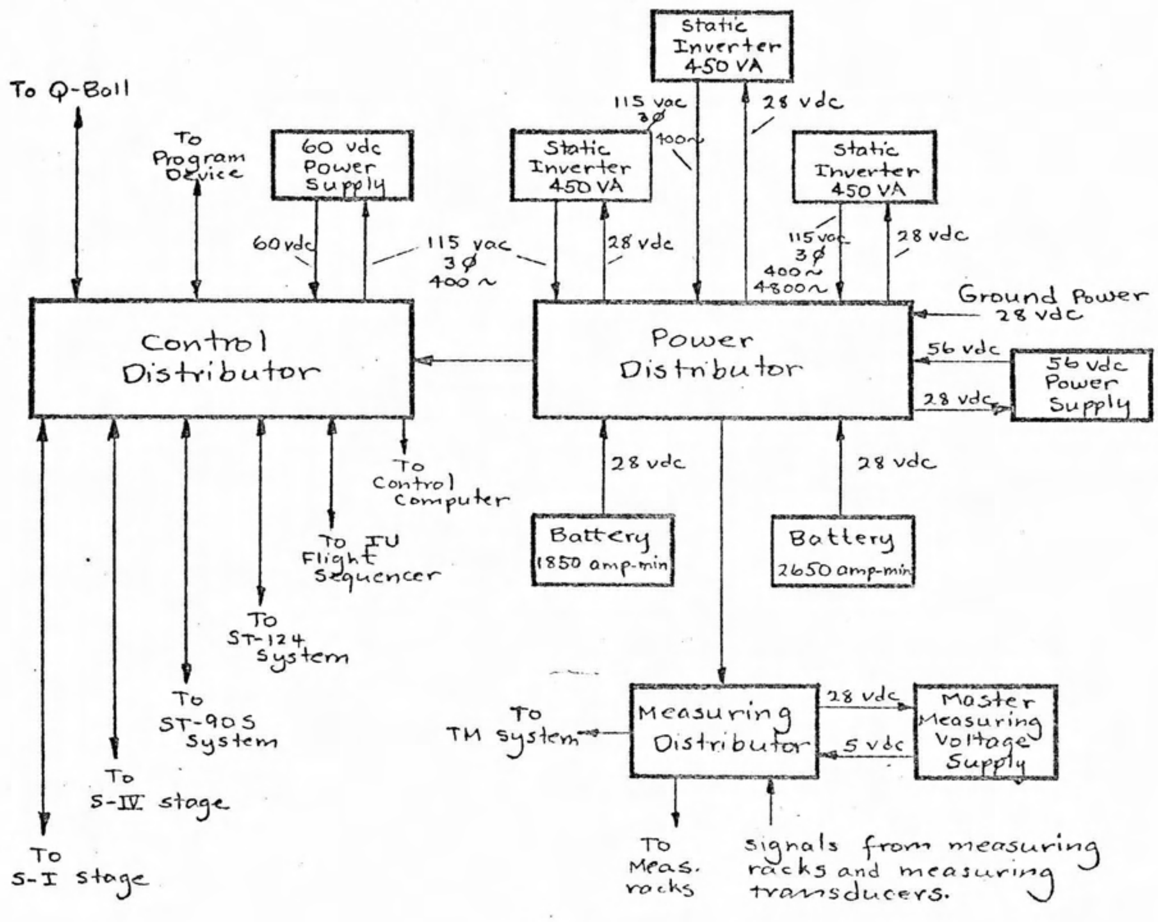
Pressure -- ~ 15.1 psia on ground
~ 15.8 to 16.9 psia inflight

Temperature -- 0 to 50°C overall; 25 ± 5°C for ST-124 & ST-90S areas; 20 to 40°C for batteries (ground and inflight)



Antenna Arrangement

Figure 31 — Instrument Unit Details



These components located in a pressurized area.

Figure 32 Instrument Unit Electrical Power and Distribution System

Telemetry System Capacity		
System	Data Channels	
	Continuous	Commutated
F5	27	—
F6	25	8
PCM	—	83
S3	2	42
Total	54	133
Grand Total = 187 channels		

PAM/FM/FM (Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation)

SS/FM (Single Sideband/Frequency Modulation)

PCM (Pulse Code Modulation)

DDAS (Digital Data Acquisition System)

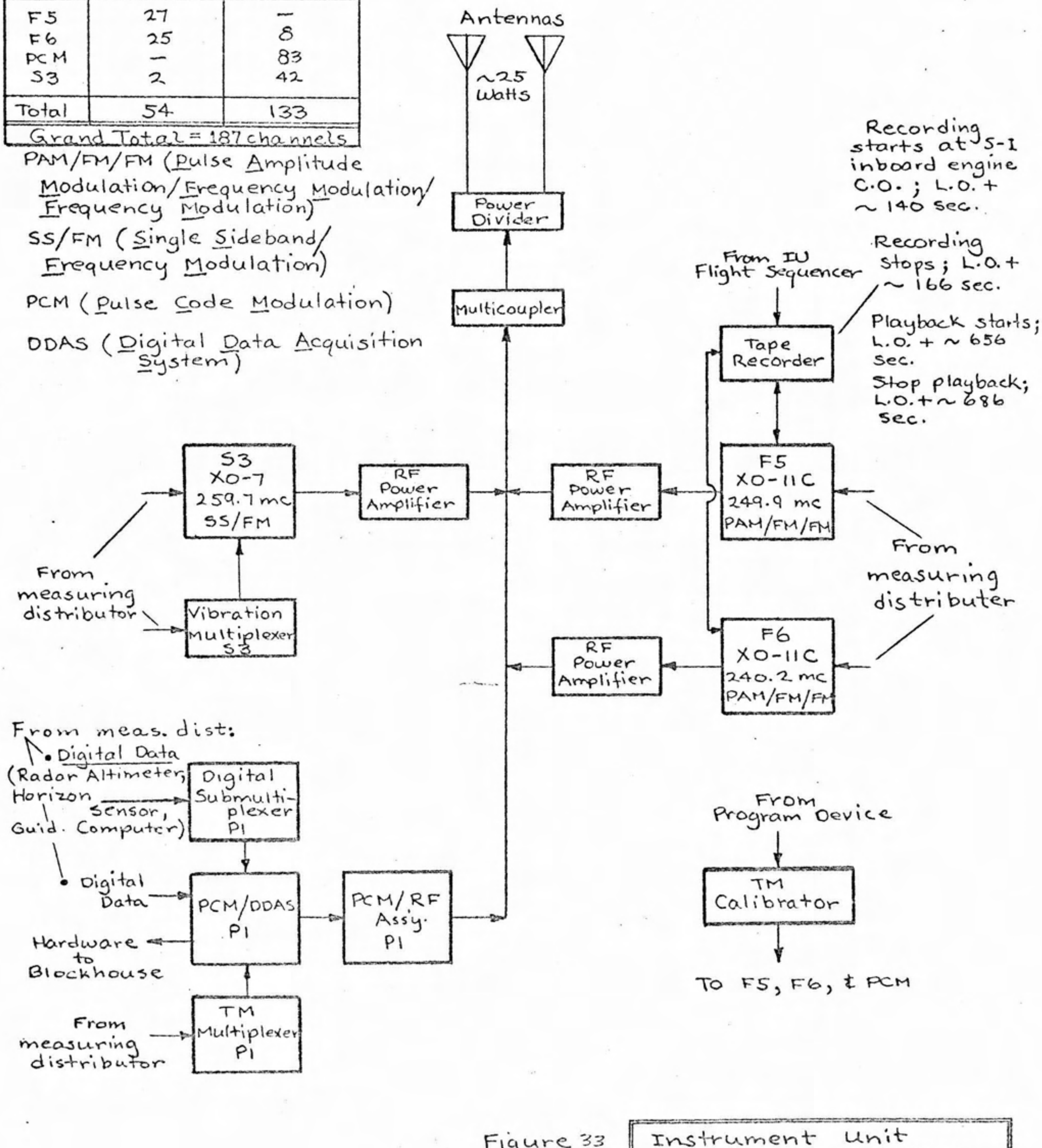
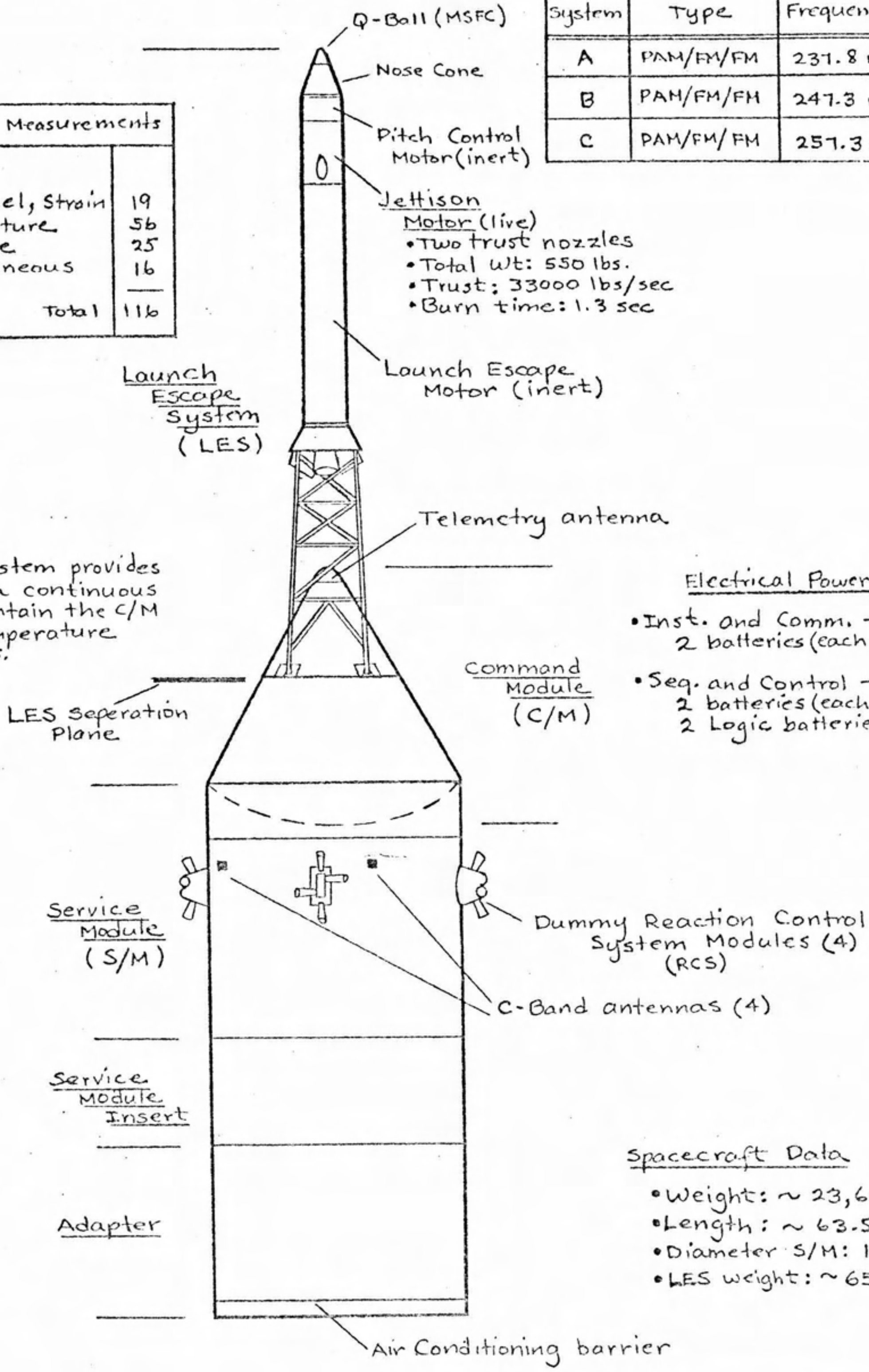


Figure 33 Instrument Unit Telemetry System

Telemetry System		
System	Type	Frequency
A	PAM/FM/FM	237.8 mc
B	PAM/FM/FM	247.3 mc
C	PAM/FM/FM	257.3 mc

Summary of Measurements	
Inflight	
1. Vib, Accel, Strain	19
2. Temperature	56
3. Pressure	25
4. Miscellaneous	16
Total	116



A cooling system provides cool air in a continuous flow to maintain the C/M ambient temperature at 80 ± 10° F.

Electrical Power System

- Inst. and Comm. -- 2 batteries (each 120 amp-hr.)
- Seq. and Control -- 2 batteries (each 5 amp-hr) 2 Logic batteries

Spacecraft Data

- Weight: ~ 23,600 lbs.
- Length: ~ 63.5 ft.
- Diameter S/M: 154 in.
- LES weight: ~ 6500 lbs.

Note:
No separation of the spacecraft.

Figure 34

APOLLO Spacecraft "Boilerplate" 13