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CENTAUR D-1A SYSTEMS SUMMARY

GENERAL DYNAMICS
Convair Aerospace Division

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Prepared by
CONVAIR AEROSPACE DIVISION OF GENERAL DYNAMICS
San Diego, California

DOCUMENT APPLICABILITY

The systems descriptions provided herein are applicable to Atlas SLV-3D and Centaur D-1A vehicles starting with vehicle number AC-30. Certain mechanical redundancy features are planned for incorporation starting with vehicles for the FLEETSATCOM Program (AC-36 and on).

X73-10297

FOREWORD

I am pleased to present this Systems Summary of the Centaur D-1A, the high energy upper stage of the National Aeronautics and Space Administration. This summary describes the various systems of the Centaur D-1A and, briefly, its first stage booster, the Atlas SLV-3D.

The Convair Aerospace Division of General Dynamics is proud of its role as prime contractor for both Centaur and Atlas. We have prepared this brochure for your use at the direction of the NASA Lewis Research Center, Technical Director for Centaur. (Contract NAS3-13514, Technical Directive Number CPO-37.)

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GLOSSARY OF ACRONYMS AND ABBREVIATIONS

AC	Atlas/Centaur (usually followed by vehicle number)
A/D	analog to digital
AFETR	Air Force Eastern Test Range
AGE	aerospace ground equipment
AKM	apogee kick motor
BECO	booster engine cutoff
CCLS	computer controlled launch set
CG	center of gravity
CCU	central controller unit
CCVAPS	computer controlled pressure and venting system
D/A	digital to analog
DAT	design acceptance test
DCU	digital computer unit
DET	design evaluation test
EED	electro-explosive device
ETR	Eastern Test Range
FAP	flight acceleration profile
FLSC	flexible linear shaped charge
GDCA	Convair Aerospace Division of General Dynamics
GHE	ground handling equipment
GHe	gaseous helium
GSE	ground support equipment
GOAS	guidance optical alignment shelter
H_2O_2	hydrogen peroxide
Hz	Hertz (cycles per second)
IMG	inertial measurement group
I/O	input/output
IRU	inertial reference unit
LC	launch complex
LeRC	Lewis Research Center
LHe	liquid helium
LH_2	liquid hydrogen
LO_2	liquid oxygen
LV	launch vehicle
MDF	mild detonating fuse
MECO	main engine cutoff
MES	main engine start
NASA	National Aeronautics and Space Administration
PCM	pulse code modulation
PL	payload

psia, g, d	pressure, pounds per square inch absolute, gage, differential
PU	propellant utilization
P&W	Pratt & Whitney
RCS	reaction control system
RF	radio frequency
RFI	radio frequency interference
RGU	rate gyro unit
RMU	remote multiplexer unit
RSC	range safety command
SAMSO	Spacecraft and Missile Systems Organization (USAF)
SC	spacecraft
SCU	sequence control unit
SECO	sustainer engine cutoff
SEU	systems electronic unit
SIU	servo inverter unit
SLV	space launch vehicle
Sta	station
TLM	telemetry
T/W	thrust-to-weight ratio
VECO	vernier engine cutoff

1

CENTAUR D-1A PROGRAM SUMMARY

1.1 INTRODUCTION TO ATLAS/CENTAUR

Booster: Atlas SLV-3D

Second Stage: Centaur D-1A

Third Stage: (optional) TE-M-364-4
Solid Motor

Launch Site: Eastern Test Range
Complexes 36A and 36B

Status: Atlas SLV-3C/Centaur D Operational Since 1966.
Atlas SLV-3D/Centaur D-1A Operational Starting in Late 1972

The Atlas/Centaur launch vehicle, shown in Figure 1-1, is a key element in the nation's space program. Atlas/Centaur has provided the boost into orbit for a variety of spacecraft on missions of scientific and lunar and planetary exploration. These versatile, reliable, and accurate space boosters will continue to contribute to many significant space programs well into the shuttle era.

Centaur D-1A is the latest version of the nation's first high energy upper stage. Major improvements in astrionics and payload structure have increased mission flexibility at lower cost. Liquid hydrogen/oxygen propellants and pressurized stainless steel structure provide a top performance vehicle. Centaur D-1A with one of the Intelsat IV series is shown in Figure 1-2.

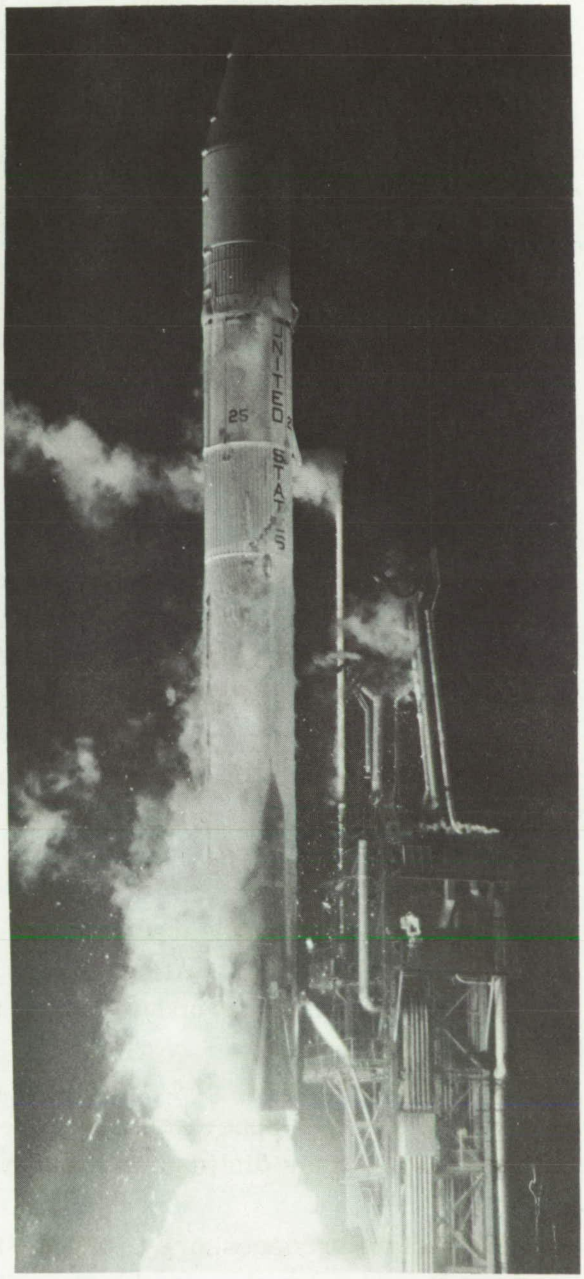


Figure 1-1. Atlas/Centaur liftoff.

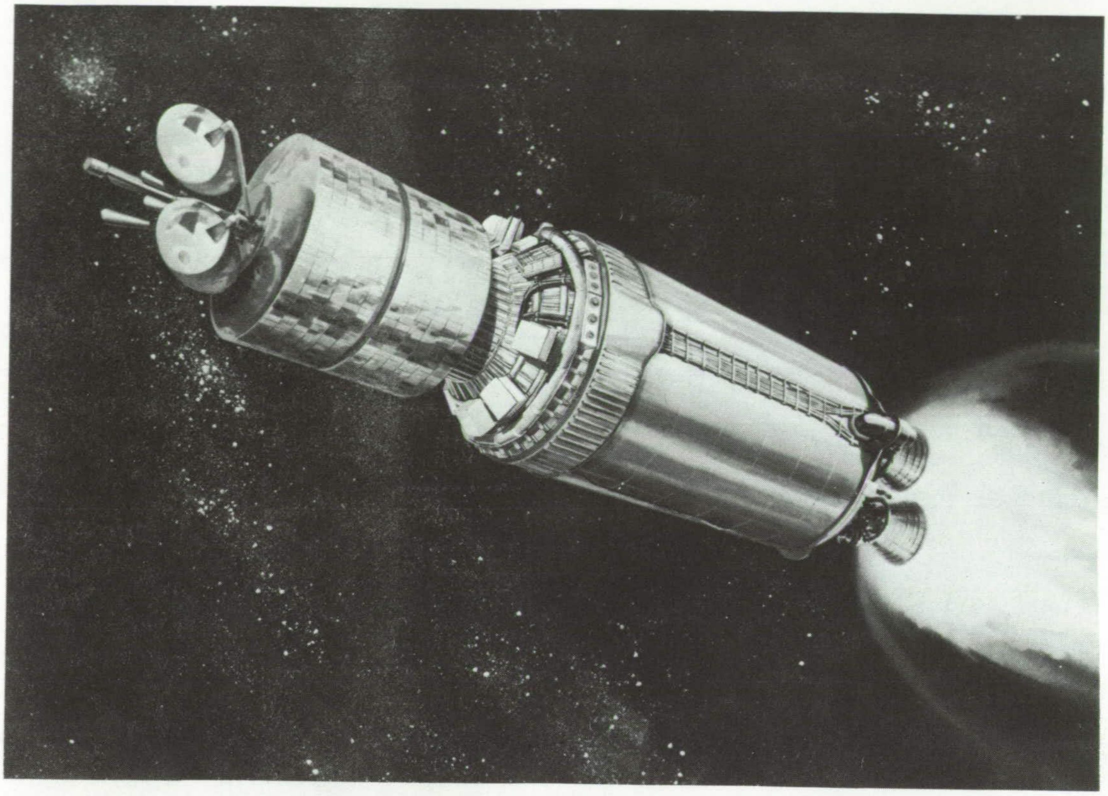


Figure 1-2. Centaur D-1A with Intelsat IV spacecraft.

Centaur D-1A is the designation for the model that flies with Atlas SLV-3D. The D-1A will be operational in late 1972 and is scheduled to fly additional Intelsat IV's, Pioneer G, Mariner-Venus-Mercury '73, and FLEETSATCOMs. Certain mechanical redundancy features will be phased in with the FLEETSATCOM vehicles.

Centaur D-1T is the upper stage for the Titan IIIE. This combination is scheduled for a proof flight launch early in 1974; it is the launch vehicle for the Viking and Helios missions.

Atlas SLV-3D has incorporated the latest improvements in propulsion, tank length, and various subsystems. Atlas lineage traces back to the Atlas weapon system. Numerous modifications have updated Atlas to today's advanced booster.

Atlas SLV-3D has a sister vehicle, the SLV-3A, which is the booster for Agena, Burner II, or OV-1 upper stages. The SLV-3A tapered nose sections and its guidance and control systems distinguish it from the SLV-3D.

Immediate predecessors to Atlas SLV-3D/Centaur D-1A were the Atlas SLV-3C/Centaur D. This booster combination flew eighteen successful operational missions and completed its last flight in August 1972. Atlas SLV-3C/Centaur D achievements include the launches of Surveyors, Applications Technology Satellite, Orbiting Astronomical Observatories, Mariner Mars '69 and '71, Intelsat IVs, and Pioneer F.

1.2 MANAGEMENT RESPONSIBILITIES

Overall management and mission integration of the Centaur Program is provided by the Lewis Research Center (LeRC) of the National Aeronautics and Space Administration.

The Atlas SLV-3D is purchased by LeRC from the USAF Space and Missile Systems Organization (SAMSO).

General Dynamics Convair Aerospace Division is the prime contractor for Centaur D-1A and its Atlas SLV-3D booster. Convair Aerospace also makes the interstage adapter, insulation panels, and nose fairing, and is launch site integrator.

Associate contractors for the Centaur D-1A are:

Pratt and Whitney Aircraft Division of United Aircraft Corporation	Engines
Teledyne Systems Company of Teledyne, Inc.,	Digital Computer Unit
Honeywell, Inc.,	Inertial Platform and Electronics

Associate contractor for the SLV-3D engines is Rocketdyne, a division of North American Rockwell Corporation.

1.3 FACILITIES

Atlas and Centaur are assembled by Convair Aerospace in San Diego, California. Final assembly of both vehicles is done at the Convair Aerospace Kearny Mesa facility. (Figures 1-3 and 1-4).

Figure 1-3. Convair Aerospace Division of General Dynamics Kearny Mesa Plant.

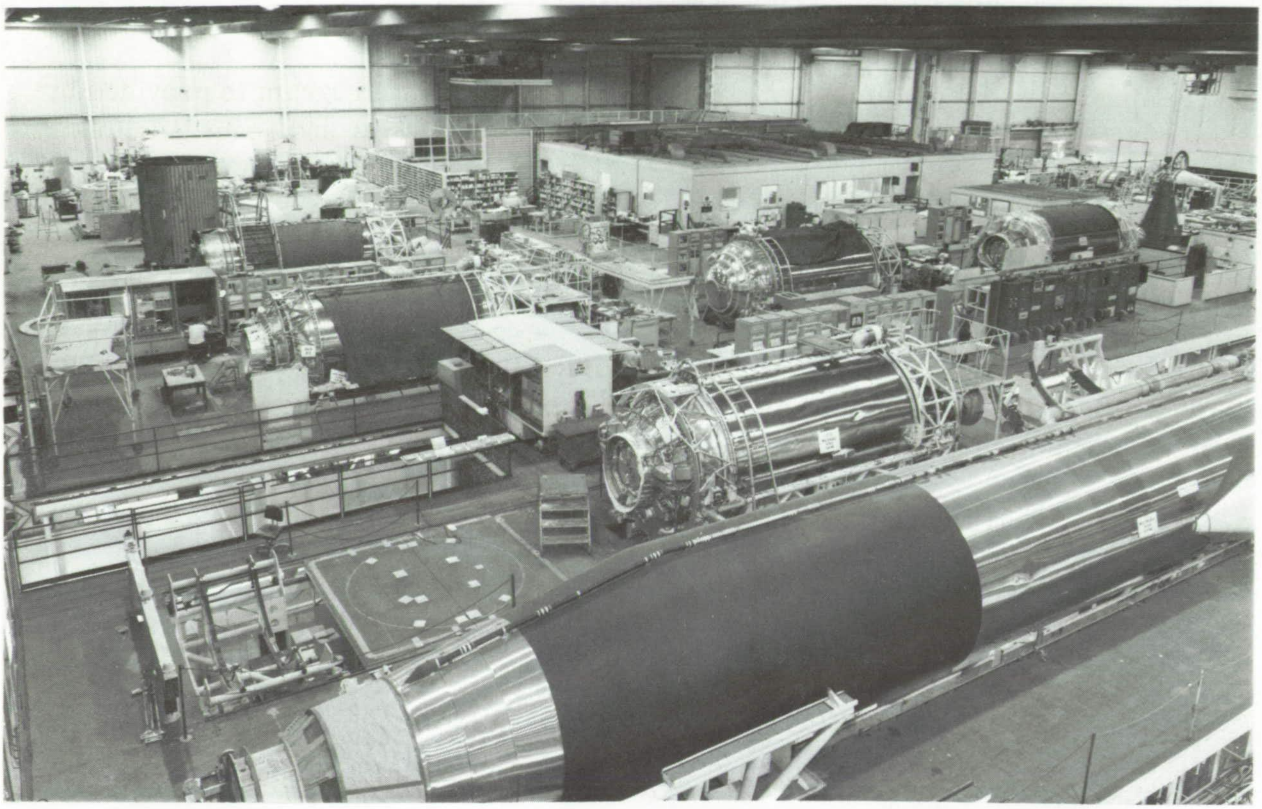


Figure 1-4. Convair Aerospace Atlas and Centaur assembly area.

Flight vehicles are transported from the Convair Aerospace Plant to Miramar Naval Air Station, and from there are flown via Guppy or C-5A aircraft to the Eastern Test Range (ETR).

Atlas/Centaur will be launched from launch complexes 36A and 36B at ETR. (Figure 1-5). These two pads share a common blockhouse, instrumentation, and launch control equipment.

Figure 1-5. Launch complexes at ETR.

2

SYSTEM AND OPERATIONAL SUMMARY

2.1 ATLAS/CENTAUR

The Atlas SLV-3D/Centaur D-1A combination is illustrated in Figures 2-1 and 2-2. The interstage adapter supports the Centaur atop Atlas and remains with Atlas at separation. The Centaur standard fairing covers the spacecraft, which is mounted to its payload adapter.

2.2 CENTAUR SYSTEM SUMMARY

Length:	30 ft (without fairing)	
Diameter:	10 ft	
Guidance:	Inertial	
Propulsion:	P&W RL10A-3-3	
Rated Thrust:	30,000 lb	
Rated I_{sp} (vac):	444 sec	
Propellants:	LO_2	24,840 lb
	LH_2	4,910 lb
Centaur Jettison:	4,400 lb	

The Centaur D-1A vehicle is shown in the final assembly and checkout docks at Convairs Kearny Mesa Plant in Figure 2-3.

Structure and Insulation. The tank structure (Figure 2-4) is made from pressure-stabilized stainless steel, 0.014 inch thick in the cylindrical section. A double-walled, vacuum-insulation intermediate bulkhead separates the liquid oxygen (LO_2) tank from the liquid hydrogen (LH_2) tank. The aft and intermediate bulkheads form a 1.38:1 ellipsoidal LO_2 tank. The forward bulkhead of the LH_2

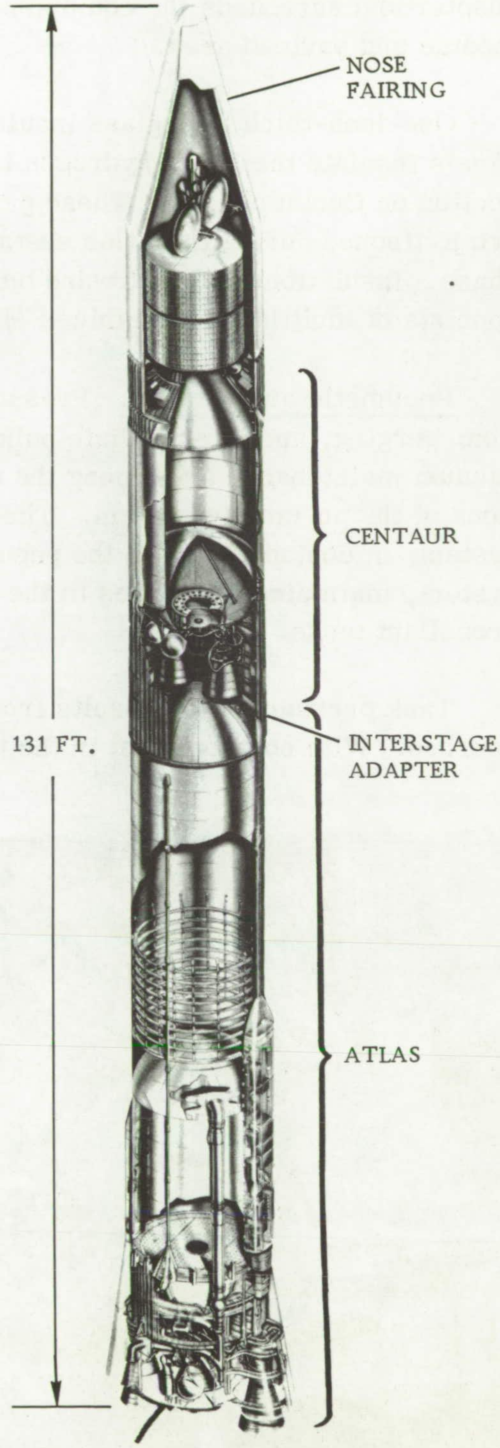


Figure 2-1. Atlas SLV-3D/Centaur D-1A.

tank combines ellipsoidal and conical sections. The forward equipment module, an aluminum conical structure, attaches to the tank by a short cylindrical stub adapter. The fairing also attaches to the stub adapter and surrounds the equipment module and payload area.

One-inch-thick fiberglass insulation panels insulate the liquid hydrogen tank section on Centaur D-1A. These panels are jettisoned during the Atlas sustainer phase. Insulation on the forward bulkhead consists of multilayer aluminized Mylar.

Pneumatic and Venting. Pressurization, purging, and intermediate bulkhead vacuum maintenance are among the functions of the pneumatic system. The vent system, in conjunction with the pneumatic system, maintains pressures in the main propellant tanks.

Tank pressurization results from two sources. With no propellant in the tank,

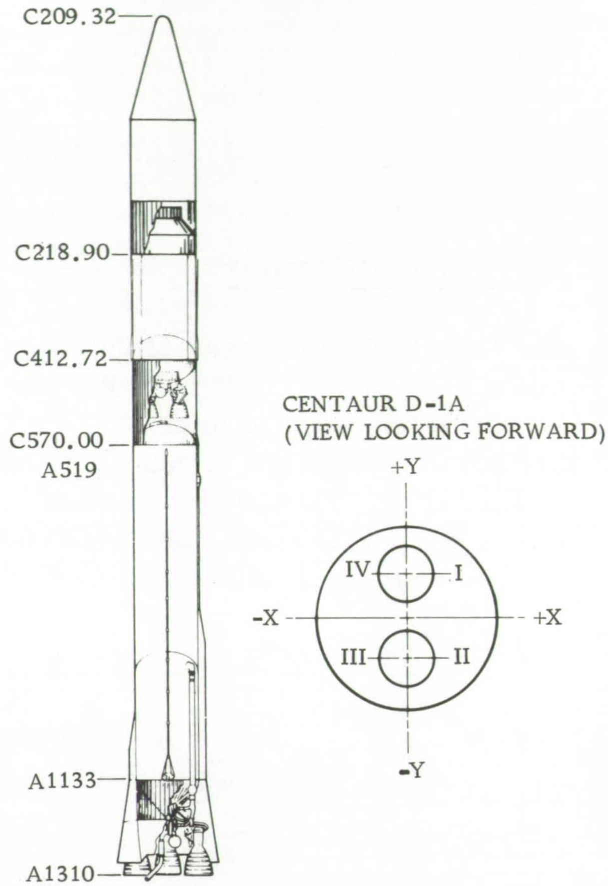


Figure 2-2. Reference stations, axes, and quadrants.

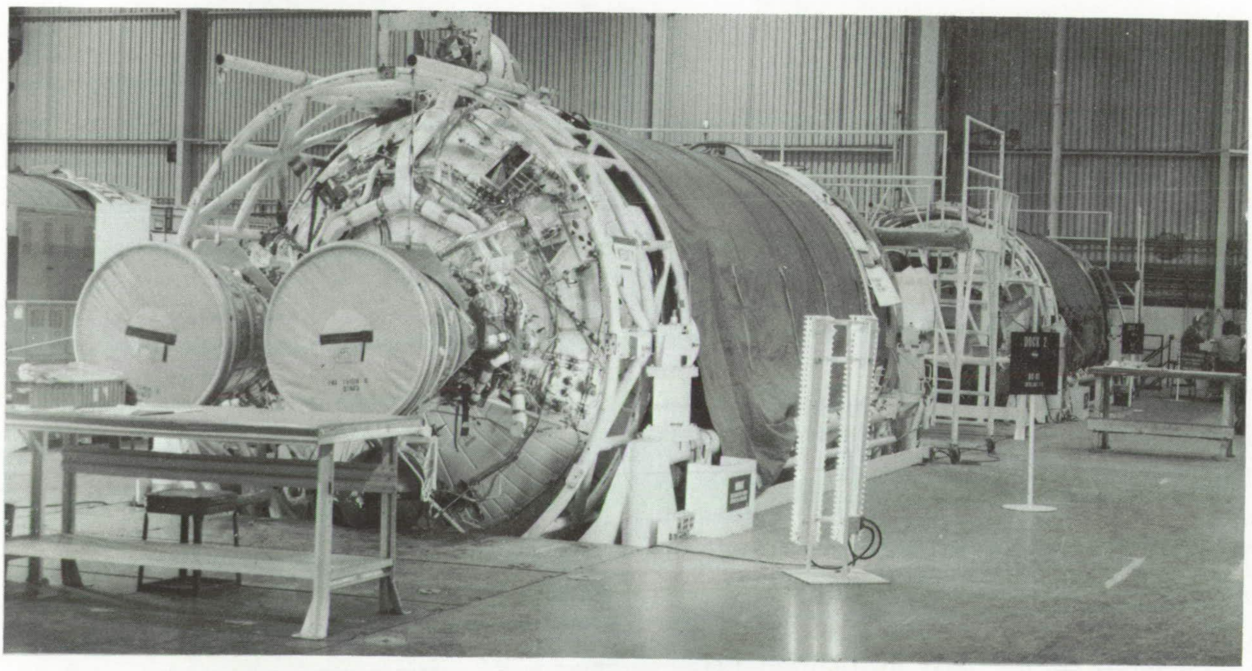


Figure 2-3. Centaur D-1 in final assembly and checkout dock.

pressure is provided by a gaseous helium system. After propellants are loaded, propellant boiloff provides the pressure. During flight the airborne helium system provides supplementary pressure when necessary. This same system also provides pressure for the hydrogen peroxide and engine controls systems.

Purging with gaseous helium prevents moisture from entering cryogenic systems and causing icing. Many vehicle systems and components are purged both before and during flight.

Propulsion. Primary thrust is provided by two Pratt and Whitney RL10A-3-3 engines that develop 30,000 pounds total thrust. These engines are regeneratively cooled and fed by turbopumps mounted on the engines. The propellants are delivered to the main engine turbopumps by boost pumps. The boost pumps are driven by turbines fueled by hydrogen peroxide.

Hydraulic. Two identical and separate hydraulic power supply systems provide the force to gimbal the Centaur main engines, one system for each engine. A power-package assembly and two actuators are the main components of each system. Each power package contains two pumps that supply pressure to the actuators. One pump, coupled to the engine turbine drive, operates while the engines are firing. During coast phase another pump, electrically-powered and thermostatically-controlled, circulates the hydraulic fluid through the system.

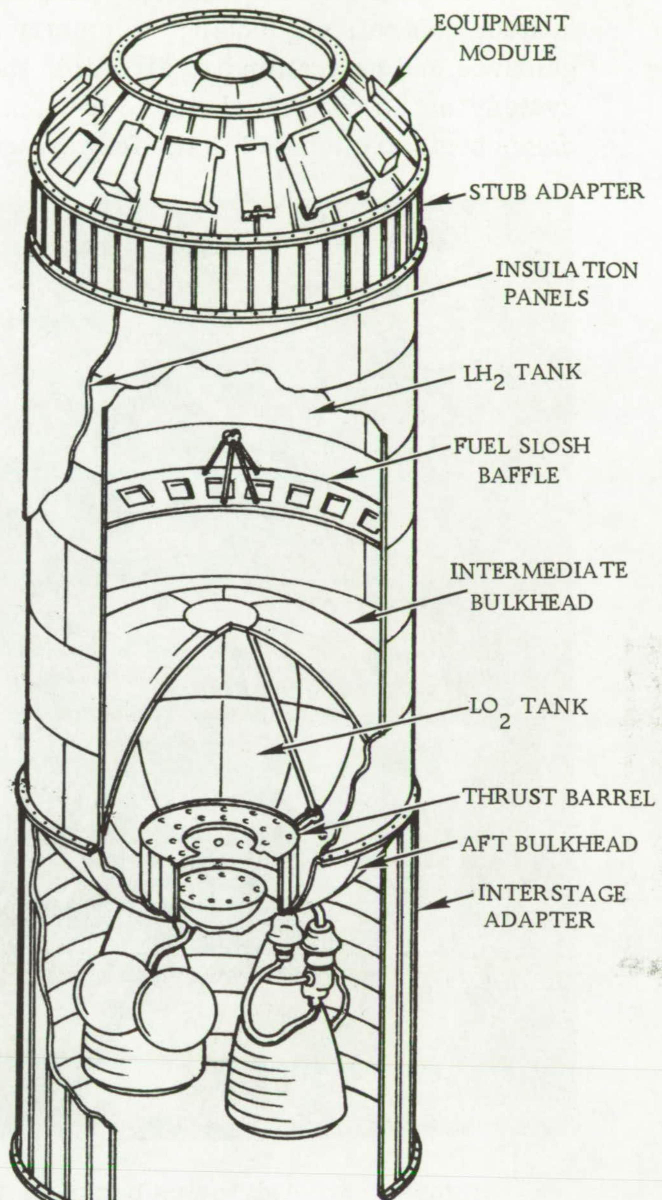


Figure 2-4. Centaur D-1A.

Reaction Control. During coast, separation, and retromaneuvers, attitude control and propellant settling are provided by a hydrogen peroxide-fueled thrust system. This system consists of small engines (3 to 50 pounds thrust) attached to the aft bulkhead. H_2O_2 is supplied to the boost pumps and the attitude control engines from the same storage sphere. A 300 psi helium system pressurizes the H_2O_2 bottle.

Astrionics. The Centaur D-1 astrionics system integrates many former hardware functions into the airborne computer software. Digital autopilot, maneuvering attitude control, sequencing, telemetry formatting, propellant management, plus guidance and navigation are all within the software scope. This results in a flexible system that is readily adaptable to mission or vehicle changes. Most of the astrionics components are located in the forward equipment module (Figure 2-5).

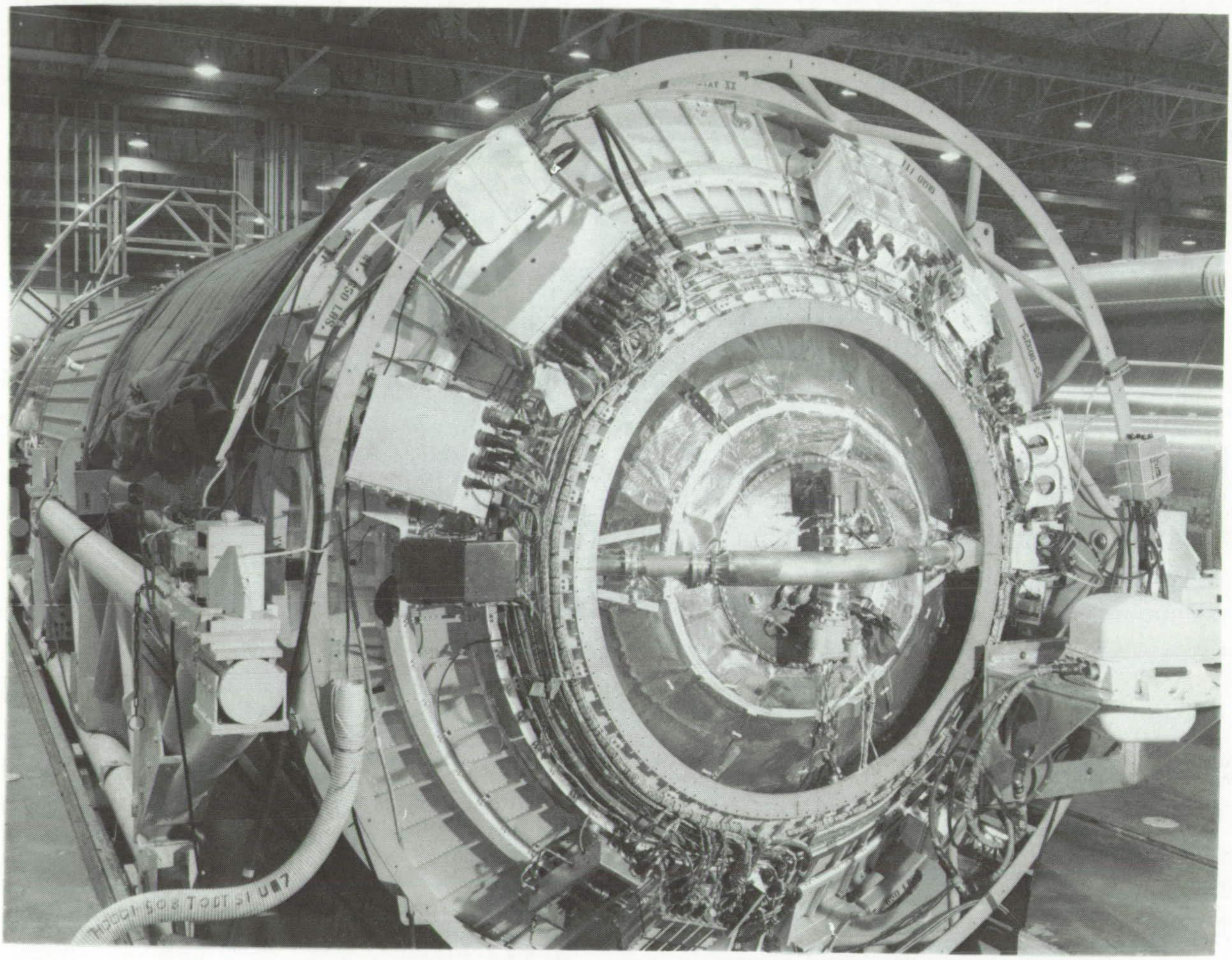


Figure 2-5. Astrionics packages mounted on Centaur equipment module.

Guidance and Control. The Teledyne digital computer unit (DCU) is an advanced, high-speed computer with extensive input and output capabilities. Its fast execution speed and 16,384-word random access memory allow its many functions to be performed with accuracy and with a comfortable margin of memory and duty cycle. The DCU provides discretes to the sequence control unit. Engine commands go to the Centaur and Atlas servo inverter units through six digital-to-analog channels.

The Honeywell inertial reference unit (IRU) contains a four-gimbal, all-attitude-stable platform. Three gyros stabilize this platform, on which are mounted three pulse-rebalanced accelerometers. A prism and window allow for optical azimuth alignment. Resolvers on the platform gimbals transform inertial vectors into vehicle coordinates. These vectors originate in the DCU a-c digital to analog (D/A) converters. They are frequency multiplexed and output to the IRU, which provides the a-c reference voltages for the a-c D/As. The IRU also contains a crystal oscillator, the primary timing reference.

The system electronic unit provides conditioned power and sequencing for the IRU. Communication from the IRU to the DCU is through three analog-to-digital channels (for attitude signals) and three incremental velocity channels. Attitude rate information is digitally derived from attitude signals within the DCU.

The Centaur D-1A system also provides stabilization and guidance for the Atlas booster.

Flight Software. The flight software is modularized into several special-purpose subroutines that operate under the control of a real-time executive program. The executive calls subroutines to perform the various tasks, with the software system recognizing hardware interrupts that demand servicing. The system allows interruptable subroutines to be coded separately with significant advantages in speed and cost of development, modification, and validation.

The flexibility of the flight software allows a variety of mission ascent modes or sequences to be considered for performance or operational improvement. The system design and module library allows a high degree of specific mission tailoring with a minimum amount of time and cost in programming or validation.

Telemetry. The central controller for the Centaur pulse code modulation (PCM) telemetry system is housed in the same package as the DCU. This arrangement simplifies communications and provides software-selectable stored PCM formats. Data from the information sensors is converted to digital words for transmission to the ground station via an S-band transmitter. System capability is 267,000 bits per second, of which about 140,000 are currently used. The central controller can service four remote multiplexer units, three of which are currently used for Centaur D-1A flights.

Propellant Utilization. A propellant utilization system controls the LO_2 flow rate to ensure that both tanks will be emptied simultaneously. Probes are mounted within the fuel and oxidizer tanks.

Tracking and Range Safety Command. The tracking system provides data to determine position and velocity information for use by range safety at the Eastern Test Range. When necessary, the range safety officer terminates the flight via the range safety command system. The airborne part of the system on Centaur D-1A is nearly completely redundant and consists of antennas, destructor and related equipment. The C-band tracking system provides ground tracking of Centaur during flight.

Locations of tracking stations include Cape Kennedy, Grand Bahama, Grand Turk, Antigua, and Bermuda. The airborne transponder returns an amplified radio-frequency signal when it detects a tracking radar's interrogation.

Power. Centaur uses a basic d-c power system, provided by batteries and distributed via harnessing. The battery arrangement is flexible, with up to three main batteries planned for some missions. If a system requires other than the 28 volts d-c, it internally converts the d-c to whatever its specific power needs are. AC power, 26 and 115 volts single phase, 400 Hz, is supplied by the servo inverter unit.

2.3 ATLAS SLV-3D SYSTEM SUMMARY

Length:	72 ft
Diameter:	10 ft
Liftoff Weight (lb)	—
Guidance:	By Centaur
Propulsion:	Rocketdyne MA-5
Rated Thrust Sea Level (lb)	
Booster	370,000
Sustainer	60,000
Verniers (axial)	1,040
Rated I_{sp}, Sea Level/Vacuum (sec)	
Booster	257/292
Sustainer	221/312
Propellants (lb)	
Oxidizer, LO_2	184,470
Fuel, RP-1	83,570
Jettison Weight (lb)	
Booster	7,510
Sustainer (including interstage adapter)	8,230

The Atlas concept is shown in Figure 2-6.

The liftoff thrust (431,040 pounds) is indicative of the many improvements that have made the SLV-3D the advanced booster it is today. Atlas was originally developed as a U.S. Air Force weapon system with a total thrust of 359,000 pounds. Early in its development period, Atlas became a versatile and highly reliable space booster. It has since undergone a series of improvements, including tank lengthening, engine performance increases, and system updating.

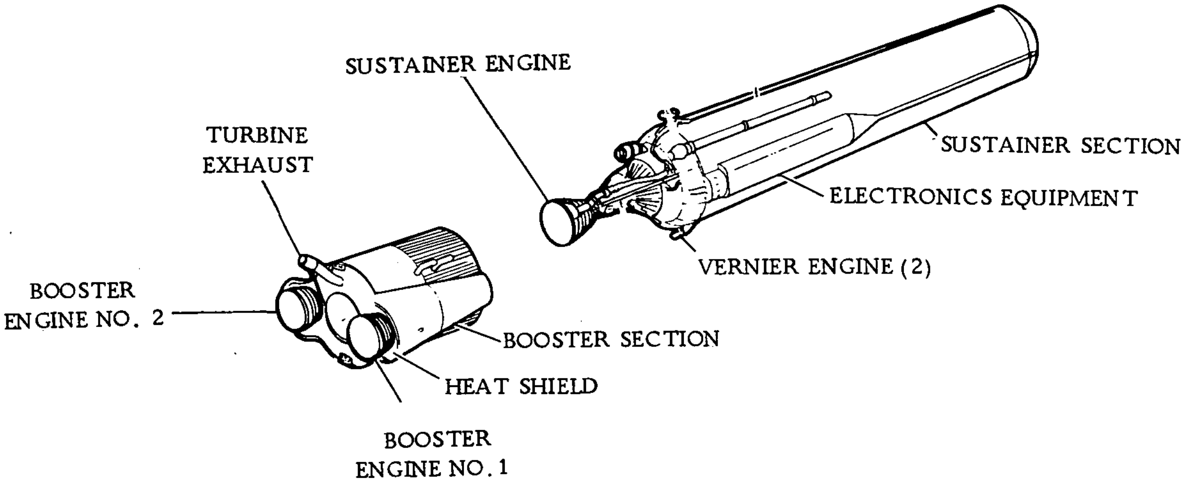


Figure 2-6. Atlas concept.

Atlas has a constant ten-foot-diameter tank up to the attach point of the interstage adapter. Total length from the adapter is 72 feet. Equipment is mounted within a pod on the side. A helium pressure system maintains structural integrity and turbopump pressure head during flight.

All Atlas engines of the Rocketdyne MA-5 propulsion system are ignited prior to liftoff. The two booster engines and the single sustainer engine share the same propellant tank. The booster engines are jettisoned about two and a half minutes into flight at about 5.7g acceleration. The sustainer engine burns until propellant depletion. Two small vernier rockets assist in the early roll maneuver to the desired azimuth and provide roll control during sustainer phase. Propellants for all engines are LO_2 and RP-1.

Attitude control is maintained by gimbaling the vernier and main engines, under the direction of the autopilot and guidance equations in the Centaur digital computer unit. The sustainer engine gimbals during the sustainer phase only. Open loop pitch and yaw programs are selectable based on the launch-day winds.

2.4 ATLAS/CENTAUR FLIGHT SEQUENCE

Trajectory data and sequences of events differ widely for the variety of missions flown by Atlas/Centaur D-1A. Spacecraft weights, direct ascent or parking orbit, mission or tracking requirements, and constraints all lead to a trajectory and sequence tailored to the specific mission needs.

The flight compendium illustrated in Figure 2-7 and discussed in the following paragraphs is for an Atlas/Centaur mission to place a spacecraft into a synchronous apogee transfer orbit. Actual times and durations are mission-peculiar.

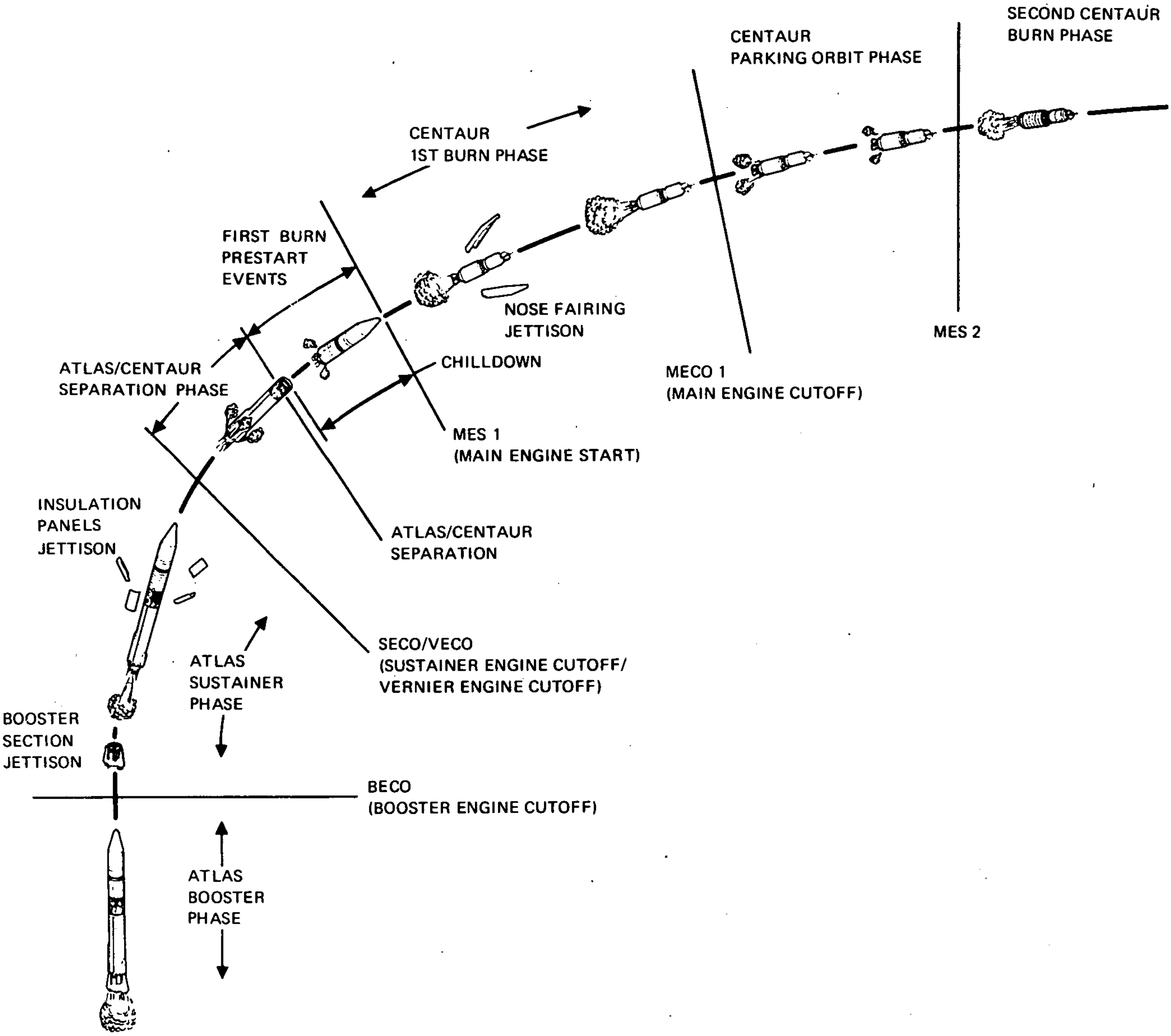
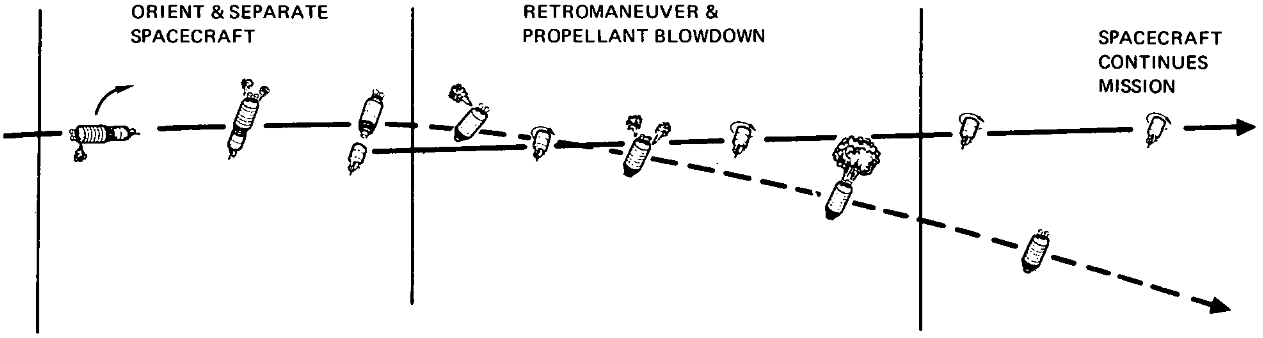


Figure 2-7. Atlas/Centaur parking orbit mission delivering a spacecraft to synchronous apogee transfer.



Preliminary Atlas/Centaur D-1A Sequence of Events

EVENT	BASIS	APPROX. TIME FROM LIFTOFF (SEC.)
LIFTOFF	2-IN. MOTION	0
ROLL PROGRAM	LIFTOFF + 2 SEC.	2-15
BECO	5.7g	153
BOOSTER PACKAGE JETTISON	BECO + 3.1 SEC.	156
JETTISON INSULATION PANELS	BECO + 45 SEC.	198
SECO	PROP. DEPLETION	251
SEPARATION	SECO + 1.9 SEC.	253
MES 1	SECO + 11.5 SEC.	263
JETTISON NOSE FAIRING	MES 1 + 12 SEC.	275
MECO 1	PARKING ORBIT (GUID.)	586
MES 2	GUIDANCE	2,086
MECO 2	HOHMANN APOGEE	2,200
SEPARATION	MECO 2 +Δt (VARIES)	

Figure 2-7. Atlas/Centaur parking orbit mission delivering a spacecraft to synchronous apogee transfer. (Contd)

"Liftoff" officially begins at 2-inch motion (Atlas/Centaur two inches off the launch pad). All Atlas engines are operating at liftoff. At two seconds after liftoff, Atlas starts a preprogrammed roll program which continues for thirteen seconds. The booster engines gimbal to perform the roll maneuver.

The Atlas pitch and yaw programs are initiated at liftoff +15 seconds and continue to Atlas booster engine cutoff (BECO). Booster engines perform these maneuvers as the sustainer engine is locked to null. The specific pitch and yaw programs flown are selected on launch day to minimize wind-induced loading on the vehicle.

BECO occurs at an accelerometer reading of 5.7g. The booster package is jettisoned 3.1 seconds after BECO.

During sustainer phase, pitch and yaw is controlled by gimbaling the sustainer engine. Roll is controlled by the vernier engines. Closed-loop guidance steering starts 8 seconds after BECO and continues to sustainer engine cutoff (SECO).

The Centaur insulation panels are jettisoned 45 seconds into sustainer phase. SECO and vernier engine cutoff (VECO) occur when propellant depletion is sensed by either an LO_2 pressure switch or a sensor in the fuel tank. About two seconds after SECO, the pyrotechnic system releases the Centaur from the interstage adapter. Separation is achieved by firing the retrorockets mounted on the aft end of the Atlas sustainer tank.

The Centaur control system begins 3.5 seconds after SECO. At this time Centaur engine prestart valves are opened, providing control capability and initiating the propulsion system chilldown. In this process the fuel and oxidizer flow through the propellant feed system and engine pumps are vented overboard through discharge valves (LH_2) and the thrust chamber (LO_2).

Centaur main engine start (MES 1) occurs about 11.5 seconds from SECO. The flight control system holds a constant vehicle attitude through nose fairing jettison, 12 seconds after MES 1. Closed loop guidance starts at MES +20 seconds and continues until Centaur cutoff. Attitude maneuvers are accomplished by gimbaling the two Centaur main engines.

Centaur main engine cutoff (MECO 1) is directed by guidance when the proper parking orbit is achieved. A propellant settling phase follows during which the 50-pound thrust engines are firing.

During Centaur coast phase, vehicle attitude and propellant retention is controlled by the smaller reaction control motors. Shortly before MES 2, the reaction motor system orients the vehicle to the proper attitude for ignition. The 50-pound thrust motors then again fire to settle the propellants. A second chilldown process occurs for 17 seconds before MES 2.

MES 2 is initiated by guidance shortly before the first equatorial crossing for the synchronous mission. The second burn continues until guidance calculates that the vehicle is properly on the transfer ellipse.

After MECO 2 the 3-pound thrust motors burn for 10 seconds to settle the propellants. The cluster motors (6 and 3.5-pound thrust) then orient the vehicle to the attitude required by the spacecraft. Spacecraft separation is directed by Centaur guidance.

After separation, Centaur is turned away from the spacecraft. (For missions with a third stage, Centaur is merely backed away.) Then the 50-pound motors fire for eight seconds to move Centaur further away from the spacecraft. The 3-pound motors burn for another hundred seconds until propellant blowdown. In this process the remaining propellants flow through the main engines until all blowdown pressure is expended. This may require several minutes to accomplish, during which time the Centaur is on an orbit considerably different from the spacecraft.

3

CENTAUR D-1A MECHANICAL SYSTEMS

The section presents descriptions of the following Centaur D-1A mechanical systems:

3.1 STRUCTURAL SYSTEM

ELEMENTS:	Subsection
• Tank Structure and Stub Adapter	3.1.1
• Equipment Module	3.1.2
• Interstage Adapter	3.1.3
• Payload Adapter	3.1.4
• Bolt-on Structures	3.1.5

FUNCTIONS:

- Contain Main Propellants
- Establish Primary Integrity
- Provide Support for All Airborne Systems and Components
- Provide Atlas/Centaur Interface
- Support Payload

3.1.1 TANK STRUCTURE. The tank structure (Figure 3-1) contains the main propellants (liquid hydrogen and liquid oxygen). It establishes primary structural integrity for the Centaur vehicle, and provides support for all second stage airborne systems and components. The tank includes a stub adapter to support the fairing system and the equipment module. The propellant tanks are of monocoque construction formed by a series of short stainless steel cylinders welded together (Figure 3-2). Ends of the tank are formed by stainless steel bulkheads. The fuel and oxidizer tanks are separated by an intermediate bulkhead.

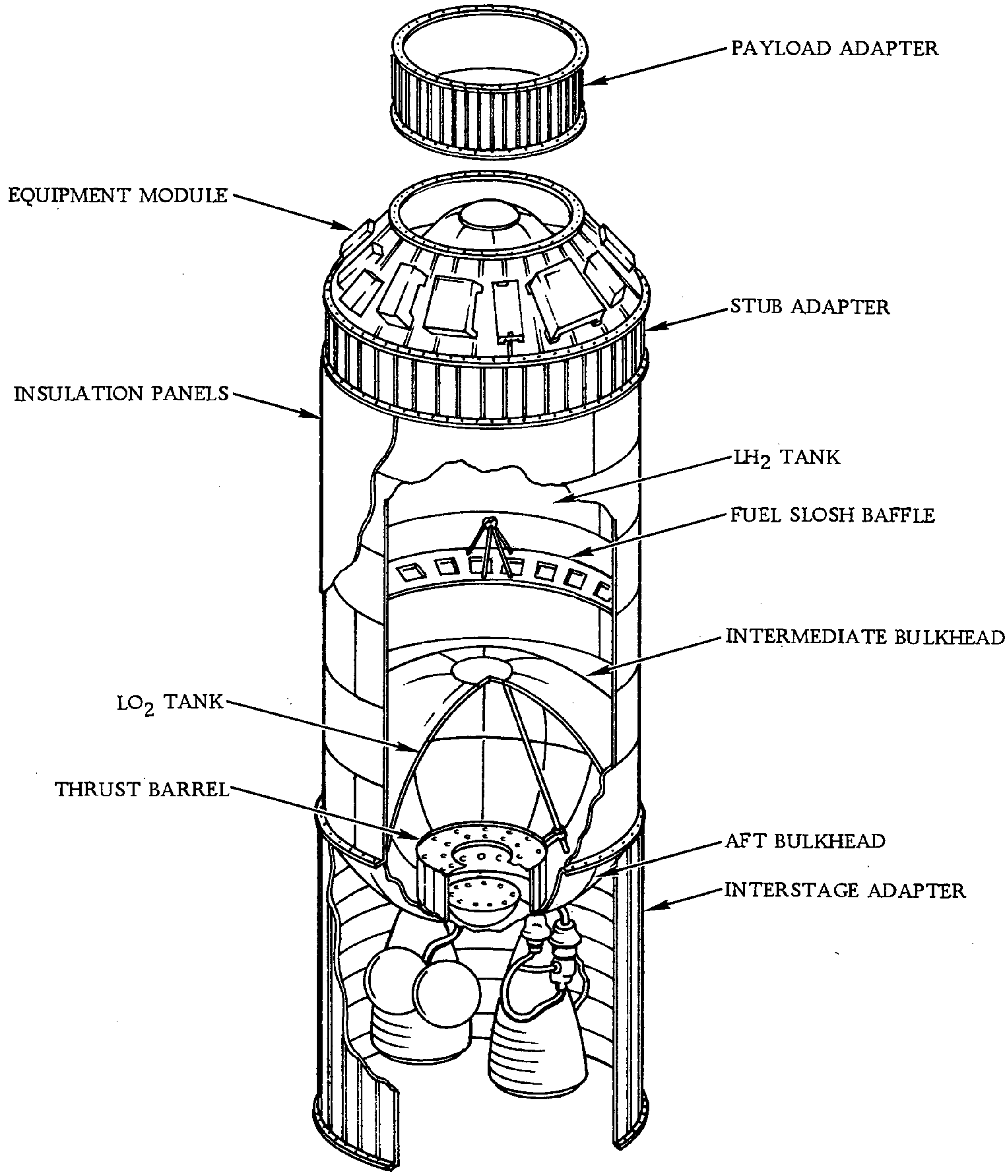


Figure 3-1. Centaur structural components.

Total Tank

Dimensions	120 in. dia. × 291 in. long
Oxidizer	Liquid Oxygen 376 ft^3 max.
Fuel	Liquid Hydrogen 1265 ft^3 max.

Cylindrical Tank Skins (6)

Material	301 Cres Stainless Steel (extra hard)
Dimensions (each)	0.014 in. thick × 32 to 34.5 in. long × 120 in. dia.

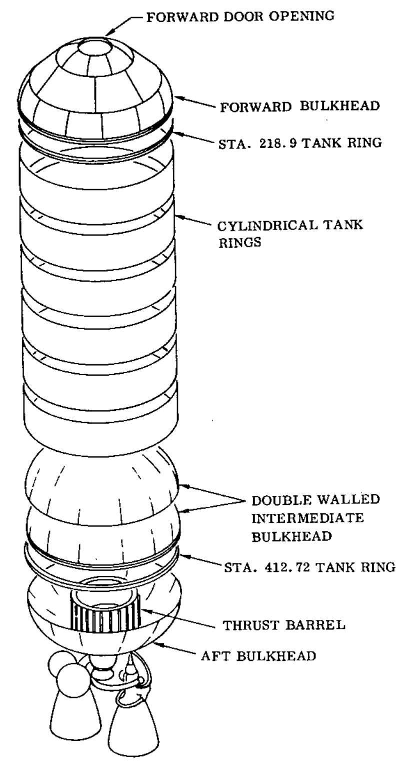


Figure 3-2. Tank fabrication concept.

Brackets welded to the tank walls support the various tank-mounted systems; flanged tank weldments facilitate attachment of propellant ducts. Internal brackets welded to the tank support the slosh baffles. The stub adapter and the interstage adapter attach to machined Cres rings at either end of the tank cylindrical portion.

An oxidizer sump is mounted on a machined ring on the aft bulkhead. The aft end of the sump contains a flange for mounting the oxidizer boost pump. A 3-inch-diameter boost pump discharge duct provides a connection for the main oxidizer supply duct. A bellows in the discharge duct allows differential contraction of the oxidizer tank and the propellant ducts.

A fuel sump is mounted to a 13-inch-diameter ring in the aft portion of the fuel tank. The aft end of the sump contains a flange for mounting the fuel boost pump.

The tank skin is stabilized at all times by internal pressure or by the application of mechanical stretch. During assembly and certain checkout operations support is provided by applying stretch to ground handling adapters attached to the forward and aft cylindrical tank rings. After erection, structural integrity is maintained by standby pressure of 5 psig in the fuel tank and 10.5 psig in the oxidizer tank. Emergency stretch is available at all times until the nose fairing is installed.

Intermediate Bulkhead. The intermediate bulkhead (Figure 3-3) forms the interface between the fuel and oxidizer tanks. This is a doublewalled, ellipsoidal structure. It consists of a structural bulkhead and an insulation bulkhead, separated by plastic mesh and fiberglass matt insulation. A vent tube leads to atmosphere from between the bulkheads. Both bulkheads are made from stainless steel gore sections welded together. In nonwelded areas the bulkheads are chem-milled down to 0.013 inch.

The volume between the bulkheads is evacuated, leak-checked, and back-filled with gaseous nitrogen prior to tanking. When the cryogenic propellants are loaded, the trapped nitrogen condenses and a vacuum is formed by the cryopumping effect. A check valve in the vent tube ensures that air does not enter the volume between the bulkheads.

The structural bulkhead is welded to the fuel tank and to a doubler outside the tank. The insulation bulkhead is welded in a similar manner on the oxidizer tank just forward of the Station 412.72 ring. A spring-type expansion ring is formed by the lower edge of the insulation bulkhead.

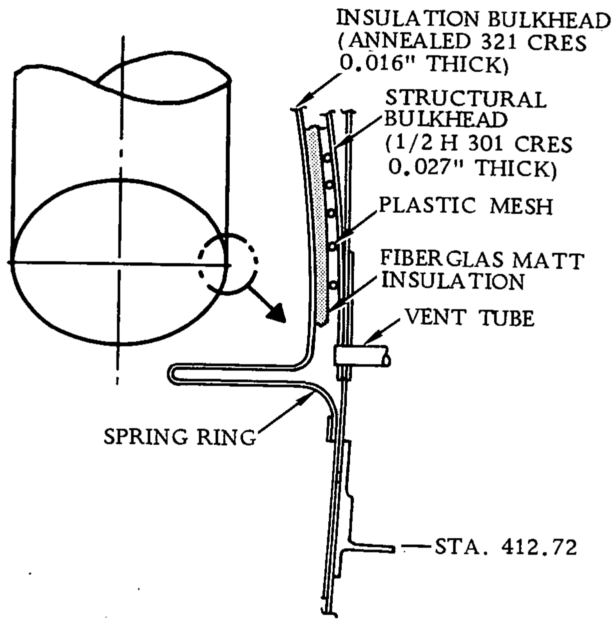


Figure 3-3. Intermediate bulkhead sectional view.

Forward Bulkhead. The forward bulkhead is made up of three sections welded together. The sections are an ellipsoidal transition aft section, a conical center section, and an ellipsoidal forward center section (Figure 3-4). Welded to the forward end is an annealed 301 Cres flanged ring.

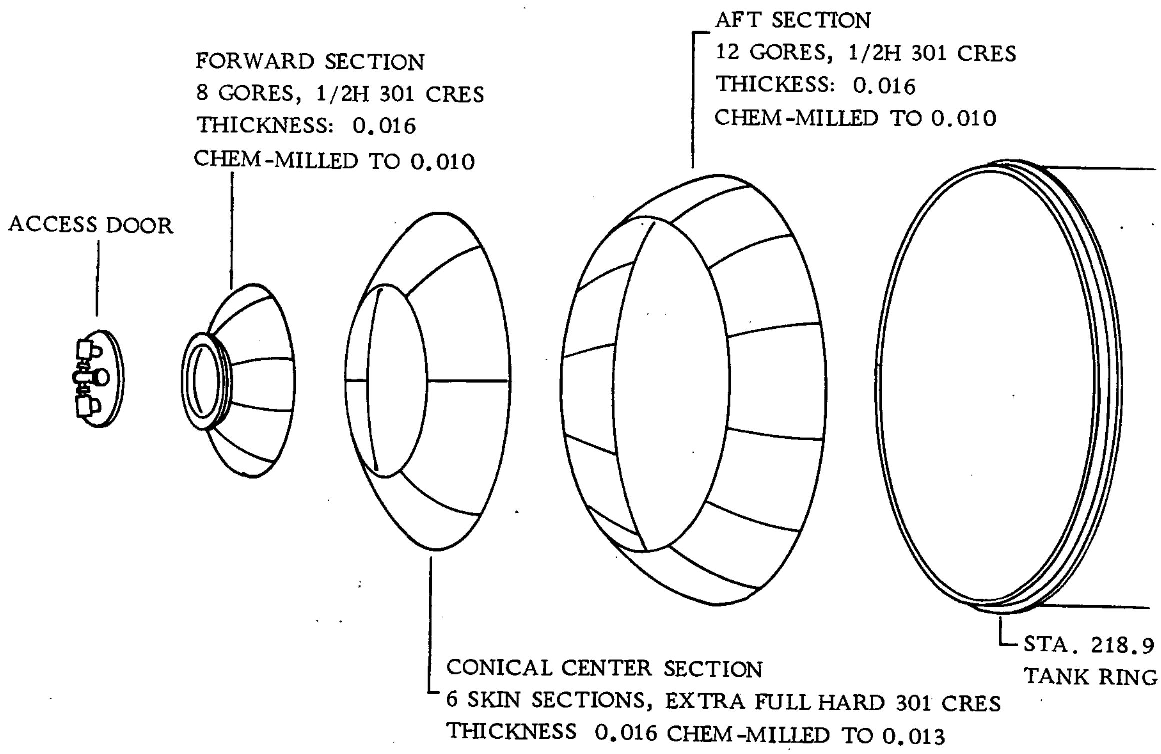


Figure 3-4. Forward bulkhead.

The forward door mounts to this flanged ring. The door is a shallow spherical section made of 0.032 thick 1/2 H 301 Cres with an annealed 321 Cres ring. The door incorporates two flanged outlets for attachment of the primary and secondary LH_2 vent valves, mounting brackets for the LH_2 vent ducts, a helium pressure inlet diffuser and bosses for various LH_2 tank PU and instrumentation connections.

Aft Bulkhead and Thrust Barrel. The aft bulkhead contains mounting provisions for the engine thrust barrel and for the reaction-control engine system. The ellipsoidal bulkhead is made from 0.018-inch-thick gore sections. The 3/4 H Cres gores are joined by welded splice doublers that are contoured to provide reinforcement for mounting brackets. Central caps, doublers, and flanges complete the aft bulkhead.

The thrust barrel is a 50-inch-diameter cylinder of aluminum sheet and stringer construction. It is riveted to a 301 Cres tee-flanged ring welded inside the aft bulkhead. Engine mounting blocks are bolted to the exterior of the bulkhead; all engine loads and thrust are reacted to by this arrangement. The thrust barrel is symmetrically located about the longitudinal axis, and contains numerous holes to provide drain passages for the oxidizer and to minimize vortexing.

Flanged outlets on the aft bulkhead provide mounting surfaces for the oxidizer boost pump sump, an aluminum oxidizer vent valve standpipe, and the oxidizer fill-and-drain valve. Bosses for various wiring and tubing for tank pressurization, liquid oxygen recirculation, and ullage pressure sensing are included on the standpipe elbow. Brackets welded to the aft bulkhead accommodate attachment of the helium bottle, engine actuators, separation bumper guides, the hydrogen peroxide system, instrumentation boxes, a pneumatic panel, radiation shield, LO_2 boiloff vent system, miscellaneous wiring and tubing, the umbilical disconnect panel, and the remote multiplexer and signal conditioner.

Blast Shield. A stainless steel blast shield protects the Centaur liquid oxygen tank from fragments that may result at Centaur separation. This shield attaches around the periphery of the tank directly in line with the shaped charge located on the interstage adapter. A fiberglass pad is located between the blast shield and the liquid oxygen tank skin.

Stub Adapter. The stub adapter (Figures 3-5 and 3-6) is installed on the forward tank ring. It supports the payload, payload adapter, and equipment module structures and the nose fairing and split barrel structures. The stub adapter is a cylindrical structure approximately 120 inches in diameter and 25 inches long. It is made from aluminum skin and stringers with machined aluminum forward and aft rings and a formed aluminum midframe. Shear pins and eight separation fittings provide for attachment of the nose fairing/split barrel structure. Nutplates provide for attachment of the equipment module.

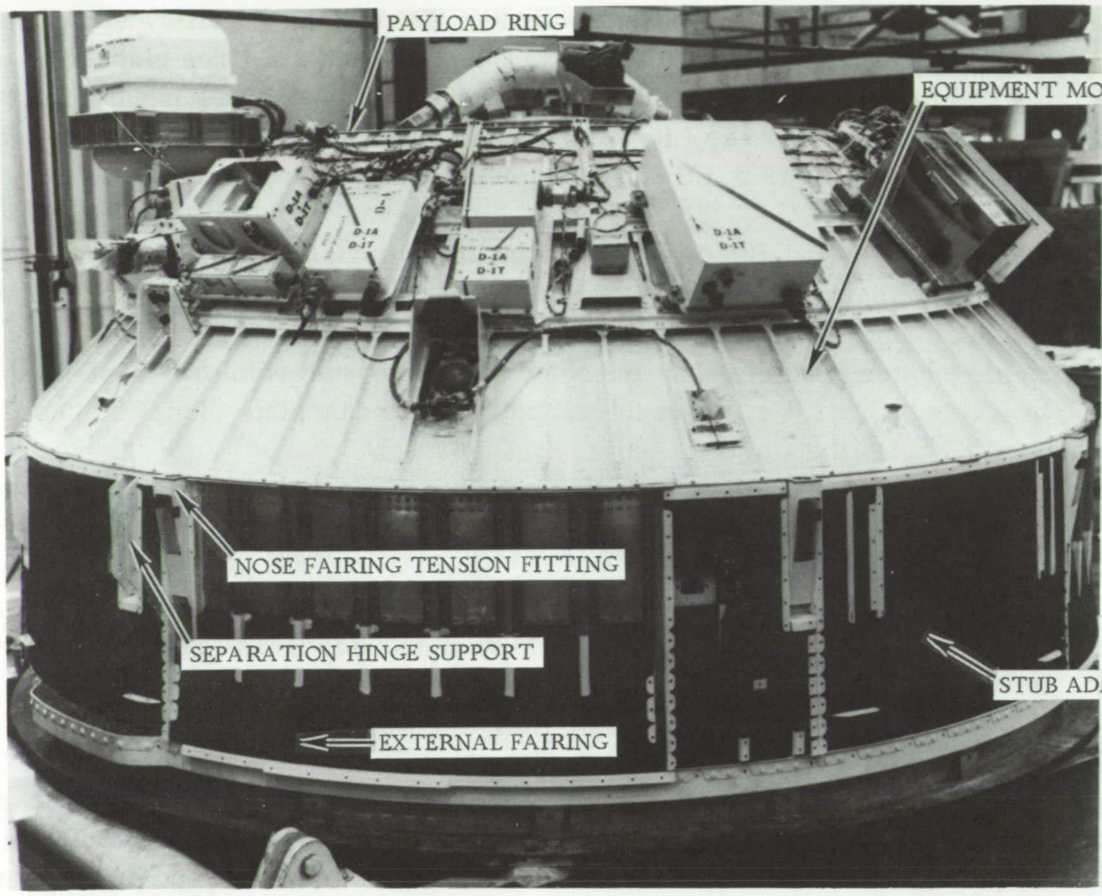


Figure 3-5. Centaur D-1A stub adapter and equipment module.

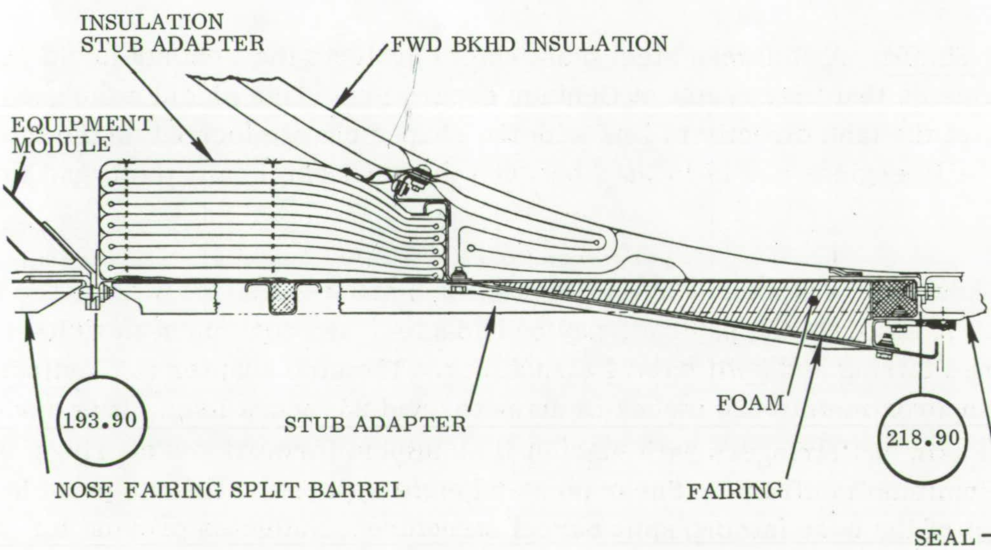


Figure 3-6. Stub adapter cross-section.

A door allows for venting the insulation panel helium purge gas. (See Subsection 3.2.2 for a description of this system.)

The stub adapter incorporates the following:

a. Thermal insulation (multilayer) on the inboard surface of the forward half. This insulation prevents the forward end of the stub adapter from becoming cold enough to produce water condensation.

b. Thermolag insulation of external surfaces not covered by aerodynamic fairings. This insulation is required to prevent excessively high temperature of the aluminum skin due to aerodynamic heating.

c. Dual detonator fairings.

d. A severance system for the Centaur insulation panels.

e. Two (2) S-Band antennas and associated ground planes.

f. Aerodynamic fairings to protect the Centaur insulation panel seal, vent system, severance system, and externally routed harnesses.

3.1.2 EQUIPMENT MODULE. The equipment module is a conical aluminum skin-stringer structure mounted on the forward end of the stub adapter. (Figure 3-5).

Its functions are to:

a. Provide mounting for various electrical/electronic components.

b. Provide mounting for the electrical harnesses servicing the mounted components.

c. Provide mounting for the forward umbilical panel.

d. Act as a supporting structure for a payload adapter carrying payloads of 4000 pounds or less.

e. Provide mounting for electrical connectors to the nose fairing.

f. Provide mounting for LH_2 balanced thrust venting system line and nozzles.

The module (Figure 3-7) is constructed in the shape of a 90-degree included angle truncated cone about 30 inches high with a base diameter of about 10 feet. It consists of a skin riveted to a machined ring at either end, longitudinal stiffeners riveted to the forward (outer) side of the skin and two circular frames riveted to the aft (inner) side of the skin. Two circumferential, fiberglass hatsection equipment mounting rails are attached to the stiffeners (Figure 3-8).

The outboard surface and one leg of the upper rail are covered with a conductive material (0.002-inch Cres foil) to provide an RF grounding path to structure. The module incorporates mounting provisions at its forward end for the wiring harnesses serving the mounted equipment. The forward end of the module incorporates provisions for attachment of both a thermal diaphragm and a payload adapter (Figure 3-9).

Two penetrations and support provisions (180 degrees apart) for the GH_2 vent duct are provided in the module skin.

The equipment module contains the helium purge gas between it and the tank and prevents flow of gas into the equipment area (forward face of module). The circular area at the forward end of the module is sealed by a thermal diaphragm. A vent tube, mounted on the equipment module, vents purge gas from beneath the module during prelaunch operations.

A vent door is installed in the module skin. When open, it allows equalization of pressures between the volumes above and below the module. This door is held positively closed during the prelaunch operations. It is unlatched by a pyrotechnic device in response to a signal from the launch ladder sequence. The door is spring-loaded so that it will immediately open fully upon unlatching and will remain fully open during launch and flight.

A mounting structure is provided on the equipment module for the inertial reference unit. Transmitters/transponders requiring heat dissipation are mounted on aluminum heat-sink/radiators. All other equipment and heat-sink/radiators are mounted on the fiberglass rails.

3.1.3 INTERSTAGE ADAPTER. The interstage adapter (Figure 3-10) provides a physical connection between the Atlas and the Centaur vehicles. The adapter is an aluminum skin-stringer cylindrical section 10 feet in diameter and 13 feet long. The adapter is designed for bolt attachment to both the Centaur aft ring at Station 412.72 and to the Atlas forward ring at Atlas Station 519.00 (Centaur Station 570.00).

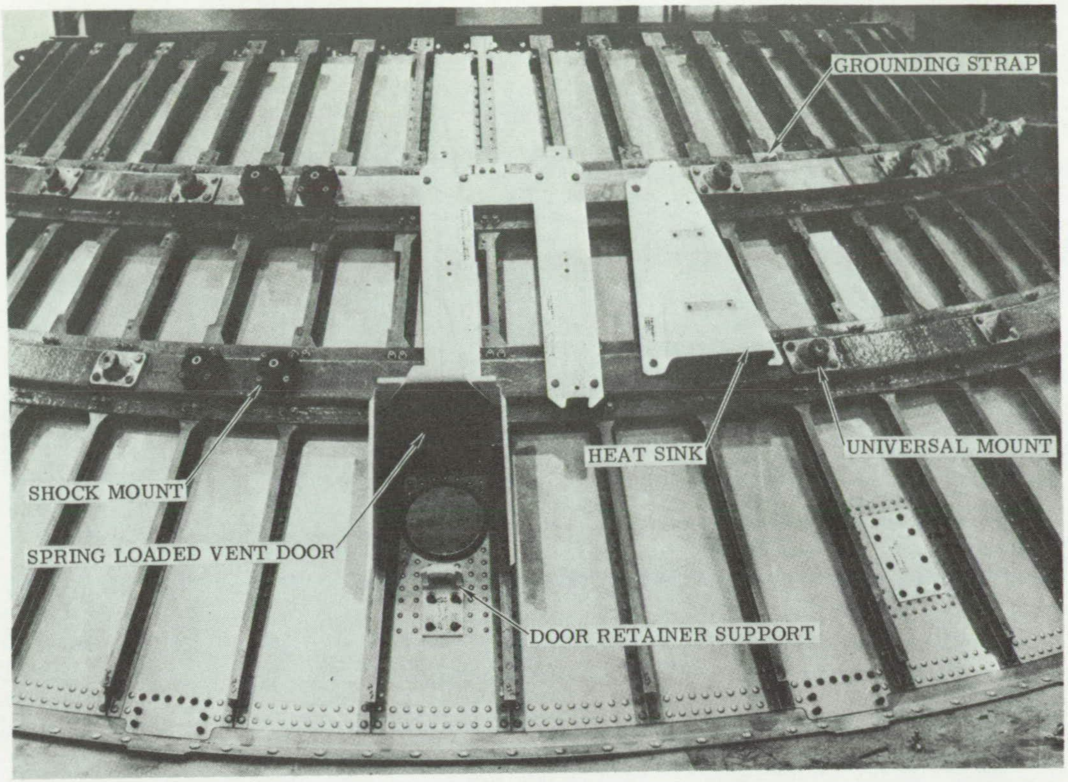


Figure 3-7. Equipment module structure.

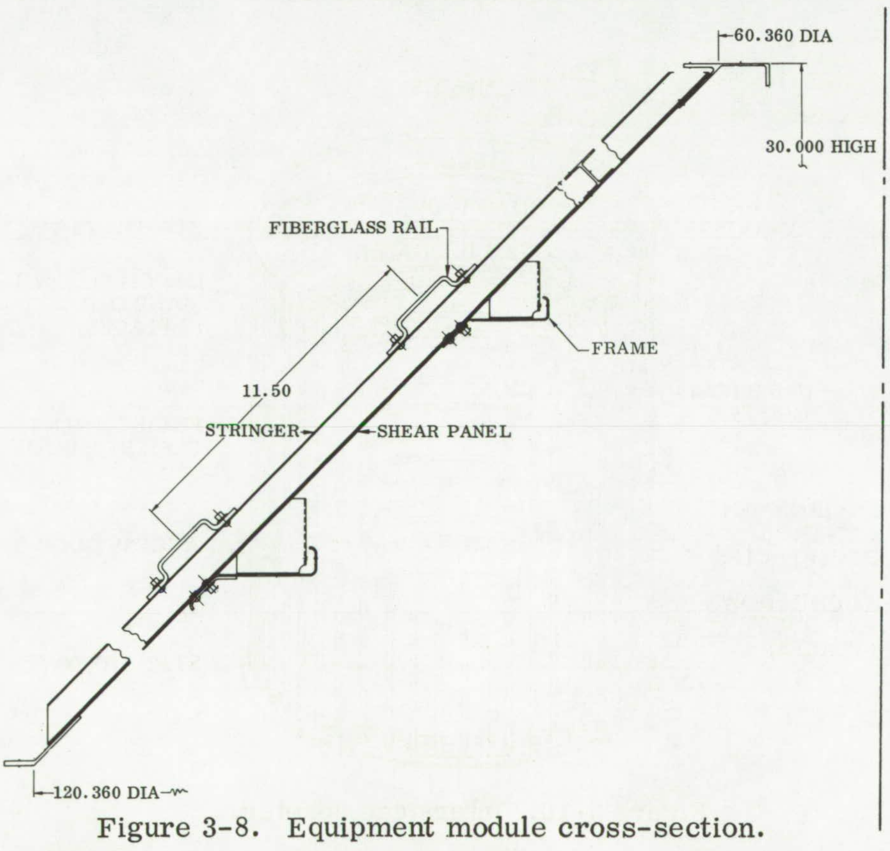


Figure 3-8. Equipment module cross-section.

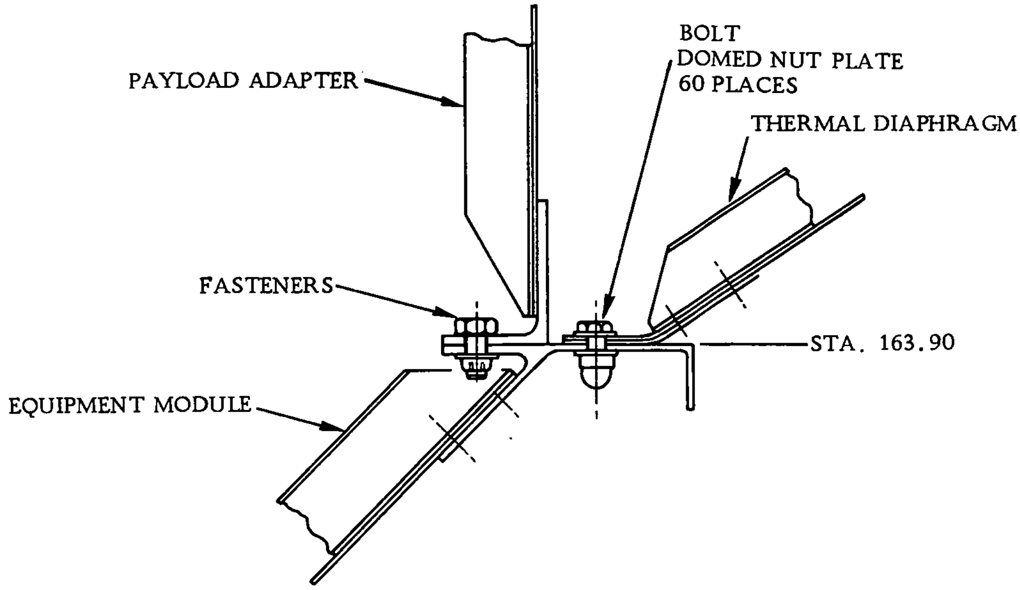


Figure 3-9. Equipment module, payload adapter, and thermal diaphragm interface.

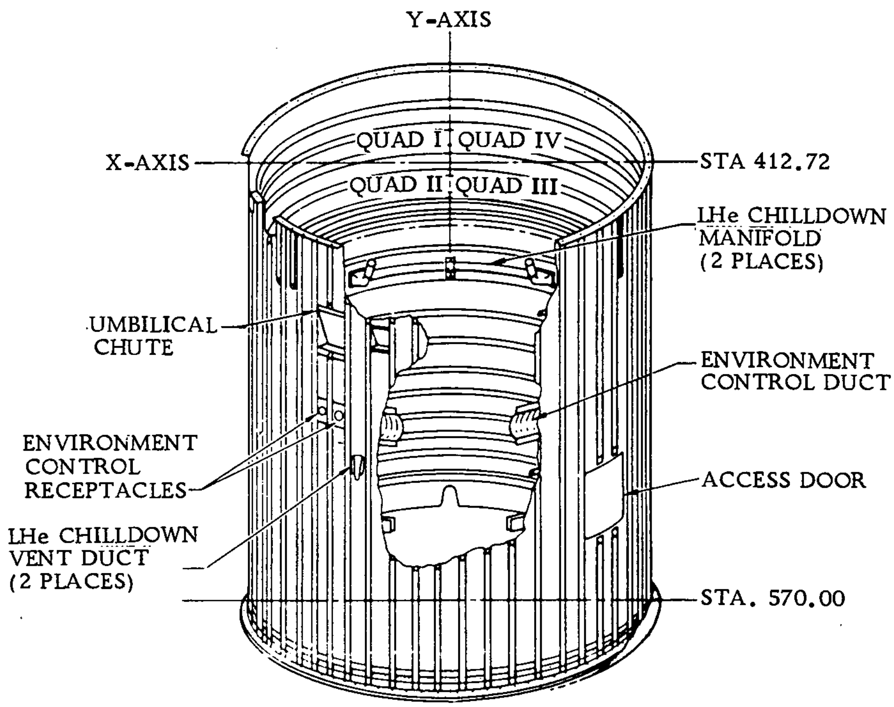


Figure 3-10. Interstage adapter.

A flexible linear shaped charge (FLSC) system (Figure 3-11) is provided to separate the Centaur vehicle from the Atlas booster vehicle by cutting the interstage adapter near the forward end. The charge is located on the forward interstage adapter flanged ring, approximately 0.5 inch aft of the interfaces of the Centaur mating ring and the interstage adapter. The separation signal is supplied from the Centaur sequence control unit to the two detonators. Each detonator has two outputs through which the charge is ignited. Power is supplied by the Atlas power supply. The detonators are located on opposite sides of the interstage adapter near the forward end. A pyrotechnic control unit remotely controls the initiation of the Atlas/Centaur separation system pyrotechnics. The pyrotechnic control unit contains relays and current limiting and isolation resistors. When actuated (closed) a power relay completes the circuit that initiates the associated pyrotechnics. Isolation resistors provide isolation of pyrotechnic devices, so that a short on one device will not inhibit the firing of another. The pyrotechnic control unit is located on the inside surface of the interstage adapter.

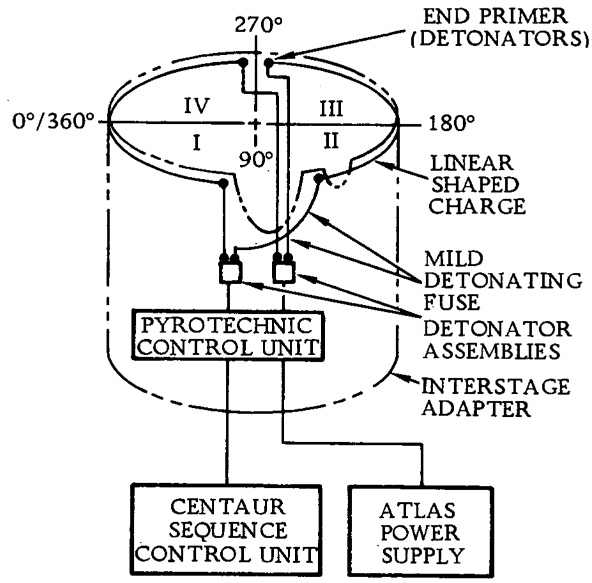


Figure 3-11. Atlas/Centaur separation system arrangement.

Physical separation of the two stages results from retrorocket forces that retard the forward motion of the Atlas. Four separation bumper guides assist in guiding the Centaur aft section out of the interstage adapter during the first 10 inches (approx.) of separation.

Engine chilldown vent valves are attached by flexible lines to two manifolds in the interstage adapter. Each manifold empties into a chilldown vent duct. The two ducts are located 180 degrees from each other so as to produce a net nonpropulsive effect. The vent ducts are in the shape of fins to direct the flow of chilldown helium or hydrogen away from the vehicle boundary layer airflow during atmospheric flight. Breakaway connections are provided for four flexible teflon tubes that connect between the engine's helium chilldown tubing and the interstage adapter helium chilldown collector manifold. Slip-joint disconnects are provided for the LO_2 tank vent duct and the H_2O_2 emergency dump line.

An environmental control duct runs circumferentially inside the adapter. Heated gas (air prior to propellant tanking, gaseous nitrogen thereafter) is pumped into the duct through the environmental control receptacles to maintain a temperature controlled low humidity environment. The nitrogen also prevents formation of combustive mixtures in the interstage adapter during tanking.

Two large access doors near the aft end of the interstage adapter allow access for general maintenance of Centaur systems after mating. Near the forward end, 16 small doors allow for reaction-control engine servicing. One door permits servicing of the H_2O_2 storage bottle. Cutouts in the forward portion accommodate electrical harnessing, miscellaneous tubing, and fill-and-drain duct clearance.

The aft umbilical disconnect panel is attached near the forward end of the adapter. Additional attachments include the airborne helium purge bottle and the aft halves of the eight insulation panel jettison hinges.

A cylindrical fiberglass heat shield is attached to the aft end of the interstage adapter at the Station 570 ring. The shield extends six inches on either side of the ring to reduce the effect of aerodynamic heating at the Atlas/interstage adapter interface.

3.1.4 PAYLOAD ADAPTERS. The payload adapter supports the payload and distributes the loads imposed by the payload to the equipment module. Three different payload adapters are currently available for the D-1A Centaur. All attach to the forward ring of the equipment module and therefore have a common machined aluminum aft ring approximately 60 inches in diameter with a bolt hole pattern to match the equipment module. All three have a similar basic structure, conventional skin-stringer design with a formed aluminum midframe and machined aluminum forward and aft rings. The forward ring of each adapter incorporates mounting provisions for the specific payload for which it was designed. Other payload adapters can be designed that may vary in length and forward ring design to accommodate different payloads.

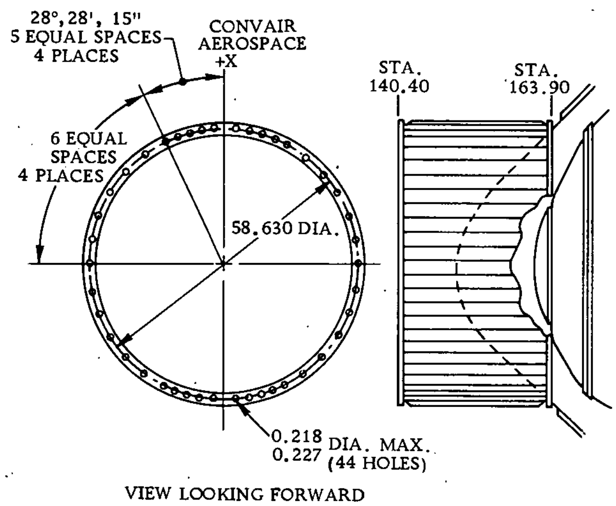


Figure 3-12. Pioneer G adapter.

Pioneer G Payload Adapter. (Figure 3-12) This adapter is cylindrical, approximately 60 inches in diameter and 23.5-inches long.

Intelsat IV Payload Adapter. (Figure 3-13). This adapter is conical; it has an aft diameter of approximately 60 inches, a forward diameter of approximately 56 inches, and is 18.75 inches long.

Mariner-Venus Mercury '73 Payload Adapter. This adapter is conical; it has an aft diameter of approximately 60 inches, a forward diameter of approximately 52 inches, and is 18.75 inches long.

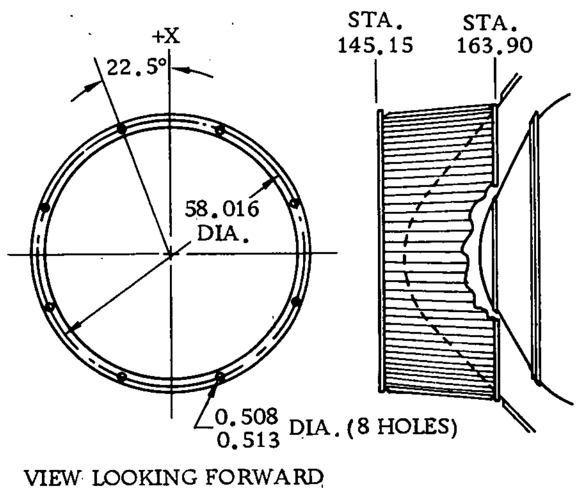


Figure 3-13. Intelsat IV adapter.

3.1.5 BOLT-ON STRUCTURES. Major bolt-on structure items are as follows:

a. An aft umbilical panel which includes provisions for mounting aft disconnects.

b. Main engine actuator support structure which provides mounting interfaces on the aft bulkhead for the yaw actuators and on the liquid oxygen sump for the pitch actuators.

c. Attitude control engine mounts for attaching the hydrogen peroxide attitude control engines. They incorporate a thermal barrier between the engines and the cold propellant tank.

d. Aft bulkhead bolt-ons:

   1. Wiring and tubing support installation.

   2. Aft bulkhead bottle supports (helium and hydrogen peroxide).

   3. Aft umbilical panel seal installation.

   4. Signal conditioner and multiplexer mountings.

   5. Instrumentation package mounting.

e. The aft wiring tunnel bulkhead is of glass fiber phenolic laminate construction and is insulated on its forward face. It bolts to the tank aft ring, seals off the area under the fuel elbow, and provides penetrations and mounting holes for various harnesses and tube fittings. It also incorporates the staging disconnect mounting.

f. Aft seal plate, in conjunction with the aft wiring tunnel bulkhead, seals the annular area between the tank aft ring and the insulation panels.

g. Forward wiring tunnel bulkhead acts as a closure for the wiring tunnel and as a wiring and tubing interface at Station 219. It also provides attachments for the aft end of the umbilical island fairing.

h. Four separation bumpers are attached to weldments on the aft bulkhead, and assist in guiding the Centaur out of the interstage adapter during vehicle separation.

3.2 INSULATION SYSTEMS

ELEMENTS:	Subsection
• Forward Bulkhead and Stub Adapter Insulation	3.2.1
• Insulation Panels	3.2.2
• Intermediate Bulkhead Insulation	3.2.3
• Aft Bulkhead Radiation Shield and Membrane	3.2.4

FUNCTIONS:

- Protect Components from Excessive Cooling
- Insulate Cylindrical Tank Section on Ground and During Ascent to Limit LH_2 Boiloff
- Minimize Heat Transfer Between Tanks

3.2.1 FORWARD BULKHEAD AND STUB ADAPTER INSULATION. (Figures 3-14 and 3-15) This insulation system is comprised of three major portions, the forward tank bulkhead insulation, the tank access door and LH_2 vent system insulation, and the stub adapter insulation.

Insulation covers the forward tank bulkhead surface from just forward of the tank forward ring to the access door mounting flange. This assembly consists of two separate blankets each approximately 3/4 inch thick laid one on top of the other to produce a total assembly thickness of 1-1/2 inches. Each blanket consists of a total of 23 sheets of aluminized Mylar arranged as follows:

Top (outer) Layer:	1 flat sheet with scrim, approx. 7 mils thick
Intermediate Layers:	12 dimpled sheets, 1/2 mil thick
	11 flat sheets, 1/4 mil thick
Bottom (outer) Layer:	Same as top layer

The dimpled and the flat intermediate layers alternate. The dimpling produces a separating gap of 0.06 ± 0.02 inch.

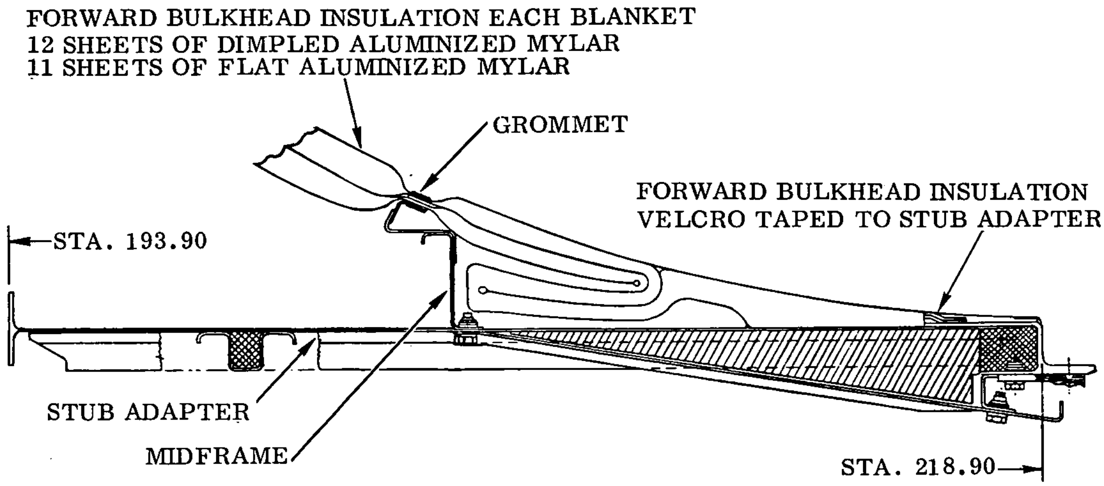


Figure 3-14. Forward bulkhead insulation.

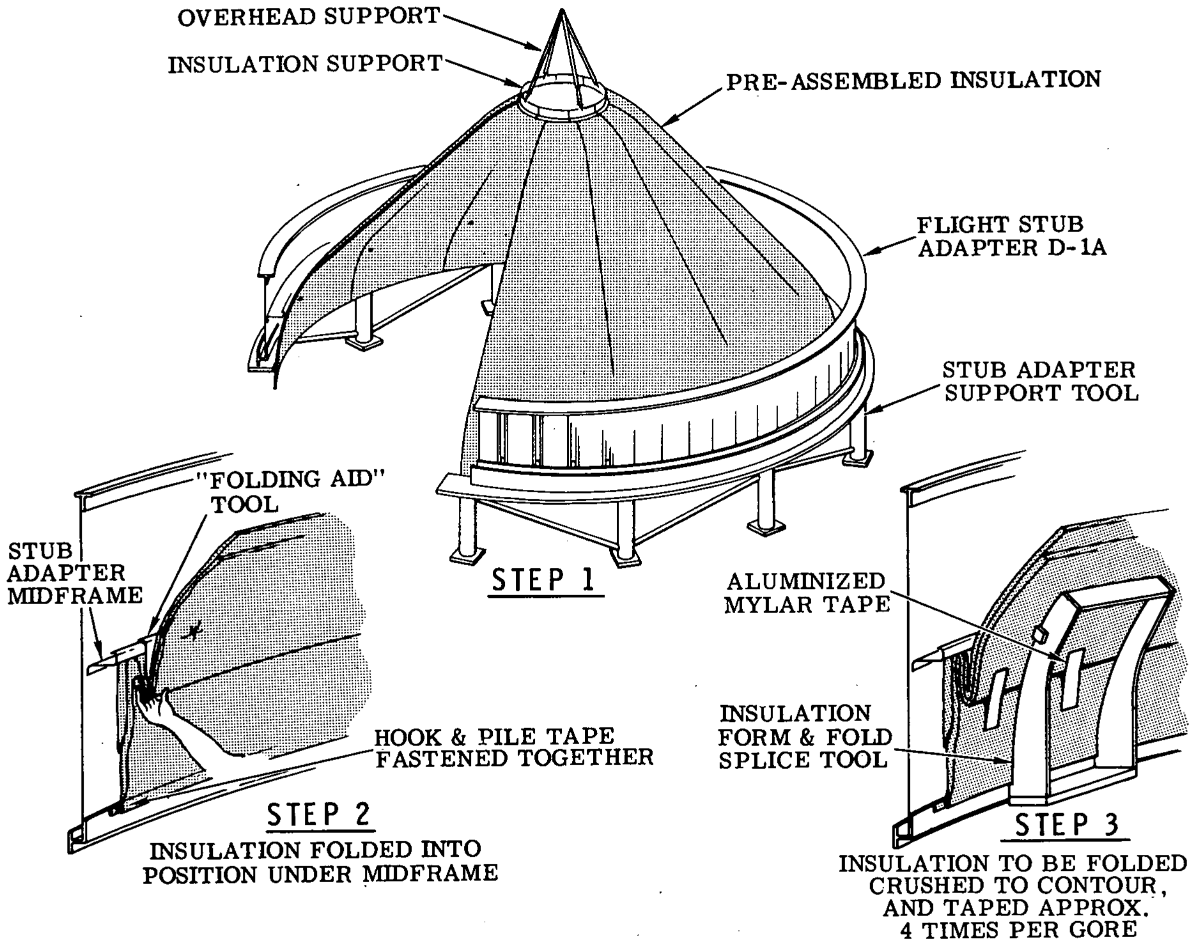


Figure 3-15. Forward bulkhead insulation installation procedure.

Each blanket is fabricated in 12 separate gores. The layers of each gore are held together by plastic (polycarbonate) grommets arranged along both of its longitudinal edges. The forward and aft ends of each gore are sealed with aluminized adhesive tape. The gores are fastened to each other with pin type fasteners inserted in the plastic grommets. Each gore incorporates a single brass grommet near one longitudinal edge located opposite and slightly below the stub adapter midframe. The two blankets attach to each other and to clips on the stub adapter midframe by 12 bolts, each of which passes through a brass grommet in each blanket and then fastens to a midframe clip. Aluminum foil is interleaved with the gore layers at the brass grommet to ground each layer to the stub adapter via the grommet, the bolt, and the clip at each of the 12 attach points. The aft end of the inner blanket is attached to the stub adapter aft ring with velcro strips, one strip is bonded to the aft ring and the mating strip is sewn to the blanket assembly.

The forward bulkhead insulation (blanket assembly) is installed on the stub adapter before the stub adapter is installed on the tank. The wedge-shaped area formed by the tank and the portion of the stub adapter below its midframe is filled with insulation by prefolding the blanket assembly while it is being installed on the stub adapter.

The forward end of the blanket assembly does not attach to any structure with fasteners, but is held down onto the bulkhead surface by the tension from the midframe attachment and by compression from the LH_2 tank vent ducts.

A separate insulation cover, 1-1/2 inches thick, is provided for the tank access door and LH_2 tank vent ducts. This cover goes around, not over, the two vent valves. A split epoxy fiberglass pan is installed on each vent valve mounting flange to ensure free gas circulation space around the vent valves by preventing the insulation cover from contacting the valves. An oval shaped doubler having a central hole to fit around the access door flange is bonded to the blanket assembly. This doubler is a single sheet of aluminized Mylar the same as the blanket outer layer. The doubler has a strip of velcro material (hooks) sewn to it around its outer edge. The cover has a mating velcro strip sewn to it. The doubler extends on either side under the LH_2 tank vent duct. The cover extends on either side to cover the vent ducts up to the point where they turn outwards.

A 1-1/2-inch thick multilayer wrap of aluminized Mylar insulation covers the LH_2 tank vent ducts from the duct support on the equipment module to the point where the ducts penetrate the equipment module.

The aft half of the stub adapter is insulated by the aft portion of the forward tank bulkhead insulation described above. An insulation assembly is installed on the inboard surface of the stub adapter between the forward ring and the midframe. The assembly consists of 12 segments, each covering a 30-degree arc. Each segment is made up of a basic blanket consisting of five dimpled and six flat sheets of Mylar identical to those used as intermediate layers in the forward tank bulkhead insulation. This blanket is repeatedly folded back upon itself to form a finshed segment of eleven blanket thicknesses. The segment is then encased in a "pillow slip" type cover made of Mylar/scrim material identical to the outer layer of the forward tank bulkhead insulation. Each segment is secured to the stub adapter skin by velcro tape.

3.2.2 INSULATION PANELS. The insulation panels (Figure 3-16) cover the fuel tank from the stub adapter to the aft cylindrical ring to reduce LH_2 boiloff on the ground and during atmospheric ascent. They prevent formation of liquid air on the tanks, and serve as aerodynamic fairings over tank-mounted equipment and structure.

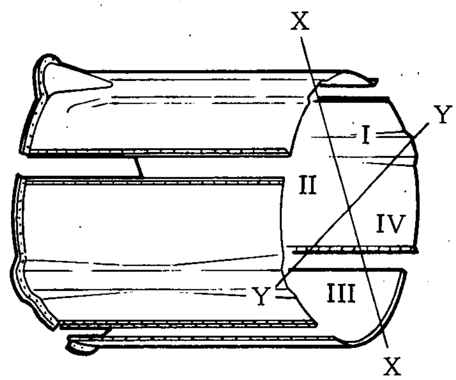


Figure 3-16. Jettisonable insulation panels.

The panels are fabricated of a sandwich construction with fiberglass-reinforced skins and a fiberglass honeycomb core filled with closed-cell polyurethane foam. Formed in four longitudinal quarters approximately 193 inches long, the assembled panels make an annular shell with an approximate 120-inch inner diameter.

The insulation panels are jettisoned approximately 45 seconds after Atlas booster engine cutoff. Each panel rotates about two separation hinges attached to the interstage adapter. As shown in Figure 3-17, flexible linear shaped charges are located

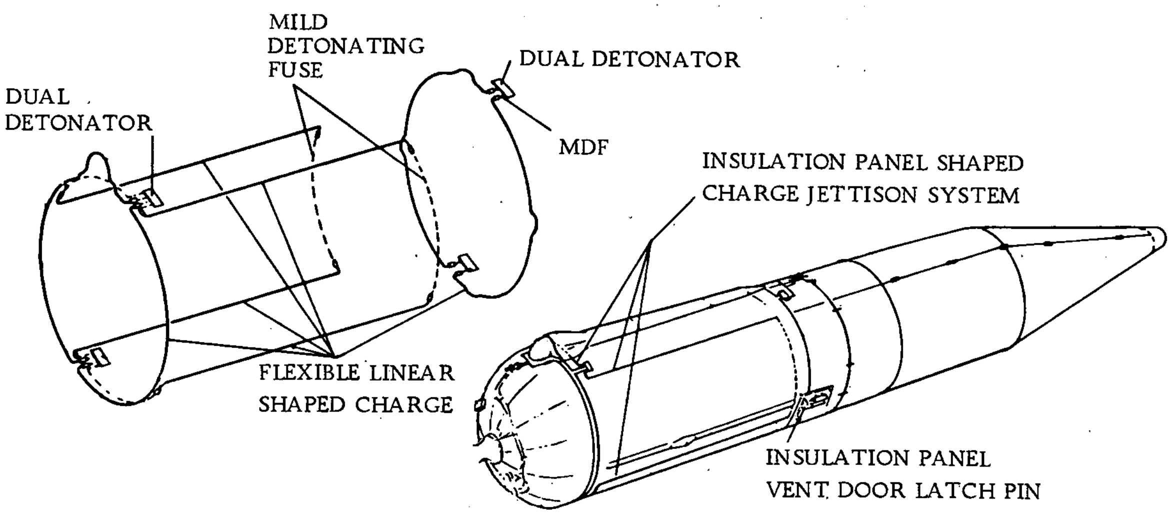


Figure 3-17. Centaur ordnance items.

along each longitudinal splice and circumferentially around the aft seals and the forward cylindrical tank ring. The charges are fired by detonators and mild detonating fuses activated by a pyrotechnic control unit upon signal from the Centaur sequence control unit. Power is supplied by the Atlas for the splice and aft end charges and by the Centaur nose fairing pyrotechnic batteries for the forward end charges.

Plastic foam pads, bonded longitudinally on one-inch centers, form a cavity between the panels and the tank surface. This cavity is purged with gaseous helium during tanking, prior to launch, and during flight to minimize freezing of air between the panels and tank. Purge gas enters through a distribution ring on the aft end of the tank and is vented overboard through the purge vent on the stub adapter. (Figure 3-18) A circumferential, flexible seal installed between the forward ends of the insulation panels and the aft ring of the stub adapter ensures that a positive helium gas pressure is maintained. Prior to liftoff, an actuator opens a door in the vent system, increasing the vent area so that purge gas will not cause overpressurization of the cavity during ascent.

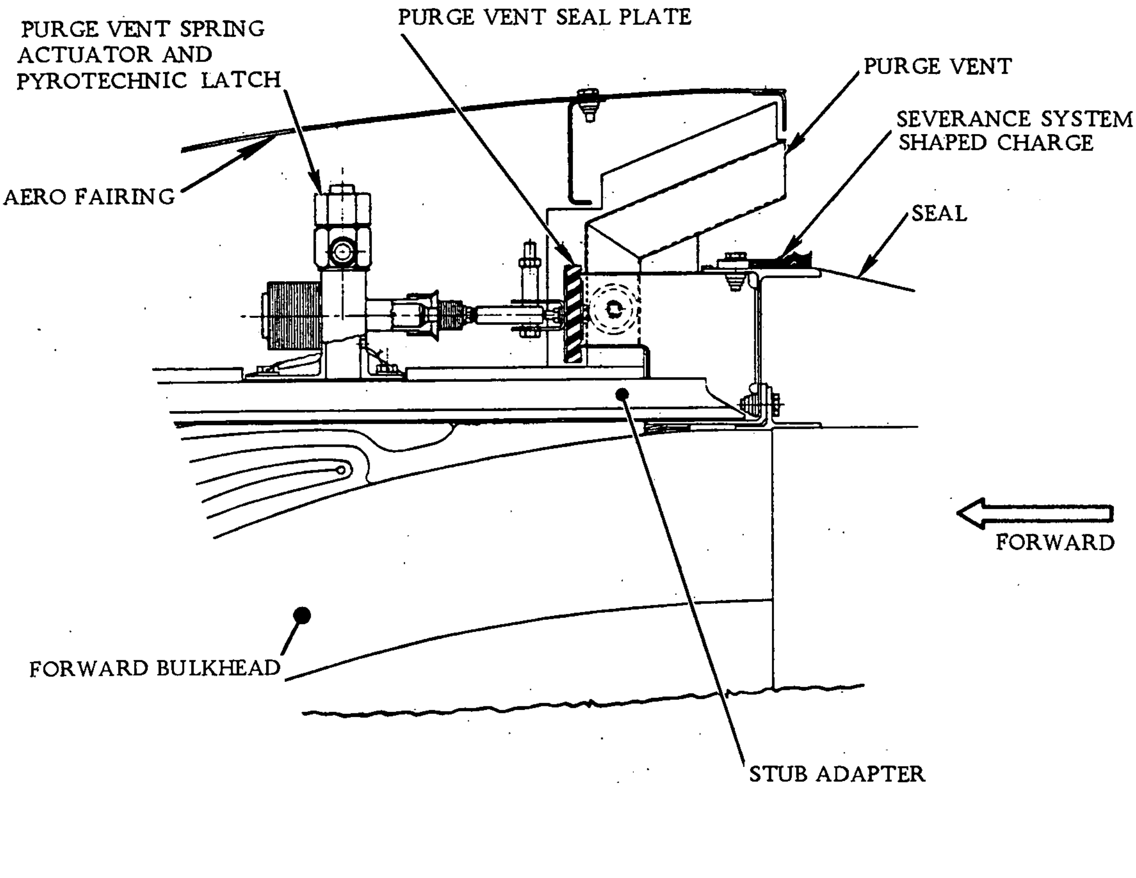


Figure 3-18. Purge vent door actuator.

3.2.3 INTERMEDIATE BULKHEAD INSULATION. This is an integral part of the intermediate bulkhead construction, see Subsection 3.1.1 for details.

3.2.4 AFT BULKHEAD RADIATION SHIELD AND MEMBRANE. (Figures 3-19 and 3-20) The radiation shield is a rigid assembly made of laminated nylon fabric with aluminized Mylar on its inner surface and white polyvinyl fluoride on its outer surface. It is made up of 12 gores forming a complete ellipsoidal half covering the aft tank bulkhead. The shield is supported on brackets which hold it one inch from the tank bulkhead surface. Cutouts are provided for the LO_2 sump and the various equipment mounting brackets.

The membrane is a layer of dacron-reinforced aluminized Mylar which is in contact with the aft tank bulkhead. All penetrations of the membrane are sealed. The membrane acts as an additional insulation/radiation shield and as a seal to contain convective gases or leakage of cryogenics to prevent their impingement on aft bulkhead-mounted equipment.

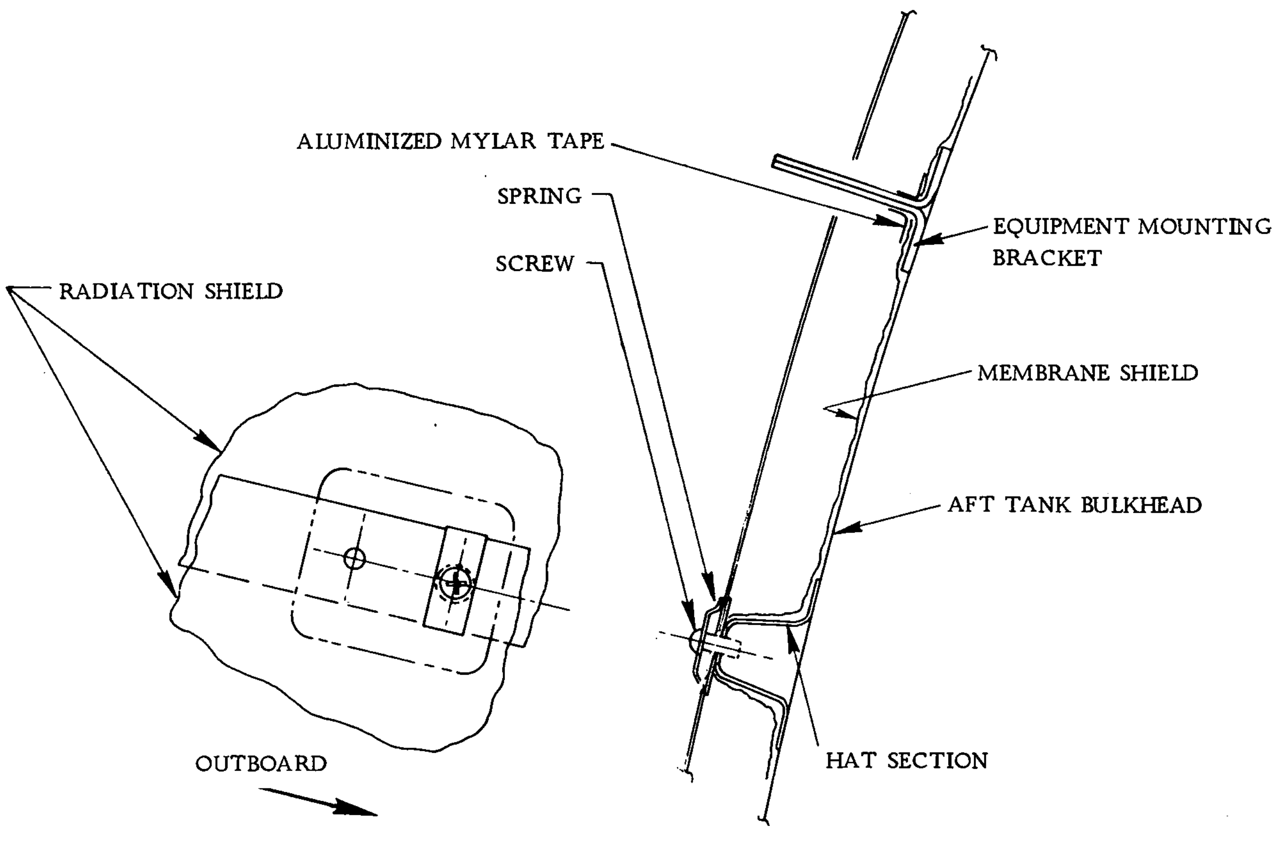


Figure 3-19. Aft bulkhead radiation shield and membrane.

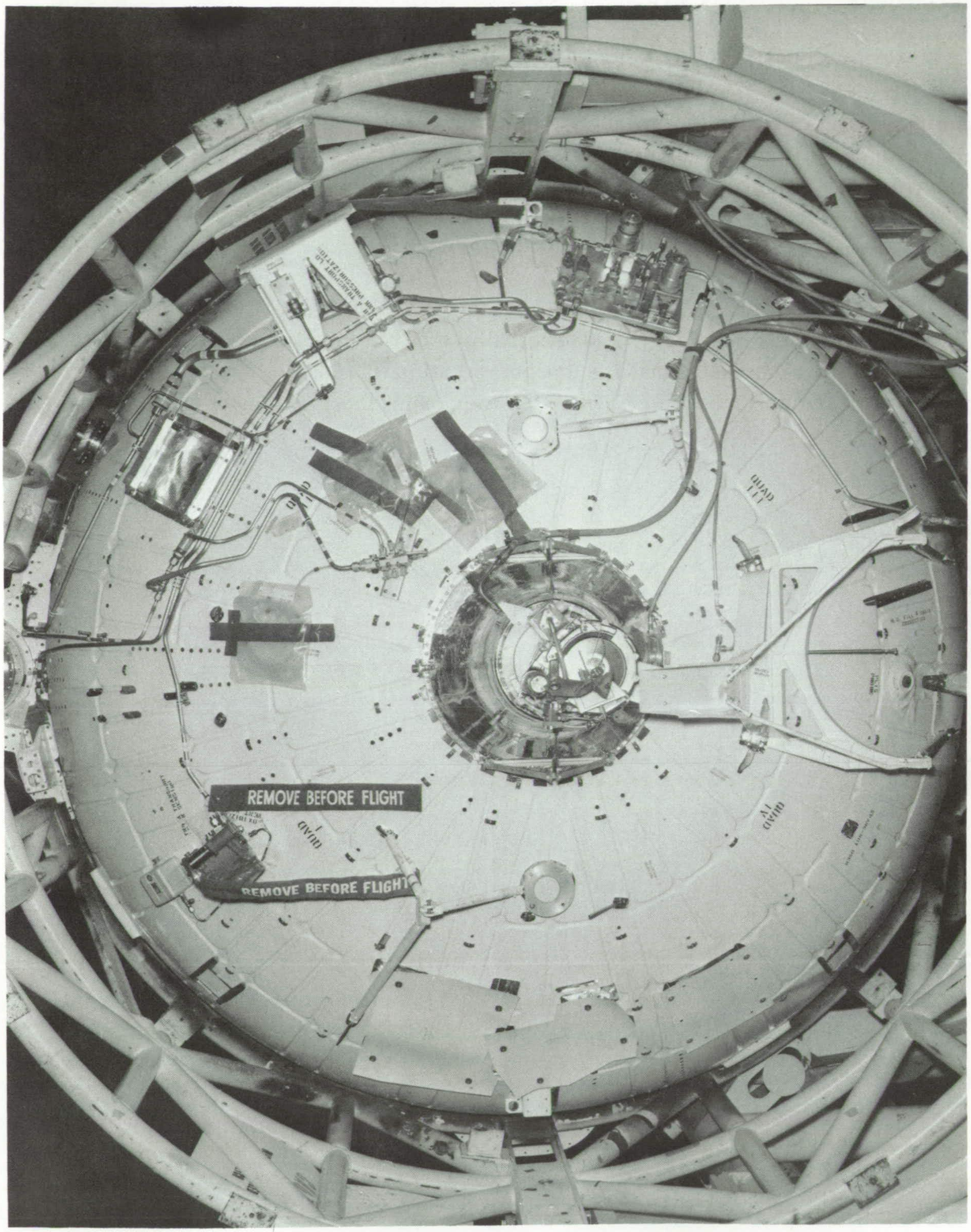


Figure 3-20. Aft bulkhead radiation shield.

3.3 CENTAUR D-1A STANDARD FAIRING

SECTIONS:	Subsection
• Fairing System Conical/cylindrical fiberglass section Aluminum split barrel section	3.3.1
• Spacecraft Envelope	3.3.2
• Separation System	3.3.3

FUNCTIONS:

- Protect the Spacecraft during Ascent Phase of Trajectory
- Permit a Controlled Environment Around the Spacecraft on the Ground

3.3.1 FAIRING SYSTEM. The Centaur D-1A nose fairing (Figure 3-21) consists of two major elements, the conical/cylindrical fiberglass section and the aluminum split barrel section. The fairing is attached to the stub adapter until the two fairing halves are jettisoned. This occurs during the first Centaur burn and is initiated by a discrete from the digital computer unit.

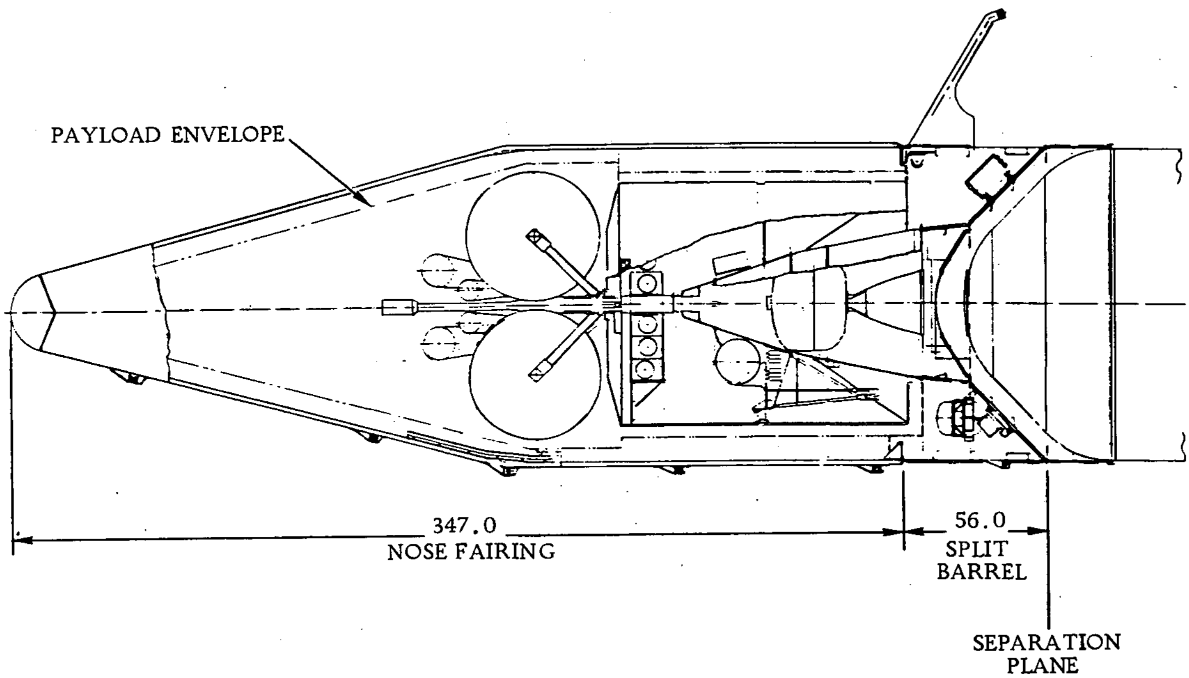


Figure 3-21. Centaur D-1A nose fairing.

The fiberglass section is nearly 29 feet long and 10 feet in diameter in the cylindrical section. Its structure consists of inner and outer skins laminated to a 1.75-inch thick phenolic core (Figure 3-22). The phenolic core is 3/8-inch over-expanded cells of 3.1 pounds per cubic foot density.

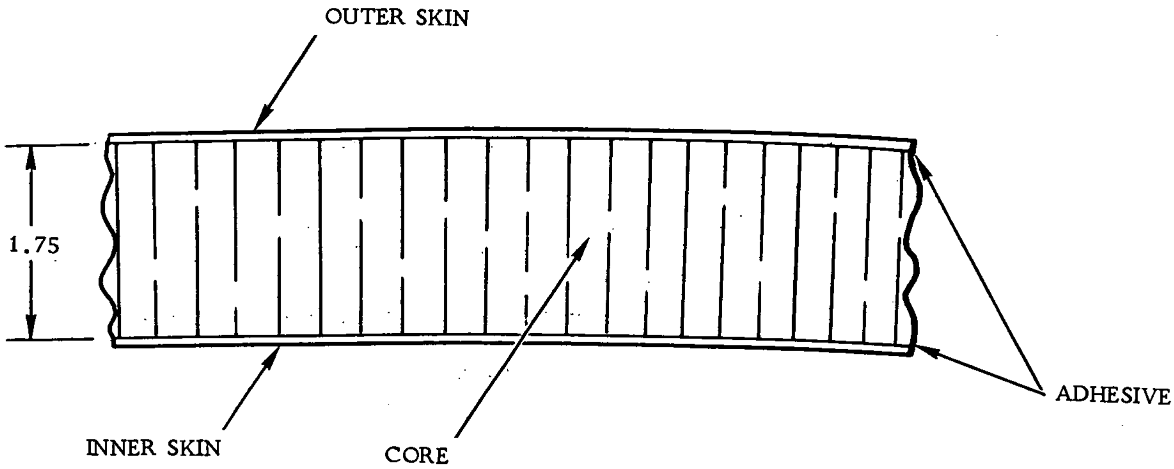


Figure 3-22. Basic shell construction.

In the basic shell area each skin is 0.04 inch thick. In areas where it is structurally required, the skin thicknesses are increased. The outer skin is 181 preimpregnated phenolic glass cloth. The inner skin is 181 preimpregnated epoxy glass cloth. The adhesive used to bond both skins to the phenolic core is 422J tape.

The split barrel (Figure 3-23) is 56 inches long and 10 feet in diameter. It is of aluminum skin/stringer construction and has four doors, one in each Quad, for access to the Centaur equipment module. The doors in Quads I, III, and IV are 39.5 inches high and 38 inches wide. The door in Quad II is 39.5 inches high and 26 inches wide.

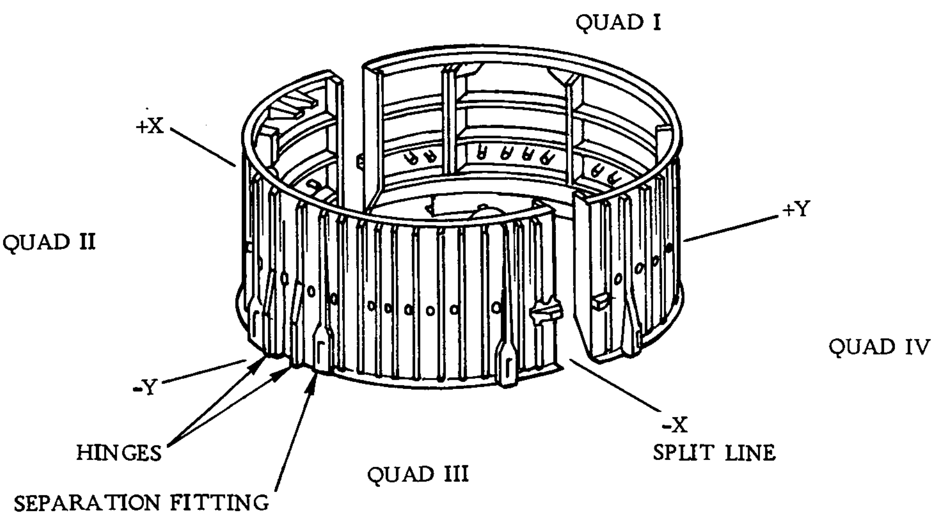


Figure 3-23. Split barrel.

Various equipment is mounted on the split barrel, including LH_2 ducting from the Centaur tank, air conditioning ducting for the equipment module area, electrical and instrumentation inflight disconnects, two pyrotechnic batteries and six pyrotechnic relays.

An air conditioning inlet for the payload compartment is located near the bottom of the fiberglass section (Figure 3-24). The ground duct is held to the airborne disconnect by an inflatable seal. The duct is released by a reduction in pressure in the seal. The umbilical boom retracts the ground duct as the vehicle lifts off the pad. The spring-loaded door closes at liftoff when the ground duct disconnects. It is held closed by a mechanical latching mechanism.

Air enters a plenum attached to the inside of the fairing (Figure 3-25). The plenum is made from 181 epoxy pre-impregnated glass cloth. Both the air conditioning duct and the plenum are mission-peculiar.

The air conditioning duct attaches to the plenum. It is made from 5-inch diameter aluminum tubing routed to meet mission requirements.

Air is supplied until 110 minutes before launch, when the system switches to gaseous nitrogen. Gas is passed through a 0.3-micron HEPA filter at a flow rate of 40 to 100 pounds per minute. Temperature is controlled to ±5°F from 60 to 100°F.

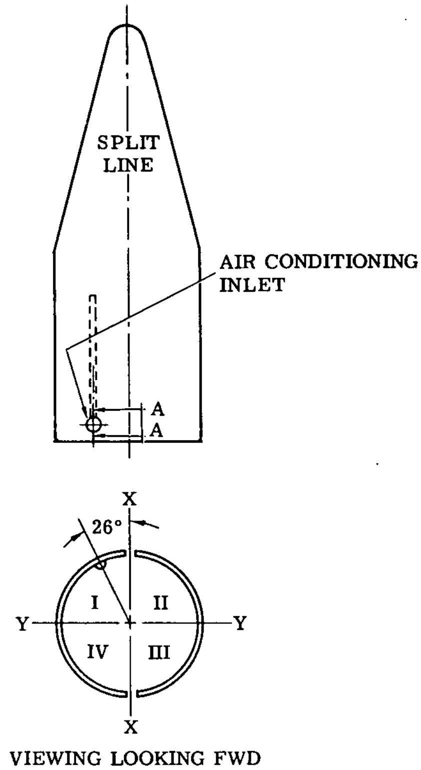


Figure 3-24. Nose fairing.

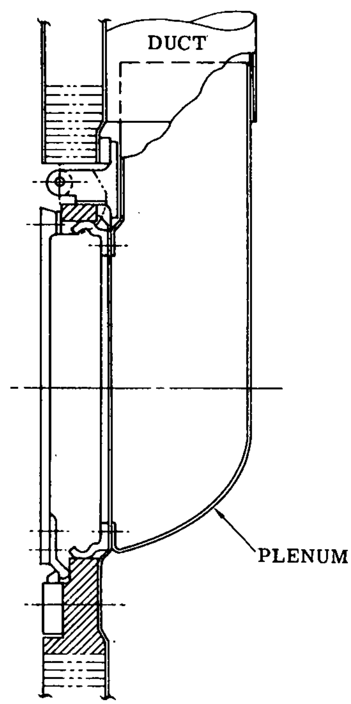


Figure 3-25. Air conditioning inlet.

3.3.2 SPACECRAFT ENVELOPE. The spacecraft envelope is shown in Figure 3-26 with the Intelsat IV spacecraft adapter. The adapter is mission peculiar, with the actual envelope length varying for each adapter. Total envelope length is about 301 inches with this adapter.

The envelope shown is for the maximum permissible dimensions of the payload. Manufacturing tolerances and dynamic motion of the fairing have been accounted for in arriving at this envelope. A minimum radial clearance of approximately one inch will exist between the envelope and fairing in its maximum configuration during ascent and fairing jettison. The spacecraft must be smaller than this envelope by the amount of its dynamic motion plus buildup tolerances.

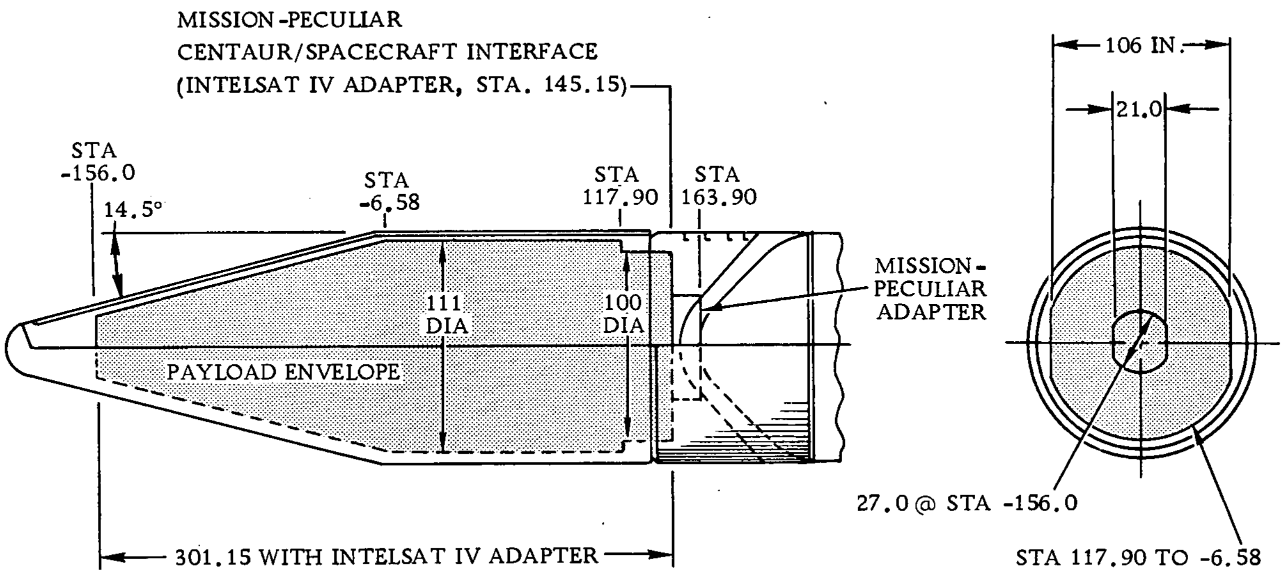


Figure 3-26. Spacecraft envelope for Centaur D-1A.

3.3.3 SEPARATION SYSTEM. The two fairing halves are held together by 12 explosive bolts mounted along the vertical split line. Eight explosive bolts attach the fairing to the stub adapter. Separation force is provided by two springs held compressed by the shroud halves and located at the forward end of the fairing.

Each fairing half swings away from the payload, rotating about two hinges attached to the stub adapter.

Longerons at the vertical split line are bonded and riveted to the basic shell (Figure 3-27). The longerons are made from number 181 phenolic preimpregnated glass cloth. Thirty-two aluminum shear pins along each split line transfer flight loads between the shell halves.

An inner seal between the longerons prevents leakage. The installation consists of a solid silicone seal bonded to one fairing half and a bulk-type seal bonded to the other fairing half. Fastening the nose fairings together compresses the rubber to form the seal. The installation is leak-tested at 0.45 psi.

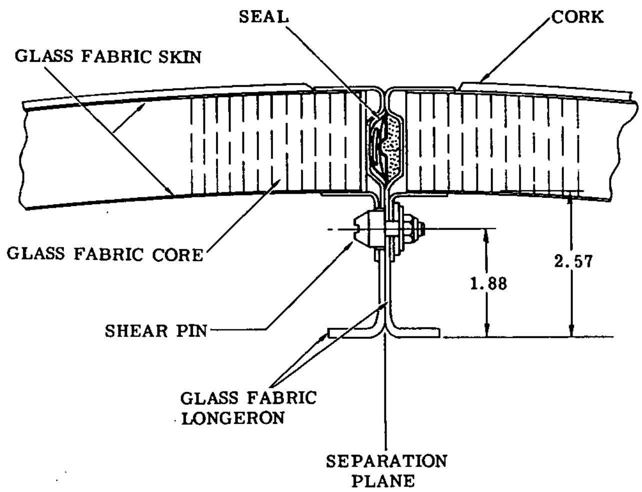


Figure 3-27. Typical vertical split line.

A typical explosive bolt installation is shown in Figure 3-28. A one amp/one watt pressure cartridge is located in each end of the bolt. Spherical seats on the bolt ends and fittings allow fitting misalignments without bolt binding. Energy absorbers reduce the impact load that occurs at bolt fracture into the structure. The bolt fracture plane is located at the separation plane of the fittings.

Each explosive bolt is mechanically redundant. The bolt will function properly when fired from either end with one electric explosive device and a tension load of 0 to 36,000 pounds applied. Its ultimate tensile load capability is 41,000 pounds and it will operate in the range from -65° to +200°F. Shelf life is five years. Each bolt (Figure 3-29) is a self-contained unit, with no additional hardware or adjustment required except electric explosive devices.

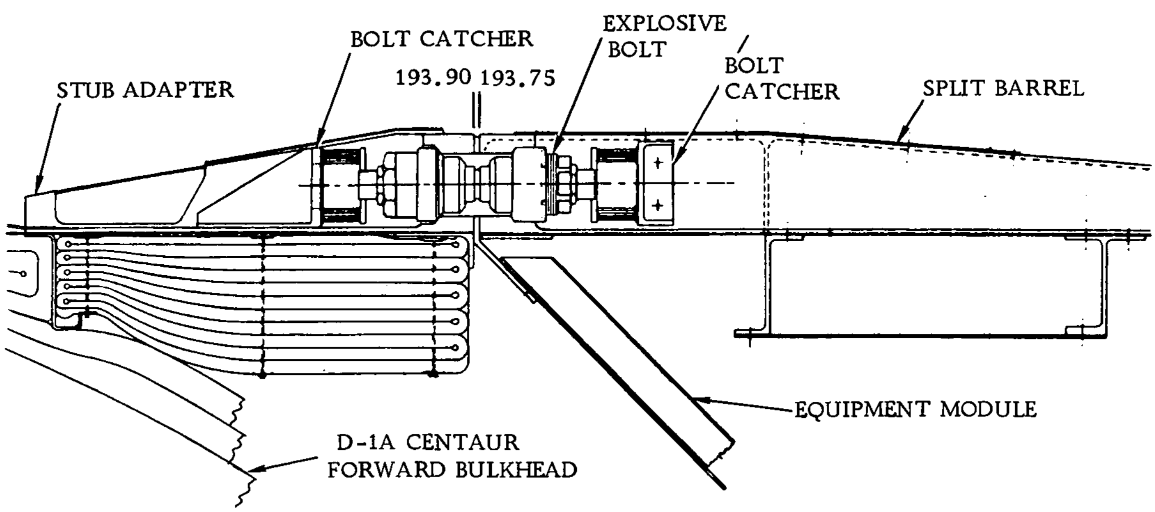


Figure 3-28. Typical explosive bolt installation.

The fairing electrical pyrotechnic system (Figure 3-30) consists of two independent subsystems, each capable of supplying power to separate the nose fairing.

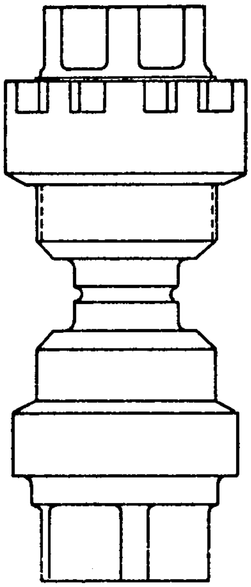


Figure 3-29. Explosive bolt.

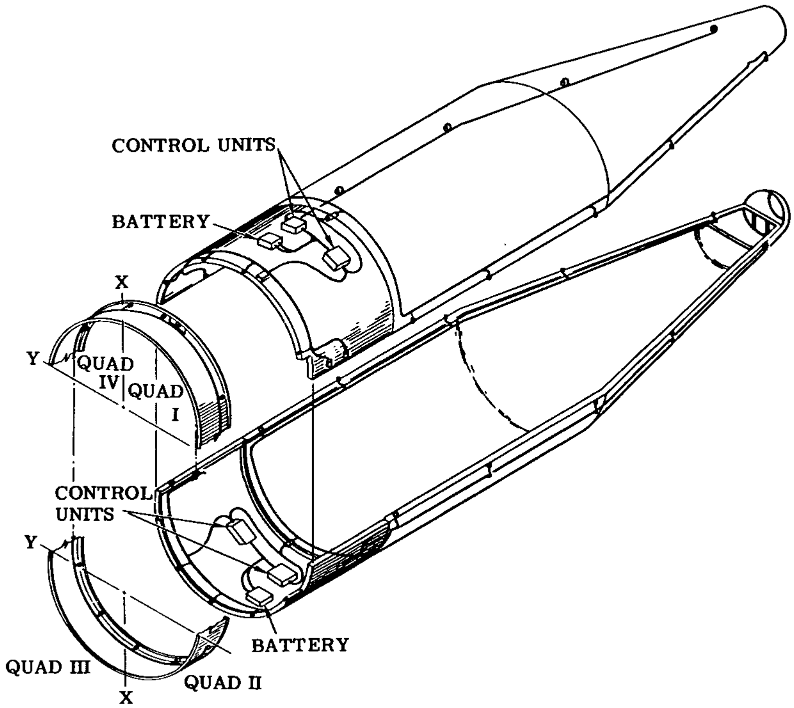


Figure 3-30. Nose fairing electrical pyrotechnic system.

Each subsystem consists of one 28 volt d-c battery, two control units, and 20 electro-explosive devices (EEDs). A minimum of five amperes of current is supplied to each EED for a minimum of 50 milliseconds. Maximum time to fire with this current is 10 milliseconds. All EEDs receive current simultaneously. Each has one amp/ one watt, no-fire capability.

The jettison actuators are loaded before liftoff. One actuator is attached to each fairing half, and is capable alone of jettisoning both halves. One end of the actuator is pinned, the other is a spherical socket (Figure 3-31).

Each actuator provides a force of from 1225 to 1450 pounds, with a stroke of 30.5 ± 1.5 inches. Compressed length is approximately 29 inches. Jettison force is applied for 2-1/2 degrees of fairing rotation. No contamination results from actuator action, as each is a self-contained unit.

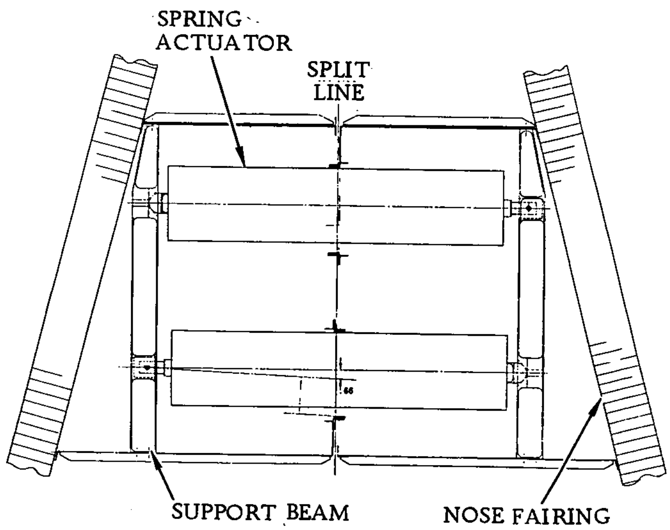


Figure 3-31. Jettison actuator installation.

Each fairing half is retained in the two hinges for about 30 degrees of rotation. After about 70 degrees the fairing halves are clear of the hinges. The hinge installation is shown in Figure 3-32.

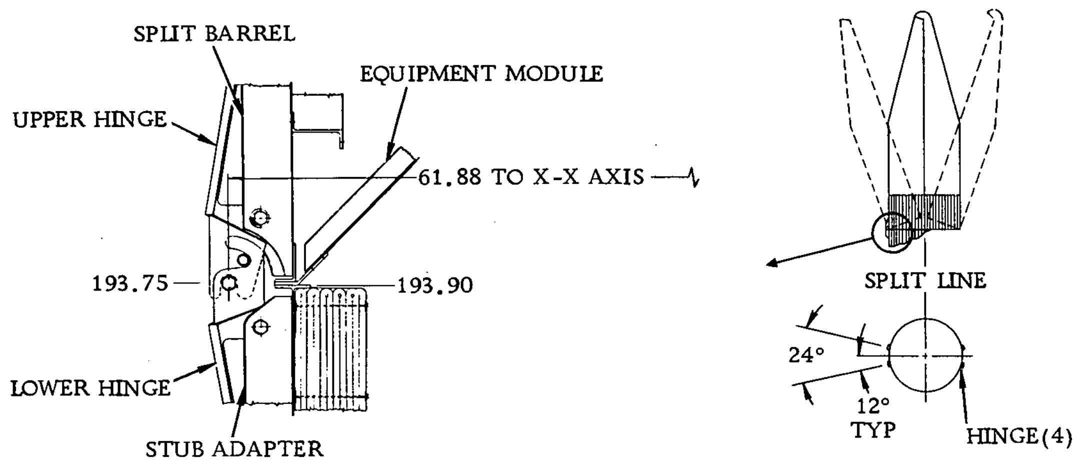


Figure 3-32. Hinge installation.

3.4 MAIN PROPULSION SYSTEM

ELEMENTS:	Subsection
• Two Main Engines	3.4.1
• Propellant Feed System (LO_2 and LH_2)	3.4.2
• Engine Chilldown System	3.4.3

FUNCTIONS:

- Provide Main Thrust For Propelling Vehicle
- Provide Thrust Vectoring For Attitude Control During Powered Flight

3.4.1 MAIN ENGINE SYSTEM. Primary vehicle thrust is provided by two government furnished Pratt and Whitney RL10A-3-3 engines. These are constant thrust, turbopump-fed, regeneratively-cooled, liquid rocket engines (Figure 3-33). The engines use liquid hydrogen and liquid oxygen as propellants and are capable of making multiple starts after long coast periods in space. The combustion process is initiated through ignition of the initial flow of propellants (gaseous) with a spark igniter which is an integral part of the engine.

Each engine is attached to the vehicle by a gimbal mount assembly. Power to operate the vehicle hydraulic system is supplied through an accessory drive pad on the engine turbopump assembly. The helium required for engine operation is provided from a storage bottle located on the aft bulkhead. A schematic of the main engine propellant flow system is presented in Figure 3-34.

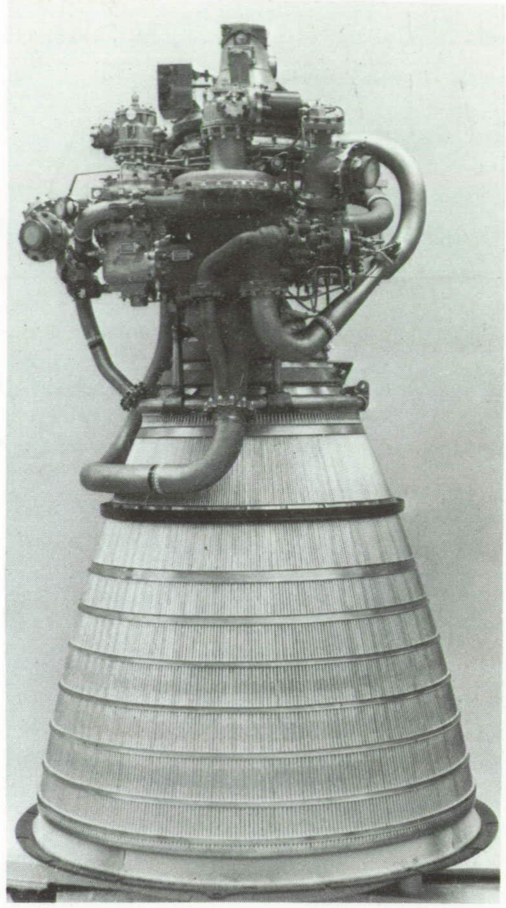


Figure 3-33. RL10A-3-3 engine.

Steady state performance at standard pump inlet conditions and at 200,000 feet altitude is:

Chamber Pressure (nominal)	400 psia
Thrust (nominal)	15,000 lb
Mixture Ratio (nominal)	5.0:1
Specific Impulse (minimum)	439 lb/lb/sec
Specific Impulse (nominal for performance)	444 lb/lb/sec
Rated Continuous Operating Duration (nominal)	450 seconds
LO_2 Flow Rate (nominal)	28.2 lb/sec
LH_2 Flow Rate (nominal)	5.6 lb/sec

The specific impulse constant includes operating ventage specified by Pratt and Whitney Aircraft of 9.0 lb/minute (liquid hydrogen) maximum and 2.7 lb/minute (liquid oxygen) maximum.

Standard pump inlet conditions are:

LH_2 Temperature	38.3°R
LH_2 Total Pressure	30.0 psia
LO_2 Temperature	175.3°R
LO_2 Total Pressure	60.5 psia
Accessory Drive Pad Speed (nominal)	12,100 (+600, -400) rpm
(Minimum at 5.7 mixture ratio)	11,500 rpm
Accessory Drive Pad Running Torque (maximum)	20 lb-in.
Permissible Engine Gimbaling Angle (square pattern)	±4 degrees about engine geometric centerline

Gaseous helium required for engine controls is at 470 ±30 psia and at 300°R to 600°R.

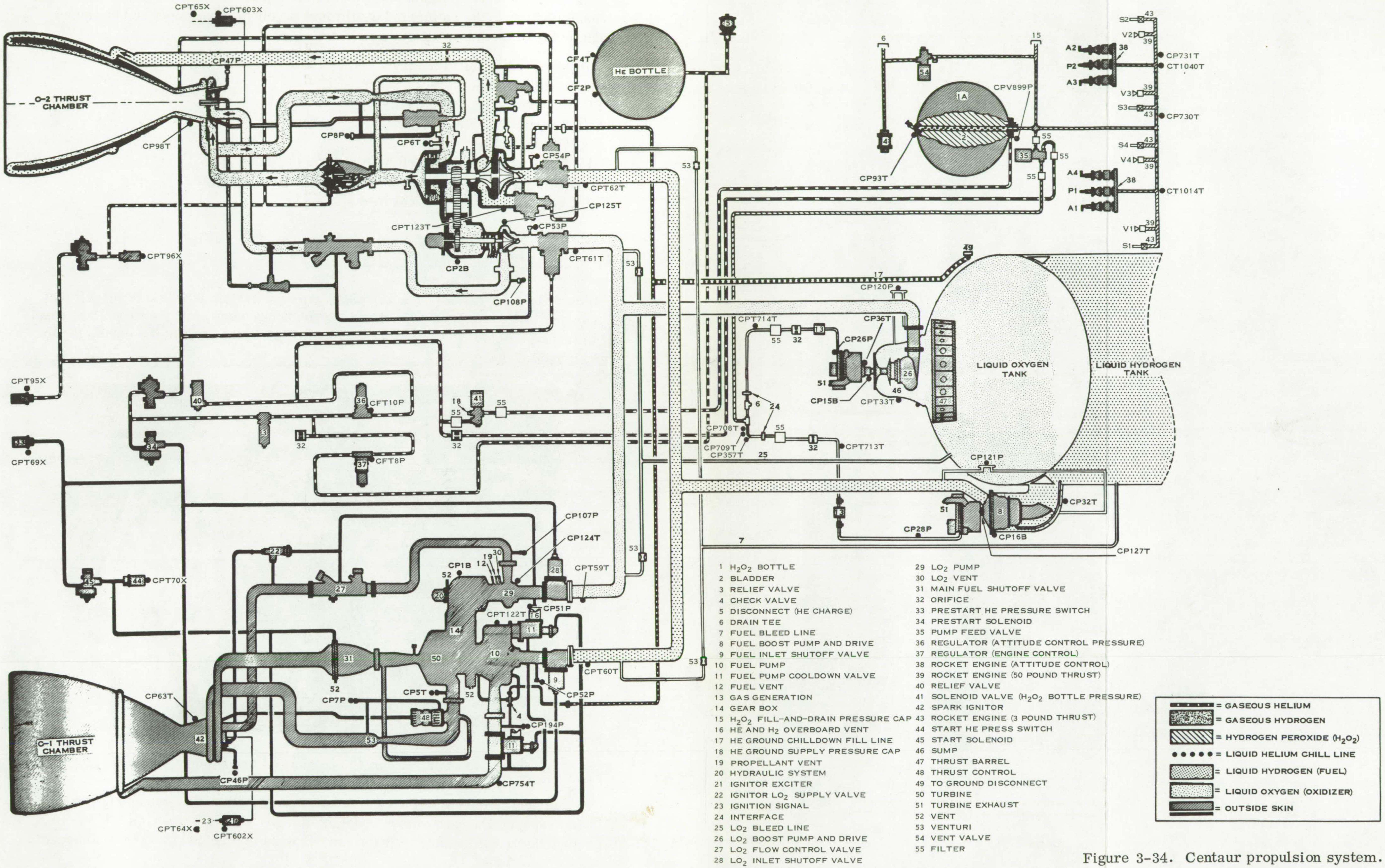


Figure 3-34. Centaur propulsion system.

Engine systems are purged with gaseous helium prior to liftoff to preclude moisture contamination. The helium enters through a vehicle disconnect and is routed through tubing to the following areas:

Thrust chamber injector faces
Engine pump seals
PU (mixture ratio) valve cavity
LH_2 feed-duct insulation
Boost pump seals
Interstage adapter liquid helium collector manifold vent
Hydraulic power package engine accessory drive interface
Turbopump, gearbox and fuel feed system

3.4.2 PROPELLANT FEED SYSTEM

Liquid Oxygen System. The liquid oxygen feed system consists of the fill and drain valve, liquid oxygen sump, liquid oxygen boost pump, and the propellant supply duct to transfer liquid oxygen from the vehicle tank to the inlet of the engine liquid oxygen pumps (Figure 3-35).

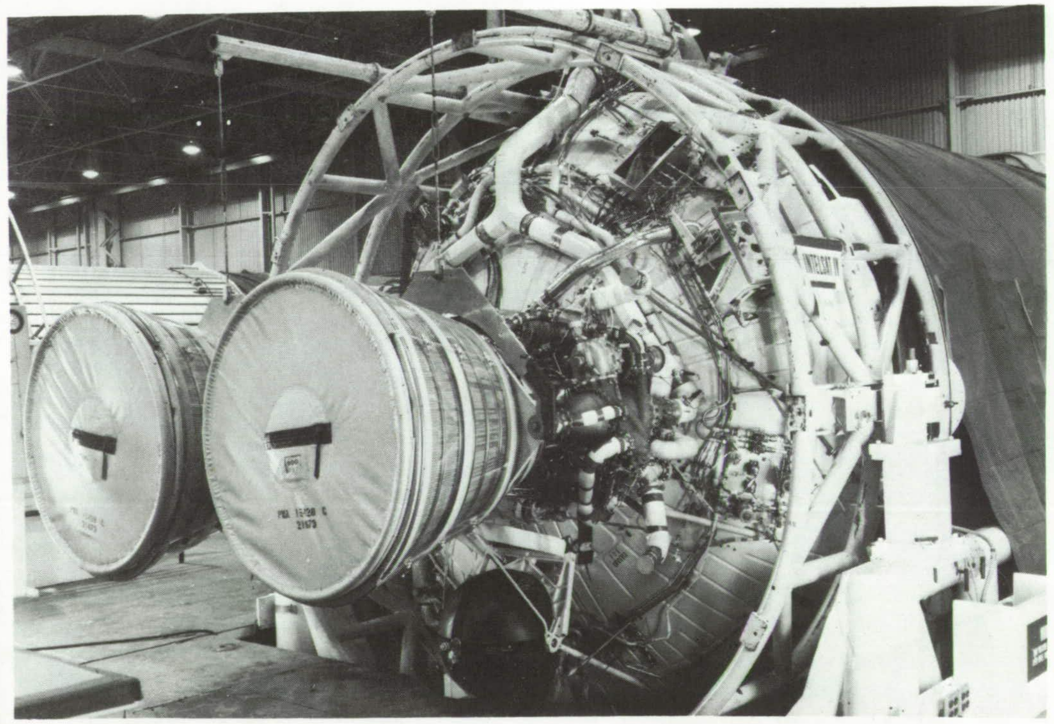


Figure 3-35. Centaur main engines and propellant feed system.

The turbine-driven liquid oxygen boost pump operates while completely submerged in liquid oxygen. The turbo-drive consists of a gas generator, turbine, gear train, and associated electrical systems. Hydrogen peroxide is directed into the decomposition chamber (gas generator) and decomposed by a silver screen catalyst bed. The products of decomposition flow through nozzles into the turbine and then are exhausted overboard. A continuous power, 40-watt, 28v d-c coil heater is assembled on the catalyst bed to heat the bed. Heating the bed improves the starting characteristics of the gas generator.

The liquid oxygen boost pump turbodrive is controlled by fixed metering orifices to maintain constant hydrogen peroxide flow to the catalyst bed. The liquid oxygen boost pump turbodrive gear box speed-reduction ratio is 9.1:1. The gear box is grease lubricated. An illustration of the oxidizer boost pump system is shown in Figure 3-36.

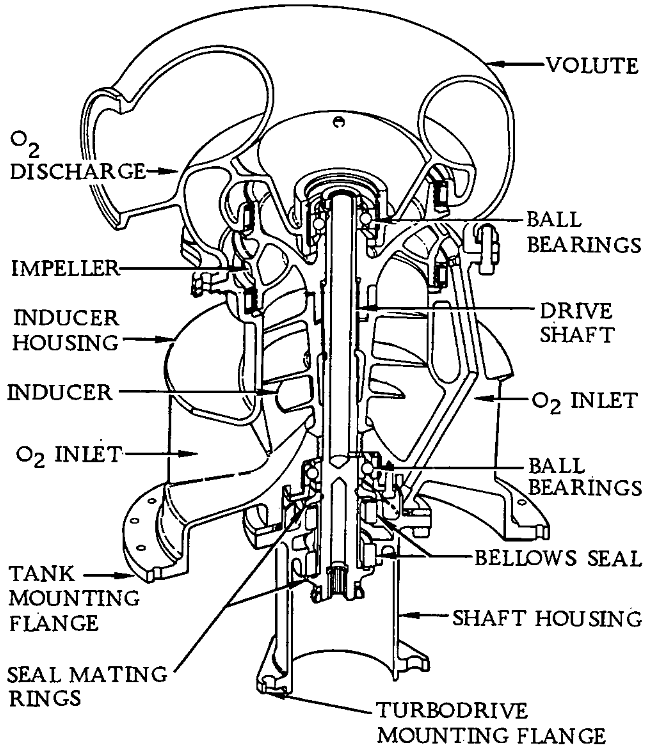


Figure 3-36. Oxidizer boost pump cutaway.

Liquid oxygen is transferred from the boost pump to the engines through 2.5-inch diameter stainless steel propellant ducts. These ducts contain gimbal bellow joints to allow engine gimbaling movement. The exterior surface of the ducts is covered with foam insulation to minimize conduction heat transfer to the liquid oxygen while the vehicle is on the launch pad. The exterior surface of the foam is provided with radiation heat transfer control to minimize the heat transferred to the liquid oxygen in flight.

Bleed lines attached to each engine propellant duct bleed gas from the gas trap in the duct, and aid in propellant duct chilldown and boost pump acceleration by recirculation flow back into the LO_2 tank.

Liquid Hydrogen System. The turbine-driven centrifugal liquid hydrogen boost pump assembly operates while completely submerged in liquid hydrogen. A diagram of the LH_2 boost pump system is shown in Figure 3-37. The inducer bell is vaned to prevent fluid pre-rotation.

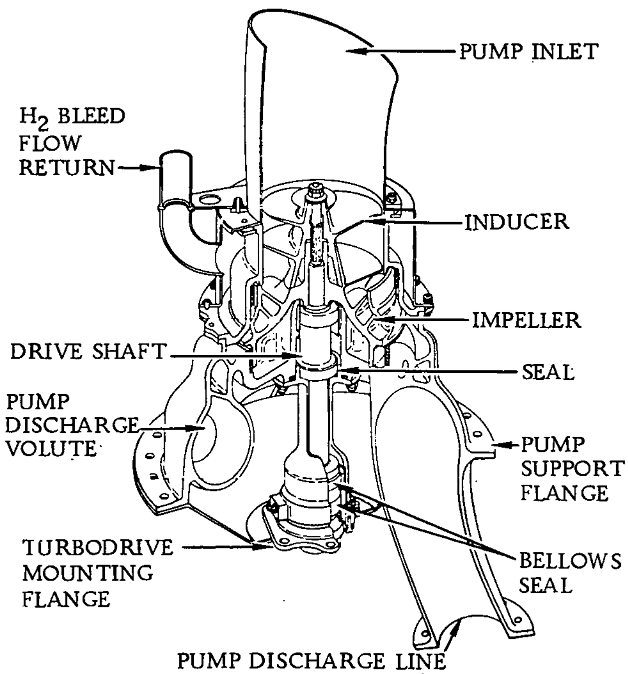


Figure 3-37. Fuel boost pump cutaway.

The turbodrive external assembly and the turbodrive controls are similar in design and operation to those of the liquid oxygen pump assembly. Speed reduction from turbine to pump is accomplished by a grease-lubricated gear box with a reduction ratio of 5.97:1.

Liquid hydrogen is transferred from the boost pump to the engines through propellant lines identical in design to the liquid oxygen duct except for a different line venting. A hydrogen bleed line serves the same function as the corresponding oxygen bleed line.

3.4.3 ENGINE CHILLDOWN SYSTEM. The propellant turbopumps must be chilled and primed before each engine operation to prevent excessive heating of the propellants (and resulting in vaporization during pump operation transients). Main propellants are used in flight to chill the turbopumps to their operating temperatures by dumping LH_2 and LO_2 through the engine. To reduce the amount of flight propellant loss for this chilldown, the turbopumps are prechilled prior to launch by pressure-feeding 1.5 to 2.5 lb/min. of liquid helium (LHe) from a ground source. The LHe enters the system through the prelaunch chilldown check valve on the fuel turbopump at a temperature below -390°F. The ground chilldown requires maintenance of fuel turbopump housing temperatures at or below -310°F for at least 15 minutes prior to launch (until approximately 8 seconds before liftoff) .

The vehicleborne portion of the ground helium chilldown system consists of transfer lines, supply tubes, connecting fittings, vent ducts, collector manifold, vent fin, and helium chilldown and staging-disconnect fittings. LHe is supplied from a pressurized dewar on the ground to an airborne disconnect, where it is routed into the fuel turbopump on each engine via a prelaunch chilldown check valve. Helium enters the inlet to the LH_2 turborpumps. Ventage flows out the gearbox vent and the chilldown valves through staging ducts to the interstage adapter collector manifold, where it is exhausted to atmosphere.

Inflight chilldown of the fuel system is accomplished following Atlas/Centaur separation during the prestart cycle prior to each engine ignition. The cycle is initiated (eight seconds for first burn and seventeen seconds for second burn) prior to main engine start by actuating the prestart solenoid valve to the open position, allowing pneumatic pressure to open the fuel inlet shutoff valve. Fuel forced by the boost pump, passes through the first stage of the turbopump and into the turbopump interstage chilldown valve where a portion of fuel is vented overboard through vent ducts on the vehicle interstage adapter. The remaining fuel passes through the second stage of the turbopump and into the discharge chilldown valve where the remaining fuel is vented overboard. Flow through the thrust chamber is precluded by the closed main fuel shutoff valve. Simultaneous with turbopump chilldown, a small portion of the fuel flows directly to the turbopump bearings and gearbox where it cools and acts as a lubricant.

Inflight chilldown of the oxidizer feed system is accomplished concurrently with fuel chilldown. During this period, LO_2 from the oxidizer boost pump flows to the thrust chamber through the oxidizer inlet valve, turbopump, oxidizer flow control valve, and injector.

3.4.4 ENGINE OPERATION. When prestart (chilldown) is concluded, the start cycle is initiated by actuating the start solenoid to the open position, allowing helium (pneumatic) pressure to partially close the turbopump interstage chilldown valve, completely close the turbopump discharge chilldown valve, and open the main fuel shutoff valve. As fuel begins to flow, expansion of gas across the turbine stages causes rotation which (through a gearbox) drives the fuel and oxidizer pumps. Increasing fuel pump discharge pressure is fed back into the actuator of the turbopump interstage chilldown valve, causing it to close completely. Throughout engine operation, the turbopump gearbox is cooled and lubricated by fuel. Gaseous hydrogen flows from the turbine through the shutoff valve to the injector. The oxidizer passes through the oxidizer flow control valve which maintains the desired mixture ratio.

Each propellant is injected into the thrust chamber where combustion is initiated by an electrical spark igniter. Heat of combustion vaporizes the fuel as it passes through the thrust-chamber tubes. This expansion of gas is the energy source that operates the turbine.

Part of the gaseous fuel bypasses the turbine by flowing through a thrust-control valve assembly. This valve provides a means to operate the engine at a constant thrust level; as it senses thrust chamber pressure it varies the amount of fuel allowed to pass through the turbine. The amount of fuel passing through the turbine controls the speed (power) which, in turn, varies the speed of the propellant pumps and, therefore, propellant flow.

3.5 CENTAUR D-1A REACTION CONTROL SYSTEM

ELEMENTS:	Subsection
• Three Systems of Small Rocket Engines	3.5.1
• Hydrogen Peroxide (H_2O_2) Monopropellant Supply System	3.5.2

FUNCTIONS:

- Provides Thrust for Vehicle Pitch, Yaw, and Roll Control
- Thrusts to Settle Propellants during Coast
- Make Post-Injection Separation and Orientation Maneuvers

3.5.1 REACTION CONTROL ENGINE SYSTEMS.

The systems consists of:

1. Four 3-pound-thrust propellant settling engines (S1 through S4).
2. Two thrust clusters, each containing:
   a. One 6-pound-thrust pitch engine (P1 or P2)
   b. Two 3.5-pound-thrust yaw/roll engines (A1, A4 or A2, A3)
3. Four 50-pound-thrust vernier engines (V1 through V4).

The four settling engines provide thrust for simultaneous pitch/yaw control and propellant settling during the coast period, and for pitch/yaw control and retrothrust during the latter portion of the retromaneuver. The two engine clusters provide thrust for roll control during the settled coast period and retromaneuver, and for pitch, yaw, and roll control during vehicle precision pointing prior to payload separation and during reorientation after payload separation. The four vernier engines provide a higher level thrust for simultaneous pitch/yaw control and propellant settling following first main engine cutoff and prior to second main engine start, and for pitch/yaw control and retrothrust during the early portion of the retromaneuver.

During the remaining reaction control system activation periods, the vernier engines will provide thrust for pitch and yaw control if vehicle disturbances become too large for the smaller thrust engines to control.

For missions where their function is not needed, the vernier and/or settling engines can be readily removed.

The fixed axis, constant thrust, hydrogen peroxide (H_2O_2) engines are attached to the Centaur aft bulkhead. One vernier and one settling engine each are mounted on the 45-, 135-, 225-, and 315-degree axes, with their thrust vectors parallel to the vehicle longitudinal axis. The engine clusters are mounted on the 0-/180-degree axis with their thrust vectors in a plane perpendicular to the longitudinal axis. The cluster pitch engine thrust vectors are aligned parallel to the 0-/180-degree axis, and the cluster yaw/roll engine thrust vectors are aligned 25 degrees outboard from the 90-/270-degree axis (Figure 3-38). Installation for one-burn missions is mission peculiar and is shown in Figure 3-39.

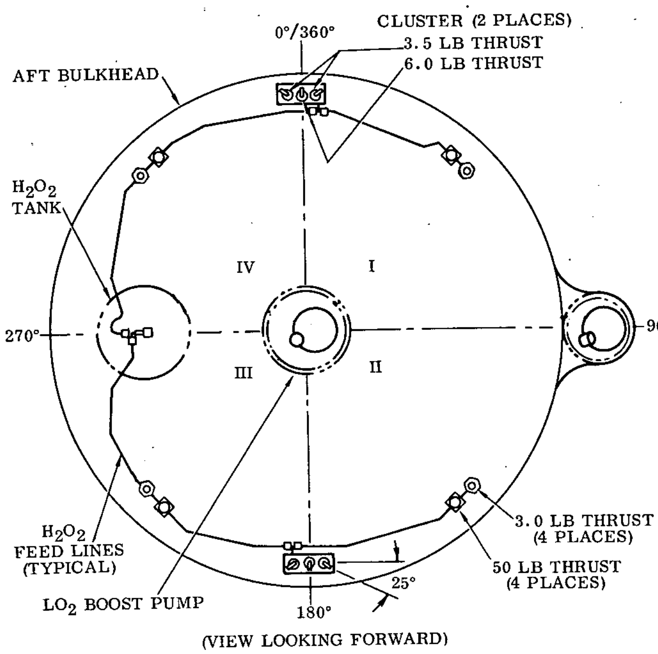


Figure 3-38. Hydrogen peroxide engine arrangement - two burn.

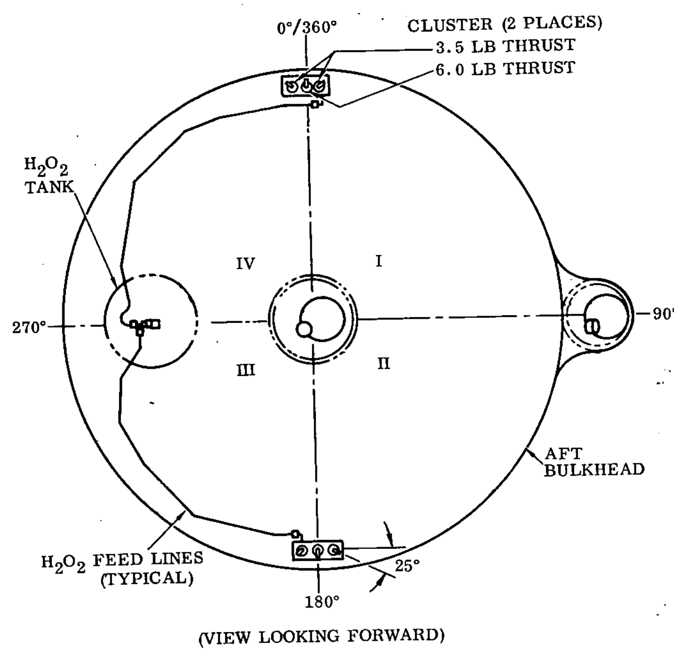


Figure 3-39. Hydrogen peroxide engine arrangement - one burn.

3.5.2 <u>HYDROGEN PEROXIDE MONOPROPELLANT SUPPLY SYSTEM.</u> The hydrogen peroxide monopropellant supply system (Figure 3-40) feeds the boost pump turbodrives and the reaction control system. The system consists of: a positive expulsion propellant tank, pressurized from an airborne helium supply; tubing for transporting the hydrogen peroxide to each boost pump turbodrive and the reaction control supply system; a relief system containing a two-way, two-position solenoid vent valve; a ground controlled three-way, two-position solenoid valve for tank pressurization; and a three-way, two-position boost pump feed valve. All feed system metallic parts contacting the H_2O_2 are manufactured of 300 series stainless steel. Stainless steel encased constant-power line heaters, silver brazed to the hydrogen peroxide supply line sections, maintain the hydrogen peroxide temperature between 40°F and 120°F.

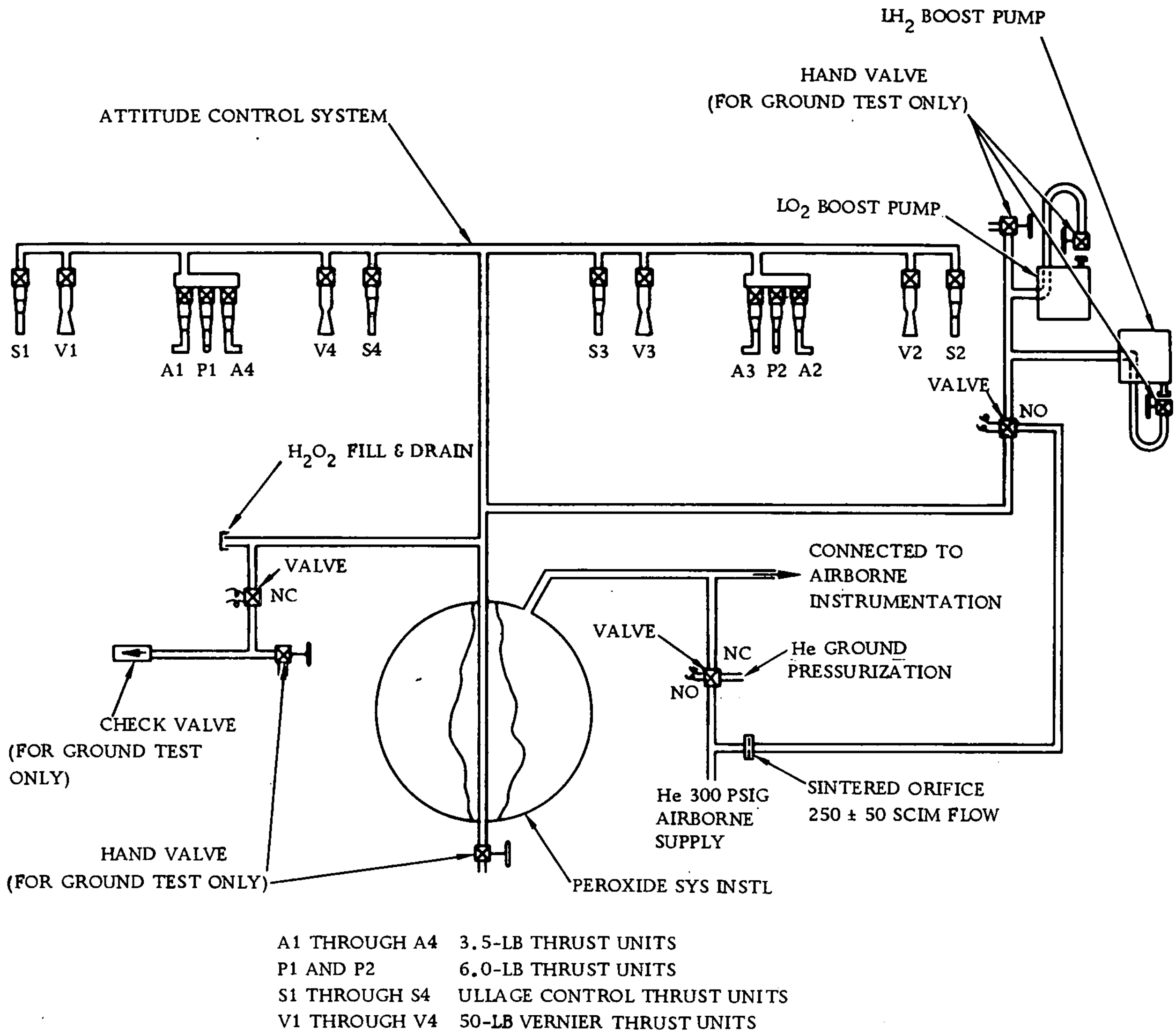


Figure 3-40. Hydrogen peroxide propellant supply system.

The positive expulsion hydrogen peroxide storage tank is pressurized to a nominal 306 psia. The hydrogen peroxide is separated from the pressurizing gas by a silicone rubber bladder. The nominal usable capacity of the propellant tank is 4870 cubic inches. The unusable residual volume of the reaction control and boost pump feed systems is less than 87 cubic inches.

A three-way, two-position solenoid valve, when energized, permits hydrogen peroxide to flow from the propellant storage tank to the boost pump turbodrives. A similar valve is used to pressurize the hydrogen peroxide propellant supply tank; pressurization occurs when the valve is deenergized. A two-way, two-position valve serves as a system vent valve and emergency dump valve.

3.5.3 OPERATION. Each thrust engine has an integral solenoid-actuated propellant feed valve operated by 28v d-c signals from the Centaur sequence control unit. Energizing the solenoid opens the valve, permitting flow of the monopropellant to the engine decomposition chamber.

3.6 CENTAUR HYDRAULIC SYSTEM

ELEMENTS:	Subsection
• Hydraulic Power Package (2)	3.6.1
• Servocontrolled Engine Gimbaling Actuator Assemblies (4)	3.6.2
• Manifold, Insulation Block, and Thermostat	3.6.3

FUNCTIONS:
• Provides Force to Gimbal Centaur Main Engines

The hydraulic system provides the mechanical force required to gimbal the Centaur main engines. A separate hydraulic system for each main engine gimbals the engines during ground checkout, prior to engine ignition after Atlas separation, and during powered flight. Each system consists of an integrated hydraulic power package, two servocontrolled engine gimbaling actuator assemblies, a manifold, and miscellaneous fittings and connecting tubing. The system operates at nominal pressures of 100 and 1000 psig under ambient temperatures ranging from -30°F to +275°F. System weight, excluding mounting provisions, is 75 pounds (for both engines). Figures 3-41 and 3-42 illustrate the hydraulic system orientation and schematic.

3.6.1 HYDRAULIC POWER PACKAGE. An integrated hydraulic power package provides power for the hydraulic system. The power package consists of a high-pressure, engine-driven pump assembly; an electric motor; a low-pressure, electrically-driven pump assembly; relief and check valves; filters; a tank assembly that includes an accumulator and a bootstrap reservoir; instrumentation; and the airborne half of the ground disconnects. Figure 3-43 illustrates the power package.

The high-pressure pump (Figure 3-44) is driven by the accessory output drive on the RL10A-3-3 engine on the propellant turbopump gearbox. A fiberglass resin block insulates the hydraulic power package from the cryogenic temperature of the propellant turbopump.

The low-pressure pump (Figure 3-45) is driven electrically by an electric motor (Figure 3-46) contained within the hydraulic power package. It provides hydraulic power upon demand when the high-pressure pump is not operating (prelaunch, coast, etc.) and is used to:

1. Check out the hydraulic system on the ground.

2. Equalize hydraulic system temperature by circulating hydraulic fluid when reduced temperatures are sensed by the thermostats in the manifold or actuator.

3. Acting on commands from the guidance system, pre-position engines prior to main-engine start so that the thrust vector will be aligned at main engine start.

4. Gimbal engines after payload separation so that thrust from residual propellants flowing through main engines aids in retromaneuver.

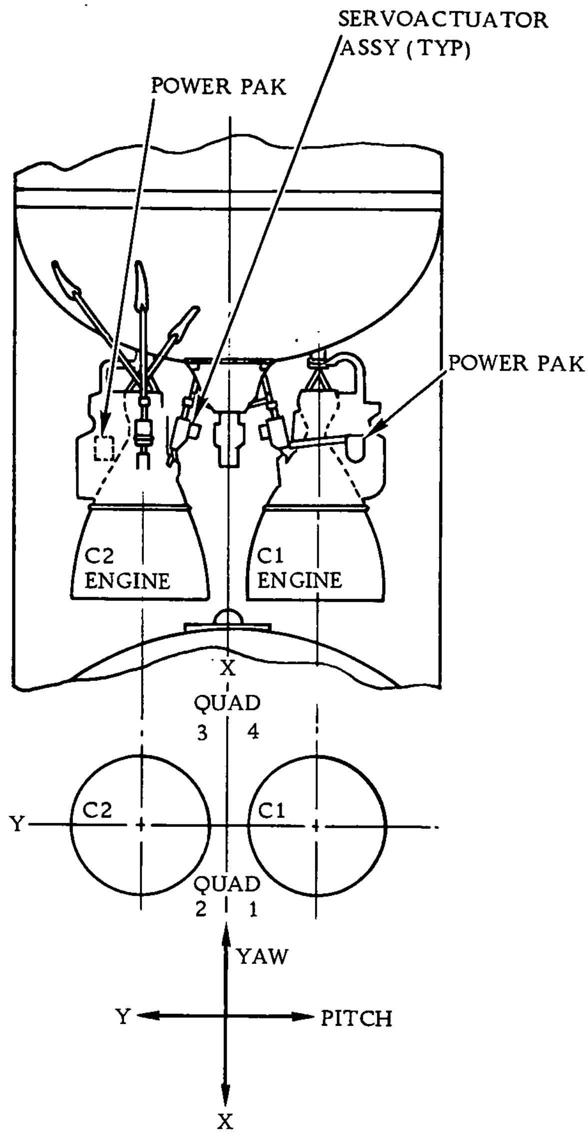


Figure 3-41. Centaur hydraulic system orientation.

The high- and low-pressure systems are functionally independent. The systems are isolated by check valves, and each has its own relief valve to control pressurization by bleeding excess flow directly into the reservoir.

Pressure from the power package is limited during the power phase of flight by the high-pressure relief valve; flow distribution is a function of pump capacity, relief valve settings, and servovalve demand. During the coast phase of flight, while the low-pressure circulating system is operating, pressure is limited by the low-pressure relief valve.

To prevent cavitation of the hydraulic pumps and overpressurization of the reservoir, pressure and volume of the reservoir are controlled by a bootstrap reservoir. The bootstrap reservoir consists of a piston assembly spring-loaded, and precharged

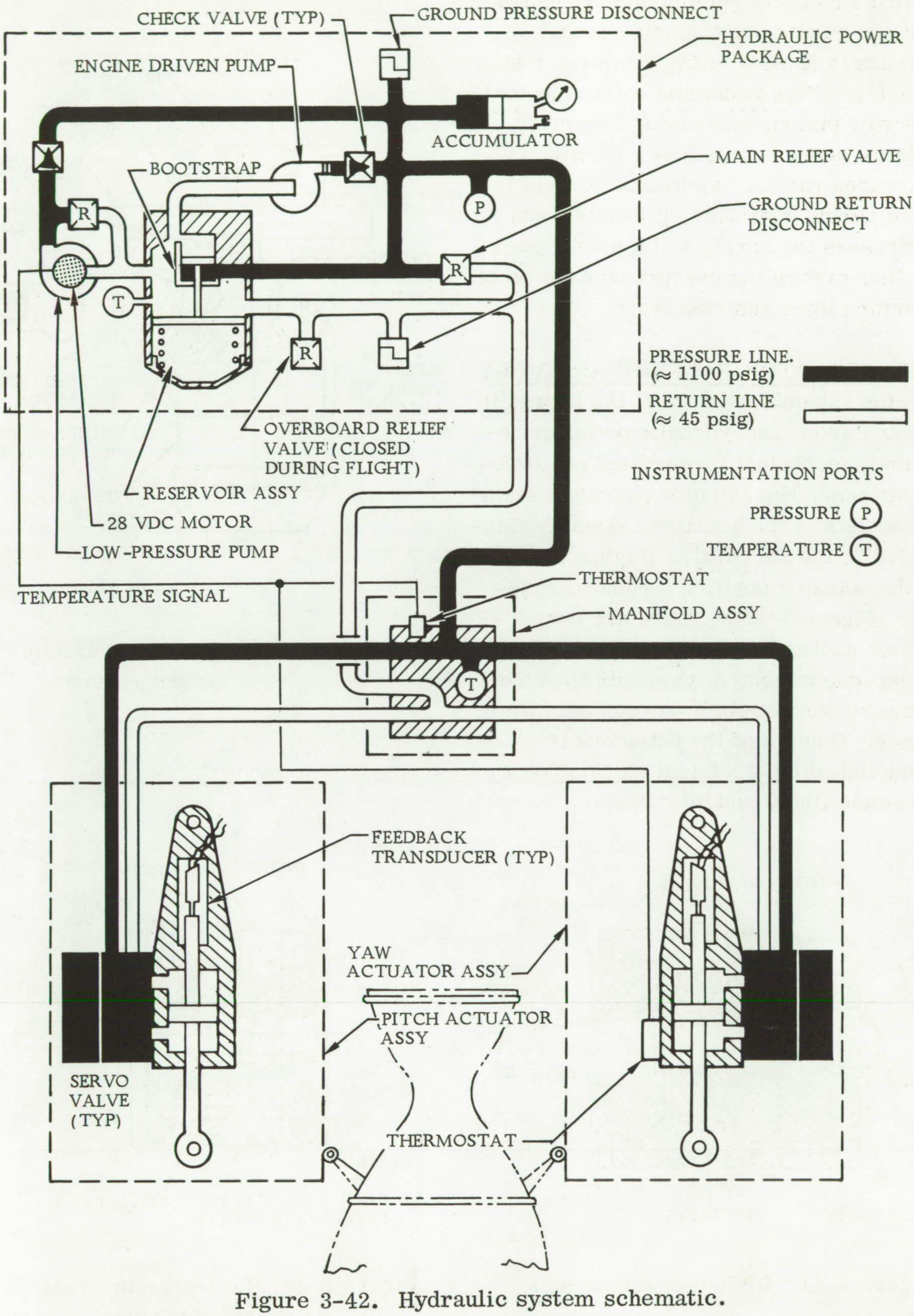


Figure 3-42. Hydraulic system schematic.

with air at one atmosphere to provide nominal reservoir volume and pressure. When the pumps start, pump-discharge pressure is applied to the bootstrap piston assembly. This transmits a force to the reservoir piston, increasing reservoir pressure above nominal as a function of piston area ratios. As hydraulic fluid is heated during operation, it expands and compresses the spring and the entrapped air, thus preventing overpressurization of the return lines and reservoir.

3.6.2 ACTUATOR PACKAGE ASSEMBLY. Actuator assemblies convert the hydraulic pressure from the hydraulic power package into mechanical force which positions the engines. The actuator package assembly consists of the actuator assembly (Figure 3-47), the servovalve (Figure 3-48) and the actuator position feedback transducer (Figure 3-47). There are two actuator package assemblies on each main engine: one to control yaw/roll movement, and one to control pitch movement of the engines. Control of the actuators is accomplished by electrical signals from the vehicle flight control system.

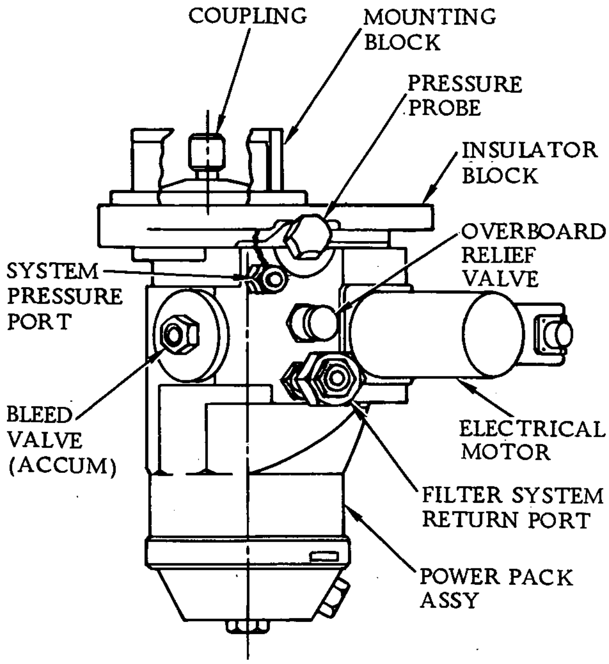


Figure 3-43. Centaur hydraulic power package.

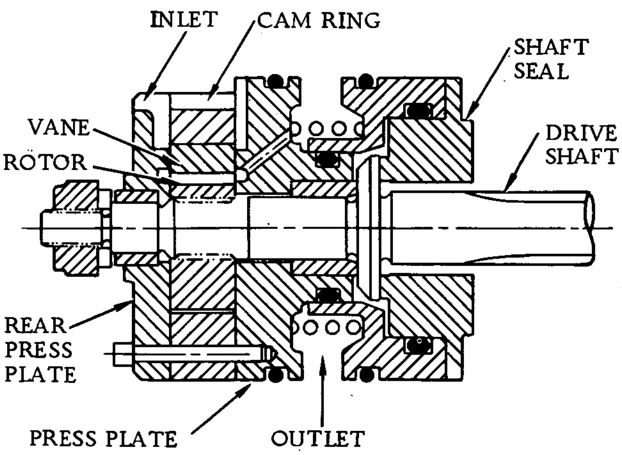


Figure 3-44. High-pressure pump.

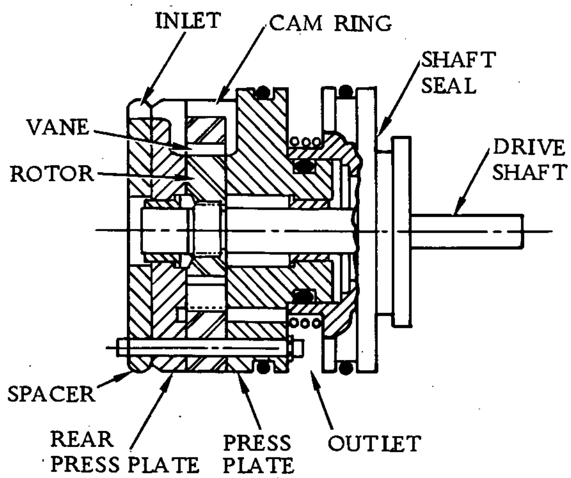


Figure 3-45. Low-pressure recirculating pump.

Thermal insulation is provided to limit heat transfer from the hydraulic fluid to the actuator mounting clevis. A hermetically sealed thermostat for sensing actuator temperatures is provided for each yaw actuator.

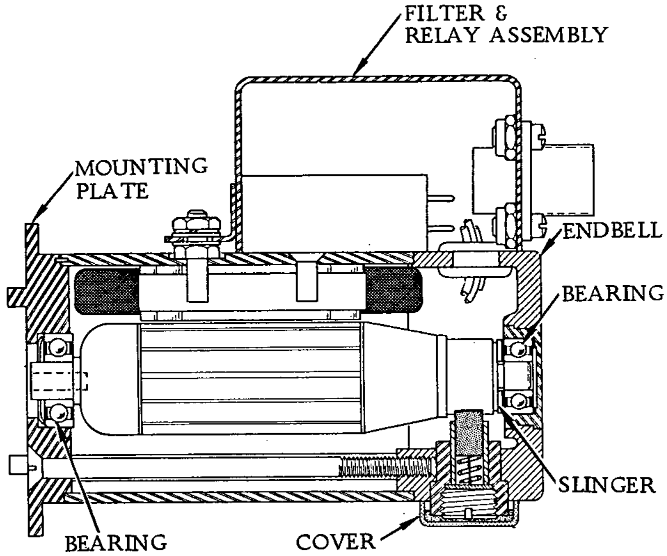


Figure 3-46. Electric motor.

The servovalve, a derivative-pressure-feedback device, is mounted externally on the actuator body. It controls the flow rate from the hydraulic pump to the actuators in response to a signal from the flight control servoamplifier unit. The servovalve consists of the following major components: a torque motor, first stage amplifier, a second stage slide valve, integral filters, and a derivative-pressure-feedback network.

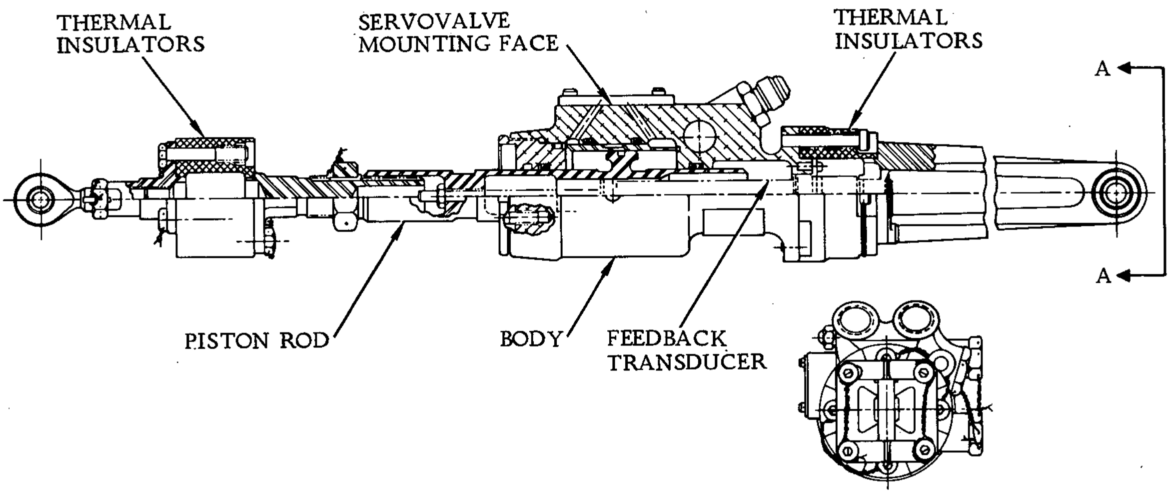


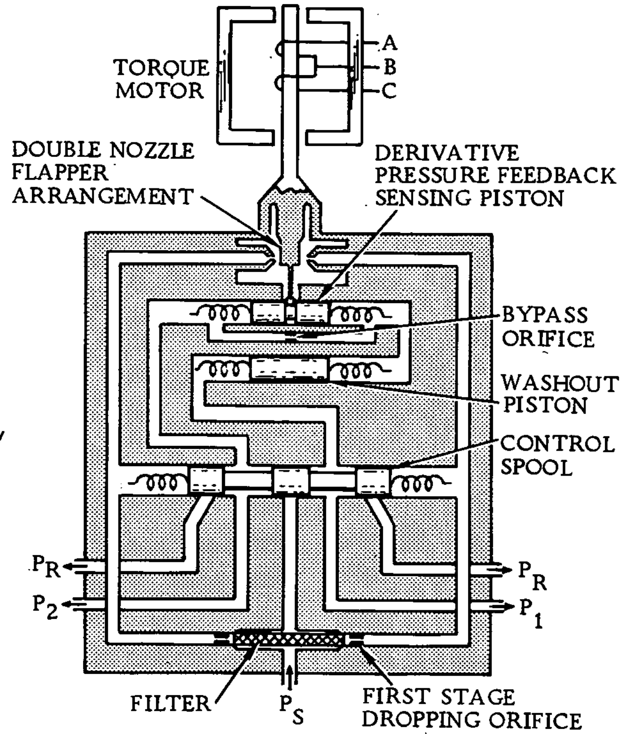
Figure 3-47. Hydraulic actuator cross section.

The feedback transducer is a position indicator that is mounted internally on the actuator piston rod. The transducer consists of a main body, a movable probe, and an electrical connector.

The servoactuator stroke of ±0.75 inches limits the engine gimbal angle to ±3.15 degrees from the neutral position. The gimbal rate requirement (minimum) for the system is 7 deg/sec pitch and 4 deg/sec yaw. The force output of the actuator varies as a function of load velocity, temperature, etc., but nominally will exert 750 pounds at 6 degrees per second gimbal velocity. Maximum stall force is 1678 pounds. Static gain relationships for the servosystem components in the actuator assembly are as follows:

1. Actuator: 2.62 deg/sec/in.^{3}/sec
2. Servovalve: 0.385 in.^{3}/sec/ma when external load = 1000 pounds
3. Feedback Transducer: 2.04 volts/deg

3.6.3 MANIFOLD, INSULATION BLOCK, AND THERMOSTAT. A manifold connects the pressure and return lines between the hydraulic power package and the pitch and yaw actuators. Provisions are made on the manifold for mounting a thermostat and an instrumentation temperature transducer. The manifold is designed for an operating pressure of 1200 psig. Mounts are provided for attaching the manifold to a bracket on the engine(s) equidistant from the pitch and yaw actuators.



P_S - SUPPLY PRESSURE P_1 - CYLINDER 1

P_R - RETURN PRESSURE P_2 - CYLINDER 2

Figure 3-48. Servovalve schematic diagram.

An insulation block insulates the hydraulic power package from the engine accessory drive pad mounting; the temperature of the drive can approach LH_2 temperatures (approximately -420°F). The cast fiberglass-epoxy block is mounted between the power pack body and the mounting block that mates with the engine accessory drive pad mounting holes.

Thermostats are provided on the manifolds and yaw actuators. When temperatures reach 10°F, signals from the thermostats activate a relay that starts the electrically-driven circulation pump. When temperatures increase to 30°F, the thermostats deactivate the relay.

The operating parameters of major components of the hydraulic system are given in Table 3-1. Table 3-2 identifies the system cleanliness requirements.

Table 3-1. Operating parameters of major hydraulic system components.

Major Hydraulic Pump (Main Engine Driven):	
Flow (minimum):	1.58 gallons per minute (gpm)
Pressure:	975 psi differential (psid)
Speed:	12,400 rpm
Type:	Fixed volume displacement, vane
Low Pressure Hydraulic Pump (Electric Motor Driven):	
Flow:	0.2 gpm
Pressure:	100 psid
Speed:	6000 rpm
Type:	Fixed volume displacement, vane
Electric Motor:	
Voltage:	28v d-c
Speed:	6000 rpm
Current:	3 amps
High-Pressure Hydraulic Relief Valve:	
Full Flow Pressure:	1100 psid at 2.0 gpm
Reseat Pressure:	980 psid
Low-Pressure Hydraulic Relief Valve:	
Full Flow Pressure:	110 psid at 0.3 gpm
Reseat Pressure:	100 psid
Hydraulic Accumulator:	
Gas Volume:	1.35 cubic inches
Precharge:	600 psig at 70°F
Return System Filter:	
Micron Rating:	40 microns (98%) 100 microns (100%)
Thermostats:	
Activated:	10 ± 4°F
Deactivated:	30 ± 4°F
Hydraulic Fluid:	
MIL-H-5606 Fluid, Red	

Table 3-2. Hydraulic system cleanliness requirements.

Size (microns)	Number of Allowable Particles
26 - 50	1000
51 - 100	70
101 - 200	12
201 - 500	3
501 - 1000	0
1000	0
	Fiber Count
100 - 1000	13
1001 - 2000	0
2000	0
Water Content	0.010% max.
Flash Point	200°F
Viscosity at 130°F	-12 cs

3.7 CENTAUR D-1A PNEUMATIC SYSTEM

ELEMENTS:	Subsection
• Propellant Tank Pressurization System	3.7.1
• H_2O_2 Bottle and Engine Controls Pressurization System	3.7.2
• Purge System	3.7.3
• Intermediate Bulkhead Vacuum System	3.7.4

FUNCTIONS:

- Provide Gaseous Helium for Propellant Tank, Engine Controls, and H_2O_2 System Pressurization
- Provide He Purging Gas to Prevent Moisture From Entering Cryogenic Systems and Causing Icing
- Prevent Inflow of Air to Intermediate Bulkhead to Ensure Desired Vacuum is Achieved

3.7.1 TANK PRESSURIZATION SYSTEM. The tank pressurization system (Figure 3-49) establishes and maintains the propellant tanks at their proper pressures prior to engine start. The system consists of one helium storage bottle, two solenoid valves (tank pressurization valves), two check valves, flow restrictors, inlet filters, a pressure switch (for LO_2 tank first-burn pressurization), and associated tubing. During the countdown, helium for pressurization is stored in a 7365-cubic-inch titanium bottle at about 3300 psig, at ambient temperature. Prior to each main engine start, this helium is used to increase the pressure in both propellant tanks so that the boost pumps will not cavitate during the engine start sequence.

For all engine prestart pressurizations, tank pressures are increased by opening a solenoid valve which has a control orifice downstream. The flight control system operates the valve, which stays open for a preset time and then is closed. The exception is the first LO_2 tank pressurization sequence, at which time the flight control system enables an LO_2 tank pressure switch. The switch cycles the LO_2 tank pressurization solenoid valve open and closed as required to maintain the tank pressure between 38 and 40 psia until first main engine start.

3.7.2 H_2O_2 BOTTLE AND ENGINE CONTROLS PRESSURIZATION SYSTEM. The H_2O_2 bottle and engine controls pressurization system (Figure 3-49) consists of high- and low-pressure regulators, relief valves, filters, checkout fittings, and associated tubing. This system is supplied by the airborne storage bottle, and

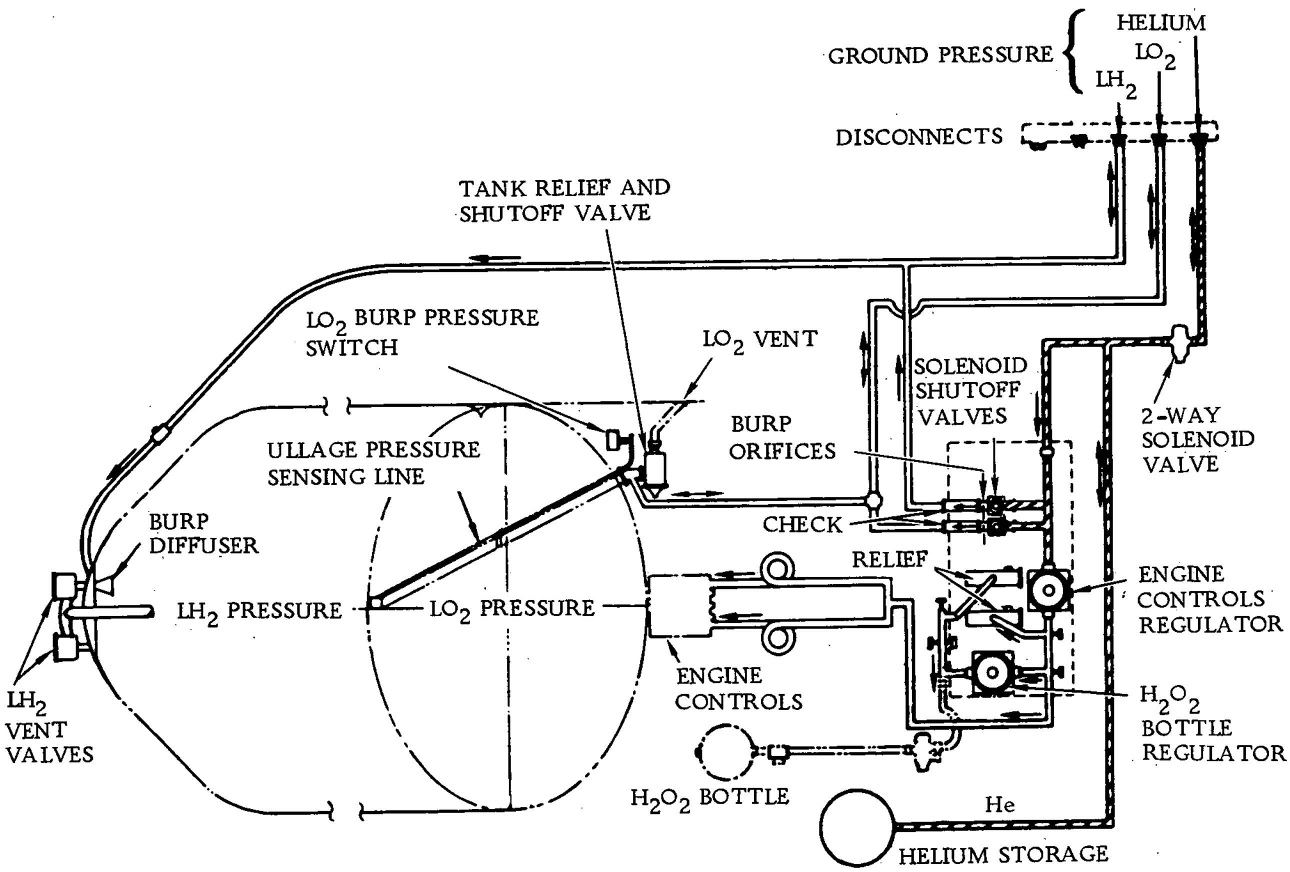


Figure 3-49. Pressurization for Centaur D-1A tank, H_2O_2 bottle, and engine controls.

provides regulated helium pressure to the H_2O_2 bottle and engine controls system. The engine controls system is a one-stage pressure reduction system that uses the high-pressure regulator and its associated relief valve to operate at approximately 450 psig. The engine controls regulator provides pressure to the engine start and prestart valves in the engine control system.

The H_2O_2 bottle pressurization system is a two-stage pressure reduction system in which the main engine controls system is the first stage. The low pressure regulator and its associated relief valve is the second stage. With operating pressure of about 300 psig, the system provides helium pressure to force H_2O_2 from its supply bottle to the boost pumps and H_2O_2 engines. A bleed orifice downstream of the H_2O_2 pressurization regulator purges the H_2O_2 lines to the boost pumps with gaseous helium when they are not operating. This eliminates residual H_2O_2 and prevents freeze up during long coast periods.

3.7.3 PURGE SYSTEM. (Figure 3-50) Purging of various Centaur systems is done during ground operations and during atmospheric flight. Purging prevents moisture from entering the cryogenic systems and causing icing. Systems purged include some external portions of the hydraulic system, the servopositioner, oxidizer pump seal cavity, engine injector, fuel-duct insulation, boost pump seals, the space between fuel tank wall and insulation panels, the destructor pod, the helium childown vent ducts, and the space between the forward tank bulkhead and the equipment module. During

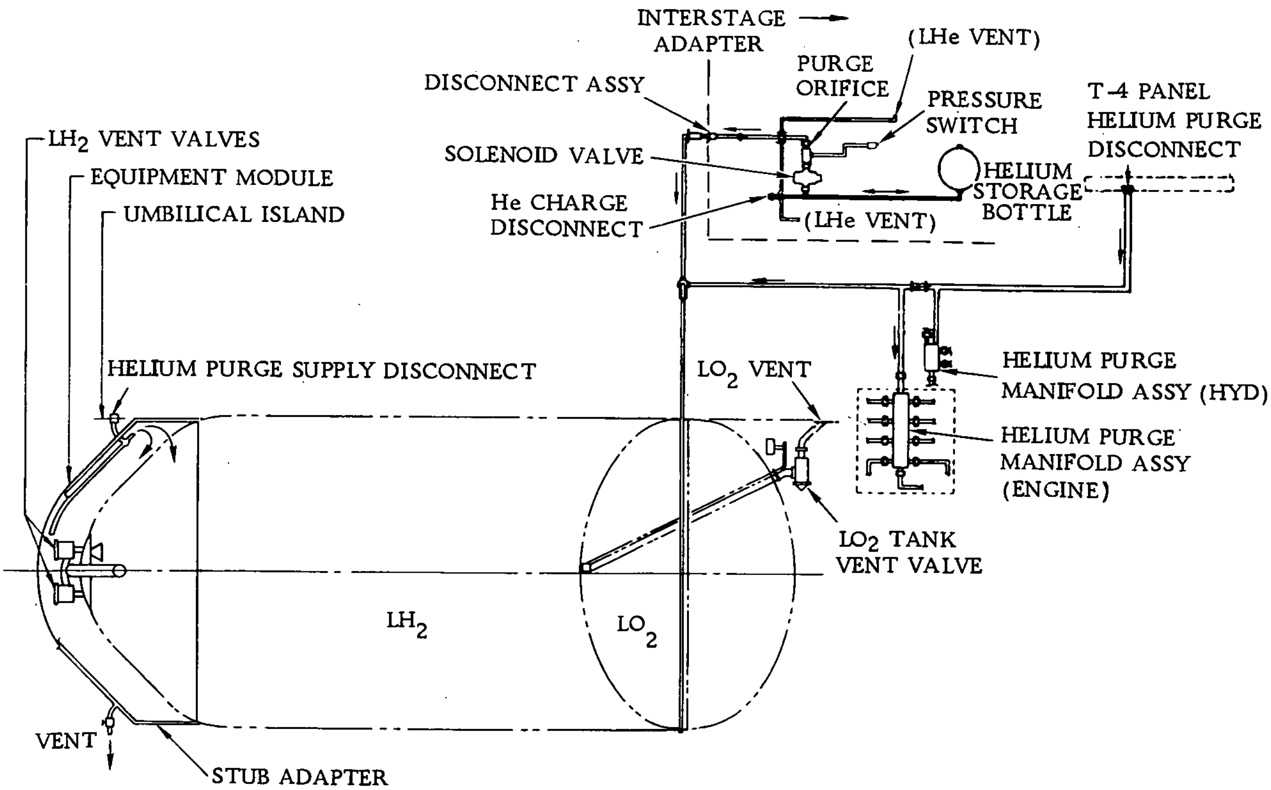


Figure 3-50. Centaur D-1A purge systems.

the countdown, helium purge gas is supplied to the vehicle through disconnects in the T-4 panel and the forward umbilical panel. A 4650-cubic-inch titanium bottle mounted in the interstage adapter is charged with helium to 2400 psig during the countdown. Just prior to liftoff and during the portion of booster flight within the atmosphere, the storage bottle discharges through a solenoid valve and control orifices to provide helium purge to various engine areas and beneath the insulation panels.

3.7.4 INTERMEDIATE BULKHEAD VACUUM SHELL. The intermediate bulkhead vacuum system, by preventing the inflow of air, ensures that gaseous nitrogen in the intermediate bulkhead cavity cryopumps during tanking to the desired vacuum (< 0.05 torr). The system (Figure 3-51) consists of a check valve, pressure transducers, and tubing connecting the intermediate bulkhead cavity and the aft umbilical panel. If the intermediate bulkhead leaks during a quad tanking and the desired vacuum cannot be attained, the airborne system, in conjunction with a ground vacuum pump, prevents a pressure buildup within the cavity.

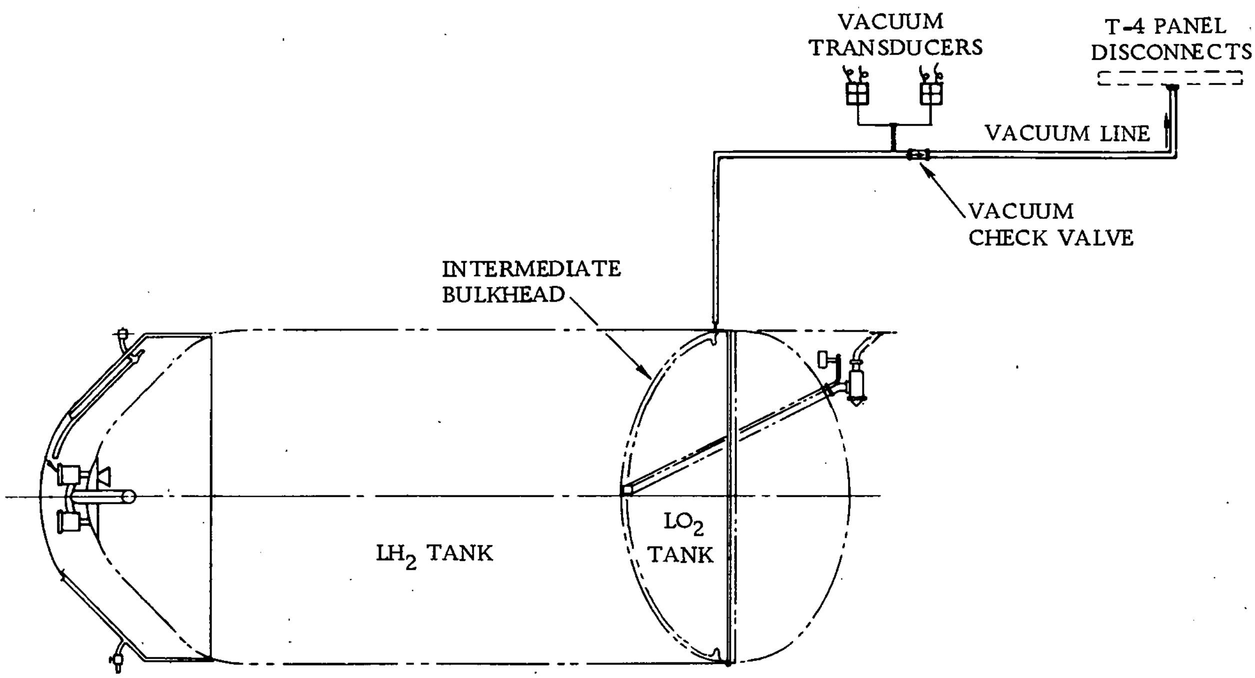


Figure 3-51. Centaur D-1A bulkhead vacuum system.

3.8 CENTAUR D-1A MAIN TANK VENT SYSTEM

ELEMENTS:	Subsection
• LH_2 Vent System	3.8.1
• LO_2 Vent System	3.8.2

FUNCTION
• Maintain Proper Pressure in Propellant Tanks

Venting of the propellant tanks is accomplished as required to maintain proper pressures in the tanks. The vent system (Figure 3-52) consists of two solenoid-operated vent valves (LO_2 and primary LH_2 valves), one non-solenoid operated vent valve (LH_2 secondary), two inflight hydrogen vent nozzle/disconnects, one inflight oxygen vent disconnect, and associated ducting.

The LO_2 vent valve is mounted on the LO_2 tank aft bulkhead. Venting occurs through a disconnect and a duct which penetrates the interstage adapter. The LH_2 primary and secondary valves are mounted on the LH_2 tank. A common duct between the valves and nozzle vent system provides a balanced thrust during coast phase venting.

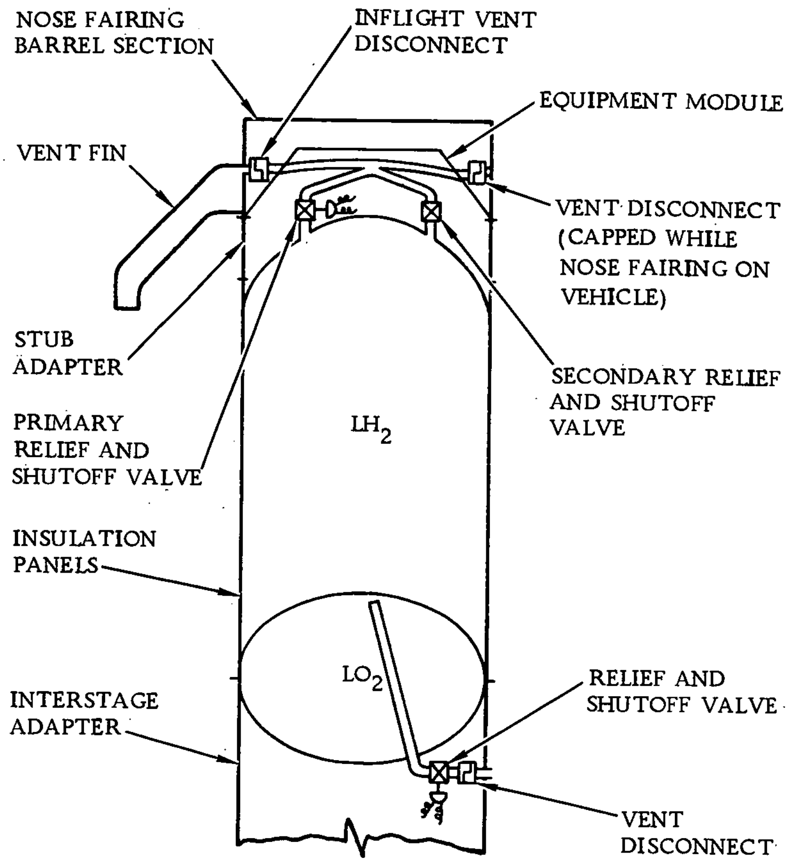


Figure 3-52. Centaur D-1A main tank vent systems.

The solenoid type relief valves are operated by tank pressure. An internal control valve is adjusted to a predetermined pressure setting, referenced to an evacuated chamber within the valve controller. Energizing of the solenoid puts the normally open vent valves in the shutoff position. The flight control system positions these valves to the proper position to maintain required propellant tank ullage pressures.

3.8.1 FUEL TANK VENT SYSTEM. LH_2 tank venting is controlled by the primary and secondary vent valves mounted on the LH_2 tank access door (Figure 3-53). The vent system consists of a plenum chamber, located between the two vent valves, with two identical outlet legs located diametrically opposite on the Y-Y axis. The vent legs penetrate the equipment module between the two circumferential fiberglass equipment support rails. LH_2 tank venting after nose fairing jettison is through balanced thrust nozzles mounted on the outboard side of the equipment module. LH_2 tank venting during ground operations and during ascent, prior to nose fairing jettison, is through the minus Y axis nozzle and split barrel mounted disconnect and ducting.

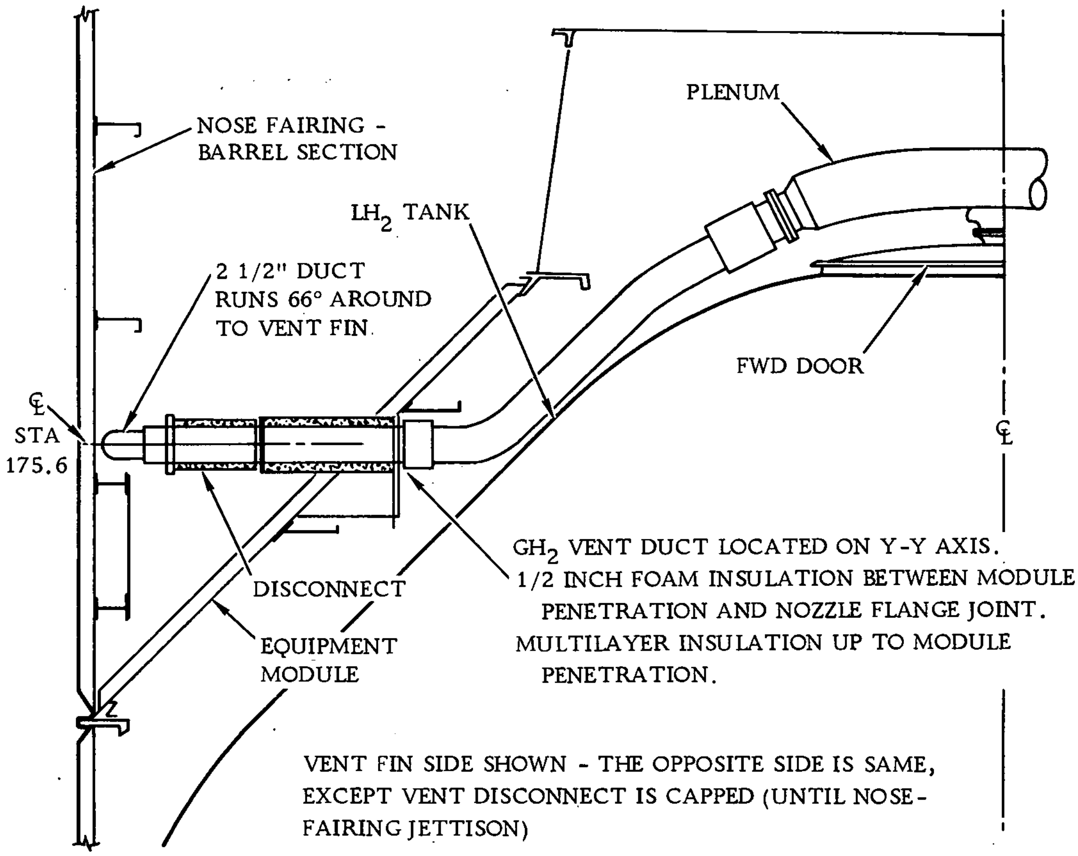


Figure 3-53. Centaur D-1A LH_2 vent system.

The split-barrel-mounted vent system connects the minus Y axis nozzle to the H_2 vent fin located in Quad 2 at 24 degrees from the X axis. The plus Y axis disconnect is identical in configuration except for a cap over the exit hole to prevent flow from the nozzle. The split-barrel mounted ducting is 2.50-inch diameter, 321 Cres with a 3.00-inch diameter omega joint section at the sheet metal transition section adjacent to the vent fin inlet flange.

A helium purge disconnect configuration is mounted adjacent to the minus Y axis H_2 vent nozzle to supply helium to the H_2 vent disconnect bellows covers. The inboard half of the disconnect is connected to the helium filled cavity under the equipment module.

The vent system ducting located in the helium environment of the forward bulkhead cavity is insulated with multilayer insulation. The plenum chamber and a portion of the two outlets is under a multilayer blanket that covers the forward door configuration. The split-barrel-mounted ducting, and the equipment-module-mounted ducting outboard of the equipment module penetration, are insulated with stafoam and Mylar tape. The bellows in the minus Y axis disconnect and the bellows located in the two plenum legs, outboard of the equipment module penetration, are covered with fiberglass bellows covers.

3.8.2 OXIDIZER TANK VENT SYSTEM. The LO_2 vent system is mounted on the standpipe flange on the aft bulkhead in Quad 1 at 30 degrees from the +X axis. The outlet flange of the vent valve points radially outboard toward the interstage adapter. A short section of 2-1/2-inch diameter aluminum alloy ducting is attached to the vent valve outlet flange and a short teflon tube section is clamped to the other end of the duct. The outlet end of the teflon tube engages an adapter section in the interstage adapter skin. This allows the gaseous oxygen to be vented overboard and provides a disconnect interface for interstage adapter separation.

4

CENTAUR D-1A ASTRIONICS SYSTEMS

This section presents descriptions of the following Centaur D-1A astrionics systems.

The airborne software contained in the digital computer unit is an integral part of the astrionics systems. Software is discussed in Section 5.

The forward astrionics packages for Centaur D-1A are mounted on the equipment module. In Figure 4-1 the inertial reference unit is on the left, the sequence control unit on the right. Layout of all packages is shown in Figure 4-2.

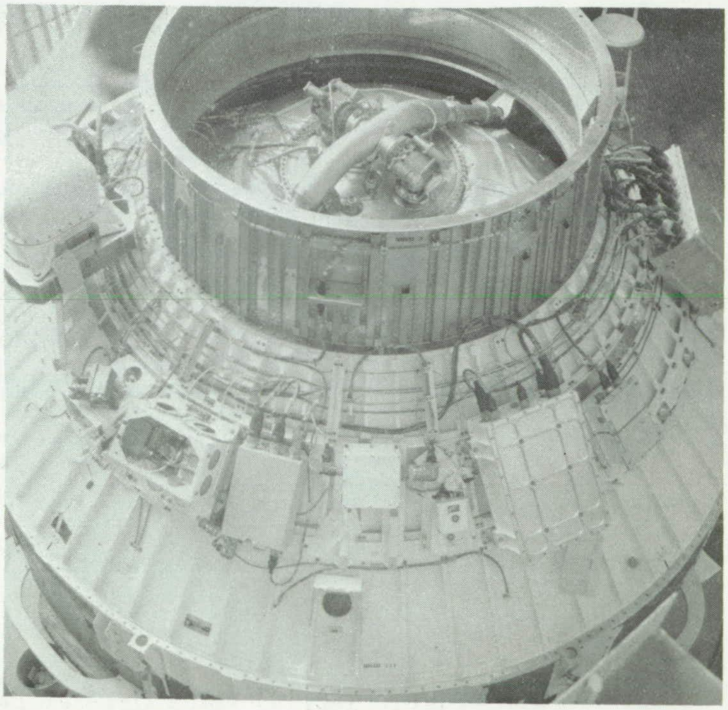
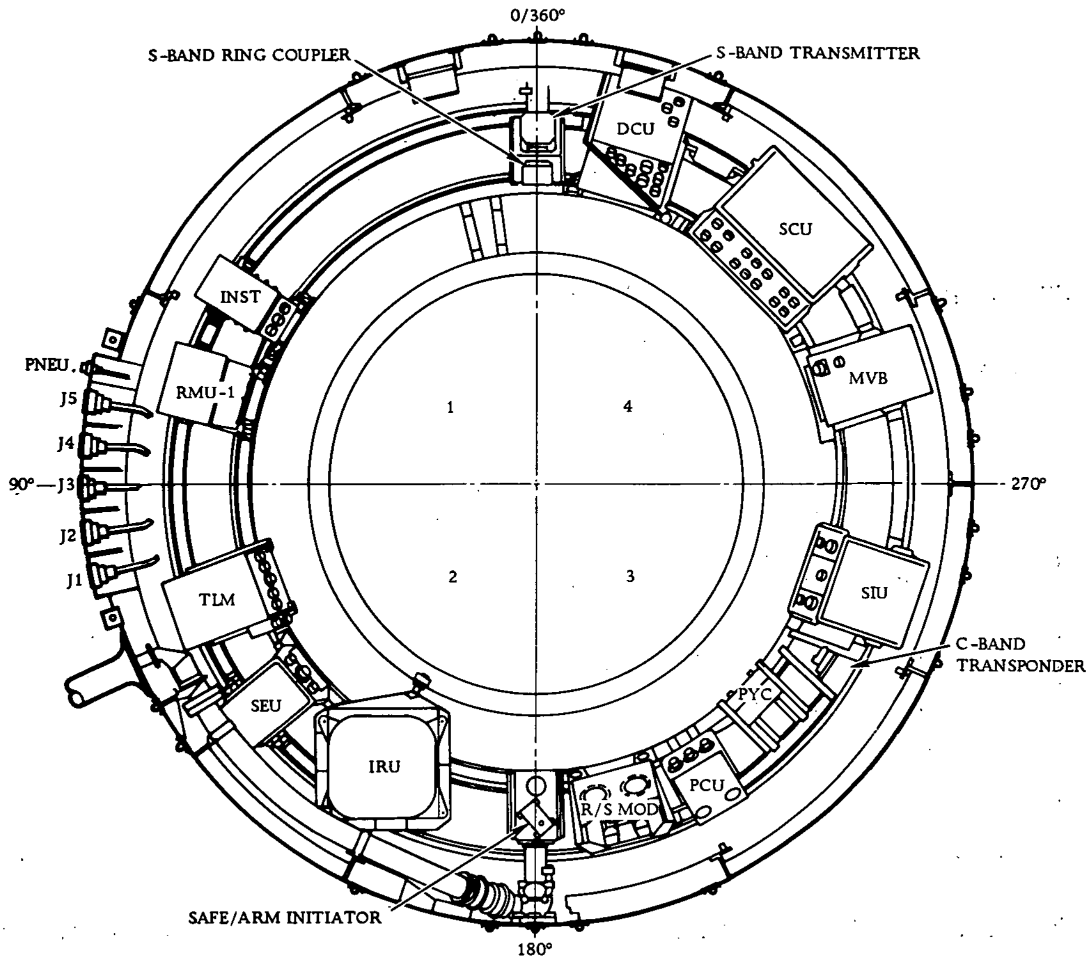


Figure 4-1. Astrionics packages mounted on equipment module.



ABBREVIATIONS

DCU	DIGITAL COMPUTER UNIT	IRU	INERTIAL REFERENCE UNIT
SCU	SEQUENCE CONTROL UNIT	SEU	SYSTEMS ELECTRONIC UNIT
MVB	MAIN VEHICLE BATTERY	TLM	TELEMETRY UNIT-FM
SIU	SERVO INVERTER UNIT	RMU-1	PCM REMOTE MULTIPLEXER UNIT NO. 1
PYC	PYROTECHNIC CONTROL UNIT		
PCU	RANGE SAFETY POWER CONTROL UNIT	INST	INSTRUMENTATION BOX
R/S MOD	RANGE SAFETY MODULE		

FORWARD UMBILICAL PANEL DISCONNECTS

J-1	VEHICLE	T-4 EJECT
J-2	VEHICLE	T-4 EJECT
J-3	SPACECRAFT	T-4 EJECT
J-4	SPACECRAFT	T-0 EJECT
J-5	VEHICLE	T-0 EJECT
PNEU	PNEUMATIC	T-0 EJECT

Figure 4-2. Centaur D-1A astrionic package layout.

4.1 DIGITAL COMPUTER UNIT

ELEMENT:

- Teledyne Digital Computer Unit (DCU)
  16,384 Word Capability

FUNCTIONS:

- Receive Data
- Process Data using Prestored Program
- Provide Data and Commands to Other Systems

A major characteristic of the D-1 Centaur is the integration of several functions through the use of a powerful airborne digital computer unit (DCU). The DCU plays a role in several functions: navigation, guidance, control, sequencing, propellant utilization, propellant tank pressurization, and instrumentation and telemetry. Its speed, storage capacity, and input/output also provide additional capability for growth.

The use of a powerful digital computer has permitted many functions to be done by software, which are normally done by other hardware. In particular, mission-peculiar requirements are handled by software where practical, permitting changes to be expressed in mission constants.

4.1.1 TASKS. The task of the DCU is to receive data, process this data in accordance with a prestored program, and output the resulting data and commands. Functions associated with the DCU are:

Navigation. Determine, on a continuing basis throughout flight, the position and velocity of the Centaur vehicle.

Guidance. Determine the proper orientation of the vehicle, and output it in terms of pitch and roll axes. Determine the proper times to turn the main engines on and off, and command these events.

Control. Determine and issue proper commands to position the main engines or actuate the H_2O_2 engines to orient the vehicle in the proper attitude.

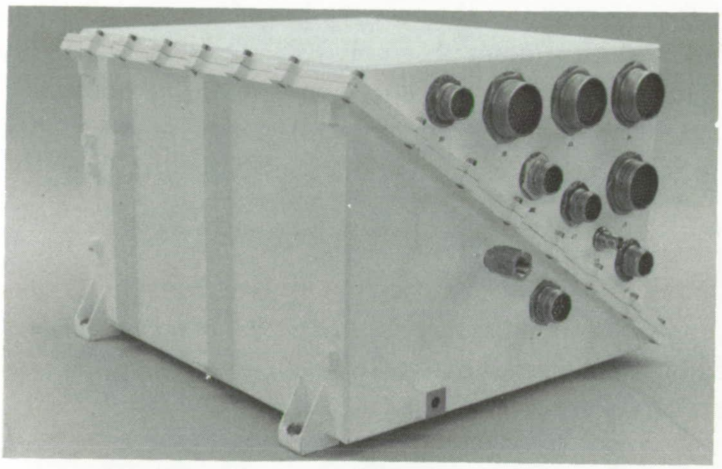
Sequencing. Determine when discrete events should occur, and command them to occur.

Propellant Utilization. Determine the proper positions of the propellant utilization (PU) valves, and command these valves to the desired position.

Propellant Tank Pressurization. Determine pressurization or venting action desired for the propellant tanks, and command this action.

Instrumentation and Telemetry. Request telemetry data in the desired sequence, format this data and output it at a selected bit rate.

4.1.2 DESCRIPTION AND CAPABILITY. The DCU (Figure 4-3) is a stored program, random access core machine. Memory is composed of 16,384 words of 24 bits each. Hardware interlocks prevent changing the contents of 12,280 of these words, however, when special laboratory equipment is connected the contents can be changed. The flight program and telemetry formats are loaded into this area, and cannot be altered on the vehicle. Memory cycle time is nominally three microseconds.



Weight: 60 lb
Size: 16" × 14" × 10"
Power: 160 watts (28v d-c)

Figure 4-3. Digital computer unit.

Instruction Set. The DCU has an instruction set of 25 hardware instructions in addition to input/output instructions. Typical execution times are shown in Table 4-1.

Table 4-1. DCU execution times.

Event	Time	
Add/Subtract	6 μ sec	Absolute Execution Times
Transfer	6 μ sec	Absolute Execution Times
Input/Output	6 μ sec	Absolute Execution Times
Multiply	22.5 μ sec	Absolute Execution Times
Divide	40.5 μ sec	Absolute Execution Times
Shift	75 μ sec	maximum execution time

Indexing. The DCU has three hardware index registers which can be addressed to modify instructions.

Interrupts. The DCU has five interrupt channels. The priority of processing these is: (1) power dropout, (2) power on, (3) DCU telemetry data, (4) ground support equipment (GSE), and (5) real-time. The first two, power-associated, interrupts cannot be disarmed; the last three are program-maskable and provide data handling capability during normal operation of the DCU.

The power dropout and power on interrupts permit orderly handling of DCU tasks when these situations occur. The DCU telemetry data interrupt is generated each time data is placed on the pulse code modulator (PCM) bit stream. After transmission, this interrupt is used to load the next DCU data word to be telemetered into a dedicated memory cell.

The GSE interrupt is used to accept and process data transferred from the ground support equipment.

The real-time interrupt is used as the timing reference for navigation, and guidance and control. In normal use, it is exercised at a rate of 50 Hz.

Pulse Code Modulation. The DCU includes a central controller unit (CCU) for formatting telemetry data into a PCM bit stream. This bit stream is in accordance with a stored format and at a selected bit rate. Up to four formats may be stored in unalterable memory, with the one in use at any time being selected by the DCU program, as is the bit rate.

Input/Output. The input/output capabilities of the DCU are shown in Table 4-2.

Table 4-2. DCU input/output capabilities.

Inputs	Comments
Serial - 24 Bits	GSE Uplink
Parallel - 12 Bits	28v d-c (Spare Capability)
Discretes - 8 Channels	28v d-c Testable (SKD's)
Incremental - 4 Channels	ΔVs and Real Time
Analog - 12 Channels	11 Bits plus Sign, ± 12v d-c
Telemetry -	
Remote Multiplexer Units - 4 Channels	1 Spare with Atlas Booster
External Serial - 1 Channel	Spare Capability
Outputs	**Comments**
Parallel - 36 Bits	22 for Switch Selection
Strobe - 1 Channel	Programmable Delay
DC Analog - 6 Channels	8 Bits plus Sign, ± 5v d-c
AC Analog - 6 Channels	11 Bits plus Sign, 3.5v rms
Telemetry - 2 Channels	1 to Transmitter, 1 to Downlink

4.1.3 GROWTH POTENTIAL. In the Centaur D-1 usage with an Atlas booster, the DCU has excess capability available for future growth. This capability is in both computational reserves and input/output.

From a computational point of view, both duty cycle and storage reserves are of interest. Using the Pioneer G mission as a reference, the maximum duty cycle is nearly 70%, leaving about 30% duty cycle reserve for growth. The corresponding storage requirement is about 7000 words of permanent (unalterable) storage and 2000 words of temporary (alterable) storage. This leaves a reserve of about 5000 words of permanent and 2000 words of temporary storage.

Spare input capability with the Pioneer G configuration includes four analog to digital (A/D) channels, five SKD discretes, and the twelve-bit parallel input channel. Fourteen additional output discrete bits are also available on this mission.

When propellant tanks pressurization control is added to the DCU tasks (AC-36 and on), the four A/D input channels will be used to bring in a primary tank pressure measurement and two redundant switched backup measurement sets.

4.1.4 CHECKOUT AND LAUNCH. The DCU is checked out, with its flight program, in a FAP simulation at Convair Aerospace in San Diego. This simulation presents the DCU with electrical inputs such as it will see during its planned flight, and its outputs are monitored to show the DCU has properly responded (worked the right problem).

During preparation for launch, the DCU participates in the checkout of other hardware, and is, itself, further checked out. Memory sum and instruction execution tests are made to verify both logic and stored program. Finally, launch day constants are loaded and the launch takes place.

4.2 INSTRUMENTATION AND TELEMETRY

ELEMENTS:
• Digital Computer Unit (DCU)
• Remote Multiplexer Units (RMUs)
• Signal Conditioner
• Telemetry Transmitter (S-band)
• Ring Coupler and Antennas

FUNCTIONS:
• Collects and Transmits Vehicle Status
• Reports Status during Vehicle Checkout
• Provides Data for Post Flight Analysis

Pulse code modulated (PCM) telemetry characterizes the instrumentation and telemetry function on Centaur D-1. Measurements that are not digital in nature are converted to digital representations and transmitted in a serial format. Measurements of several types are accommodated by signal conditioners that convert measurements to voltages usable by the PCM formatting hardware.

The PCM format (sequence of transmission) and bit rate are each selectable from four choices by the program being executed by the DCU. Flexibility in formatting permits any reasonable choice of sequence. For prelaunch checkout of several airborne packages and general astrionics status monitoring, the PCM data are sent to the computer controlled launch set (CCLS) via a hardwire link through the PCM ground station.

The S-band telemetry ground station network is provided and operated by the Eastern and Western Test Range. The facilities used depend on the specific mission profile and needs, and can include downrange ground stations, ships, or aircraft.

4.2.1 SYSTEM OPERATION.

(Figure 4-4) The instrumentation and telemetry function collects, digitizes, and sends to the ground measurements made on the airborne systems. These measurements are sent in a preprogrammed PCM format. During the prelaunch activity, this PCM data is used to check out and establish the flight readiness of most of the on-board astrionics. For this function, the data downlink is via a coax in one of the umbilical cables. During flight, the PCM data are sent to the ground by radio frequency (RF) link for post-flight analysis.

Measurements, made directly or by transducers, are collected by the remote multiplexer unit (RMU). Analog signals are scaled by the signal conditioners before reaching the RMUs, where they are digitized. Event signals go directly to the RMUs. The RMUs identify signals by addresses, and when addressed by a central controller unit (CCU) send these measurements to a central controller unit.

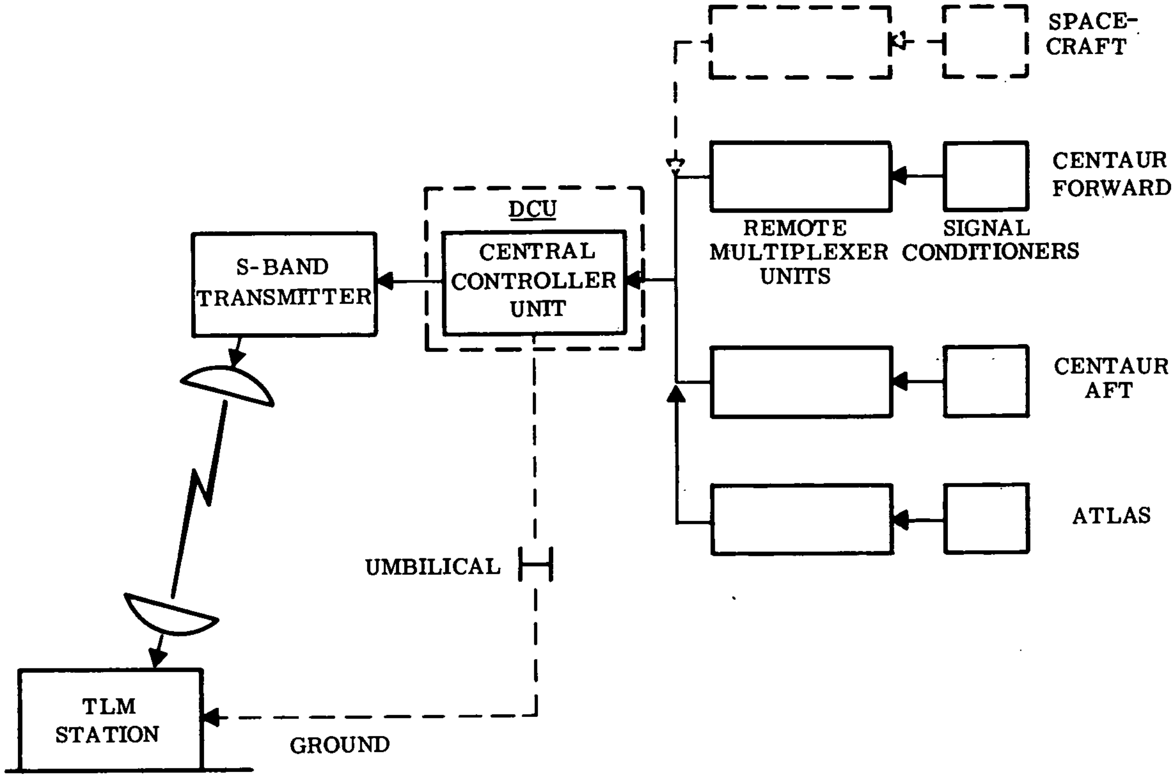


Figure 4-4. Instrumentation and telemetry system.

The CCU, part of the DCU, determines the format (sequence of measurement addresses) by reading and interpreting a segment of the DCU memory. Measurements are put on the PCM bit stream by the CCU in the order of the format addresses. This PCM bit stream goes to the transmitter and, before launch, by landline to the PCM ground station.

The PCM bit stream modulates the output of the transmitter. This output is split by the ring coupler and feeds both antennas to provide good ground coverage.

4.2.2 SYSTEM CAPABILITY. The instrumentation and telemetry function can be configured to match the vehicle or mission, with as many as 1536 measurements that can be individually addressed. DCU internal data can also be addressed.

Measurements are both analog and digital types, with the analog measurements being further broken down by voltage range as "low level" (0 to 30 mv), "mid-range" (-0.5v to +0.5v), and "high level" (0 to 5.0v). For signals not lying reasonably within one of these ranges, signal conditioners rescale them to match.

Digital, or "bi-level" signals are nominally 0 to +28v, and are used to record the occurrence of events.

Each PCM data word has eight bits. Analog signals are converted to digital by eight-bit converters that cover the signal range being serviced (low, mid, or high). Events are grouped in clusters of eight, all of which are reported when that group is addressed.

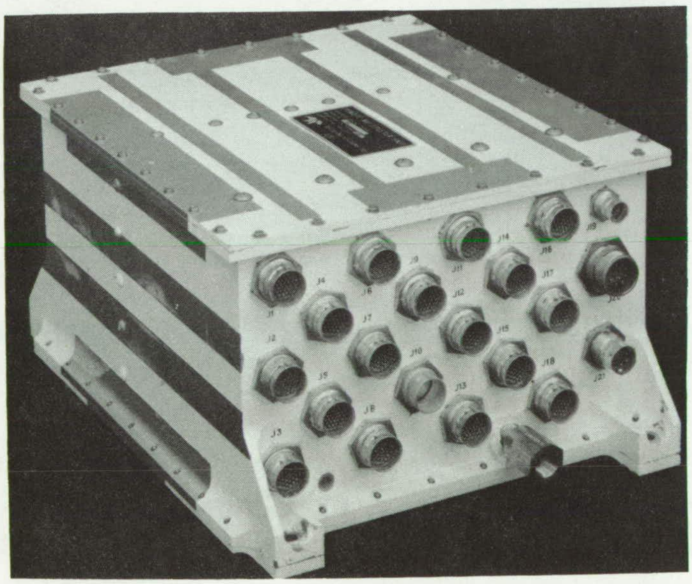
Formatting, or the sequencing of PCM addresses, is done by programming a dedicated area of DCU memory. Up to four formats may be stored, and the one in use at any time is selected by the DCU program being executed.

Four bit rates may be selected by the DCU program. The maximum selectable is 267K bits per second. This provides the capability to send about 30,000 measurements per second.

Growth. Over 1000 measurements are available for use in a format that can accommodate an additional 4000 measurements per second. These can be accommodated by adding circuit boards or RMUs and extra signal conditioners and by the inclusion of these measurements in the PCM format.

4.2.3 COMPONENT DESCRIPTIONS. The instrumentation and telemetry function uses the DCU (See Subsection 4.1), up to four remote multiplexer units and signal conditioners, a telemetry transmitter, a ring coupler, and two antennas. The following paragraphs describe each of these components.

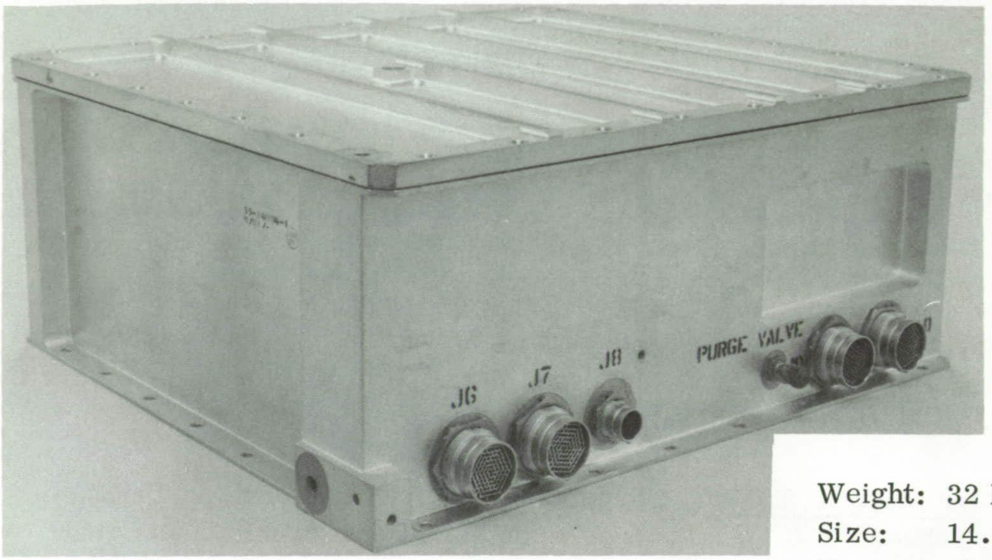
Remote Multiplexer Unit (RMU). Each RMU (Figure 4-5) is tailored to its requirements by the inclusion or omission of circuit boards that service the various types of signals. The RMU contains A/D converters, registers, logic, and power supplies. Its power consumption is rated at 25°C, above which a 60-watt heater cycles on.



Weight: 19-1/2 lb
Size: 12" × 10" × 6"
Power: 20 watts (28v d-c)

Figure 4-5. Remote multiplexer.

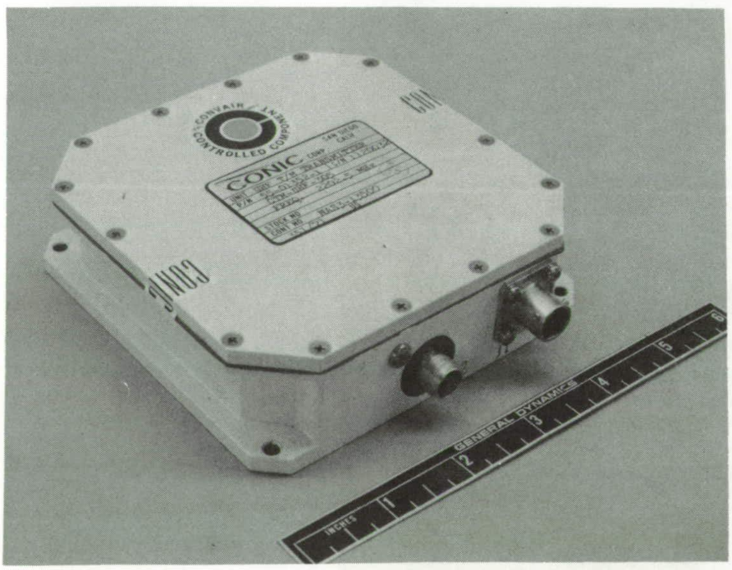
Signal Conditioners (up to 4). Like the RMU, each signal conditioner (Figure 4-6) is tailored to its specific tasks. It contains circuitry to convert measurements to signal voltages compatible with the analog ranges available in the RMU. Transducers are selected from a standard list to convert physical measurements, such as temperature, pressure and acceleration, into corresponding voltages.



Weight: 32 lb
Size: 14.6" × 13.3" × 6.5"
Power: 56 watts (28v d-c)

Figure 4-6. Signal conditioner.

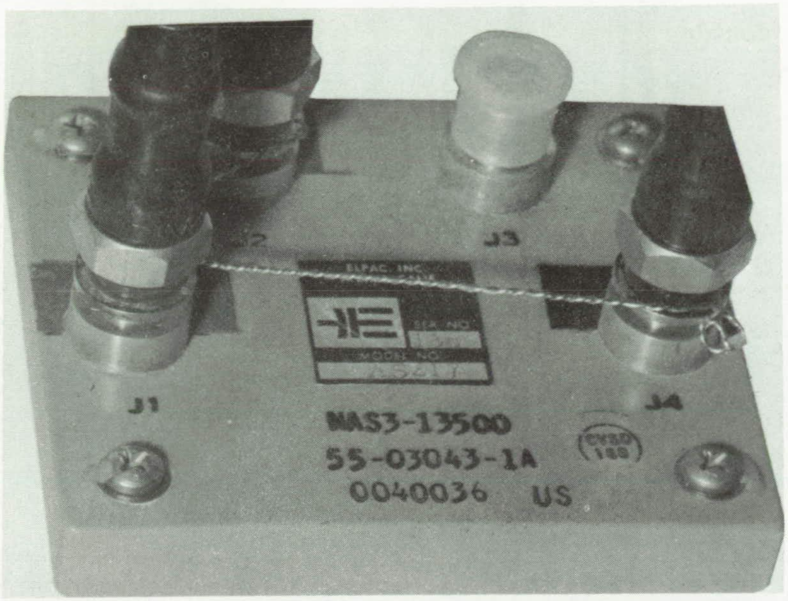
Telemetry Transmitter. The telemetry transmitter is shown in Figure 4-7.



Weight: 2 lb
Size: 4.6" × 4.6" × 1.3"
Power: 45 watts (28v d-c)
Output: 6 watts at S-band

Figure 4-7. Telemetry transmitter.

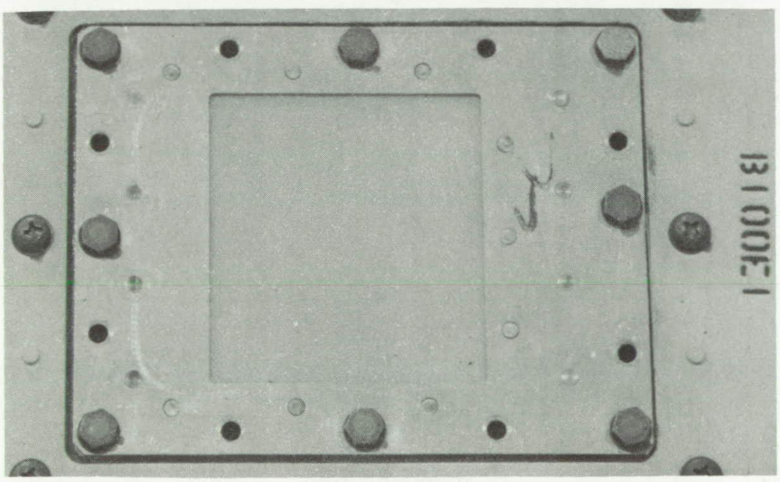
Ring Coupler. The ring coupler (Figure 4-8) divides the transmitter output power and directs half to each antenna.



Weight: 1 lb
Size: 3.25" × 3.75" × 0.5"

Figure 4-8. Ring coupler.

Antennas (2). The radiation pattern of each antenna (Figure 4-9) is such that the two antennas give continuously adequate coverage of the telemetry ground stations during all flight phases when transmission is desired.



Weight: 1 lb (each)
Size: 4.6" × 6" × 2"

Figure 4-9. S-band antenna.

4.2.4 CHECKOUT AND LAUNCH. The PCM elements of the instrumentation and telemetry system are checked out by the computer controlled launch set through the PCM data station, using the landline link to the DCU. RF telemetry signal radiation to local ground stations is required to verify flight readiness of the telemetry system and vehicle.

4.3 COMPUTER CONTROLLED LAUNCH SET

ELEMENTS:
• Ground Computer • Display System • Digital Interface Electronics
OPERATING WITH
Optical Alignment GSE Telemetry Ground Station

FUNCTIONS:
• Ensure Astrionics Launch Readiness • Monitor Status of Vehicle • Provide Diagnostics for Use During Testing • Abort Launch if Conditions are not Satisfactory

Testing of the D-1 Centaur astrionics is controlled and monitored by the computer controlled launch set (CCLS) (Figure 4-10). The CCLS functions include: functional testing of the vehicle astrionics package, calibration and final alignment of the inertial measurements group, loading of computer temporary storage, and both end-to-end and Flight Events Demonstration testing of the entire astrionics system. Additional space is available on the CCLS rapid access drum to permit testing of other astrionics equipment should any be added to Centaur or required for any spacecraft system. The CCLS operates in conjunction with the telemetry ground station.

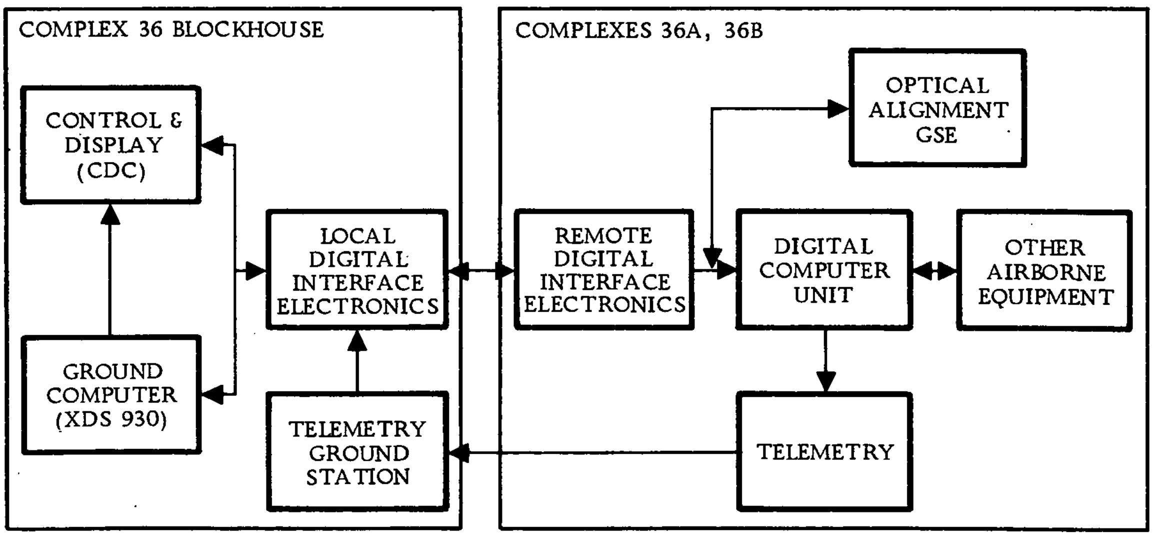


Figure 4-10. Computer controlled launch set layout.

The use of CCLS permits: (1) rigorous checkout, repeatable for every vehicle; (2) bit-for-bit verification of DCU memory contents with error detection and correction for the data transmission links, filtering and correlation processes for analog data; and (3) an interactive operator interface wherein the operator controls the test, is apprised of status, but is spared routine tasks that might distract him.

4.3.1 CCLS FUNCTIONS. The CCLS generates commands and data for transmission to the digital computer unit (DCU), receives data from the DCU and processes the data for output to the operator via the display system.

Inertial Measurements Group (IMG). CCLS monitors the IMG velocity, time, steering, and gimbal demodulator data throughout the testing to establish: (1) performance as required by specification of the IMG inertial components and their associated electronics (including calibration), (2) orientation and alignment of the platform gimbals, including final alignment, and (3) performance as required by specification of the IMG coordinate transformation resolver chain. CCLS also controls and monitors a ground theodolite system to permit optical alignment of the IMG.

Digital Computer Unit (DCU). The CCLS communicates with the DCU to permit: (1) proper DCU initialization so that various tests and test programs can be conducted, (2) functional testing of its performance by loading an instruction test program, (3) bit-for-bit verification of its permanent storage, (4) functional testing of its analog signal interfaces with other astrionics units, and (5) loading and verification of temporary storage locations prior to flight with just updated calibration, wind, and time data.

Telemetry. The CCLS can monitor and process pulse code modulated (PCM) telemetry to produce strip charts, digital printouts, and correlations with time.

Sequence Control Unit (SCU). The CCLS: (1) functionally checks all SCU switches, (2) monitors switch states to assure proper state prior to flight, and (3) controls RF power and permits attitude engine testing. For switch control, CCLS loads the switch commands into the DCU via the hardwire DIE uplink. The DCU then issues the commands to the SCU via the parallel output bus which is the control path used during flight. For switch monitoring, CCLS receives switch status information through the PCM telemetry data.

Servo Inverter Unit (SIU). The CCLS tests the Atlas and Centaur SIU's to ensure that: (1) the engine servoamplifiers are operating correctly, (2) the propellant utilization error detection circuitry is operating correctly, and (3) the inverters are generating 400-Hz power correctly. The CCLS also stimulates the SIU via the DCU during hydraulics checkout and monitors status prior to liftoff.

Atlas SLV-3D Rate Gyro Unit (RGU). The CCLS tests the RGU to verify that: (1) the gyros are functioning correctly, and (2) the gyros are running at liftoff.

Combined Systems Testing. The CCLS initializes and monitors the entire astrionics system during combined systems testing to: (1) provide an end-to-end test of the propellant tank pressurization control functions, and (2) simulate sequencing and engine commands during simulated flights (Flight Events Demonstrations).

4.3.2 CAPABILITY AND DESCRIPTION. Computer and Display. The CCLS uses a Xerox Data Systems 930 computer with 16,384 words of core storage. The computer is equipped with the following XDS peripherals: a rapid access drum (RAD), two magnetic tapes, a card reader, a paper tape punch and reader, a line printer and teletype. A block diagram of the CCLS computer functions is presented in Figure 4-11.

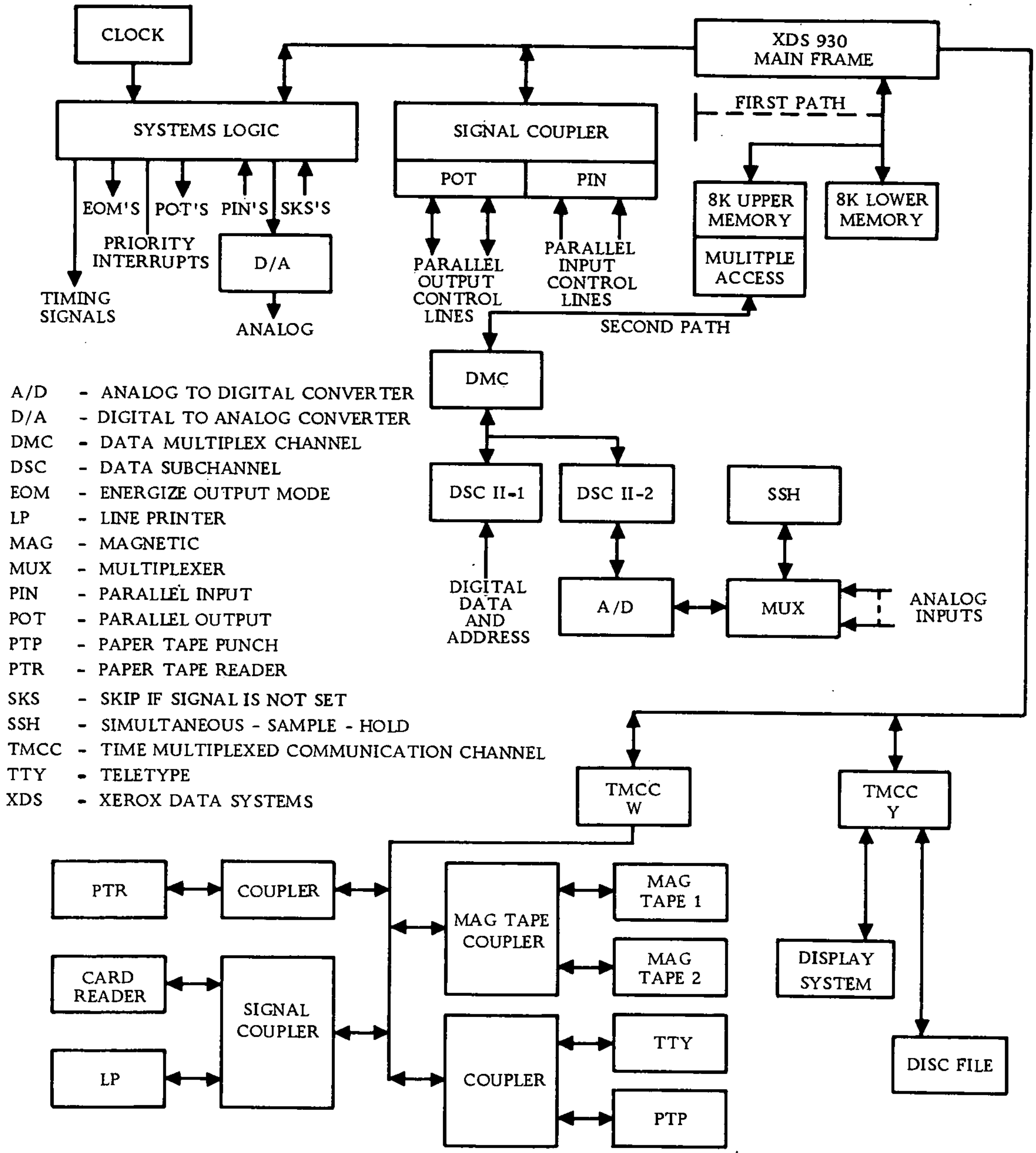


Figure 4-11. CCLS block diagram.

A Control Data Corporation CRT display (Figure 4-12) informs the operator of up-to-date status. The RAD has a storage capacity of two million 6-bit bytes. All programs are stored on the RAD, and then overlaid in core or into the DCU upon operator command. RAD access time for a program is typically three or four seconds; core memory cycle time is 1.75 microseconds.

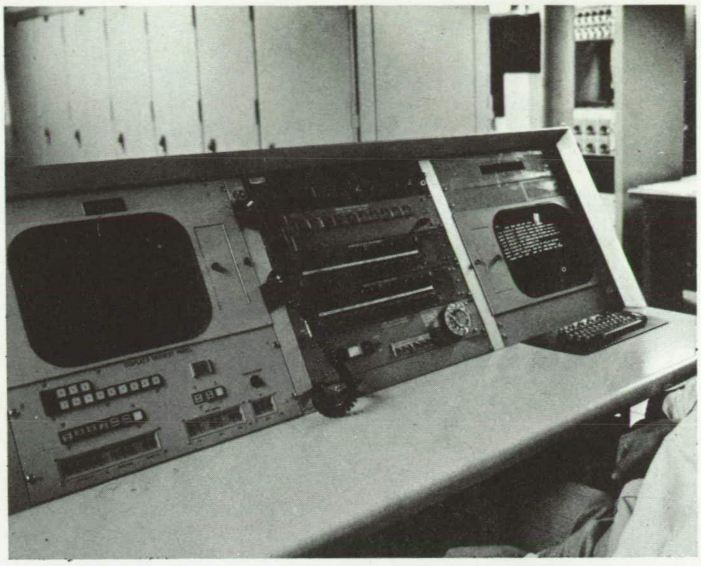


Figure 4-12. CCLS display.

Other computer-related characteristcs are:

a. Instruction Set. The XDS 930 has an instruction set of 61 instructions.

b. Indexing and Indirect Addressing. One index register is provided. The instruction words have one bit for indexing and one for indirect addressing, which is multilevel.

c. Interrupts. Four buffer control, 2 power control, and 32 system interrupts permit standard and specialized input/output operations with considerable multiplexing capability. All except the power interrupts are individually armable.

d. PCM Data Input. A data subchannel provides a 24-bit input register to allow PCM data to be loaded into the CCLS memory automatically, so that a continuously updated table is maintained. Once initialized by the program, no software time is wasted storing the data; a second path to memory is used.

e. Time. A Hewlett/Packard HP101A precision timing reference provides clocks for software timing control. Its accuracy is better than one part in 10^7.

Digital Interface Electronics (DIEs). These two electronic cabinets provide an interface with the launch site. The Local DIE (LDIE) in the blockhouse communicates with the Remote DIE (RDIE) at the launch pad by sending a serial bit stream to the RDIE and receiving a serial data stream from it. The communication hardware detects errors and causes retransmission to avoid processing bad data. The LDIE also accepts a serial range time bit stream and conditions PCM data for input to the XDS 930. The RDIE communicates with the LDIE and DCU, as well as interfacing with the guidance optical alignment shelter (GOAS) and other manual ground support equipment used for power and attitude engine control.

Data Handling, Operation, and Software. The CCLS method of handling data allows for errors in data transmission, and provides two modes of hard copy printout. Digital data communicated to, and read from the DCU is checksummed, analog data is sampled and filtered, and digital SCU data is sampled and voted upon, discarding nonsimilar data points.

During the testing of any portion of the system, the CCLS teletype maintains a hard copy typeout of the various tests the operator has selected, as well as a log of major test events. In this way the operator can review a brief typeout and refresh his memory of events. If a more extensive printout is needed, the CCLS software prints out data on the line printer rather than typing it.

The CCLS software itself is written largely in Fortran, with only the executive and often-used library routines written in assembly language. The executive is a time-sharing type: up to three different test programs can run at one time in timeshare, allowing simultaneous checkout of several subsystems.

4.4 NAVIGATION

ELEMENTS:

- Inertial Measurements Group (IMG)
  - Inertial Reference Unit (IRU)
  - System Electronics Unit (SEU)
- Digital Computer Unit (DCU)

FUNCTION:

- Determine Vehicle Position and Velocity for Use by Guidance and Control

Navigation is the process of determining vehicle position and velocity. This is done on board the Centaur D-1 by inertially measuring accelerations, computing a gravity correction, and integrating.

A stable platform and its electronics unit make up the inertial measurement group (IMG). The IMG measures acceleration and provides a time reference for the digital computer unit (DCU) to make the navigation computations.

4.4.1 SYSTEM OPERATION. The navigation function is to measure sensed acceleration, compute gravity contributions to vehicle accelerations, determine time, and integrate the net vehicle accelerations to maintain continuous values of position and velocity. These values are stored for use by other functions.

A reference inertial coordinate system is maintained in the IRU by gyros mounted on a platform. The platform is the inner gimbal of a 4-gimbal assembly (Figure 4-13). It is kept fixed in space by the movement of the other gimbals with respect to it, each other, and the vehicle. The platform itself has unlimited 3-degrees-of-freedom movement. If the platform rotates from the inertial reference, the gyros sense this and put out an error signal. The error signal is then used to drive the platform back to its proper orientation.

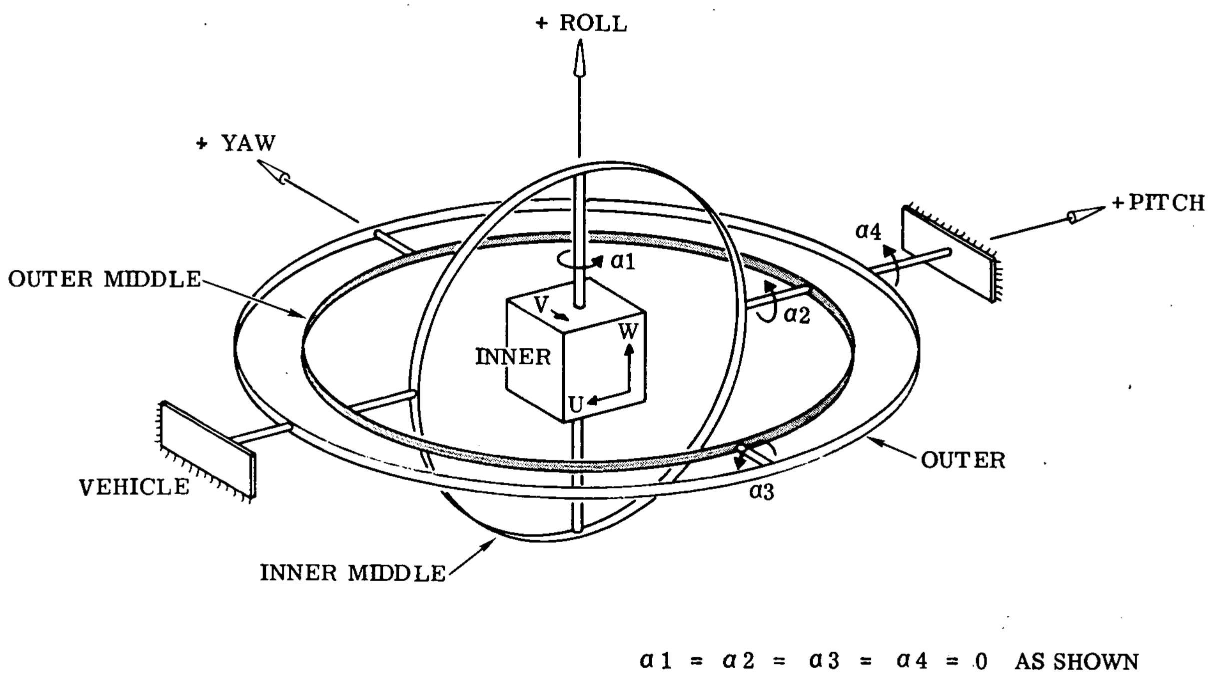


Figure 4-13. Stable platform 4-gimbal assembly.

In response to disturbances, the platform is stabilized by the three orthogonally-mounted single-degree-of-freedom gyros (U, V, W) which drive the appropriate gimbals (Figure 4-14) through resolvers (U and V) and torque motors. No resolving of W gyro output is required, since the W axis is held fixed in space by the inner middle and outer middle gimbals. The torque motors are d-c, direct drive.

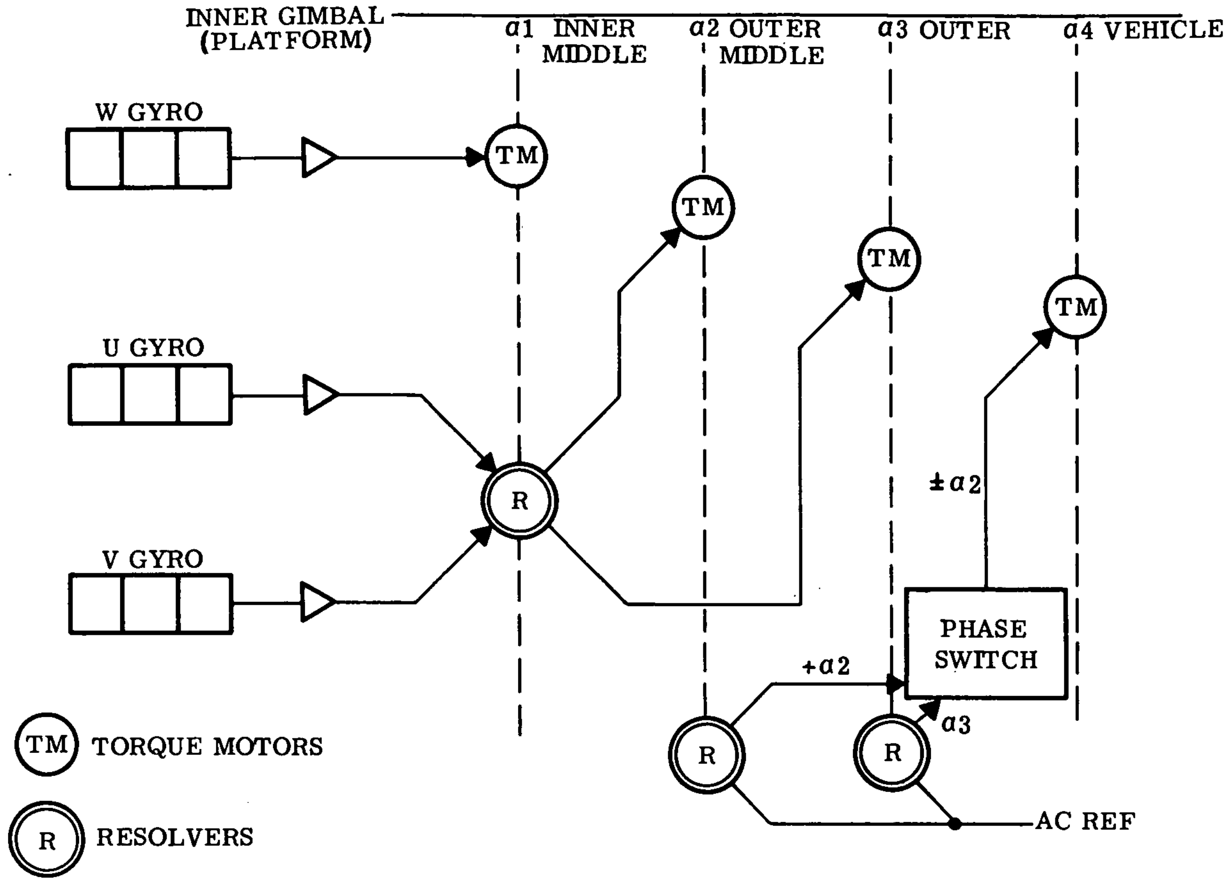


Figure 4-14. Platform stabilization mechanization.

Velocity. Three orthogonal accelerometers mounted on the platform sense acceleration and generate pulses at rates proportional to the sensed accelerations. The accelerometers are continuously torqued to maintain their gimbals near null. The torquing pulses gate clock pulses, which represent velocity increments.

These pulses are sent as delta-velocity pulses to the DCU, where they are accumulated to determine velocity (Figure 4-15). A crystal oscillator in the IRU drives a circuit which counts its frequency down to 800 Hz and sends these pulses to the DCU for maintaining real time against which to integrate accelerations and velocities.

The DCU software processes the delta velocity and real time pulses, computes the effects of gravity, and updates velocity. Position data are calculated by software by integrating the velocity information.

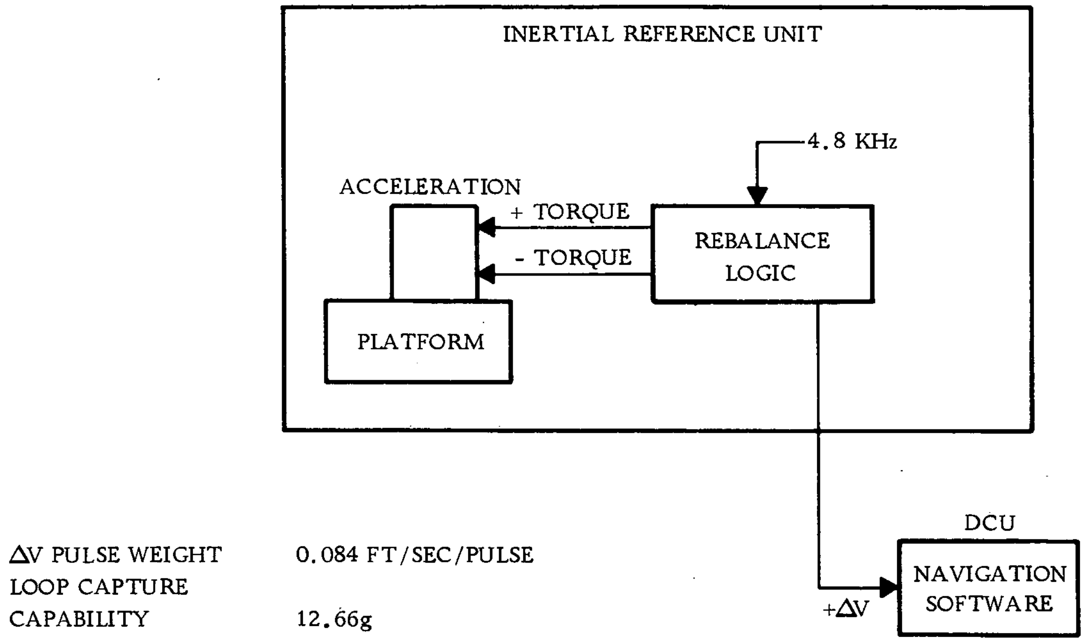


Figure 4-15. Velocity measurement.

4.4.2 SYSTEM CAPABILITY. Capability of the Centaur D-1 navigation function can best be described in terms of mission accuracy. Table 4-3 shows this accuracy for some missions flown by Centaur, using navigation sensors of essentially the same design. Improved on-pad calibration techniques are expected to provide a slight improvement in accuracy.

Table 4-3. Centaur D flight experience.

Mission	Flights	Parameter	Preflight 3 σ Value	Flight Value
Surveyor	7	MCR (m/sec)	17	4 (average)
Mariner Mars	3	MCR (m/sec)	10	2 (average)
Pioneer F	1	MCR (m/sec)	117	14
OAO	1	Apogee Minus Perigee (n.mi.)	5	1
ATS	1	Transfer Ellipse Apogee (n.mi.)	130	50
Intelsat IV	4	Transfer Ellipse Apogee (n.mi.)	80	6 (average)

4.4.3 COMPONENT DESCRIPTION

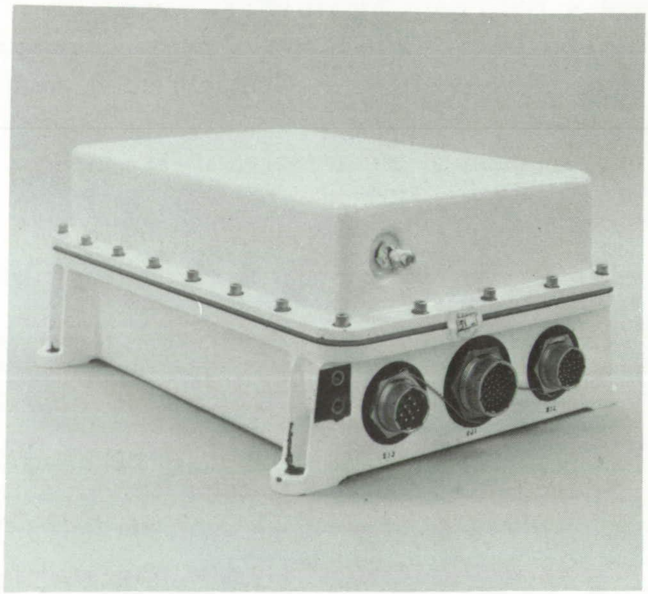
Weight: 63 lb
Size: 18 1/2" × 18" × 13 3/4"

Figure 4-16. Inertial reference unit (IRU).

Digital Computer Unit (DCU) (Subsection 4.1)

Inertial Reference Unit (IRU). All power required by the IRU (Figure 4-16) comes from the systems electronic unit (SEU). The IRU contains a 4-gimbal, all attitude, gyro-stabilized platform which supports three orthogonal, pulse-rebalanced accelerometers, and a prism for alignment. Resolvers provide transformation vectors from the inertial to the vehicle coordinate system. The IRU contains all pulse-rebalance, gimbal stabilization, gyro torquing, and resolver chain electronics. There is also a crystal oscillator in the IRU which is the navigation function primary timing reference.

Systems Electronic Unit (SEU). The SEU (Figure 4-17) contains filters, power supplies, and mode control relays for the IRU, and supplies all power required by the IRU.



Weight: 25 lb
Size: 13 3/4" × 8 3/4" × 5 7/8"
Power: 185 watts (28v d-c)

Figure 4-17. Systems electronic unit (SEU).

4.4.4 CHECKOUT, CALIBRATION, AND LAUNCH

Elements of the navigation function are prepared for flight by the computer controlled launch set (CCLS). Programs loaded into the CCLS ground computer, and by the CCLS into the DCU, perform checkout and calibration of the IRU, and orientation and final alignment of the platform gimbals. The DCU permanent storage contents are verified and proper operation of the DCU is checked. Launch day flight constants are loaded into DCU temporary storage and verified just prior to flight.

4.5 GUIDANCE FUNCTION

ELEMENT:
• Digital Computer Unit

FUNCTION:
• Determine Proper Vehicle Attitude and Issue Steering Commands to the Control System

4.5.1 OPERATION. The vehicle position and velocity, together with information about the desired trajectory, provide input to the guidance function. The output is the desired vehicle attitude, stated in terms of roll and pitch axes in inertial coordinates.

The DCU performs computations to determine the desired attitude, expresses this attitude in inertial coordinate components of vehicle roll and pitch axes, and outputs these components via the a-c D/A converters. Figure 4-18 is a block diagram of the guidance system.

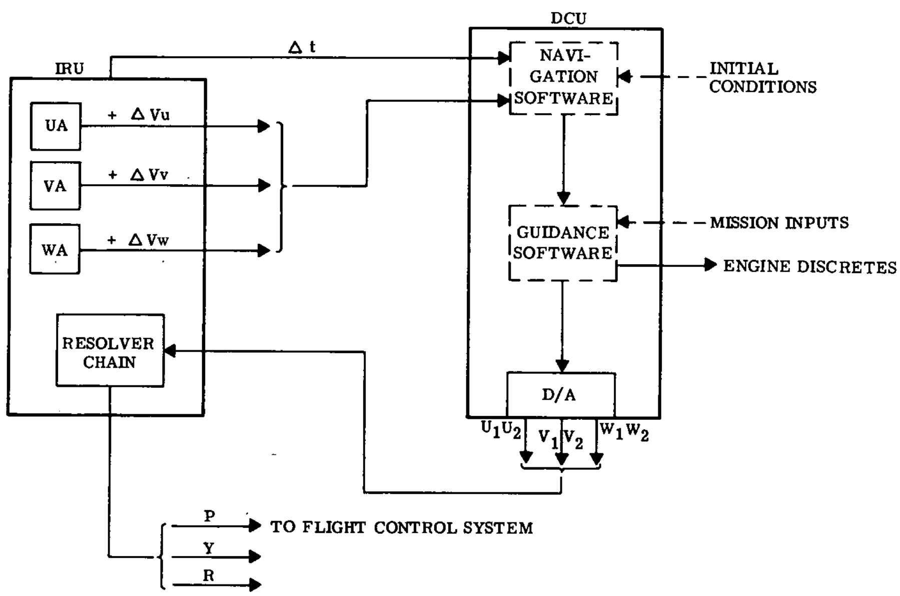


Figure 4-18. Guidance system block diagram.

4.5.2 DESCRIPTION. The digital computer unit is described in Subsection 4.1.

4.5.3 CHECKOUT. The guidance function is checked out by a FAP simulation (page 5-21) in San Diego and by memory sum tests prior to launch.

4.6 CONTROL SYSTEM

ELEMENTS:
• Digital Computer Unit (DCU)
• Inertial Measurements Group (IMG)
Inertial Reference Unit (IRU)
System Electronics Unit (SEU)
• Servo Inverter Unit (SIU)
• Vehicle Main Engine or Reaction Motor Systems

FUNCTIONS:
• Provides Vehicle Attitude Stabilization
• Points the Vehicles in Response to Guidance Steering Commands

Signal processing and commands for vehicle control are mainly functions of the DCU. It receives analog attitude outputs from the inertial measurements group (IMG), converts them to digital form, operates on them as the software and mission require, generates vehicle output commands (pitch, yaw, and roll), and converts them to analog signals for powered flight control and digital commands to the coast phase control system.

4.6.1 SYSTEM OPERATION. A block diagram of the control system is presented in Figure 4-19.

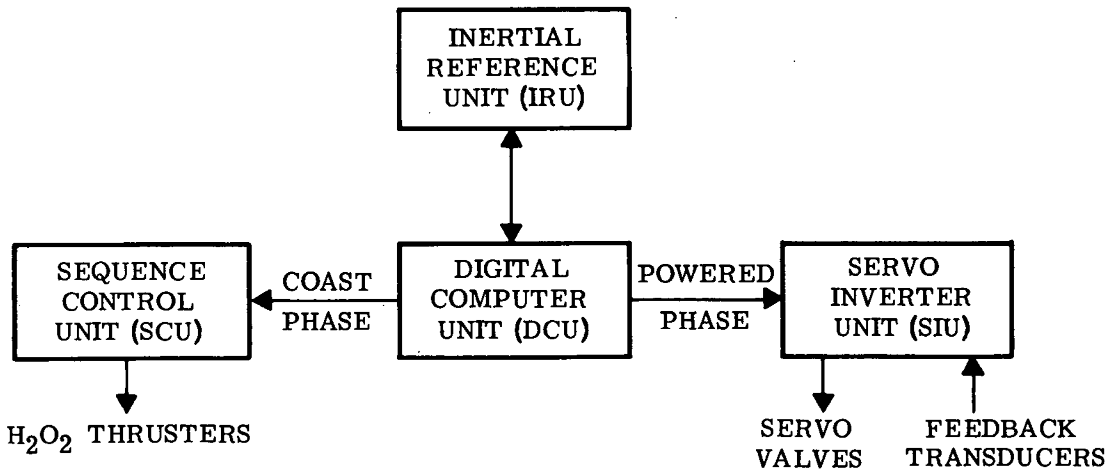


Figure 4-19. Control system.

Powered Flight Control. The DCU issues three analog output signals (P, Y+R, and Y-R) which correspond to a desired engine position to the servo inverter unit (SIU)

The mechanical engine control device is a hydraulic actuator controlled by a hydraulic servovalve. An electronic input from the SIU to the servovalve causes hydraulic fluid flow and engine movement. An engine position signal is provided by a feedback transducer.

Each of the two Centaur engines has two actuators, one to do pitch movement and one to do yaw. Because of vehicle orientation during flight, both engines move together to provide pitch motion. Yaw directional control is achieved from identical engine side motions and roll is a result of equal and opposite vertical motions.

The computer pitch command drives two position servoamplifiers which control the pitch servovalves on each engine. The yaw command is combined with roll to provide a yaw plus roll command to one engine yaw actuator and a yaw minus roll command to the other.

Coast Phase Control. During coast, vehicle rate and displacement are detected and measured by the IMG, just as in powered flight. These signals are processed by the DCU and translated to vehicle command requirements.

However, basic vehicle control during this flight period is accomplished by torques provided by 14 monopropellant (H_2O_2) thrusters mounted in an orthogonal arrangement at the perimeter of the Centaur thrust section. They are strictly on-off devices, pointed in a single direction. They fire in short bursts for control, to hold the vehicle in a rate-displacement limit cycle; the characteristics of which are determined by the vehicle dynamics and the rate displacement threshold stored in the DCU.

Since engine control is strictly on-off, it is simply accomplished by 28v d-c on-off commands from a single group of SCU switches. The DCU executes these commands via the normal DCU-SCU interface and no special provisions are required.

4.6.2 SERVO INVERTER UNIT OPERATION. The Centaur servo inverter unit (SIU) contains four standard D-1 type printed circuit boards, each of which is a servoamplifier. Each servoamplifier operates in conjunction with a servovalve, actuator, and feedback transducer to form the engine position servo.

The amplifiers operate in the current feedback mode and provide ripple filtering for both the DCU input signals and the demodulated feedback signal from the a-c engine position feedback transducers.

Engine Position Servo Control. The servoamplifier provides a servovalve current in response to an error signal input. The error signal can be thought of as the difference between the DCU command and the engine position feedback signal. Therefore, the zero current output represents the engine being in the proper position. If they do not agree, engine motion will be commanded (hydraulic overflow) until the feedback and command currents are equal.

The control loop contains the summing resistor for the d-c command input, and a half-wave synchronous demodulator and summing resistors for the a-c position feedback signal. The active amplifier is a monolithic integrated circuit operational amplifier (UA741 type). Input and demodulator ripple filtering is accomplished by an amplifier feedback equivalent resistance-capacitor (RC) network which provides an active filter single lag at approximately 16 Hz. Output current sensing and feedback causes the amplifier to operate in the current mode, making its characteristics relatively independent of load characteristics. Figure 4-20 is a block diagram of this configuration.

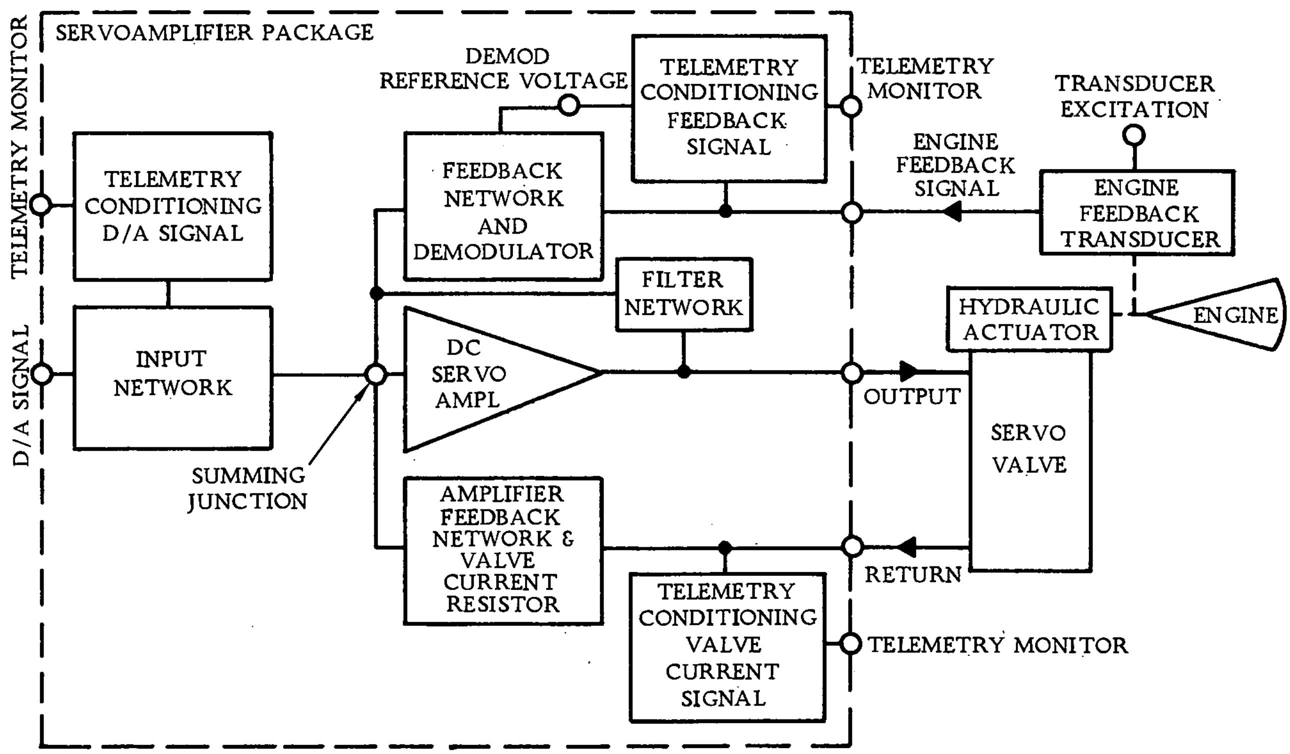


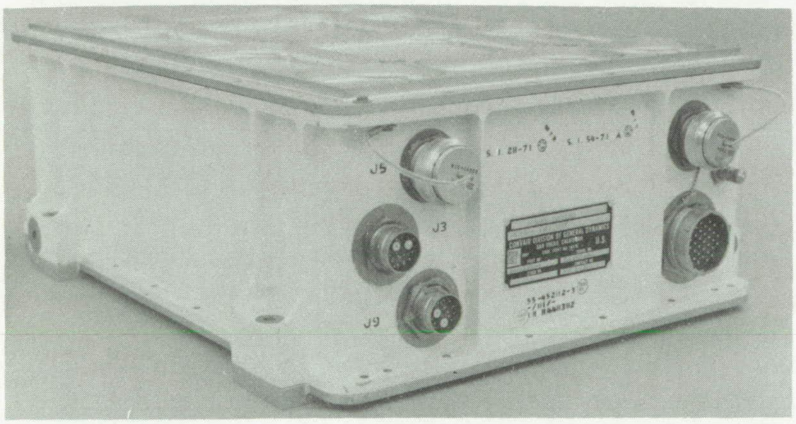
Figure 4-20. Centaur D-1A signal flow block diagram of a typical servo package channel.

Transformation and Scaling. Input voltages (command and feedback) result in an output current to the servovalve. The following defines the input and output relationship and capabilities:

Command Input:	Analog from -5.0v d-c to +5.0v d-c. Minimum input impedance, 10K Ω
Position Feedback:	Analog from -12.0v rms to +12.0v rms. Minimum input load, 15.0K Ω
Output Current:	± 4.0 ma d-c (linear range) ± 5.0 ma d-c (maximum range)
Command Gain:	2.7 ma d-c/v d-c ±5.0%
Static Gain:	0.736 deg/v d-c ±3.0%
Servoamplifier Gain (feedback):	1.8 ma/v a-c ±5.0% (assuming a transducer gain of 2.04 v/deg)
Null Offset:	±0.028 deg maximum (engine position)

4.6.3 COMPONENT DESCRIPTION. The digital computer unit and inertial measurement group are described in Subsections 4.1 and 4.4 respectively. The servo inverter unit is shown in Figure 4-21.

The SIU also contains the D-1 Centaur propellant utilization electronics and the inverter which supplies vehicle a-c power.



Weight: 45 lb
Size: 12.5" × 17.3" × 6.4"

Power:
Servos - 0.10 watt
Total SIU - 250 watts

Figure 4-21. Servo inverter unit (SIU).

4.6.4 CALIBRATION, CHECKOUT, AND LAUNCH. Checkout and calibration at the unit level in San Diego is accomplished using an SIU special purpose test set, while exposed to normal flight environments (temperature and vibration). Vehicle tests are conducted prior to launch to verify basic engine response and qualitative system function and overall gain.

4.7 VEHICLE SEQUENCING

The Centaur D-1 digital computer unit provides all the basic flight data evaluation and processing functions, and the basic timing and switching information required for flight. The interface between the DCU output registers and the vehicle systems that require switched and/or timed commands is the sequence control unit (SCU).

ELEMENT:
• Sequence Control Unit (SCU)

FUNCTION:
• Converts DCU Outputs into Commands Usable by Vehicle System

4.7.1 OPERATION. The SCU receives data from the DCU as a 22-bit parallel output word, and decodes that word to operate output switches in groups of 16 each time it receives an execute command (strobe). Its output is comprised of 96 sets of relay contacts in 6 groups of 16 each. Six sequential input-word strobe commands are required to change the state of all 96 output switches.

The output switches provide +28v d-c, 26v rms a-c, 115v rms a-c, and isolated contact closures. The mix of output types is designed to run the Centaur vehicle and provide sufficient spares to accommodate reasonable mission peculiar changes.

In addition, the SCU provides for airborne-to-ground power changeover and arm-safe functions on critical outputs via multipole motor driven switches. It also acts as the main junction box for power distribution, and the SCU housing is the vehicle ground plate. Since all power returns terminate at the unit housing in the D-1 single-point ground system, shunts used for current monitoring of the various vehicle systems are also included in the unit.

A block diagram of the SCU is shown in Figure 4-22. The operation of one switch (Switch 1 in Group 1) is shown in Figure 4-23.

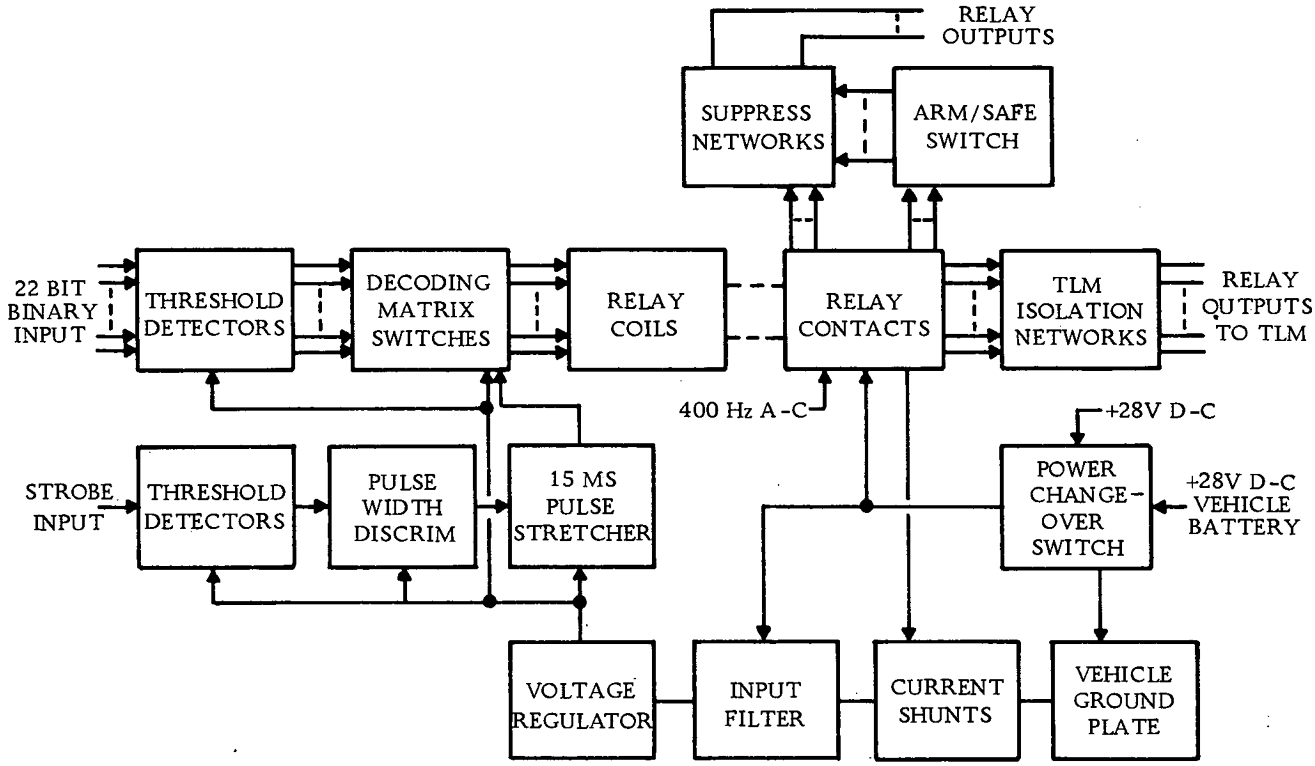


Figure 4-22. Sequence control unit block diagram.

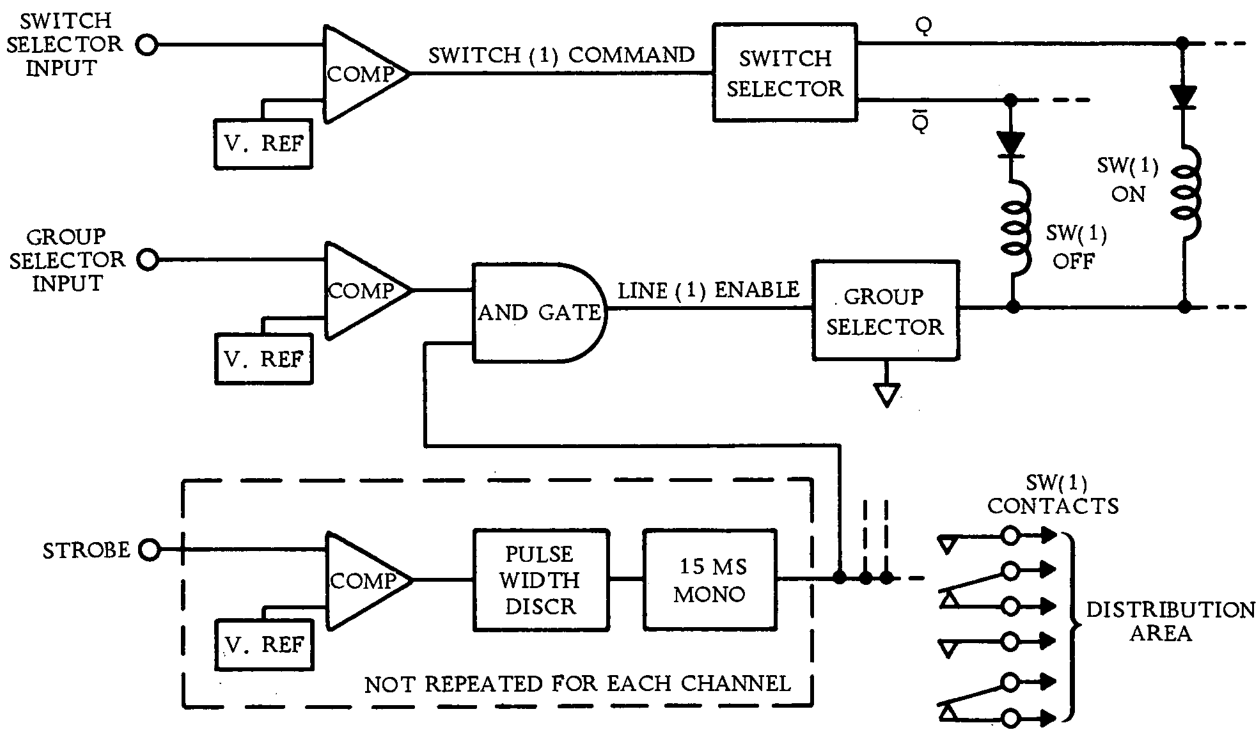


Figure 4-23. Functional channel description power matrix.

The output relays are magnetic latching types and are used for final load switching. They are arranged in a 6 × 16 matrix with the set or reset coil at the cross points of the row and column lines. They provide the basic SCU memory capability. The SCU interprets the 22-bit command in two parts. Part 1, group selector, consists of six bits, bits 0 - 5; Part 2, switch selector, consists of 16 bits, bits 6 - 21.

$$\overbrace{0\quad 1\quad 2\quad 3\quad 4\quad 5}^{\text{GROUP SELECTOR}}\quad\overbrace{6\quad 7\quad 8\quad 9\quad 10\quad 11\quad 12\quad 13\quad 14\quad 15\quad 16\quad 17\quad 18\quad 19\quad 20\quad 21}^{\text{SWITCH SELECTOR}}$$

In operation, the group selector bits will have one bit high (1) only, at any time, with no exceptions. The switch selector bits may be either high (1) for a switch "ON" command or low (0) for a switch "OFF" command. Any arbitrary combination of inputs, including all bits "ON" or "OFF", does not cause SCU damage.

Simultaneous switching is accomplished by all switches within a group. Switches changing status upon receipt of a single strobe signal are defined as switching simultaneously.

SCU reset is accomplished by executing six sequential 22-bit commands.

Direct Current (d-c) Interface. The input amplitude comparator, which is a simple threshold circuit and the switch selector which drives the matrix columns are combined. One or the other of the relay coil outputs is always on, depending on the state of the input.

A grounding switch/AND gate sums the group selector input and the stretched strobe (strobe) inputs to provide a "line enable" signal. This signal connects the return sides of all the relay coils in a single group to ground for 15 ± 1 ms, thereby pulsing them to the proper output state.

Strobe Interface. Pulse amplitude discrimination is performed by a simple threshold input stage similar to the one used on the other DCU input interfaces.

Pulse width discrimination is provided by a simple integrator type input period measurement circuit, which provides a pulse output to drive the 15-ms mono.

The 15-ms monostable circuit provides a stretched strobe pulse with a length that regulates the period for which the relay coils are pulsed. It is a temperature compensated monostable multivibrator.

4.7.2 CAPABILITY. The basic capabilities of the SCU can be broken down into four major categories:

a. Receive and decode command data.

b. Provide vehicle timing commands in response to that data.

c. Provide vehicle power switching capability (ground or airborne).

d. Provide instrumentation outputs to assess the status of the vehicle command functions and power system.

Receive and Decode Command Data. The receiver circuitry used for the d-c interface is designed to provide noise discrimination and has a well controlled threshold. Its characteristics are:

Logical "1":	greater than 3.0v d-c
Logical "0":	less than 1.0v d-c
Overvoltage Protection:	-1.5v d-c to +10.0v d-c
Noise Rejection:	no state change for 1.0 μ sec transients or 0.5v a-c peak periodic waveform at any frequency.

The strobe interface is a high frequency arc and provides both amplitude and time discrimination for input signals. Its characteristics are:

Logical "1"	greater than 4.5v d-c
Logical "0"	less than 1.0v d-c
Frequency:	50 pps maximum
Time Discrimination:	
Must Reject	9.0v d-c for 6.0μs recurring at 8.0K Hz maximum (any waveform)
Must Accept	4.5v d-c minimum for 12.0 μs minimum
Overvoltage Protection:	-1.5v d-c to + 10v d-c

Decoding is accomplished by allocating the 22-bit input word into two sections. Bits 0 through 5 select one of six groups of 16 switches each and bits 6 through 21 command switches 1 through 16 respectively, in the selected group to be high or low.

Vehicle Timing Commands. Actual output commands are provided by relay contact closures in response to the decoding of the preceding paragraph. Those commands are executed a maximum of 7.0 msec after a strobe pulse is received and have the following characteristics and capability:

D-C Output:	Vehicle battery voltage (+28v d-c), 7.5 A max.
A-C Output:	26.0v rms, 115v rms, 3.5 A max.
Contact Closure:	Isolated switch contacts having the same capability as d-c or a-c outputs.

Vehicle Power Switching. Units or systems requiring power on-off capability when connected to the airborne supplies can be switched by connecting them to the appropriate a-c or d-c outputs. They may then be controlled by the appropriate DCU command in the same way as any output switch. Nonflight critical functions supplied from the SCU are fused to protect the airborne power system.

Airborne/ground power switching is accomplished by a ground operated, multi-pole, motor driven switch. This power changeover switch is a make-before-break type; it transfers in 200 msec maximum and has a current capability of 65 amps per pole.

Command switching is also provided by a similar pair of arm/safe switches in order that critical functions (fire pyrotechnics, etc.) may be tested without actually providing an output to the vehicle system. There are presently 31 such outputs distributed on the two switches, including nine spares. The switches are also ground controlled and may be operated independently.

Instrumentation Outputs. Outputs to determine the status of all command outputs are conditioned and provided to the instrumentation system. For a-c and d-c commands, the actual output voltages also provide the instrumentation outputs while isolated contact closures provide their indications from a second set of contacts on the same relay armature. The characteristics of these bi-level outputs are:

Low Output:	less than 0.5v d-c
High Output:	greater than 3.5v d-c, less than 35.0v d-c

Seven current monitors in the ground return lines of selected systems are provided that have full scale outputs from 2 amps (SCU electronics) to 100 amps (total battery current). Full-scale corresponds to 30mv d-c, matching the full-scale input of a low level analog telemetry channel.

Approximately 10 percent of the SCU output switches are unused for current missions and boosters. These switches can be considered spares. However, because of the distribution of switch types and individual mission/booster/payload requirements, it is usually the case that while some spares are used, other switches are not required and the true growth or flexibility potential is greater than 10 percent.

4.7.3 SEQUENCE CONTROL UNIT DESCRIPTION. The SCU is shown in Figure 4-24.

4.7.4 CALIBRATION, CHECKOUT AND LAUNCH. Checkout and calibration at the unit level in San Diego is accomplished by a general purpose digital computer (XDS 910) coupled with a special interface test set. While exposed to normal flight environments (temperature and vibration) it is tested for:

a. Input threshold levels

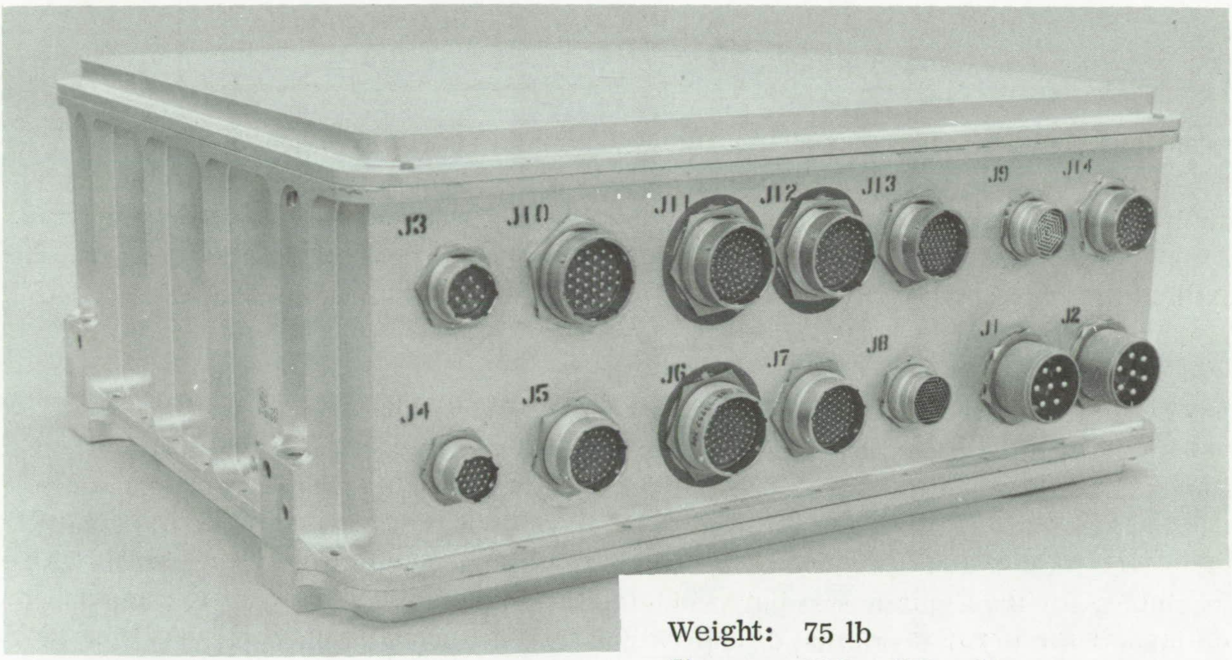
b. Output response and switch characteristics.

c. Logic functions.

d. Power switching.

e. Instrumentation and test outputs

During preparation for launch, the DCU-SCU interface is verified by mated operation, and performance of logic functions and outputs is verified by simulated flight sequences.



Weight: 75 lb
Size: 19" × 17" × 8"
Power:
During Switching: 35 watts (28v d-c)
Quiescent State: 7 watts

Figure 4-24. Sequence control unit (SCU).

4.8 PROPELLANT UTILIZATION

ELEMENTS:

- Digital Computer Unit (DCU)
- Tank Probes (LH_2 and LO_2)
- PU Electronics
- Sequence Control Unit (SCU)
- Engine PU Valves

FUNCTIONS:

- Monitors Propellant Quantities during Tanking and Flight
- Adjusts Propellant Feed Ratios to Minimize Residuals at End of Flight

To realize optimum performance in a liquid-fueled bipropellant space vehicle, it is necessary to control both propellants so as to exhaust them simultaneously. Such a simultaneous depletion both minimizes vehicle burnout weight (by not allowing any unusable amounts of one propellant or the other to remain in the tanks) and tends to maximize the mission total impulse (by utilizing all available propellant mass in engine reaction).

Two major factors influence simultaneous propellant exhaustion. The first is accurate calibration of engine mixture ratios, flow rates, and total thrust under flight conditions. This results in an inability to predict the relative propellant masses to be loaded at liftoff. Even if such a prediction were possible, uncertainties in actually determining what has been loaded on board provide the second large error source. As an example, for the Centaur two-burn vehicle, these errors would result in a maximum mass ratio error (3-sigma) of approximately 350 pounds at burnout; resulting in a loss of 350 pounds of payload capability from a mission requiring propellant depletion.

Clearly then, one way to improve total payload capability is to provide some sort of inflight system for propellant management.

The first requirement of such a system (Figure 4-25) for proper propellant utilization (PU) is to accurately measure the ratio of propellants in the vehicle tanks during the entire powered flight portion of a mission. Centaur has a shaped, concentric cylinder capacitor installed in each tank. The space between the inner and outer plates is open to allow LH_2 or LO_2 to fill the probe. When empty, the capacitors are airdielectric types and have their minimum capacitances. As the tank is filled, the propellant (having a higher dielectric constant) displaces the gas and increases the element capacitance. Probe shaping makes the increase in capacitance directly proportional to the increase in propellant mass. Therefore, if we excite the probe from an a-c source, we get an output current or voltage proportional to the empty (dry) capacitance plus the increase in capacitance (Δ) due to propellant mass.

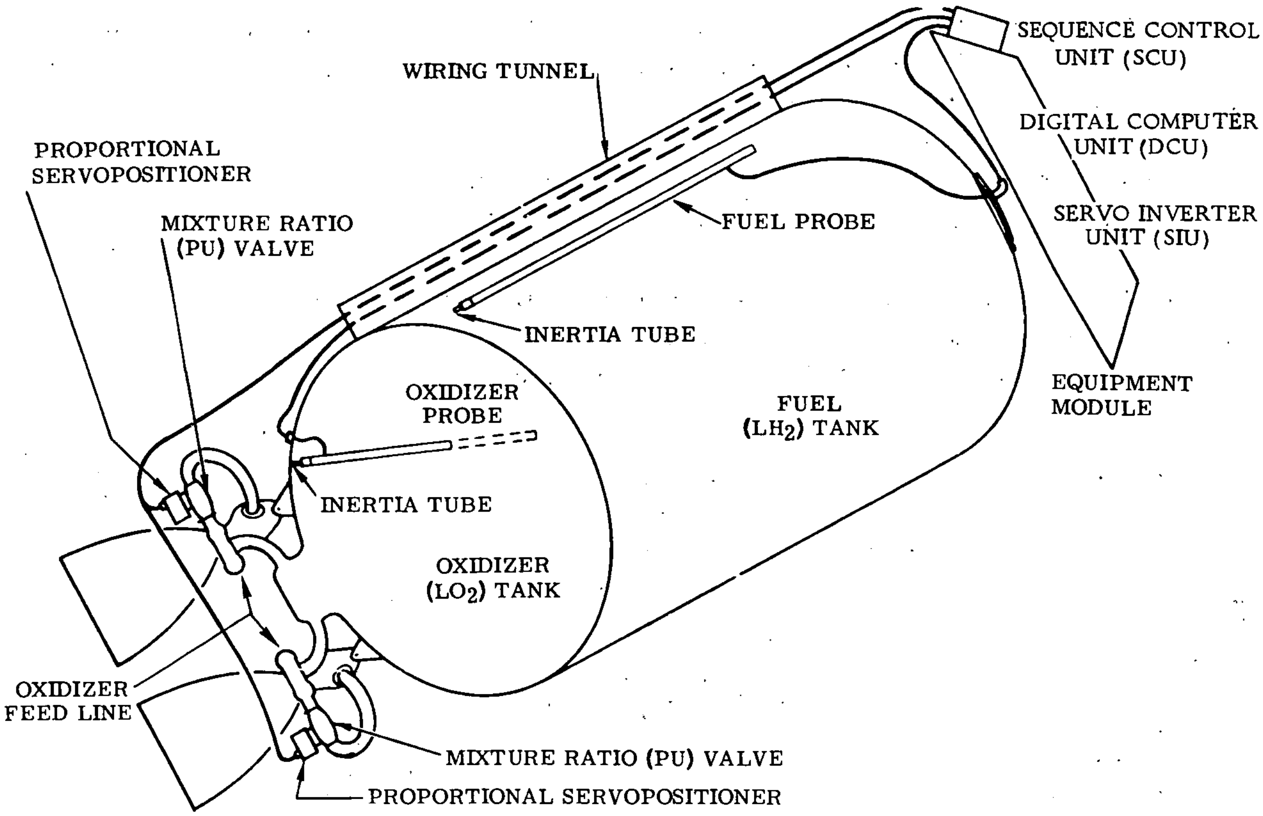


Figure 4-25. Locations of propellant utilization system components.

The second basic function of a propellant utilization system is to control the flow of propellant through the engines to adjust the ratio of the amounts of propellant remaining in the tanks.

4.8.1 SYSTEM OPERATION. In operation, the capacitance probes are excited from an a-c source and used as summing impedances to the summing junction of an operational amplifier. A diode network in series with the probes and the amplifier sees pulsed d-c currents. A third similar "compensation" input is provided to make the "dry" or empty tank LO_2 and LH_2 equal and in the same ratio as the "full" currents. The system excitation voltages, and therefore the error detector amplifier input current are adjusted to provide a zero voltage output when the ratio of propellants in the vehicle tanks is correct.

If the ratio is not correct and an error signal is produced, it is detected by the DCU via one of its A to D converters. In practice, the system's output at zero is not always zero volts and the DCU establishes the initial zero value and uses that for the system baseline. Mission related biases (error bias, coast bias, etc.) are also contained in the software and added or subtracted as required by the mission status.

If the computer decides that a tank ratio change is necessary, it commands a change in the positions of the PU valves on the engines by operating SCU switches through the normal DCU-SCU interface. The valves are motor driven and closed-loop position control is provided by the DCU which receives position feedback signals through two of its A to D converters.

System timing is set so that the tank ratio is measured and checked five times a second and the position feedback loop is monitored 50 times a second.

Amplifiers similar to the error detector amplifier provide independent instrumentation outputs that are indicators of the total amount of LO_2 or LH_2 in each tank. A block diagram of the PU system is shown in Figure 4-26.

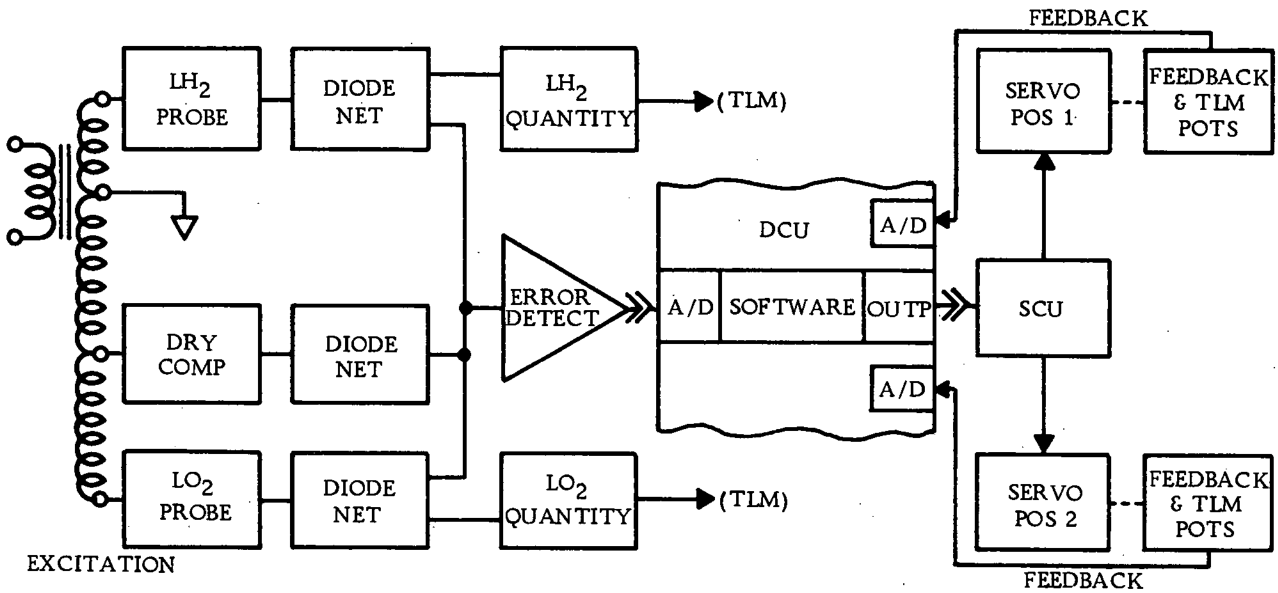


Figure 4-26. Propellant utilization system.

4.8.2 SYSTEM CAPABILITY. A discussion of system capabilities breaks down into three main areas.

a. Detection of propellant masses.

b. Calculation of mass ratio errors.

c. Propellant flow control.

Mass Detection and Ratio Calculation. The dry capacitance and the Δ capacitance for any practical probe are very small. For example, Centaur has the following values:

LH_2 (dry) = 503 pf	LO_2 (dry) = 170 pf
LH_2 (ΔC) = 115 pf	LO_2 (ΔC) = 84 pf

The measurement of small values such as these limits system accuracy insofar as mass detection or ratio detection is concerned. Inaccuracies in the probes also have a small effect on system accuracy. Additions of biases and constants is performed digitally and is very precise compared to other error sources. A summary of those error sources follows:

a. Probe errors

b. Dielectric errors

c. Excitation voltages

d. Temperature stability

e. Amplifier/rectifier errors

f. Valve position errors.

The composites of these errors limits system accuracy to ±40 pounds of LO_2 for a two-burn mission and ±20 pounds of LO_2 for a one-burn mission.

Propellant Flow Control. The engine LO_2/LH_2 mass ratio is nominally controlled at ±11.5 percent. To correct mixture ratio errors, PU valves in the engine LO_2 feed lines are commanded to change the rate of LO_2 usage. The engine performance requirements therefore limit the maximum tank mass error which can be corrected. If the PU valve were commanded to one limit for the entire flight, the resulting correction in tank error would be approximately 2100 pounds of LO_2. When considering other errors and dispersions inherent in the system, the guaranteed total correction capability is approximately 1400 pounds of LO_2.

Growth Potential. Mission peculiar changes (coast biases, error biases, etc.) are accomplished with simple DCU software changes. Other basic tank shape or size changes could be accommodated by reshaping the in-tank probes and using the same basic error detector system, making it applicable to a wide variety of vehicles.

4.8.3 COMPONENT DESCRIPTION. The PU system is made up of four standard D-1 type printed circuit boards plus two small potted modules mounted in the Centaur servo inverter unit (SIU). The SIU is described in Subsection 4.6.

The SIU also contains the electronics for the Centaur powered flight autopilot and the inverter which supplies ac power. Power required by the propellant utilization system is 65 watts. Total SIU capability is 250 watts.

4.8.4 CALIBRATION, CHECKOUT, AND LAUNCH. Checkout and calibration at the unit level in San Diego are accomplished by a special purpose manual test set. While exposed to critical flight environment (temperature and vibration), it is tested to verify basic calibration, gains, filtering, and instrumentation outputs.

Vehicle tests prior to launch are conducted to verify error detector operation and software constants and execution. Zero level constants are determined at this time.

During tanking for launch, gross system operation is verified by monitoring error signal and LO_2 and LH_2 quantity outputs, and by verifying proper output at full tank.

4.9 C-BAND TRACKING SYSTEM

The C-band tracking system determines the Centaur D-1A real-time position and velocity. The system is compatible with the AFETR ground system and provides data to support the range safety requirements.

ELEMENTS:

- Transponder
- Power Divider
- Antennas

FUNCTIONS:

- Provide Data to Determine Position and Velocity for Range Safety Purposes
- Provide Data for Vehicle Performance Evaluation

4.9.1 SYSTEM OPERATION. The C-band tracking system ground transmitters emit (through radar antennas) double RF interrogation pulses spaced six microseconds apart on a frequency of 5690 MHz. The interrogation signal is received at the vehicle by one or both of two antennas mounted on opposite sides of the Centaur tank.

The signal from the antenna is conducted through RF cables to the power divider (Figure 4-27), a tee device that connects the signals together and provides interconnection to the transponder. The transponder responds to the double pulse interrogation after a fixed delay of 2.5 microseconds. The peak power response is a minimum of 400 watts consisting of a 0.5-microsecond pulse on a carrier of 5765 MHz. The output signal, after division by the power divider, is radiated from the same antennas to the radar ground station.

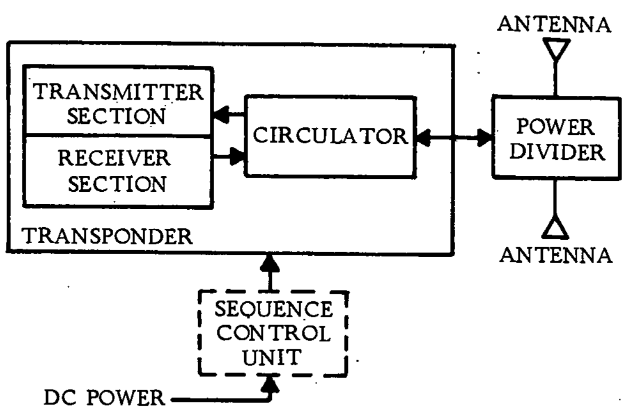


Figure 4-27. C-band tracking system block diagram.

All radar ground stations within range of the vehicle can interrogate the vehicle transponder continuously using a synchronized rapid sequential time-sharing technique. The round-trip transit time of the pulsed signal (compensated for beacon delay) is measured and used by the radar computer to determine the range of the vehicle. The elevation and azimuth angles of the returning signal are transferred to the radar computers and recorders directly from elevation and azimuth shaft encoders to determine vehicle location.

4.9.2 SYSTEM CAPABILITY. The C-band tracking system design includes the following capabilities:

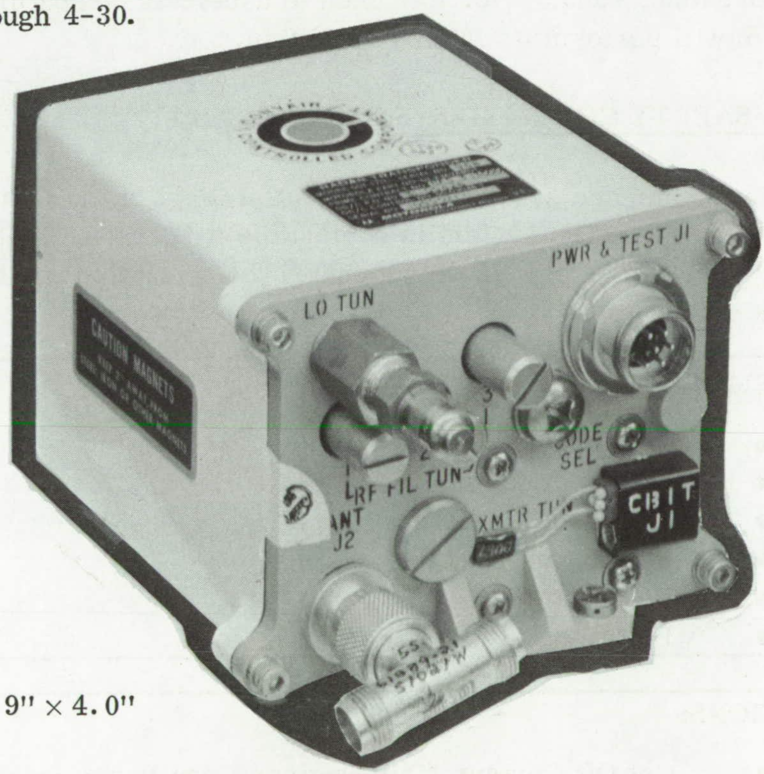
a. It is compatible with the AFETR ground system.

b. Its receiving sensitivity is at least minus 60 dbm and the power output is not less than plus 46 dbm excluding the antenna gain which varies with vehicle aspect angle.

c. It provides an antenna gain of at least minus 10 db over 95 percent of the radiation sphere when used with circular polarized ground stations.

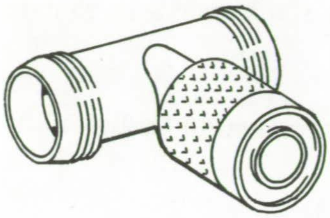
d. It can be continuously interrogated by multiple stations on a time-sharing technique to increase position measuring accuracy without exceeding the maximum system pulse repetition rate which is internally self-limited.

4.9.3 COMPONENT DESCRIPTION. C-band tracking system components are shown in Figures 4-28 through 4-30.



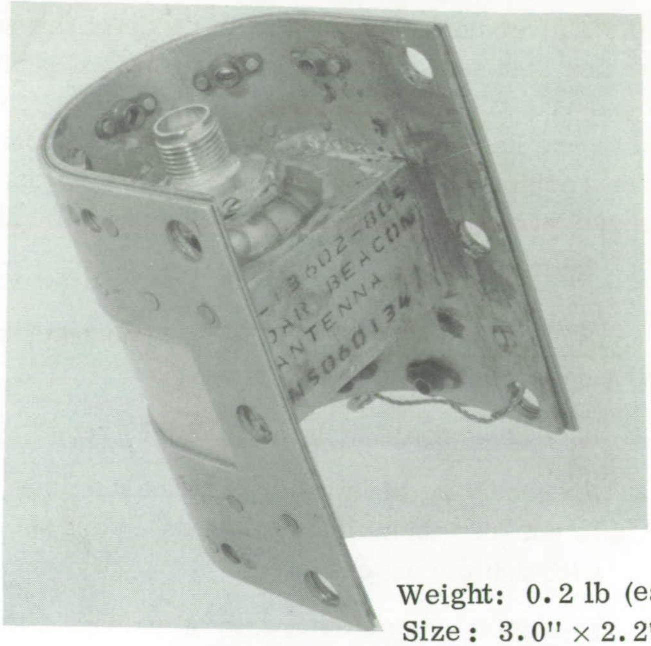
Weight: 3.3 lb
Size: 3.4" × 2.9" × 4.0"

Figure 4-28. C-band transponder.



Weight: 0.1 lb
Size: 1.3" × 1.1" × 0.5"

Figure 4-29. Power divider.



Weight: 0.2 lb (each)
Size: 3.0" × 2.2" × 2.1"

Figure 4-30. Antenna (2).

4.9.4 CHECKOUT AND LAUNCH. System level tests are performed after the individual components are installed on the vehicle. The testing is performed both at San Diego and AFETR to demonstrate proper operation. Open-loop testing is performed at AFETR with ground radar prior to launch to ascertain compatibility and assurance that the system will perform its intended function.

4.10 RANGE SAFETY COMMAND SYSTEM

The range safety command system terminates the flight of the Centaur D-1A on command from the ground. The system is compatible with the AFETR ground system and is completely redundant except in the antenna/hybrid junction combination and the high explosive tank destructor.

ELEMENTS:

- Antennas
- Hybrid Junction
- Command Receivers
- Power Control Unit
- Destructor
- Batteries

FUNCTIONS:

- Cut Off Centaur Main Engines Upon Range Safety Command
- Destroy Tank Structures Upon Command

The functions of the Centaur D-1A Range Safety Command System are as follows:

a. Cut off the Centaur D-1A main engines in response to an RF command (MECO) thus imposing a condition of zero thrust.

b. Destroy the LH_2 and LO_2 tank structures in response to an RF command (destruct) and thereby disperse the propellants.

4.10.1 SYSTEM OPERATION. As the Centaur vehicle follows its prescribed flight path, the RSC RF carrier (416.5 MHz) is transmitted from successive AFETR ground transmitting stations. The carrier is picked up by one or both of the two antennas which are mounted on opposite sides of the Centaur tank. A block diagram of the RSC system is shown in Figure 4-31.

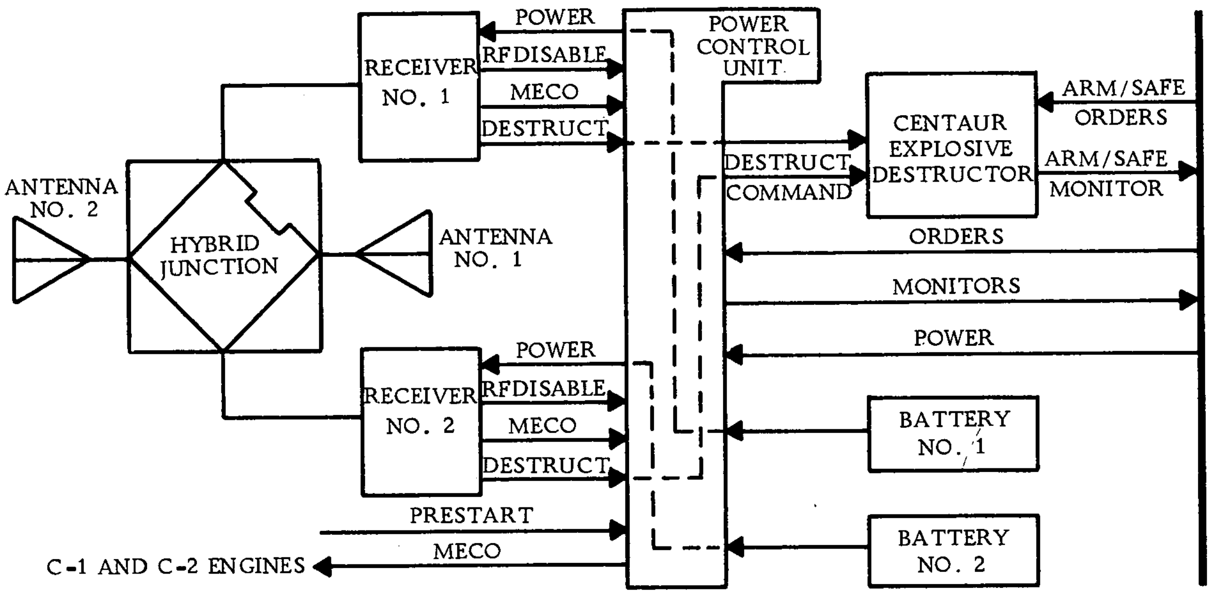


Figure 4-31. Centaur range safety command system block diagram.

The signal is conducted from the antennas through RF cables to input ports of a hybrid junction. The two output ports of the hybrid junction are connected to the two command receivers.

When the carrier is modulated with command tones, the tones are demodulated within the receivers and converted to 28v d-c commands: MECO, DESTRUCT, and RF DISABLE.

The command tones used are: tone 1 (7.5 kHz), tone 2 (8.46 kHz), and tone 5 (12.14 kHz). Tones 1 and 5 constitute the main engine cutoff (MECO) command, tones 1 and 2 (preceded by the MECO command) constitute the destruct command, and tones 2 and 5 provide the RF disable command.

The three commands are routed from the receivers into the power control unit where they are conveyed by relay switching circuits to their respective destinations: (1) the MECO command to the engine prestart circuits (via the Sequence Control Unit), (2) the destruct command to the destructor, and (3) the RF disable command to the power changeover switches (within the power control unit) to remove power from the Range Safety Command System. Power is provided by two batteries that are connected through the power control unit to the receivers and the command circuits.

4.10.2 SYSTEM CAPABILITY. The range safety command system has the following significant characteristics.

a. The system is separate and independent of any flight termination system used on the booster stage.

b. The system is compatible with the AFETR ground system.

c. The system is redundant except for the antenna/hybrid junction combination and the destructor.

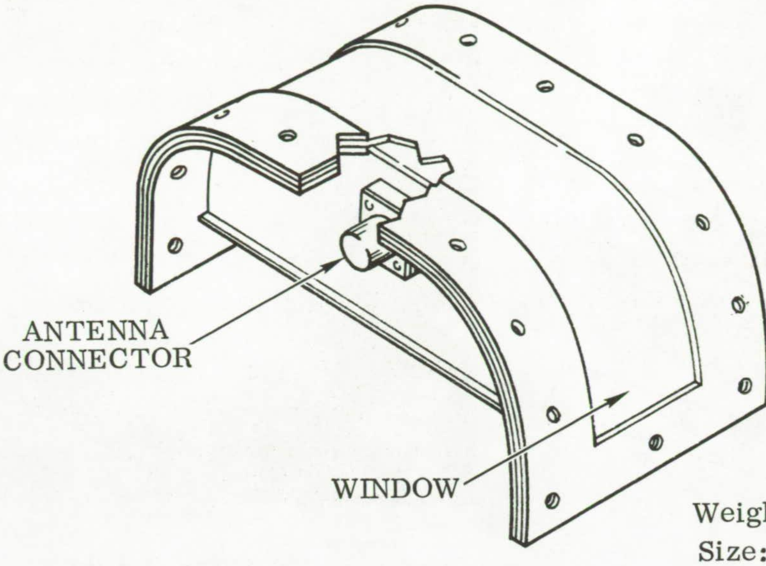
d. The system provides adequate radio command coverage over at least 95 percent of the radiation sphere. To this end, the system is capable of operating with an electromagnetic field intensity which is 12 db below the intensity provided by the Range.

e. A payload destruct capability is also available and may be used when the mission requires it. For payload destruct, a parallel destruct output from the power control unit is wired to a safe/arm initiator. This initiator contains two independent, electrically-initiated detonators which initiate a pyrotechnic chain to a payload destruct charge or a mild detonating fuse. The destruct charge may be conical shaped type supplied by Convair Aerospace or it may be furnished by the payload contractor.

4.10.3 COMPONENT DESCRIPTIONS. The range safety command system (Figure 4-31) is made up of two antennas, a hybrid junction, two command receivers, a power control unit, a destructor, and two batteries. Figure 4-32 is a sketch of the antennas and Figures 4-33 through 4-37 are photos of the remaining components.

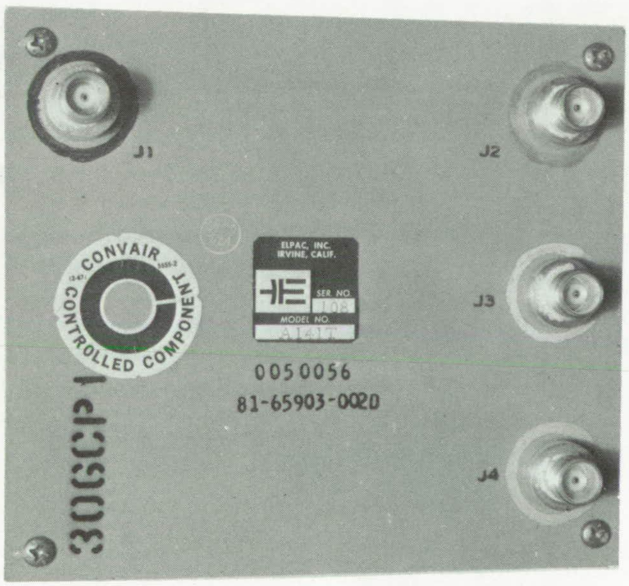
4.10.4 CHECKOUT AND LAUNCH. System level tests are performed after the individual components are installed on the vehicle to ascertain that the system will operate properly and perform its intended function. The tests are performed both at San Diego and at AFETR. An inert destructor is used in all San Diego tests and prior to the final launch countdown at AFETR.

On launch minus one day, the live destructor is installed and system testing from that point is accomplished with the live unit.



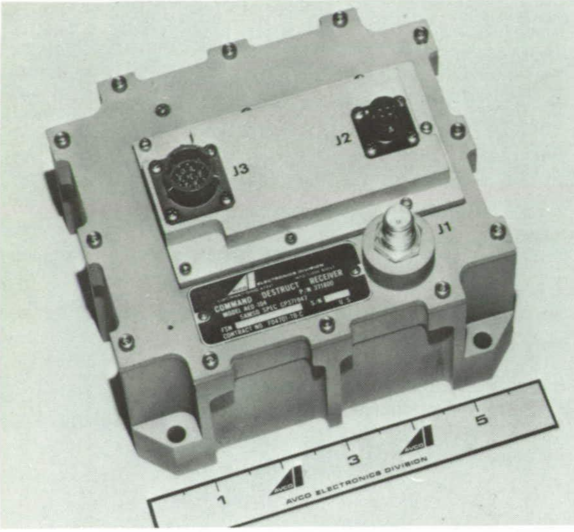
Weight: 1.4 lb (each)
Size: 4.8" × 4.0" × 8.0"

Figure 4-32. Antenna.



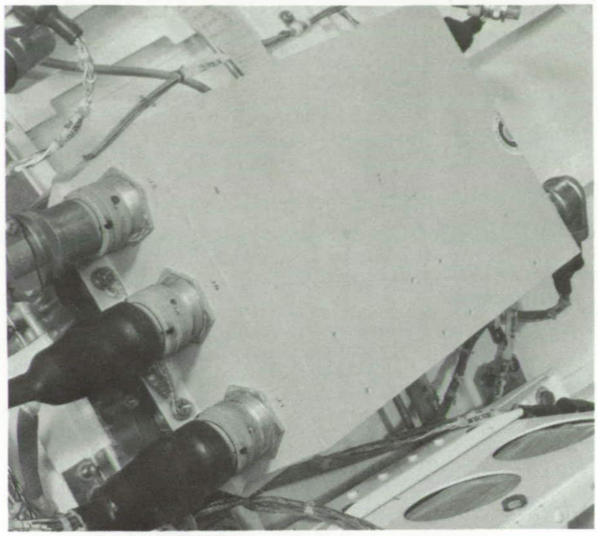
Weight: 0.5 lb
Size: 5.6" × 5.0" × 1.3"

Figure 4-33. Hybrid junction.



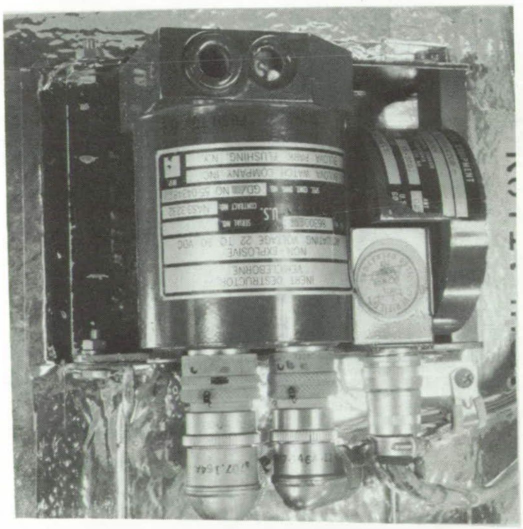
Weight: 4.5 lb
Size: 6.0" × 5.1" × 4.5"

Figure 4-34. Command receiver.



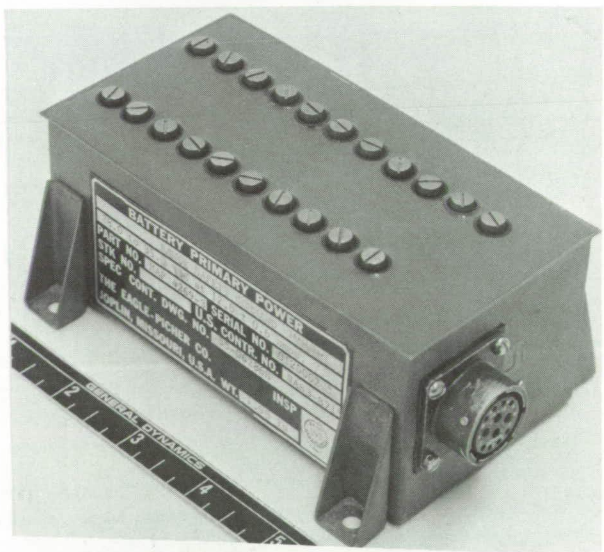
Weight: 9.8 lb
Size: 8.3" × 12.3" × 3.5"

Figure 4-35. Power control unit.



Weight: 5 lb
Size: 7.0" × 5.0" × 5.5"

Figure 4-36. Destructor.



Weight: 2.1 lb
Size: 3.7" × 2.5" × 5.2"

Figure 4-37. Battery.

4.11 ELECTRICAL POWER SYSTEM

The Centaur D-1 vehicle electrical power system consists primarily of a 28v d-c main vehicle battery and its associated power distribution system. Three separate buses distribute power from the main vehicle battery through the electrical wiring harnesses to the various vehicle loads. A power changeover switch distributes power from a ground power supply prior to flight.

ELEMENTS:
• Main Vehicle Battery
• Power Changeover Switch
• Harnessing
• Servo Inverter Unit for a-c Power

FUNCTIONS:
• Provide d-c and a-c Power
• Sequentially Apply d-c when on External

4.11.1 SYSTEM OPERATION. The electrical power system (Figure 4-38) uses the main vehicle battery to supply d-c power to the vehicle electronic systems. The main vehicle battery supplies power to three separate busses in order to isolate equipments which tend to be generators of electromagnetic interference from equipment which may be sensitive to electromagnetic interference.

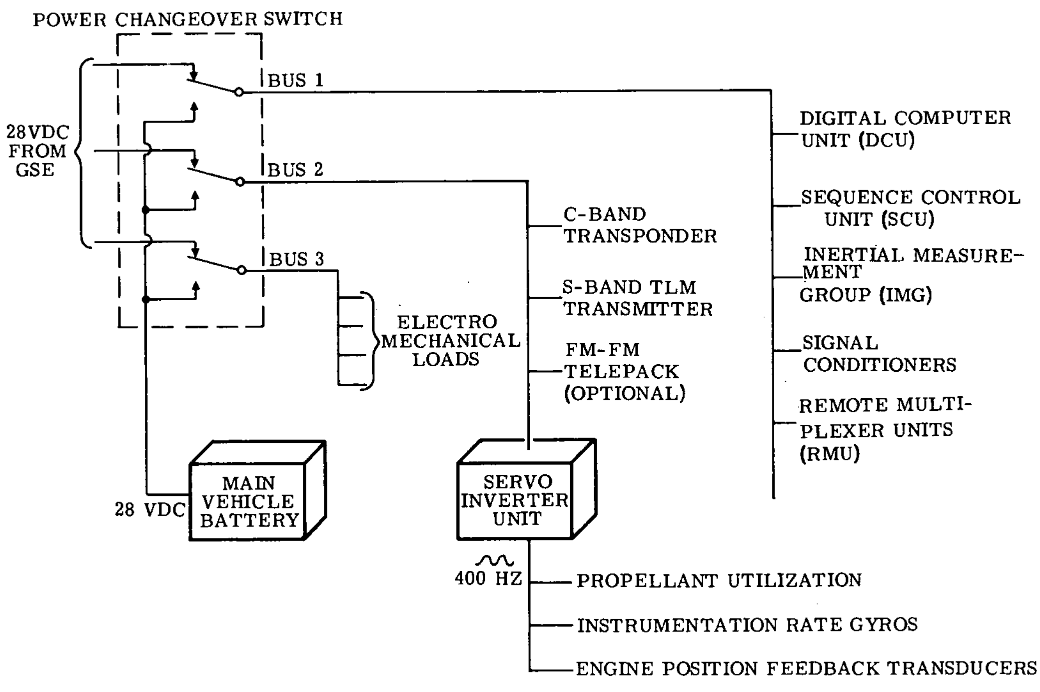


Figure 4-38. Electrical power system.

Bus 1 supplies power to the digital computer unit, sequence control unit, multiplexers, and signal conditioners. Bus 2 provides power to the S-band transmitter, C-band transmitter, propellant utilization system, servo inverter unit, and the FM/FM telepak system. The loads on Bus 1 and Bus 2 are primarily loads that tend to be sensitive to electromagnetic interference. Bus 3 supplies power to switching loads such as solenoids, relays, and motors. This bus carries loads which tend to be generators of electromagnetic interference.

Only those systems not critical for vehicle flight, such as the C-band transmitter and FM/FM telemetry, are fused. The three buses are common at the battery or ground power supply, thus minimizing the need for a filter.

A single-phase inverter in the servo inverter unit provides 400 Hz, 26v a-c needed to supply power to the instrumentation rate gyro unit and the propellant utilization servo positioners. The inverter also supplies 115v a-c for use by the propellant utilization servopositioners.

Vehicle power is provided from a ground source prior to flight. The power changeover switch is activated prior to launch and connects the internal power source (main vehicle battery) to the power distribution system. The power changeover switch features a make-before-break contact arrangement to ensure uninterrupted power to the loads during power changeover.

The electrical power system employs a single-point ground. Current monitoring for individual system usage is provided at the single-point ground bus in the sequence control unit.

Prior to power changeover, the electrical power distribution system receives power from a ground installation source. At approximately T minus 4 minutes the power changeover switch is activated and power changeover is accomplished, which connects the internal power source to the electrical distribution system. The vehicle loads then derive power from the main vehicle battery. Due to the nature of the power changeover switch, no interruption of power occurs. The main vehicle battery then continues to supply power throughout the remainder of the mission.

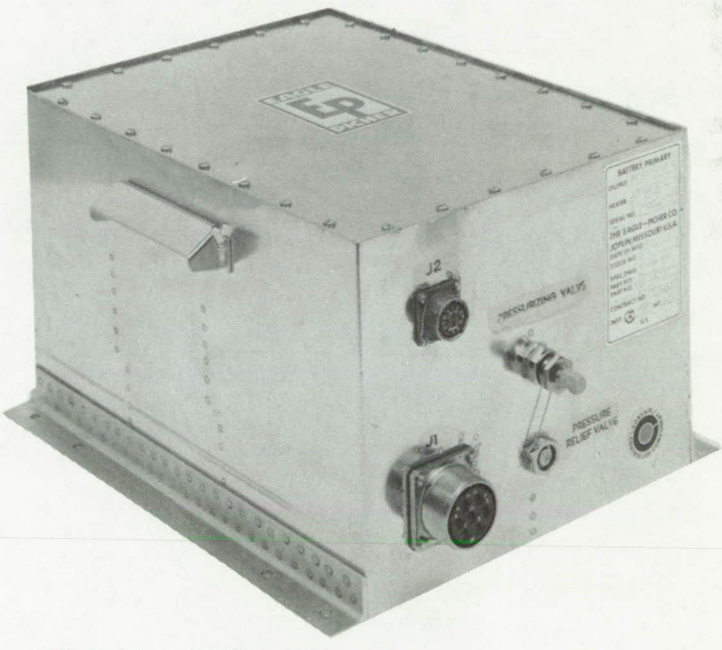
4.11.2 SYSTEM CAPABILITY. The main vehicle battery is capable of 100 ampere hours at 26 to 30v d-c output for any load between 50 and 80 amperes. Transient limitations are 37v d-c maximum open circuit voltage and 20v d-c upon load application. Within 30 milliseconds the output voltage regulates to within 25 to 32v d-c and from 26 to 30v d-c within 100 milliseconds. For a period not exceeding 30 seconds, the main vehicle battery is capable of delivering 100 amperes at 26 to 30v d-c output. During flight the average loads vary with maximums ranging between 55 and 75 amperes. Design also provides for a 150-ampere-hour battery to be used in lieu of the 100-ampere-hour battery for different Centaur D-1 configurations. The 150-ampere-hour battery has similar specifications to the 100-ampere-hour battery.

The power changeover switch is capable of carrying 65 amperes per pole continuously at 28v d-c with a voltage drop of less than 100 millivolts on each of its four poles. The switch also has a set of single pole single throw, break-before-make contacts capable of carrying 7 amperes continuous with less than a 700-millivolt drop. These contacts are primarily intended for signal type circuits, but are not used in the present Centaur D-1 configuration. Total maximum transfer time for the switch is 170 milliseconds.

4.11.3 COMPONENT DESCRIPTIONS

Main Vehicle Battery. The battery consists of 19 primary type, dry charged, manually activated, silver-zinc battery cells (series connected) enclosed in a two-piece stainless steel canister. A 0 +5-volt analog signal to the telemetry system is provided by two isolation resistors connected to form a voltage divider circuit; this telemetry signal corresponds to a battery voltage of 0 to +35v d-c. A temperature transducer mounted within the battery provides internal temperature monitoring from 0 to 200°F. Single-phase 115-volt, 400-Hz or 115-volt, 60-Hz power is required for the battery heater system during prelaunch activities. The battery is shown in Figure 4-39.

Power Changeover Switch. This switch employs a four-pole double-throw contact arrangement for the power circuit. The switching is activated by a 22 to 32v d-c signal to the driving motor of the switch. The switch weighs 2.65 pounds and has an electoless nickel finish. It is hermetically sealed, and pressurized to one atmosphere with 95 percent dry nitrogen and 5 percent helium. It is physically contained within the SCU.



Weight: 66 lb
Size: 13.2" × 11.7" × 9.7"

Figure 4-39. Battery.

Electrical System Harnessing. The electrical system harnessing consists of H-film insulated wires. The harnessing is physically segregated into three basic classifications: wires that connect equipment that tends to generate electromagnetic interference, wires that may be sensitive to electromagnetic interference, and wires that do not fall into any of the above categories. The third group, which is neither sensitive to electromagnetic interference nor tends to generate interference, is routed between the other two groups. This routing provides some isolation and minimizes interaction between the various electronic systems.

Servo Inverter Unit (SIU). The SIU is described in Subsection 4.6.

4.11.4 CHECKOUT AND LAUNCH. The electrical power system is monitored prior to flight at several key areas including the main vehicle battery and power changeover switch. It is checked out prior to launch by switching to internal power and using a battery simulator in place of the main vehicle battery. At approximately four minutes before launch the complete internal power system is activated. This connects the main vehicle battery into the onboard power distribution system allowing time to determine that the power system is working properly before liftoff. The critical restraints on the main vehicle battery, output voltage and internal temperature, are constantly monitored.

5

DCU SOFTWARE

As discussed in Subsection 4.1, the digital computer unit (DCU) plays a significant role in the Centaur D-1A astrionics system. The software is an integral part of the DCU. The software design and operational characteristics are described in this section.

5.1 INTRODUCTION

Two of the design concepts that influenced the selection of the Centaur D-1A computer were: (1) a random access memory for programming efficiency and (2) capacity for a complete command and control system.

ELEMENTS:

- Digital Computer Unit
- Flight Software Program
- Preflight Software Program

FUNCTION:

- Provide Command and Control for Many Vehicle Systems

The Centaur D-1A software incorporates many functions done by hardware on Centaur D. The Centaur D-1A capability is illustrated in Figure 5-1. This reduction in hardware allows the vehicle configuration to remain static while adding in the software the design flexibility for mission-peculiar requirements. Particularly, software has taken over hardware tasks which in the past have required these mission-peculiar modifications.

The Centaur D-1 computer, built by Teledyne, has a 16,384-word memory, over five times the memory capacity of the D Centaur computer. In addition, the Teledyne computer has a variety of I/O to serve Centaur D-1 systems as shown in Figure 5-2.

To take full advantage of the new computer, and to ensure the maximum benefits of the new Centaur D-1 Astrionics System, a new flexible software system was required.

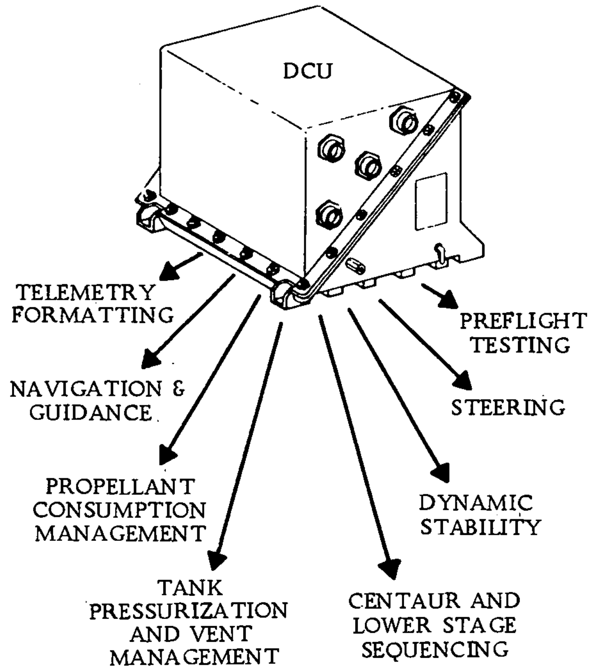


Figure 5-1. Software provides total vehicle command and control.

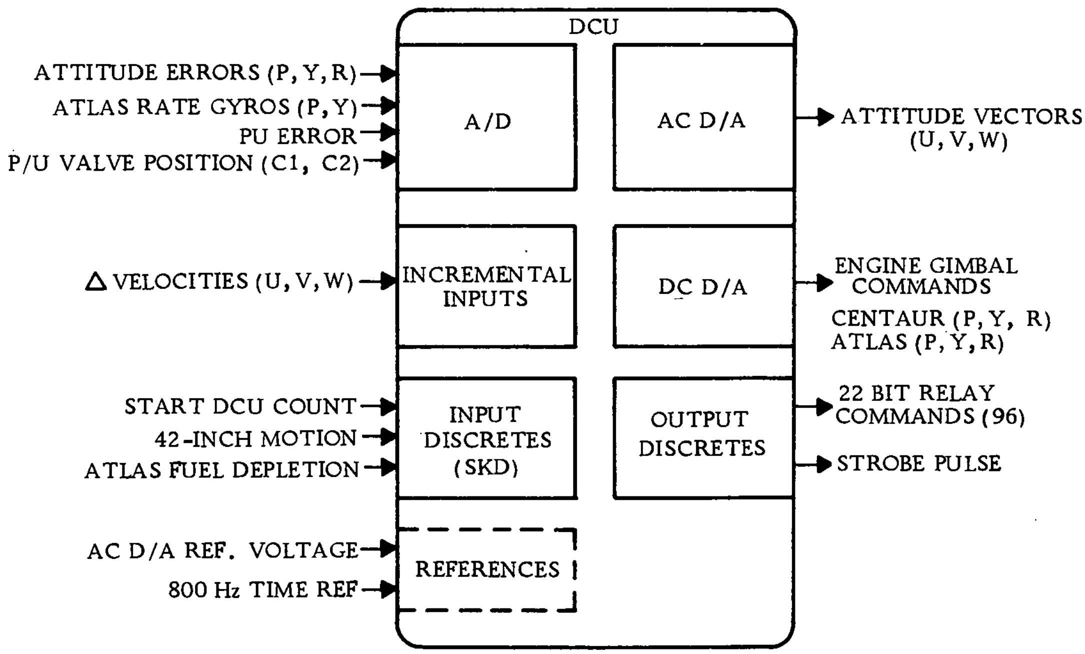


Figure 5-2. DCU input/output to serve D-1 systems.

5.2 SOFTWARE OBJECTIVES

The D-1 software was designed to satisfy specific objectives in the areas of cost, reliability, launch simplicity, response, and resiliency.

To minimize cost, many of the mission- and vehicle-peculiar changes formerly requiring hardware are now implemented via software.

To realize high reliability, the software was designed to simplify checkout and ensure an error-free flight program.

To achieve launch simplicity, the software design allows reasonable last minute changes in applicable areas and test philosophies.

For quick response, or to reduce lead time, the software is constructed in modular fashion. Thus, modular changes and checkout can occur without disturbing the configuration of other modules.

To provide resiliency, the software is designed to remain intact and functioning in the unforeseen event of failures in the external system hardware. Its task is to achieve maximum flight success in spite of system failures.

SOFTWARE DESIGN GOALS		
•	Lower Cost	Minimum Cost for Changes (Recurring Cost)
•	Reliability	100% Error Free
•	Resiliency	Forgiving of External (Vehicle) Failures
•	Flexibility	Minimum Turnaround for New Missions

5.3 SOFTWARE DESIGN CONCEPTS

The basic D-1 software concepts have been developed to achieve the following objectives:

Modularity. A modular software concept fulfills the requirements for a cost effective and flexible software system. The concept classifies software into two categories: An executive software system that remains unchanged through all mis sions, and a set of mission- or vehicle-peculiar task modules that can be selected from a library and adapted for the current mission.

This modular system is truly flexible. The task modules can be scheduled by the executive at different frequencies during the flight. They can be turned off or reactivated for different phases of flight, and interrupted at any time during their operation. Since the modules do not communicate with each other, but only through the Executive, they are assured of consistent sets of data. The system is also readily changeable. Assuming memory and duty cycle are available, new modules can be added or mission-peculiar modules exchanged with each other.

Change flexibility is enhanced by subdividing the task modules into subroutine blocks whenever possible. Program changes are inserted at the module (or subroutine) level, and checked at the module and integrated program level (all task modules operating together as a system). Flexible modularity is illustrated in Figure 5-3.

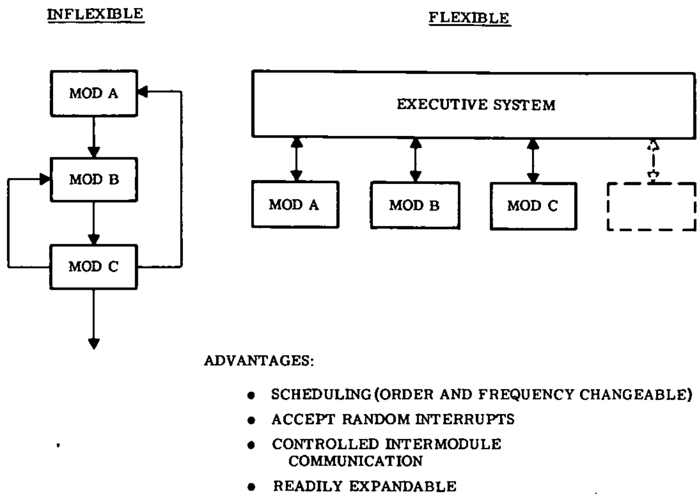


Figure 5-3. A flexible kind of modularity.

<u>Independence from Interrupts</u>. Both real-time and vehicle telemetry interrupts will occur during a flight. By design, the occurrence of an interrupt will cause the program to suspend whatever it is doing, and force it to a preset address in order to execute the interrupt subroutine. Upon completion of the interrupt subroutine, the program returns to the interrupted address and resumes whatever it was doing.

The D-1 software is designed so that <u>any</u> combination of two or more modules can be in a simultaneous state of interrupt; furthermore, the interrupt is allowed at <u>any</u> location in the modular program. Thus, the modular task programs are completely independent of the interrupts, and each one can be coded as if it were the only module in the computer.

This "independence" concept is implemented by task scheduling and real-time interrupt system service techniques, at the DCU level, and by appropriate design of support software which automatically assembles and assigns absolute addresses to all of the modules only at the final program level.

<u>Decentralized, Parallel, Design and Checkout</u>. Functional tasks are designed as separate software modules, and are developed and checked out in parallel (Figure 5-4). The checkout of the software is subdivided into the task level and integrated level. The software system structure is designed general enough so that after the first integrated checkout has occurred, revised or new modules can be incorporated with minimum effort. Once a revised module is completely checked out, it is added to the modular library.

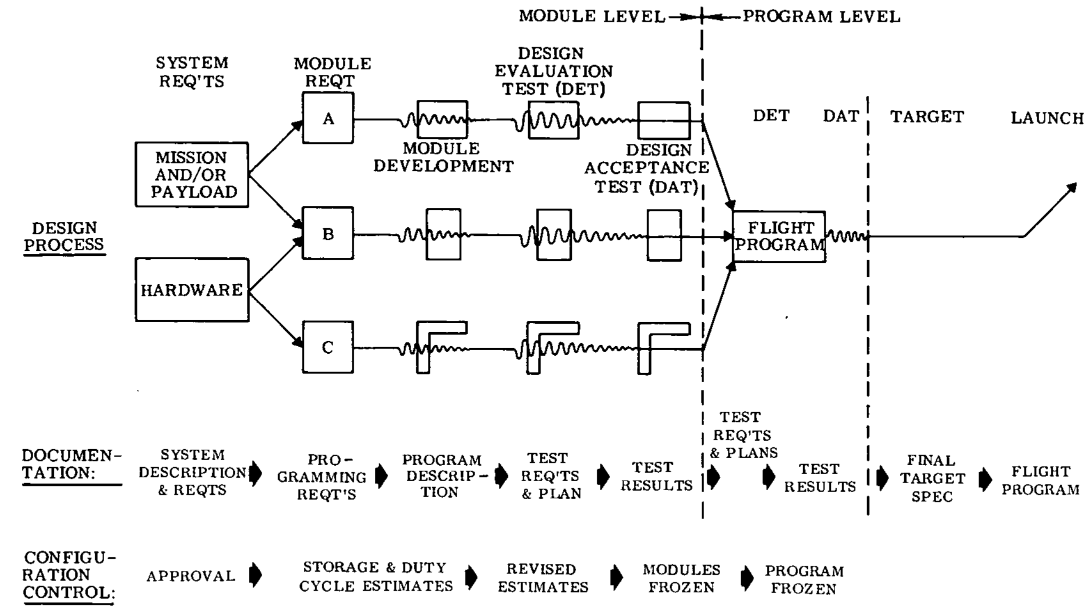


Figure 5-4. Software development — parallel module design and checkout.

The decentralized checkout concept minimizes software reaction times to changes, and promotes maximum reliability by providing detailed engineering visibility at the task and integrated task levels.

Contingency Software. In the event of certain external equipment failures (such as, unscheduled thrust termination, or failure to start a stage) the software provides the capability of selecting reasonable alternative strategies.

For nonstandard environments (such as a large steady-state drop in thrust level) the software senses the environment and makes appropriate adjustments to the trajectory. The recovery techniques are designed so that the mission is achieved within the performance capability of the launch vehicle. Every conceivable extreme nonstandard environment cannot be protected; however, a reasonable balance is achieved between software complexity and protection attained.

FLIGHT SOFTWARE DESIGNED TO BE FAILURE TOLERANT

- Accommodates Any One H_2O_2 Engine Out
- Forgives Transient Accelerometer Failure
- Advances Flight Phase for Early Engine Shutdown
- Maintains Integrity Under Abnormal Dispersions
  - 50% Pitch Program Errors
  - 50% Launch Azimuth Errors
  - 30% Thrust Errors
  - 30% I_{sp} Errors
  - 10% Thrust Sinusoidal Oscillations

Flexible Ground Computer Interface. The airborne computer operates in conjunction with a ground computer to perform many vehicle and avionics systems tests. For this preflight mode of operation, a modular DCU software concept is again used, only in this case the modules are called "tenants". The tenant programs are loaded from the ground computer to the airborne computer. A DCU tenant program operating in conjunction with its corresponding ground computer tenant, services each vehicle or avionics systems test.

The tenant regions are sectors of temporary storage (in the DCU) allocated to the test programs. Flexibility is such that any test program can be loaded into any tenant region of the DCU. The tenants work in conjunction with the resident software control system which allows each tenant to be interrupted. Thus, the tenants have the same independence from the interrupts as discussed previously.

The communication link between the ground and airborne computers also allows flexibility. A common format is used to communicate data, programs, or special requests (such as for telemetry of a specified memory cell) from the ground computer to the airborne computer. The downlink, or telemetry channel, is formatted so that any word desired can be communicated from the airborne to the ground computer.

Documentation. A management and engineering interface aid developed for Centaur D-1 software has been a flexible, decentralized, documentation system. For each software function there is a document which defines the design requirements and, after the module is developed and checked out, a document which provides a thorough functional description.

SOFTWARE OBJECTIVES ARE MET THROUGH THESE FEATURES

• Modularity	Module/Library Approach
• Random Interrupts	Programs and Subroutines Fully Interruptable
• Multi-functions	Total Vehicle Control and Information System
• Resilient Software	Software Adjusts to Nonstandard Conditions
• Decentralized Checkout	Parallel Module Development and Validation
• Dual Level Checkout	Checkout and Validation at Both Module Level and Program Level

5.4 EXECUTIVE SYSTEM

The Centaur D-1 flight program consists of subprograms in two basic categories, system routines and functional tasks. Systems routines which comprise the Executive System control program execution, manage data flow between tasks, and service the computer I/O functions. The modular functional tasks, such as navigation and guidance, perform the computations which satisfy the computer's responsibility with respect to its external world.

Clock pulses which occur at 20 millisecond intervals generate a real-time interrupt. The task currently in progress is stopped and program control given to the Executive. The Executive first services the DCU input/output functions, then it solves all the 50-Hz tasks, and finally routes the program execution to the appropriate lower frequency task.

Interrupts. Two interrupts will normally occur during flight, real-time and telemetry. The occurrence of an interrupt causes a transfer to the interrupt processor. The interrupt processor saves the variables that were being calculated by the interrupted program. The real-time interrupt occurs at precisely a 50-Hz rate (it also functions as the "software clock"); the telemetry interrupt occurs at approximately a 1000-Hz rate.

"Power Off" followed by "Power On" are two other interrupts that conceivably could occur during flight for some unplanned reasons. If these unexpected interrupts occur, the software is designed to continue functioning in a reasonable manner.

If the telemetry interrupt occurs, the program executes a subroutine which results in the emission of a telemetry word. For the real-time interrupt, the program executes the "real-time interrupt" program, a system program that provides input/output servicing, and scheduling of task programs.

Task Schedule/Task Table. The task scheduler scans the task table which contains three entries for each task. These entries are:

1. Normal task entry address.
2. Required task solution period.
3. Task start time.

The task scheduler uses this information to execute all software task programs at their required frequency and within the required time interval. Figure 5-7 shows that the task scheduler either scans the table from the top or continues the scan, depending on whether the return was via the interrupt entry or the task completion entry. Interrupted tasks have the highest priority.

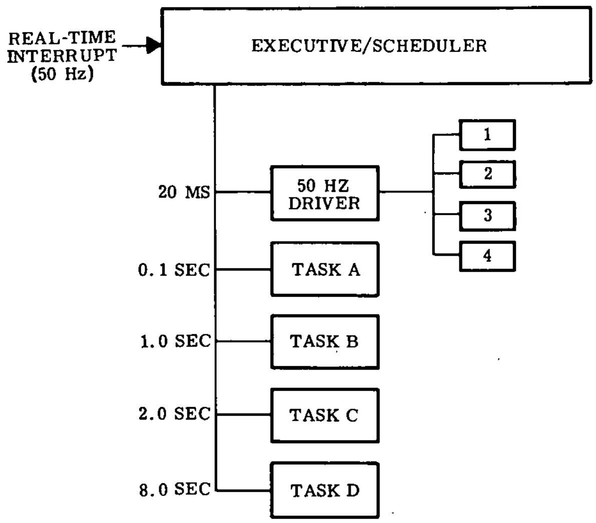
Entry From	Action	Result
Real-Time Interrupt (RTI)	Scan Table From Top	Interrupted tasks are resumed or tasks are started, whichever occurs first.
Completed Task	Continue Scan Where Left Off	Same as above.

Figure 5-5. Task scheduler.

Task Organization. A "tree" structure is used to organize the tasks (Figure 5-6). The tasks are grouped according to execution priority requirements. Each task on the "tree" must appear as an entry in the task table (except for the 50-Hz tasks).

A 50-Hz driver is used to control the execution of the 50-Hz tasks in order to save duty cycle. Thus, only one call to the 50-Hz tasks appears on the task table.

The Executive System can also turn tasks on and off simply by modifying data in the task table. The exception is the 50-Hz tasks. Since only the driver call appears in the task table, the enabling/disabling of 50-Hz tasks is controlled internally by the driver.



- EXECUTIVE ENSURES THAT ALL TASKS ARE PERFORMED ONCE PER TASK PERIOD
- EXECUTIVE ALLOWS ANY TASK TO BE INTERRUPTED WITHOUT DEGRADATION

Figure 5-6. Typical "tree" organization.

Adaptation to Real Time. Table 5-1 summarizes the software concepts used to adapt to the real-time problem. Note that time is accurate but is known only once per 20 milliseconds. If time is required by the software task to any resolution finer than 20 milliseconds, an instruction count is required or the countdown register can be implemented. The countdown register facilitates the issuing of discretes to a time resolution of 1.25 milliseconds.

Since the task programs will be coded such that they are not vulnerable to random interrupts, and since they will be minimally sensitive to start-time changes, it follows that the task programs can be coded separately as if there were no time constraints. The term "insensitive to time changes" means that if a task execution is skipped because of a temporary overload, the task will continue to function properly.

Table 5-1. Adaption to real time.

Real-Time Reference:	50-Hz Clock Interrupt
Engine Cutoff:	Precise 800-Hz Clock Countdown Register
Task Programs:	Insensitive To Being Interrupted Insensitive To Time Changes
Task Scheduler:	Uses Task Table To Determine Which Task Is Executed Task Priority Determined By Its Location On Table Tasks Are Executed At Required Frequencies Reschedules Tasks In Case Of Temporary Overload Turns Tasks "On" Or "Off" As Required Changes Task Frequencies As Required

The task scheduler will compare actual start time with desired start time and recalculate a new start time if a temporary overload occurs.

Executive System Allows Time Sharing Of DCU Tasks

Input/Output Servicing. This subroutine is executed every time a real-time interrupt occurs; it interfaces DCU software with the I/O devices. Since the task program interface with the I/O servicing software is fixed, the functional task software is independent of changes in I/O device assignments. The input software reads the attitude signals from the resolver chain and the most recent 20-millisecond accumulation of velocity pulses. The output software stores the desired attitude vectors into the resolver chain input locations and issues the discrete register bit pattern to external equipment (if required).

Telemetry Formatting. A PCM telemetry system is used that results in the DCU words being intermingled with all other vehicle telemetry data. The DCU data consumes approximately 10 percent of the PCM channel capacity, assuming that approximately 1000 24-bit DCU words are telemetered per second.

Table 5-1 shows the tasks performed by the formatting function in the DCU. The "fast" portion establishes the frame marker and counter, approximately 16 out of the 18 data words (words 19 and 20 are excess "dummy" words to take up the slack in the relative frequencies of the DCU and PCM interrupt). This "fast" portion is actually performed as part of the real-time interrupt service subroutine (I/O Service). The "fast formatter" supplements the I/O service by assigning words to the remaining two slots. To do this it selects words from the lower frequency task buffers as shown in Table 5-1.

The "slow" formatter moves the low frequency task data from the "current" to the "previous" buffer, thus preparing these buffer tables for use by the fast formatter. Data from each task is moved at a different frequency; which means that the slow formatter need only be executed at the highest non 50-Hz task frequency.

Data Communication. Intermodule data communication is accomplished by employing a Data Management Module (DMM). The DMM is a subroutine that has multiple entry points. The basic function of the DMM is to load the task's input or output buffer with the appropriate dynamic data. These data are defined in the input/output requirements of the task's documentation.

The buffer fill operation is protected by disarming the real-time interrupt (RTI) during this vulnerable phase. Since the disarm/arm sequences occur under the control of the executing task, a coherent set of the latest available input/output data is always guaranteed. Furthermore, these buffers remain static throughout the execution period of the task (i.e., until the executing task requests the DMM to refresh them again on the subsequent cycle).

The overall procedure is shown by Figures 5-7 and 5-8. Here, XXIN and XXOUT denote the entry points in the DMM used by Task XX for management of its input and output data. The section of the DMM which provides the data management for Task XX is structured as shown in Figure 5-8.

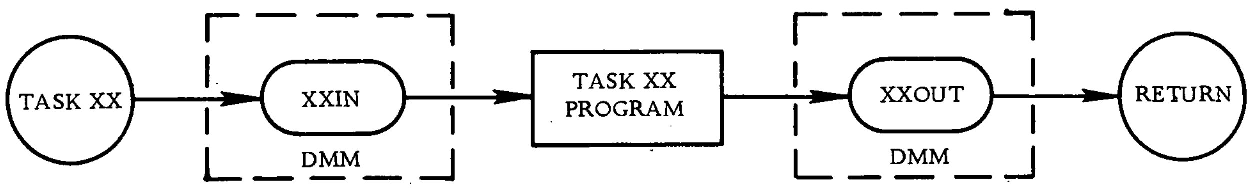


Figure 5-7. Data management module input and output data handling.

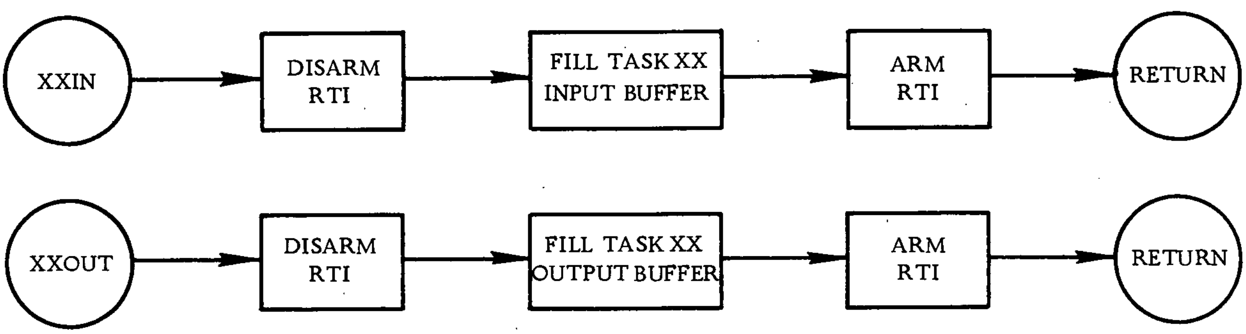


Figure 5-8. Entry points in DMM used by Task XX.

Task Table Modifications. There are occasions during a flight when it is necessary to enable or disable certain tasks, such as enabling the powered guidance task just after BECO. Appropriate use of the task table can facilitate this requirement and obviate the need for extra branches and flags. Tasks can be enabled or disabled by modifying the task start time. The frequency of execution of a task is changed in flight by modifying that number on the task table. Also, a different portion of the task can be enabled by changing the task entry address. In general, task turn on, or turn off, or task frequency change can be achieved by task table modification.

5.5 FUNCTIONAL TASKS

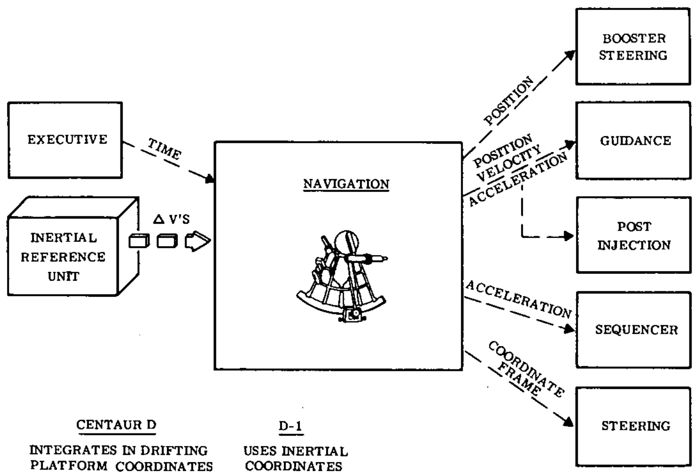
The functional tasks performed by the DCU are coded in program modules. These modules are listed in the task table from which they are called by the Executive for operation at the proper time to ensure correct module frequency. Some of the modules, e.g., Navigation, operate throughout flight, while others are scheduled for only certain phases, e.g., Powered Autopilot. The modules do not interface with each other. Data flow is controlled by the data management portion of the Executive.

A brief description of each module follows.

NAVIGATION

Function. Furnish position, velocity, and acceleration data to guidance.

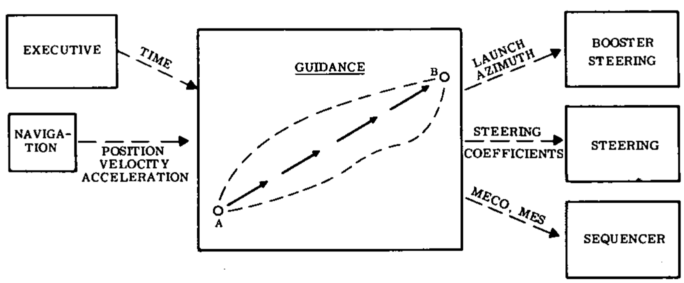
Method. Integrate in true inertial coordinate system, having pre-viously converted for known platform drifts.



GUIDANCE

Function. Determine steering coefficient data for optimizing the trajectory and furnish engine cutoff time to the sequencer.

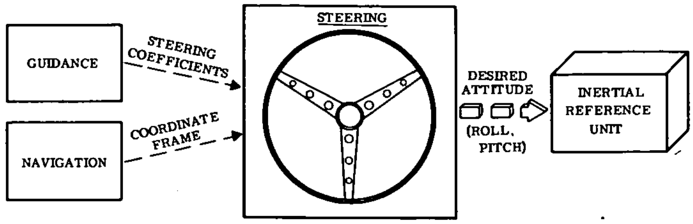
Method. Assumes a near-optimum linear tangent steering law in pitch and a calculus of variation steering law in yaw.



STEERING

Function. Furnish the desired vehicle attitude to the platform resolver chain.

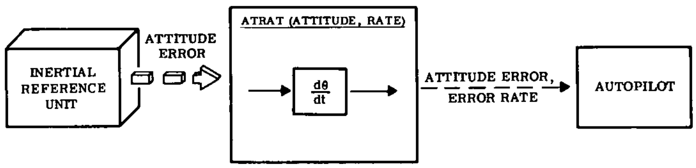
Method. Computes the desired vehicle attitude from the guidance-supplied steering coefficients.



ATRAT (ATTITUDE RATE)

Function. Furnish rate information to powered and coast phase autopilots.

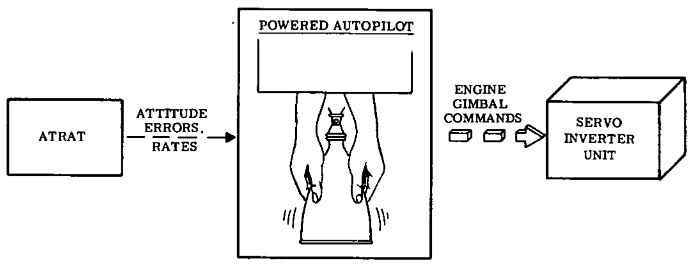
Method. Computes the time derivative of the attitude error signal.



POWERED PHASE AUTOPILOT

Function. Maintain control stability during main engine firings and control the vehicle axes to the desired attitude.

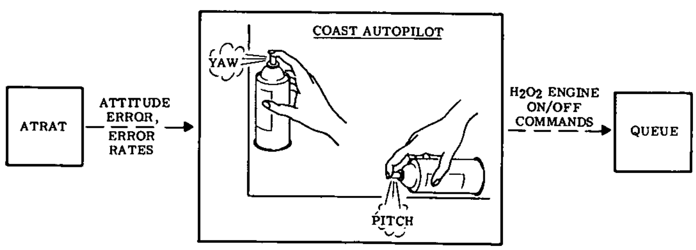
Method. Engine gimbal angle output commands are computed using attitude errors and error rates as inputs to control laws.



COAST PHASE AUTOPILOT

Function. Control the vehicle attitude during coast phase maneuvers.

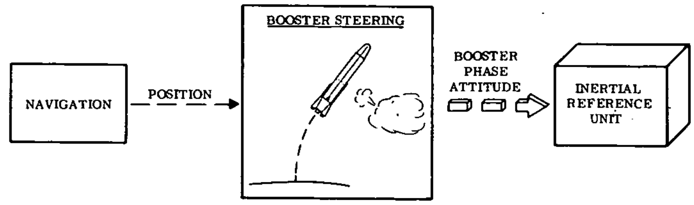
Method. Command H_2O_2 attitude control engines in on/off mode.



BOOSTER STEERING

Function. Steer the booster in pitch, yaw, and roll in an open-loop manner during ascent through the atmosphere.

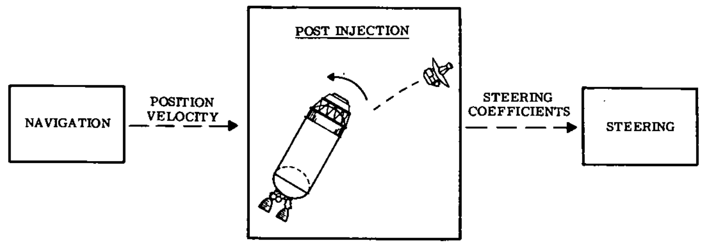
Method. Using polynomials in altitude, generate attitude as a function of altitude.



POST INJECTION (PIJ)

Function. Provides steering coefficients to point the vehicle for separation and retromaneuver.

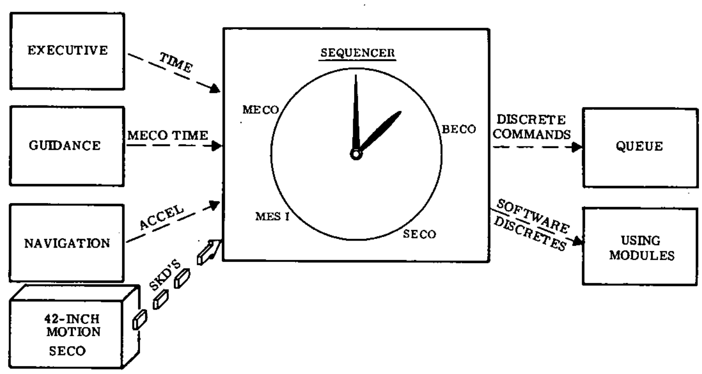
Method. Output to the resolver chain roll and pitch axes pointing vectors.



SEQUENCER

Function. Generate discretes for sequencing of all events during flight.

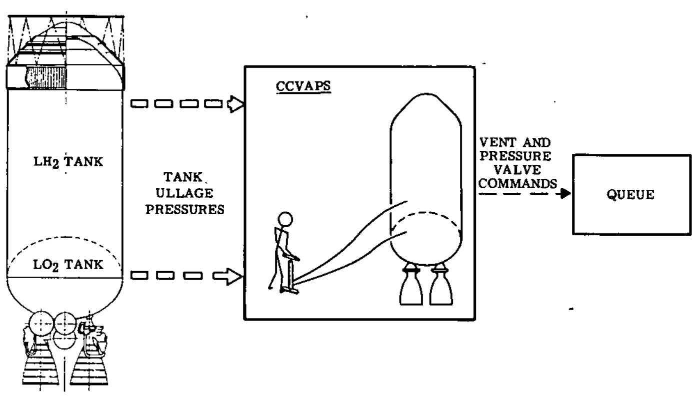
Method. Perform various tests to determine time to issue event discretes or to accept output from selected modules for module dependent discretes.



COMPUTER CONTROLLED VENTING AND PRESSURIZATION SYSTEM (CCVAPS)

Function. Maintain proper LO_2 and LH_2 tank pressures.

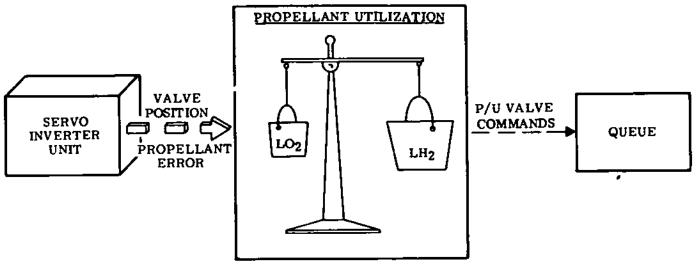
Method. Monitor tank pressures and command venting and pressurization valves to function via SCU switch commands.



PROPELLANT UTILIZATION (PU)

Function. Maintain a proper ratio of LH_2 and LO_2 in the tanks to preclude a premature depletion of one or the other.

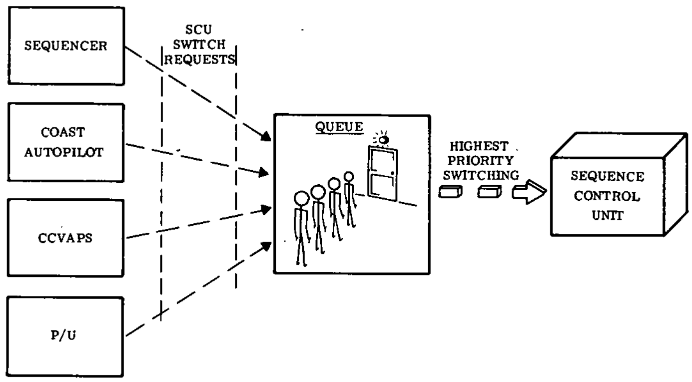
Method. Monitor the propellant ratio error signal and command the PU valves to a position which will null the error.



QUEUE

Function. Resolve any potential conflict of simultaneous requests for switch action.

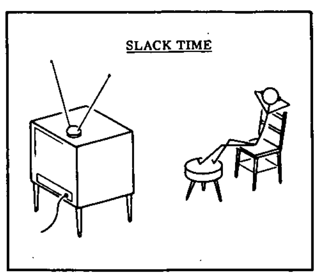
Method. Assign priorites to switch requests and command switches according to the priorities.



SLACK TIME

Function. Provide a task to perform when free time is available during a duty cycle.

Method. Sum permanent memory, determine duty cycle expenditure, and perform simple instruction tests.



5.6 SOFTWARE CAPABILITY

5.6.1 GUIDANCE EQUATIONS. The guidance equations are explicit, using a linear tangent steering law. Inherent flexibility exists in the equations in that they have multimission capability. They can accommodate both earth-orbital and earth-escape missions with either one or two Centaur burns. They can also guide for 3-burn earth orbital missions.

The explicit nature of the equations means that targeting is much simplified with a minimum number of mission dependent constants. The linear tangent steering law allows near optimum trajectory profiles.

5.6.2 STEERING. The attitude vectoring capability of D-1 Centaur is enhanced through the use of roll steering.

During powered phases, the pitch and yaw steering must be dedicated to guidance. The present function of roll steering is to align the pitch axis with the local horizontal; however, this could be modified to some other mission-peculiar roll steering technique if desired.

During coast phases, pitch and yaw steering can be used to fulfill several requirements. These requirements can be for jettison/separation attitudes, retromaneuvers, thermal maneuvers, and/or propellant retention. The roll steering as well as the pitch/yaw steering in coast phases will likely be mission/payload peculiar and can accommodate special solar or antenna pointing requirements.

5.6.3 DIGITAL AUTOPILOT. Powered phase control system software uses "gains" which are a function of measured acceleration, making it relatively insensitive to vehicle dispersions.

Coast autopilot software uses the same error inputs as powered autopilot but can operate in three modes: (1) the limit cycle mode is used for general attitude control requirements, (2) the precision mode is used when requirements dictate a precise pointing need, (3) the maneuver mode is called upon whenever large angle (>10°) maneuvers are required. All three modes can be required in one mission.

Another function of coast autopilot is to command the axial thrusting H_2O_2 engines for propellant settling and holding. These engines can also be used for pitch and yaw attitude control either for backup of the other H_2O_2 engines or simultaneously with the propellant holding mode.

5.6.4 BACKUP SOFTWARE. Backup mode software is intended to add a measure of "forgiveness" to the software. The philosophy is to provide practical alternative (sequencing) action, in the case of certain external vehicle (or system) failures.

The prime requirement of backup mode sequencing is to ensure that the switching tests will never cause a good flight to fail. This requirement is achieved even if necessary to sacrifice some thoroughness in the backup test.

The software itself is kept simple to reduce checkout costs and minimize storage required. Maximum use of prime sequencing software (already in computer) is made, and the software is mission and vehicle independent to the maximum extent possible.

Since the backup modes ultimately interface with the spacecraft, the payload user must provide requirements on a reasonable alternative action to be taken as it pertains to the spacecraft. The question is, usually, whether or not to separate the spacecraft. In any case, the alternative strategies must be consistent with the general requirements discussed above.

Some examples of backup software in the airborne computer are the tests performed in the last four seconds prior to launch and the abort action taken. These tests include a platform inertial test, a check of the resolver null calibration, acceleration checks, and a DCU memory sum and instruction test. In flight the software watches for unscheduled thrust termination and will properly stage the vehicle to the next phase if that occurs.

5.6.5 EXPANSION CAPABILITY. Considerable expansion capability exists in the D-1 Centaur Teledyne computer. For the first coded flight program (Pioneer G), 55% of temporary memory and 42% of permanent memory are unused (Figure 5-9). In the time domain (for Pioneer G), the computer is sitting idle one-third of the time in the slack time module.

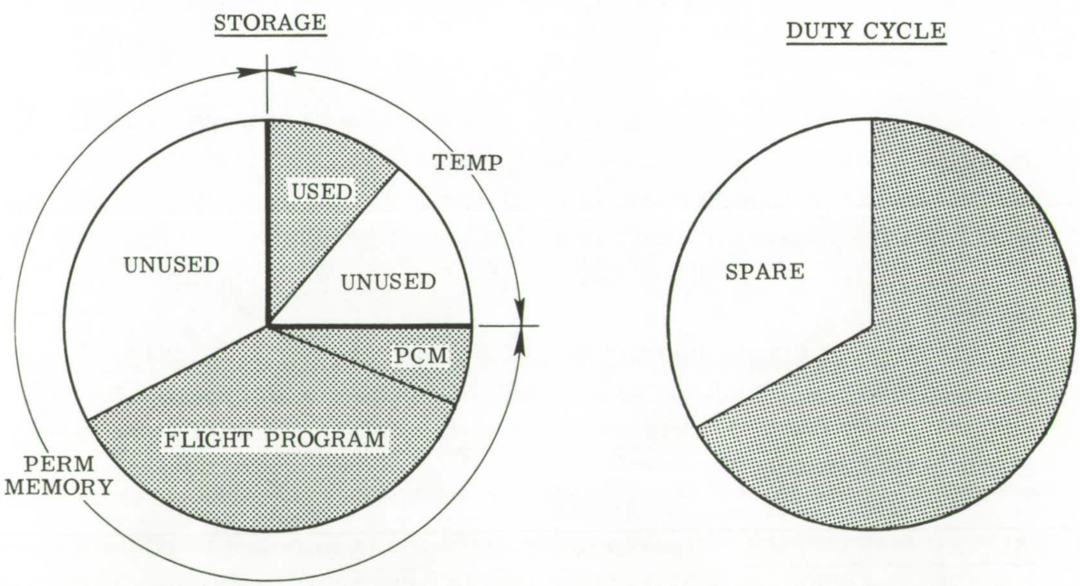


Figure 5-9. Expansion capability exists for D-1 software in Teledyne computer.

This spare capacity has allowed the addition of new functions, such as computer controlled venting and pressurization system (CCVAPS), which do not exist on Pioneer G, and the consideration of using the DCU for monitoring and controlling of redundant vehicle hardware systems or components.

An example of software modules linked together to perform a specific function (in this case Centaur powered phase steering) is shown in Figure 5-10.

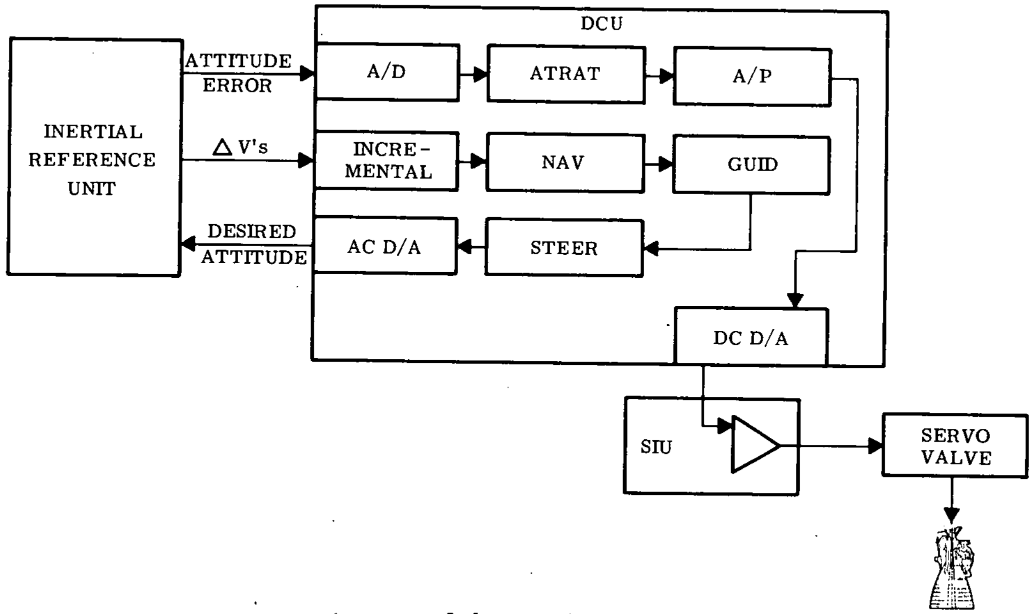


Figure 5-10. Module combination example.

5.7 PREFLIGHT SYSTEM

The preflight program encompasses both operating system programs and vehicle related special purpose (tenant) programs required to support prior-to-flight vehicle and avionics systems tests.

Configuration. The preflight system is assumed to be operative with the flight program contained in permanent memory. The systems test programs, therefore, are loaded into the temporary memory. As shown in Figure 5-11, the memory configuration in the preflight mode shares the use of the flight program interrupt processors.

The real-time interrupt software operates just as described for the inflight portion except that the task table now contains five tasks in this order: preflight resident control, 3 tenant regions, and slack time.

When the configuration changes to the inflight mode, all the memory on the right-hand side of the storage fence changes, and the task table temporary memory is configured to the inflight tasks.

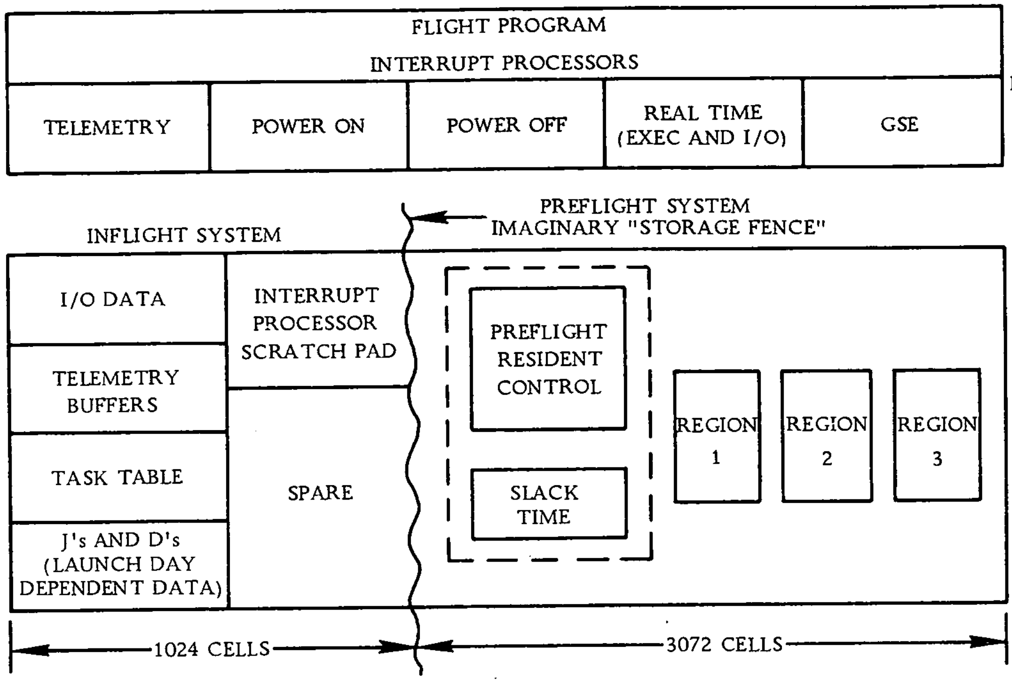


Figure 5-11. Memory configuration in the preflight mode shares the flight system software.

Resident Control. Resident control is the first task executed upon receiving the real-time interrupt because it must communicate data to and from the tenants. This software is resident in the DCU throughout the duration of all testing and vehicle countdown. The tenants, on the other hand, are shuttled in and out as needed.

The resident control software performs preflight oriented utility functions such as resolving conflicts (by the various tenants) in demands for switch setting service. The resident control program also provides input data to the tenants, and formats the preflight telemetry data.

Tenants. Tenant regions contain complete tenant programs. A tenant program contains an "argument" section, a "program section", and a "scratch pad" section, all within the 853-cell tenant region. The argument section reserves storage for input data arrays, supplied either by the ground operator or by the resident control program, and output data which needs to be communicated to either the ground or resident control program. The scratch pad section allocates space within the tenant region for temporary data private to the tenant.

Tenant programs do not always perform a complete function, although they are complete programs because they are designed to operate in conjunction with the ground computer. An example of this is the calibration and alignment tenant. This program computes the value of certain key velocity sums and provides this data, via telemetry, to the ground computer. The analogous tenant in the ground computer then completes the computation of the actual platform calibration coefficients.

5.8 FLIGHT PROGRAM VALIDATION

SOFTWARE VALIDATION PROCESS ENSURES THAT:

- Program is Mathematically and Logically Correct
- All Design Requirements Are Satisfied
- Software/Hardware Interfaces Are Correct
- Duty Cycle Margin Is Ample
- Flight Program Is Forgiving In the Presence of Anomalies
- Program Is Ready For Launch

Validation of the DCU software ensures an error free program. Validation checks that the software meets the design requirements, that the software and hardware interfaces are correct, that an adequate duty cycle margin exists, and that the software is forgiving in the event of large hardware dispersions.

The validation procedure consists of two testing phases: a Design Evaluation Test (DET) and a Design Acceptance Test (DAT). The DET is primarily a search for weaknesses in the design. The software is severely stressed to determine its limits, and to verify an adequate design margin. This is accomplished by simulating and inputing data representing failed or severely dispersed hardware systems into the software. Figure 5-12 illustrates the DET.

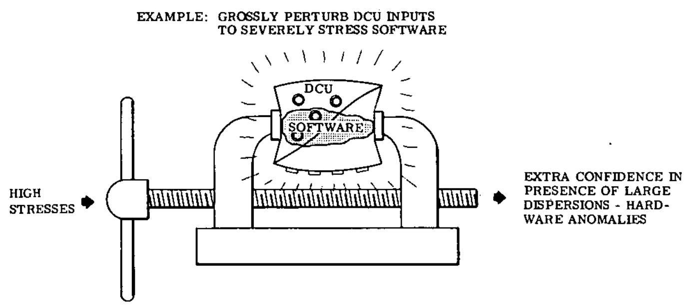


Figure 5-12. Design Evaluation Test - search for weakness.

The second validation test, DAT, is a formal procedure that verifies that the program is error-free and qualified for release. The input data is generally the limits of the acceptable flight environment, i.e., 3-sigma dispersion or combinations of dispersions. This test verifies that the logic is coded correctly and that the program meets the design requirements. The DAT is illustrated in Figure 5-13.

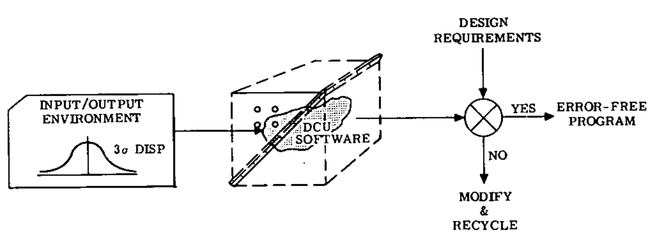


Figure 5-13. Design Acceptance Test - formal validation.

The validation of the DCU software is a multilevel function (Figure 5-4). It first takes place at the module level where it results in a library of validated modules. Modules may then be combined into subsystems for an initial check of the module interfaces. This subsystem level is not part of the formal validation procedure but serves as an interim between the formal module and program validations. Validation of the integrated flight program is then performed. Finally a verification is made using the targeted trajectories.

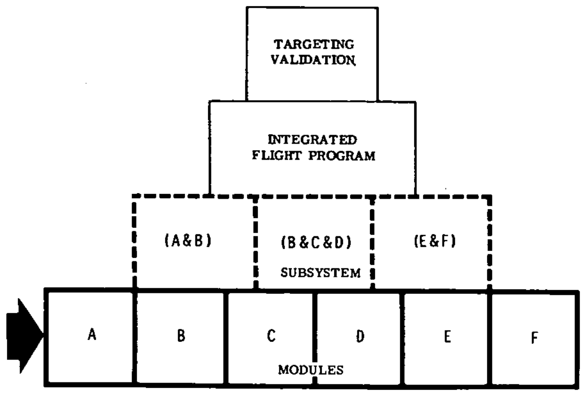


Figure 5-14. DCU software validation takes place at several program levels.

Module Validation. The first step in the validation procedure begins at the module level. Figure 5-15 is a block diagram of module-level validation. Simulation of the module is in Fortran, hence floating point. The external environment that the module interfaces with is an accurate model of the world and resides in the input and response simulation.

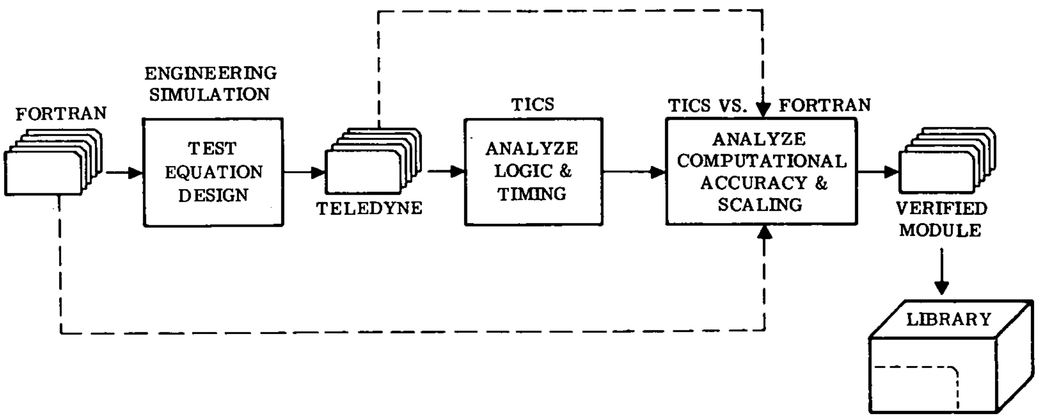


Figure 5-15. Module-level validation.

The conformance to the functional engineering design requirements in response to a changing input environment is checked. If the module checks, it advances to the next test. If it fails, it is modified as necessary and retested until it passes. The module is also driven over the limits of its design frequency range.

Next the module is coded for the Teledyne computer and tested in the Convair Teledyne Interpretive Computer Simulation (TICS) system which is a bit-for-bit simulation of the Teledyne computer on the CDC 6400 scientific computer.

The tests run with this configuration are all open-loop. Inputs are according to ICT module test plan and the outputs are checked against the outputs specified in the module test plan. Input and output buffers are the same as will be used in flight. The inputs are force-fed to cause different branching. The use of TICS allows access to the intermediate data output required for checkout.

The logic receives a thorough check by verifying that all possible branches are coded correctly. When applicable, timing constraints such as initialization, input/output, etc., are also checked at this level.

The next step in module validation compares the TICS and Fortran simulations. The input buffer will be configured to stress the computational capability of the module. The outputs are bit-for-bit compared.

The results of the tests will be differences of the output, Teledyne versus Fortran. This checks directly the computational accuracy of the module and tests the adequacy of scaling for a specified range of values. The tests also check conformance to the program requirements of the Teledyne module.

Upon completion of module level testing, the module is verified and is placed into the module library.

Flight Program Validation. Validation now moves to the integrated flight program level, where a variety of tools are used to ensure complete validation. Flight program validation is illustrated in Figure 5-16.

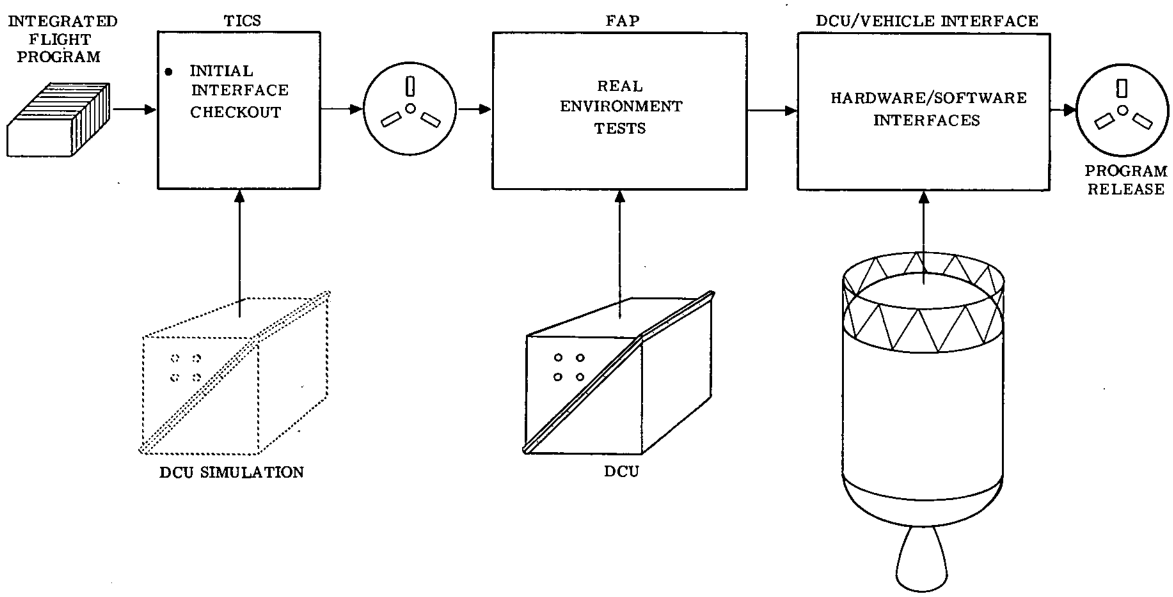


Figure 5-16. Integrated flight program validation uses many tools.

The first closed-loop simulation consists of TRAJEX (vehicle and environment) linked to TICS (Teledyne Interpretative Computer Simulation) which uses the integrated flight program. TRAJEX flies the booster/Centaur/upper stage in a simulated flight under the control of TICS.

The model in TRAJEX represents a realistic simulation of linear and rotational vehicle motion with freedom to simulate any input environment desired. It further simulates pertinent DCU interfaces.

The model for TICS loaded with the integrated flight program represents a bit-for-bit simulation of the DCU in program operation. A particular merit of this simulation is that it provides the engineer with access to intermediate calculations, in addition to the DCU telemetry bit stream.

In this configuration, correct operation of the flight program is checked under simulated environments. Scaling margins of all intermediate computations are tested.

The next closed-loop simulation uses FAP. In FAP, a model of TRAJEX resides in the XDS 930 (CCLS computer) and is controlled by the DCU which is loaded with the integrated flight program. FAP stands for Flight Acceleration Profile, a test in which the DCU operates in conjunction with the vehicle model in the XDS 930.

The XDS 930 contains a model which faithfully simulates linear motion. The rotational motion is rigid body only. The actual interrupt environment is simulated. This simulation allows the checking of many input environments at low cost. It checks out program operation in the presence of a multivariable input environment, verifies the telemetry data reduction interfaces, and provides a check on rigid body mode stability.

Following the closed-loop flight simulations, the hardware/software interface tests are run.

The DCU/vehicle interface test checks the compatibility of the DCU with the vehicle and the CCLS when going through the countdown sequences from preflight to flight.

The DCU/vehicle interface consists of the DCU linked to the vehicle under the prelaunch control of the CCLS. During countdown the umbilicals can be ejected to simulate launch conditions.

In this configuration, the DCU will be loaded with the flight program and then mounted on the electronic equipment module in the factory. The CCLS will operate as if launch is to occur. It will load the DCU with sample J's, P's, Y's, etc., and the countdown tenant. Both CCLS and the DCU will proceed in real time through the simulated launch.

The interfaces of the flight program with the vehicle and the CCLS will thus be checked out, as well as selected interfaces and phasing characteristics as desired by ground test personnel.

Upon completion of these tests, the flight program is validated and ready for release.

Targeting Verification. The final level of validation verifies that the flight program, which has been validated in a general sense, will work correctly for the specific mission to be flown. Targeting verification consists of several steps using the targeted trajectory. The targeting verification sequence is represented in Figure 5-17.

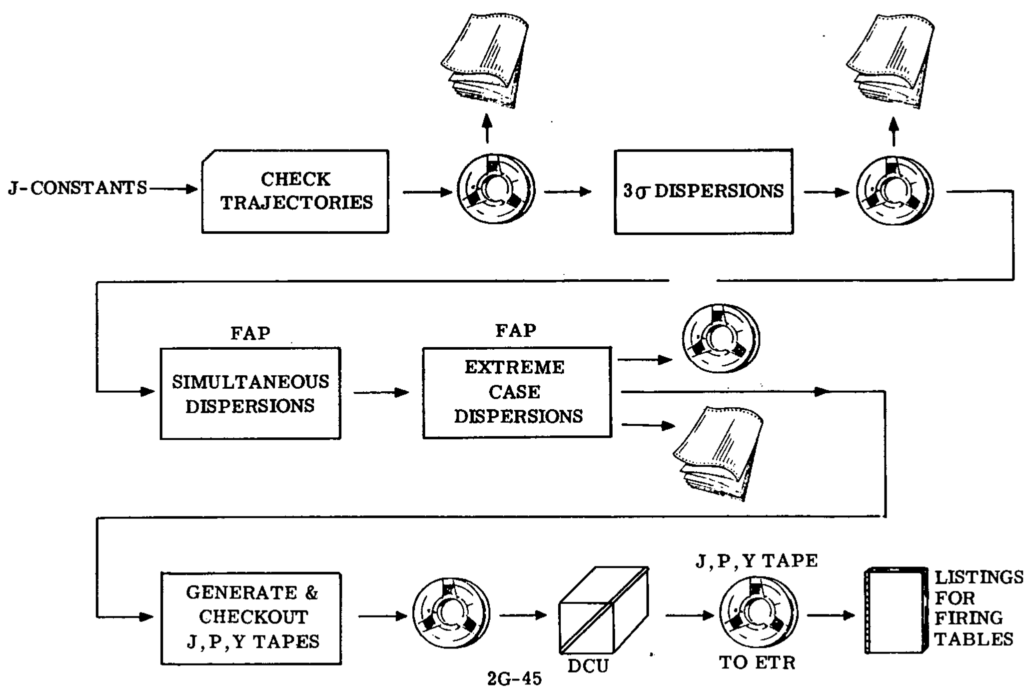


Figure 5-17. Targeting verification.

Check trajectories are run to check the J (launch dependent) constants over the launch window in the presence of varying demands of the trajectory, such as doglegs. The check trajectories also provide anchor points for the generation of injection polynomials which in turn are used to provide graphic and tabular data for an appendix to the firing tables. The check trajectories include an N-Body program which provides midcourse requirements. Copies of these trajectories called for by the target specification are sent to the mission management.

A set of 3-sigma dispersions are simulated and run either on the CDC 6400 engineering simulation or with FAP. The results are tabulated and unexpected data resolved and presented to the review boards for approval.

Simultaneously 3-sigma dispersions are then run which tend to strain the trajectory (and the flight program) to a low-left, a high-left, a high-right, and a low-right condition. Their primary function is to check scaling of the program using actual J's and to check performance of the software under these conditions.

The selected set of dispersions exercise the program at intermediate points during a launch period. For a 15-day launch period, the full set of dispersions may be run on the first and last day, and selected dispersions will be run at points in between. This is to ensure that no peculiar launch geometric characteristics are overlooked.

After the J constants have been completely checked out, a magnetic tape is written with J's, P's, and Y's. The P and Y constants are used to generate the booster steering pitch and yaw profiles. The tape is bit-for-bit compared with the deck used to generate them. This tape is then written into the Teledyne computer temporary memory. The tape is then read out, using the read locations in the firing tables, and the values are compared with the input to the firing tables.

The listings of the P's, Y's, and J's are provided as a part of the input to the firing tables. The J, P, Y tape is shipped to ETR for use during launch along with the firing tables.

5.9 SYSTEM MANAGEMENT

Centaur D-1 software is subject to controls from module inception through flight. These controls ensure that the management of the software will be thorough and complete. The Software Review Board, SRB, monitors all DCU software from inception on. This board is made up of LeRC and Convair Aerospace personnel.

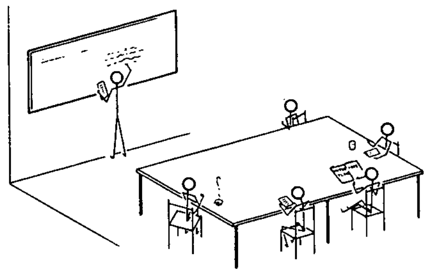
The Change Requirement System is a formalized procedure for initiating, approving, and recording changes to modules or programs. Change approval is required from the SRB chairmen.

When a module has been validated it has a unique check number assigned to it. This number is generated by the support software and will change with any change made to the module.

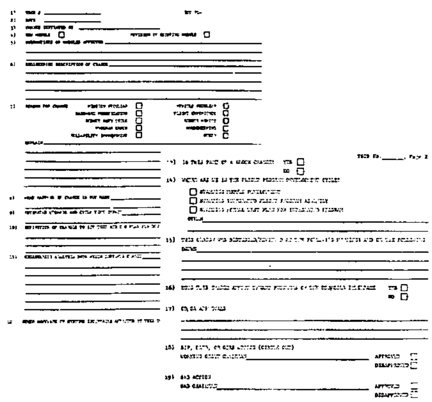
In a similar way, when modules are assembled into a program, a unique check number is calculated for that program configuration. This number is sensitive not only to the modules in the program, but also to the order in which they are assembled into the program.

The Resources Control Process is a continual review by LeRC and Convair Aerospace management of new DCU software requirements which would use more storage and/or duty cycle resources. The gains obtained by the implementation of these requests are weighed against the ever increasing load on the DCU. This review helps ensure that the DCU resources are used for only total D-1 system improvements.

CONTROLS FACILITATE DCU SOFTWARE MANAGEMENT



- SOFTWARE REVIEW BOARD →
- CHANGE CONTROL SYSTEM →
- MODULE CHECK CODE NUMBER → 5634437746
- PROGRAM CHECK CODE NUMBER → 14034104401617117037
- RESOURCE CONTROL PROCESS → TRADE STUDY
  DCU DUTY CYCLE
  DCU STORAGE
  RELIABILITY
  PERFORMANCE
  FLEXIBILITY
  COST



6

ATLAS SLV-3D SYSTEMS

6.1 STRUCTURE

The airframe (Figure 6-1) is composed of two major sections: the sustainer or tank section and the booster section. The sustainer section consists of the fuel tank, liquid oxygen tank, and the equipment pods. The sustainer and vernier engines are mounted on this structure. The booster section is composed of the thrust cylinder, engine nacelles, heat shield, and fairings. The booster engines are mounted in this section.

6.1.1 SUSTAINER SECTION. The propellant tank is the primary structure of the sustainer section. The tank is a thin-wall, fully monocoque-structure pressure vessel, which derives its rigidity from internal pressurization.

The tank body is a resistance-welded structure of corrosion-resistant stainless steel sheets (skins) which vary in thickness from 0.041 to 0.016 inch. The tank is cylindrical with an elliptical transition to a conical aft end and an elliptical surface at the forward end. The center cylindrical section is 120 inches in diameter. The tank is approximately 60 feet long. An ellipsoidal intermediate bulkhead divides the tank into two sections: an aft section of approximately 1690 cu. ft. for RP-1 fuel and a forward section of approximately 2700 cu. ft. for liquid oxygen. A thrust ring joins the conical aft section to the cylindrical portion of the tank. A webbed bulkhead is fastened to the inner flange of the thrust ring to prevent radial distortion from the stresses created by the thrust of the engines. Annular baffles in the LO_2 tank dampen propellant sloshing.

The sustainer engine and associated equipment and subsystems is gimbal-mounted to the sustainer thrust cone which is the aft end of the fuel tank.

Vernier-engine thrust chambers are gimbal-mounted on opposite sides of the tank structure on the X-X axis near the aft end of the tank at the interface between the tank and booster sections.

Liquid-oxygen engine feed lines, tank pressurization lines, and electrical cable fairings are attached to the outside of the tank structure. Equipment pods containing electrical and electronic units are also attached to the outside of the tank structure.

6.1.2 BOOSTER SECTION. The aft section consists of two booster engines, booster structure, and associated equipment and systems. This section is attached to the

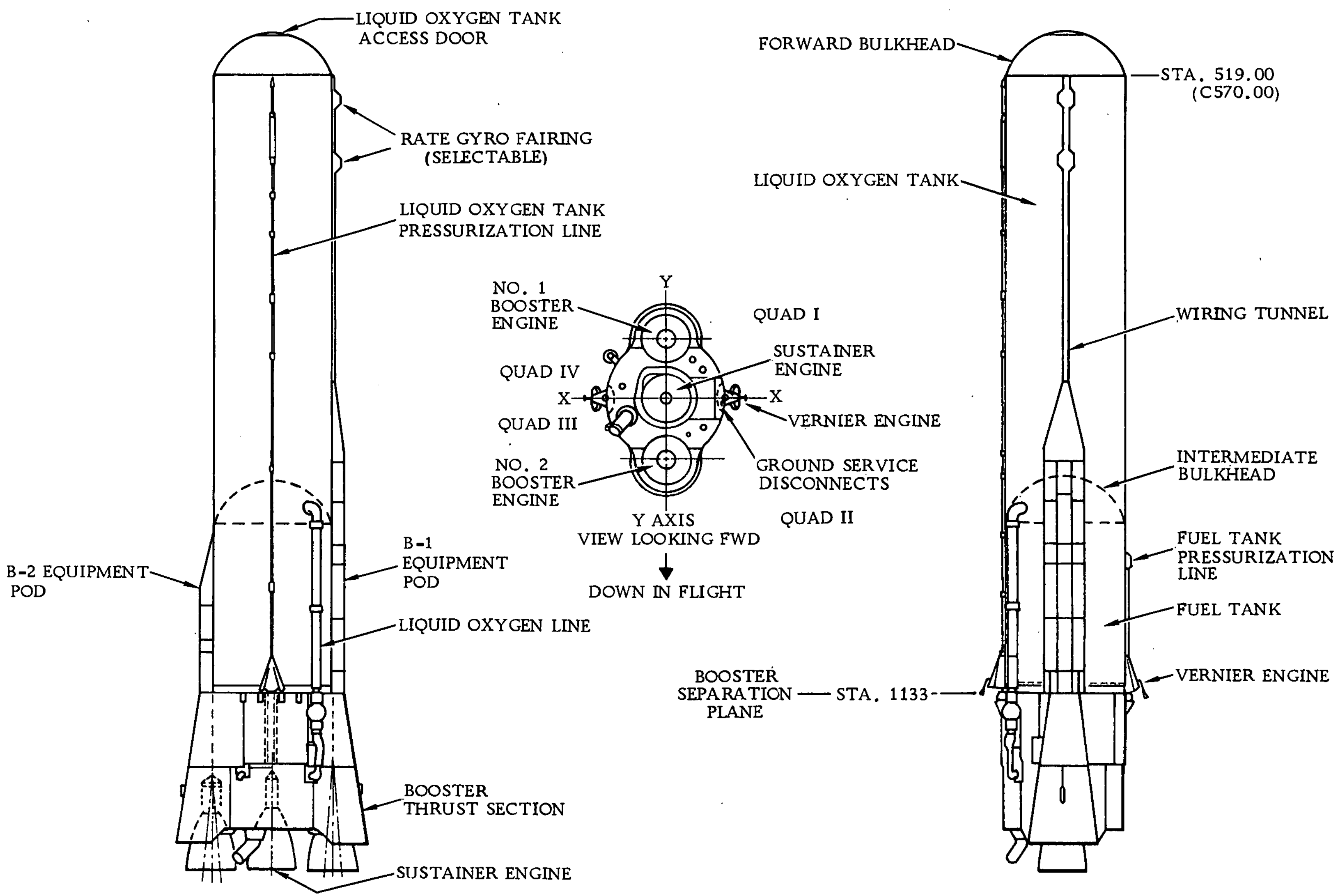


Figure 6-1. Atlas SLV-3D space launch vehicle.

thrust ring near the aft end of the tank section by 10 latches (Figure 6-2) which release it for separation, via jettison tracks, at staging.

The booster structure consists of a thrust cylinder, heat shield, and engine nacelle installation. The structure forms a single compartment to house the propulsion subsystem and its associated equipment and subsystems. The heat shield of fiberglass sandwich with a cork aft surface protects the interior aft section of the booster section from engine exhaust heat radiation.

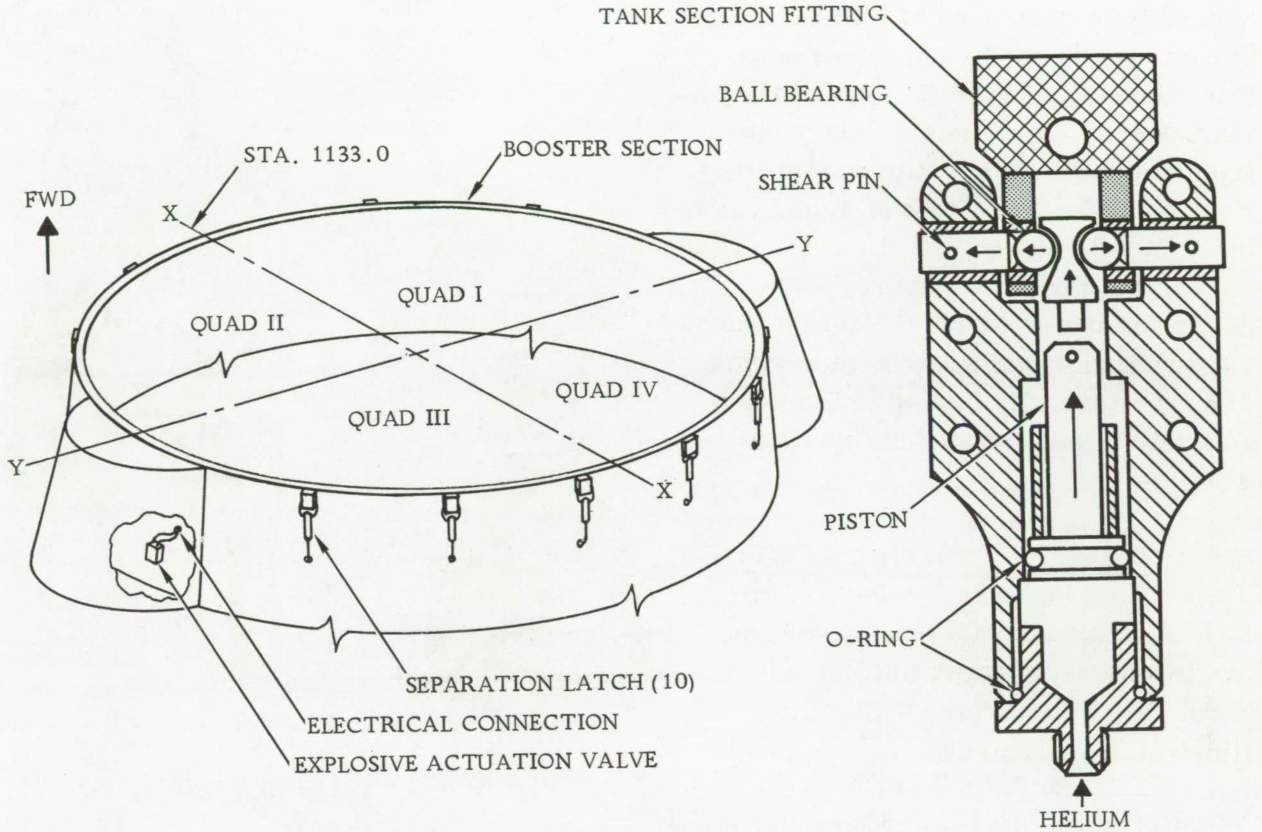


Figure 6-2. Booster section separation system.

6.1.3 SEPARATION MECHANISMS. The first stage booster section separation system (Figure 6-2) consists of ten pneumatically operated separation latches (which connect the booster structure to the tank section), a control valve assembly (with two explosively actuated pneumatic valves), a pneumatic manifold interconnecting the individual fittings to the control valve assembly, a helium pressure supply bottle (with a pneumatic line to the valve assembly), and the necessary electrical wiring to fire the explosive valves.

The booster engine separation mechanism is actuated by a command from the guidance system via the flight control subsystem, and backed up by a staging accelerometer.

6.2 ATLAS PROPULSION SYSTEM

The Rocketdyne MA-5 rocket engine group used on Atlas space launch vehicles consists of three main assemblies: (1) a YLR89-NA-7 booster engine, (2) a YLR105-NA-7 sustainer engine, and (3) two YLR101-NA-15 MD2 vernier engines. All are single-start, fixed-thrust rocket engines that use liquid oxygen and a liquid hydrocarbon fuel, RP-1, as propellants.

Each booster engine and sustainer assembly is composed of a thrust chamber or chambers, a dual turbopump and valves to control propellant flow, a start system, and the necessary electrical harnesses and interconnect lines. Each basic loop consists of a dual turbopump to deliver propellants under pressure to the thrust chambers, a gas generator loop to drive the turbopumps, and the thrust chambers. All engines are hypergolically ignited. The gas generators use pyrotechnic ignition systems.

Figure 6-3. Atlas SLV-3D booster engines.

6.2.1 BOOSTER ENGINE ASSEMBLY.

The booster engine assembly (Figure 6-3) is composed of a power package and two nearly identical thrust chambers. The total engine system is illustrated in Figure 6-4.

The power package consists of two dual turbopumps that deliver the propellants to the thrust chambers. These turbopumps are centrifugal pumps driven by a high-speed turbine through a reduction gear. High-velocity gas for driving the turbine is supplied by a gas generator. A pressurized oil supply tank and an oil pump supply oil under pressure for lubrication of the turbopump gears. The power package components are secured on a tubular mount which can be attached to the vehicle as a single unit.

The booster gas generator consists of a spherical combustion chamber and an exhaust manifold. Liquid oxygen and fuel are supplied under pressure to the combustion chamber and ignited by electroexplosive igniters. The combustion gases are then routed to the turbopump turbine wheels by the exhaust manifold. The turbine exhaust gases pass through a heat exchanger to heat and expand helium for vehicle system pressurization and are then dumped overboard.

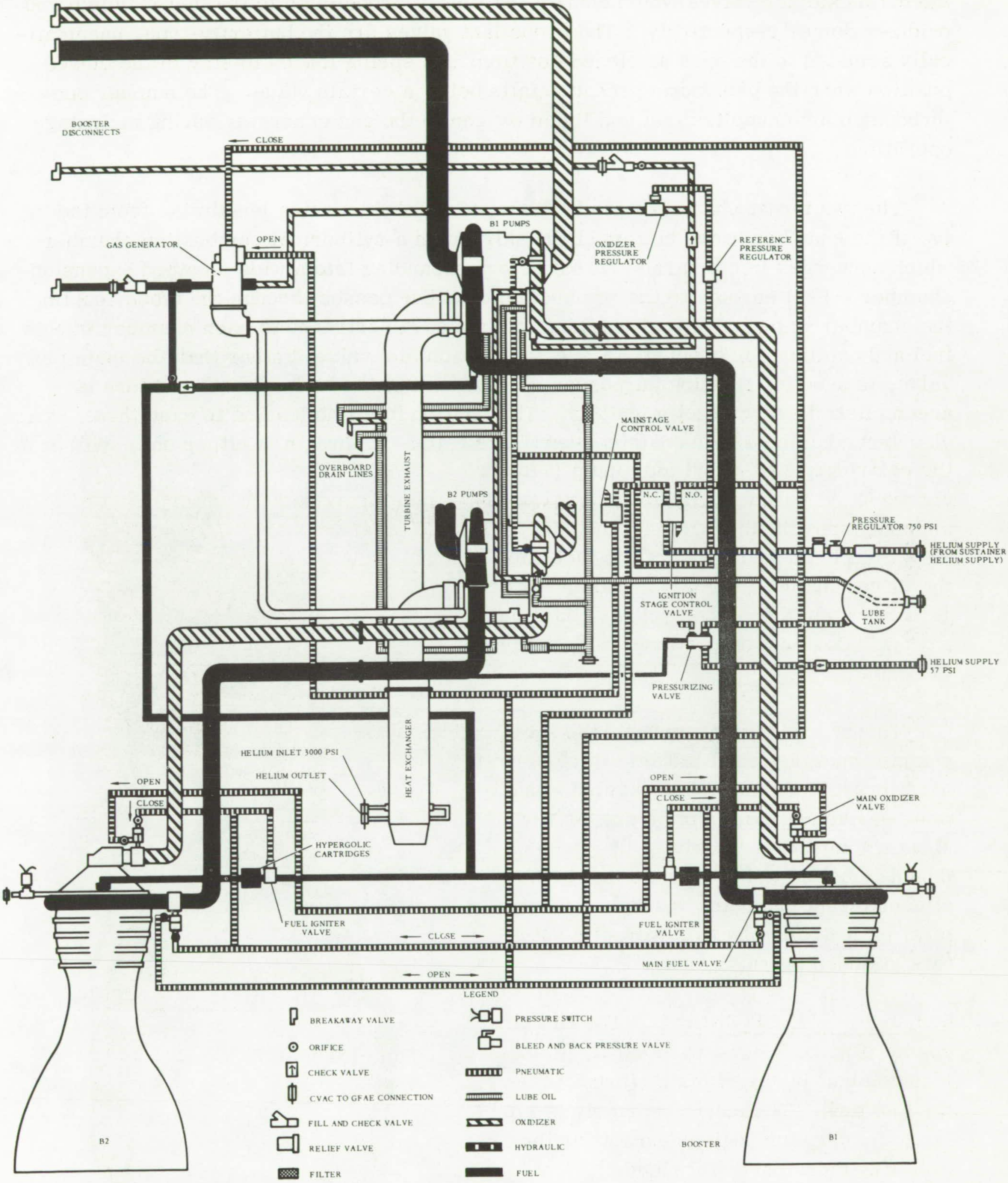


Figure 6-4. MA-5 booster engine system.

The propellants flowing from the turbopumps pass through the main fuel valve and the main oxidizer valves which control the flow of propellants to the fuel manifold and oxidizer dome, respectively. The propellant valves are the butterfly-type, pneumatically actuated to the open or closed position, and spring loaded to stay in the closed position when the pneumatic pressure falls below a certain value. The number one turbopump also supplies fuel and liquid oxygen to the gas generator during mainstage operation.

The two thrust chambers are bell-shaped. Tubes running lengthwise from the top of the chamber to the bottom of the skirt form a cylindrical combustion chamber which converges into a narrow throat before expanding into a wide mouthed expansion chamber. Fuel enroute to the combustion chamber passes through the tubes, cooling the chamber walls. A circular injection plate at the entrance to each chamber injects fuel and oxidizer for burning. A special ignition fuel valve, rather than the main fuel valve, is used for starting purposes. Combustion of the fuel-oxidizer mixture is accomplished by hypergolic ignition. The ignition fuel line leading to each thrust chamber contains a self-contained cartridge with a diaphragm at either end. Within the cartridge is tri-ethyl aluminum (TEA) and tri-ethyl boron (TEB) fluid. Fuel pressure ruptures the diaphragms; the TEA and TEB fluid is pushed ahead of the fuel into the thrust chamber where the mixture hypergolically ignites upon contact with the liquid oxygen. Combustion continues with the fuel/ LO_2 mixture.

Thrust loads are transmitted to the vehicle through gimbal mounts which permit each thrust chamber to be swiveled a maximum of five degrees in pitch and/or yaw about the missile centerline. A gimbal mount is bolted to the oxidizer inlet elbow of each thrust chamber. Chamber movement is controlled by two hydraulic actuators attached to outriggers.

6.2.2 SUSTAINER ENGINE. The sustainer engine (Figure 6-5) is an integral, gimbal-mounted unit installed on the thrust cone of the fuel tank. The entire assembly is similar to the booster engines except for the following principal differences:

1. During mainstage operation the sustainer engine dual turbopump

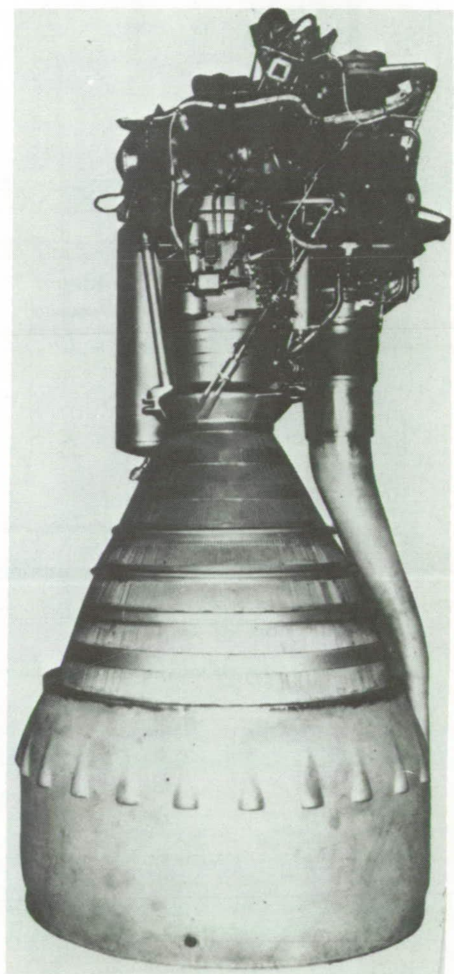


Figure 6-5. Sustainer engine.

supplies propellants to the vernier engines and gas generator as well as to the sustainer thrust chamber (Figure 6-7).

2. The valves for the propellants passing through the turbopumps to the sustainer engine are different. The sustainer engine uses hydraulically-actuated valves for controlling propellant flow to the combustion chambers. The main oxidizer (head suppression) valve is the butterfly type, hydraulically actuated against propellant pressure. The main fuel (propellant utilization) valve is hydraulically actuated to control the rate of flow to the combustion chamber.

3. The sustainer engine thrust chamber is smaller than those of the boosters.

4. Maximum gimbal deflection of the thrust chamber is 3 degrees in pitch and/or yaw about the centerline.

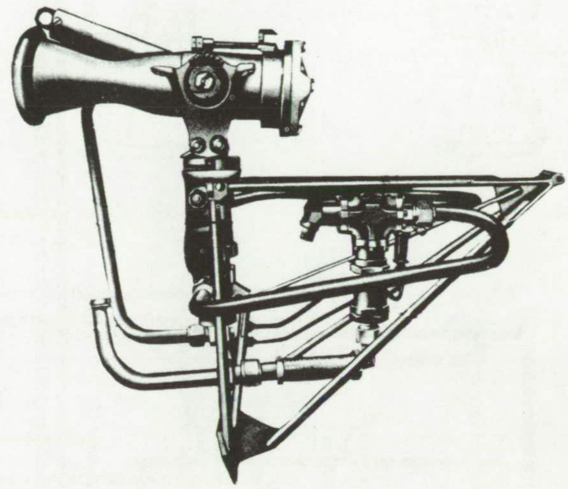


Figure 6-6. Vernier engines.

6.2.3 VERNIER ENGINES. The two vernier engines (Figure 6-6) are installed on the airframe as two completely separate units. Propellants for the vernier engines are supplied by the vernier start tanks during starting, and are supplied from the sustainer dual turbopump during mainstage operation.

The thrust chamber body is double-walled and contains a steel spiral coil between the walls. Fuel flows through the coil to cool the thrust chamber walls.

The gimbal shafts of the engine permit pitch and roll movement of the chamber through an arc of 140 degrees. SLV-3D vernier thrust chambers have no yaw controllers and have a constant bias at 45 degrees.

6.2.4 SEQUENCE OF OPERATION. The engines are started and develop rated thrust while the vehicle is held in the launcher. When the vehicle is released, the combined thrust of the booster, sustainer, and vernier engines lift the vehicle off the pad and accelerate it along its course. At staging, booster engine thrust is terminated and the booster engine section separates from the tank section. The vehicle, lightened by the jettisoning of the booster engine section and by the consumption of propellants, continues to accelerate, propelled by the sustainer and vernier engines. At propellant depletion, sustainer and vernier engine thrusts are terminated.

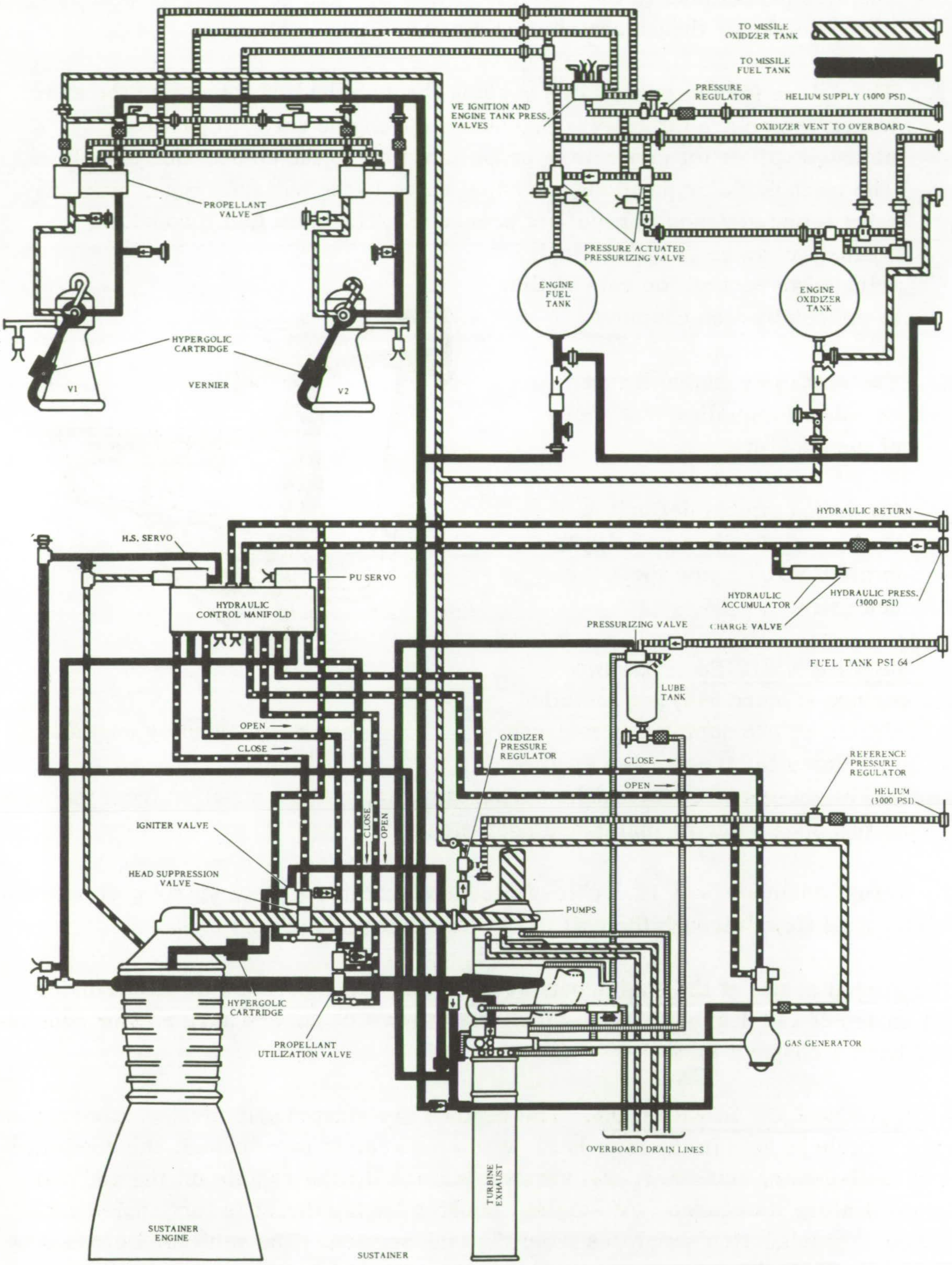


Figure 6-7. MA-5 sustainer engine system.

The sequential operation of the engines can be logically divided into eight stages, as follows:

1. Preparation stage
2. Arming and tank pressurization stage
3. Ignition stage
4. Main stage
5. Booster engine command cutoff stage
6. Vernier engine solo preparation stage
7. Sustainer engine command cutoff stage
8. Vernier engine command cutoff stage

6.3 SLV-3D PROPELLANT UTILIZATION SUBSYSTEM

The purpose of a propellant utilization (PU) subsystem in a dual-propellant vehicle is to cause propellants to be used in the proper ratio, so that there is a minimum residual of propellants at the end of powered flight. Without a PU system, a significant residual (several hundred pounds) of one propellant usually remains at the depletion of the other due to uncontrollable (or unknown) variables and/or tolerance buildups.

SLV-3D vehicles use a Convair designed PU system (Figure 6-8); it is an analog, manometer-type system. The system controls propellant flow to the sustainer engine throughout its operation by varying the position of the fuel control (PU) valve and, indirectly, the oxidizer control (head suppression) valve, ensuring a minimum of residual propellants at the end of powered flight.

The fuel control valve (PU valve) located in the sustainer fuel supply line is actuated by the propellant utilization servocontrol valve which is mounted on the sustainer hydraulic control manifold. The servovalve consists of two coils which control the position of a piston in the servovalve upon command of the electrical signal from the PU system. The piston controls the flow of hydraulic fluid through ports to the closing side of the propellant utilization valve actuator. A built-in protractor is used to match the electronic control circuits to the valve.

A position feedback transducer attached to the PU valve generates a feedback signal (proportional to valve angle) which completes the servoloop and balances the error signal to prevent over-correction of the system. The transducer is a rotary-position variable reluctance transformer, mechanically linked to the shaft of the PU valve butterfly gate. The transducer forms part of an inductance bridge, the output of which is combined with the manometer error signal and applied to the propellant utilization servovalve.

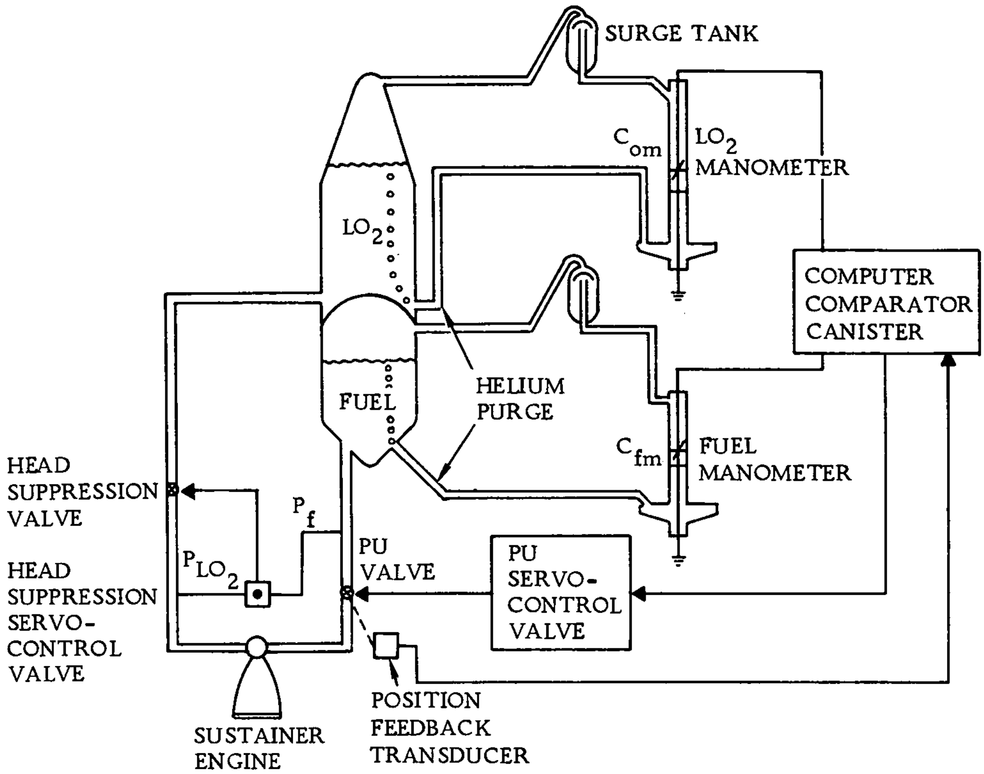


Figure 6-8. Atlas propellant utilization system.

The head suppression servocontrol valve, also mounted on the sustainer hydraulic control manifold, actuates the oxidizer control (head suppression) valve in the sustainer oxidizer supply line. The servovalve consists essentially of a bellows-actuated piston. Differential pressure acting on the bellows moves the servovalve piston, controlling the flow of hydraulic fluid to the closing side of the head suppression valve actuator.

Head suppression servovalve action is such that a differential movement is caused between the PU valve and the head suppression valve. As the PU valve opens, the head suppression valve closes and vice versa. In this manner, a propellant tank ratio error is corrected while a nearly constant total mass rate flow to the sustainer engine is maintained.

6.3.1 THEORY OF OPERATION. The analog PU system (Figure 6-9) senses the mass of propellants remaining in the propellant tanks, computes errors by comparing the mass ratio with a nominal mixture ratio, and adjusts the rate of flow of fuel by repositioning the sustainer engine fuel control valve (propellant utilization valve) throughout powered flight to maintain a proper mass ratio of residual propellants. Main components of the PU system are a liquid oxygen manometer assembly, a fuel manometer assembly, and a computer-comparator assembly.

The mass of propellant remaining in each tank (liquid oxygen and fuel) is constantly monitored by a mercury manometer. Each of the two manometers functions

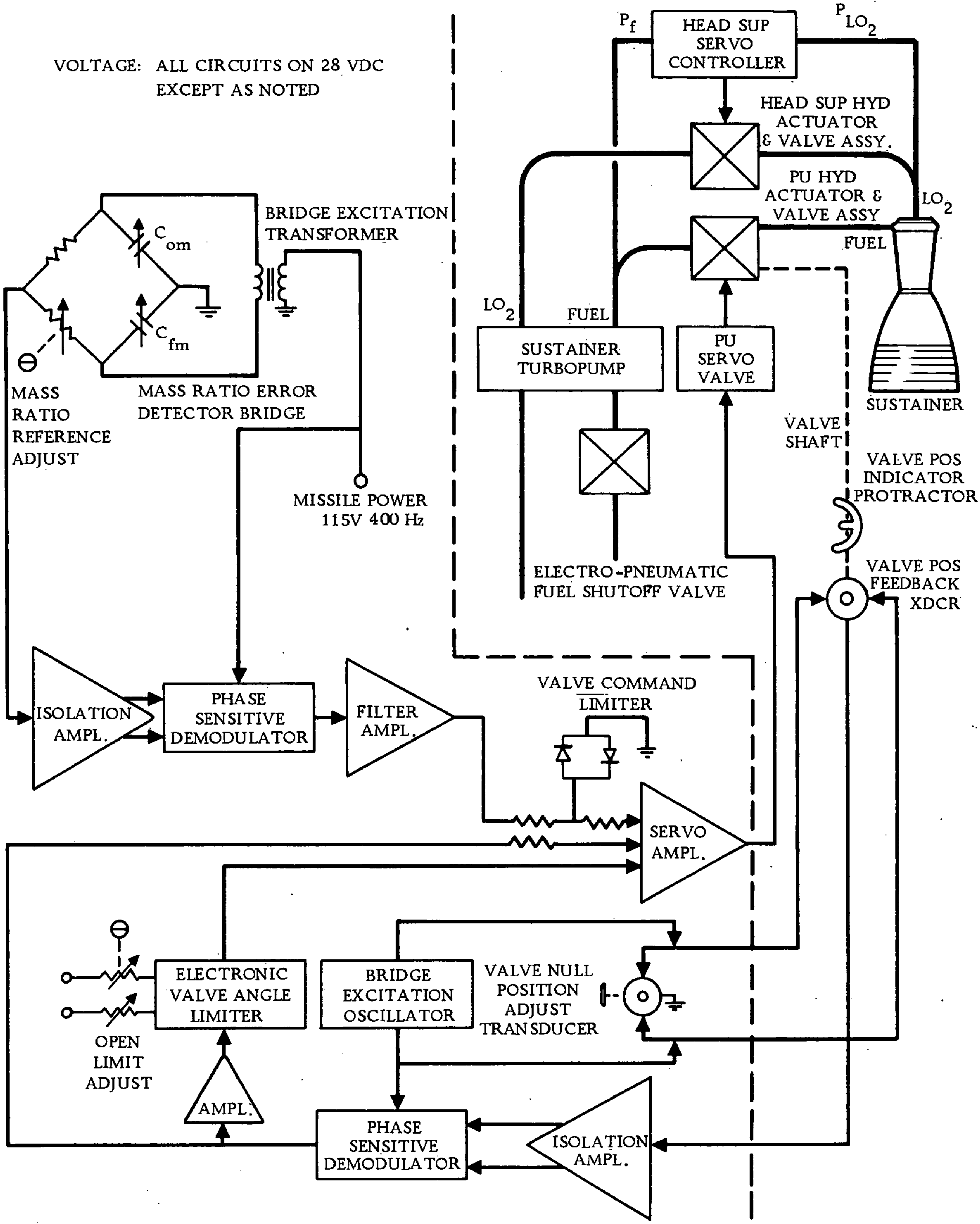


Figure 6-9. Propellant utilization system circuitry.

as a variable capacitor whose capacitance varies in proportion to the mass of propellant in the respective tank.

Manometer capacitance is applied to the computer-comparator canister and forms one-half of a capacitor-resistor bridge error-detector circuit. The output of the error detector bridge is rectified in the computer-comparator and a d-c error voltage is applied to the propellant utilization servocontrol valve.

Through changes in fuel pressure, the head suppression servocontrol valve senses movement of the propellant utilization valve and directs hydraulic pressures to the head suppression valve. The head suppression valve controls the flow of oxidizer to the sustainer engine and moves in opposition to the PU valve.

6.3.2 MANOMETERS. The liquid oxygen manometer is 40 inches long and the fuel manometer is 18 inches long. Each manometer consists of a hollow metal spindle or mandrel, a cylindrical housing of stainless steel, and a mercury reservoir. The mandrel is concentrically mounted within the stainless steel housing. The mandrel acts as the fixed plate of a variable capacitor and is coated with an enamel which acts as a dielectric between plates. Mercury rising about the mandrel serves as the variable plate of the manometer capacitance.

A pressure-sensing line leads from the bottom of each propellant tank to the mercury reservoir of its associated manometer. Because the reservoirs are not coincident with the bottoms of the tanks, and to prevent contamination of the mercury by the propellants, the sensing lines are purged. Helium from the engine control bottle passes through a pair of constant-flow valves (4 standard cu. ft. per hour) and into each of the propellant sensing lines. The sensing lines, therefore, maintain a pneumatic pressure equal to the head pressure of their respective propellant tank. It is this helium pressure that is sensed by the mercury reservoirs.

Pressure applied to the mercury reservoir forces the mercury up around the mandrel. Pressure in the ullage space of the propellant tank is connected to the top of the manometer. Manometer mercury height is therefore proportional to the height and density of propellant in the tank.

Because of the shape of the intermediate bulkhead, tank tapers, etc., the propellant tanks do not have a consistent shape from top to bottom. In order for the manometers to produce an electrical capacitance proportional to the mass of propellants remaining in the tank, the mandrels (which act as the fixed capacitor plates) are shaped to conform to the shape of their respective tanks. The manometers, therefore, present the computer-comparator with an impedance (capacitive reactance) which is inversely proportional to the mass of propellant remaining in the respective tanks.

6.3.3 COMPUTER-COMPARATOR. The ratio of the two manometers' impedances is matched against the resistance ratio of two known resistances, one of which is adjustable in order to set the desired ratio. Thus, the manometers form the variable half of a capacitor/resistor bridge whose excitation voltage is Phase B of the 400-cycle inverter.

When capacitances of the manometers are in the desired ratio, the bridge is balanced. If manometer capacitances vary, the bridge becomes unbalanced and an a-c error voltage is produced. The error voltage will be in phase or 180 degrees out of phase with the excitation voltage, depending on a fuel excess or an oxidizer excess condition in the propellant tanks.

The error voltage is amplified and fed to a phase-sensitive demodulator which converts the a-c input to a d-c output. Polarity of the d-c output is determined by the phase of the input. This error signal is combined with the valve position feedback signal. The resultant valve command is sent to the PU servovalve.

6.4 ATLAS HYDRAULIC SYSTEM

The SLV-3D hydraulic system is made up of two independent systems: booster and sustainer/vernier. These systems provide the mechanical force to position the five engine thrust chambers for directional and roll control of the launch vehicle during flight. The sustainer/vernier system also provides hydraulic power to the Rocketdyne sustainer engine control system for operating the main engine propellant valves for engine start and shutdown and for controlling the gas generator propellant valve.

6.4.1 BOOSTER ENGINE HYDRAULIC SYSTEM. The booster system (Figure 6-10) provides hydraulic power to actuate the two booster engine thrust chambers. The chambers are moved together for pitch and yaw control and moved differentially for roll control.

The major components of the system include two servoactuator assemblies on each thrust chamber, a hydraulic pump on the accessory drive pad of the B-2 booster turbopump, a hydraulic tank, an accumulator, and associated hardware. The complete booster hydraulic system is jettisoned with the booster section at staging.

6.4.2 SUSTAINER/VERNIER ENGINE HYDRAULIC SYSTEM. This system (Figure 6-11) provides hydraulic power to actuate the sustainer thrust chamber and the two vernier thrust chambers. Sustainer thrust chamber gimbaling provides pitch and yaw control from booster engine cutoff (BECO) until sustainer engine cutoff (SECO); vernier engine gimbaling provides roll control.

The major components of the system include two servoactuator assemblies for the sustainer engine, one servoactuator assembly on each vernier engine, a hydraulic

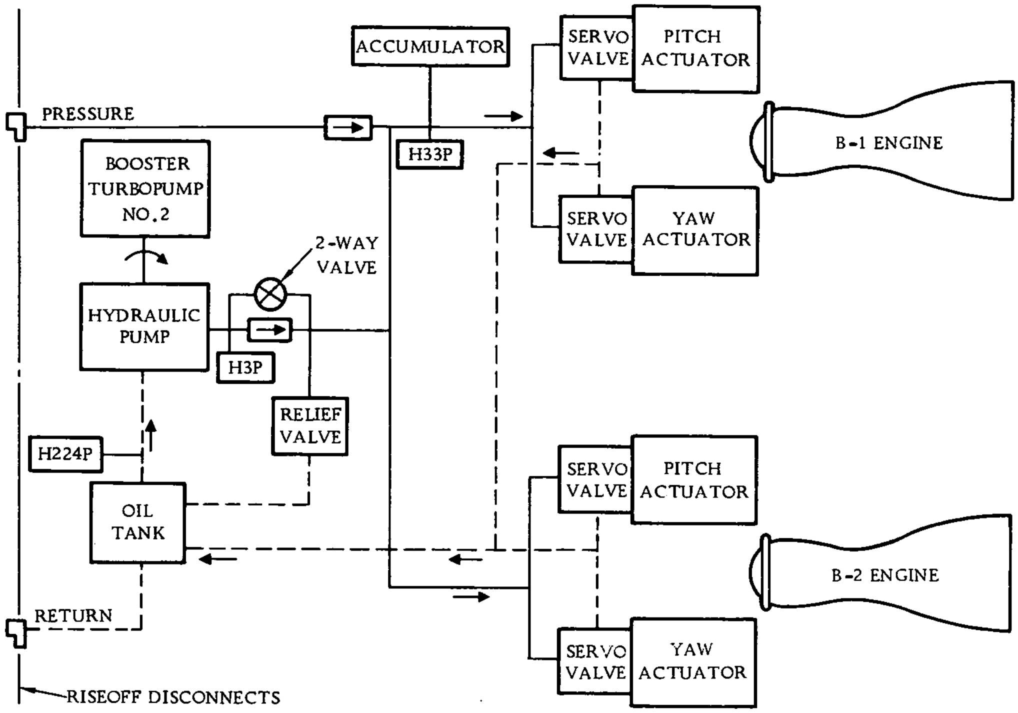
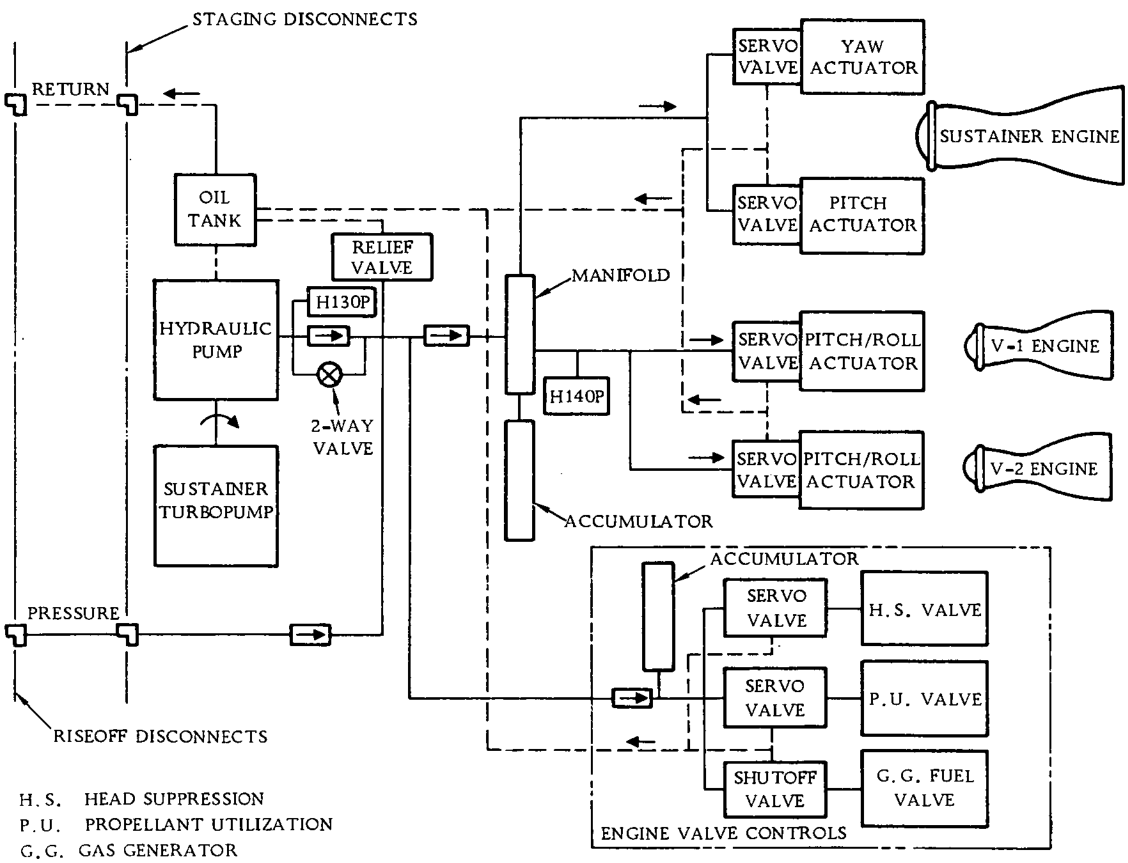


Figure 6-10. Booster engine hydraulic system.



H. S. HEAD SUPPRESSION
P. U. PROPELLANT UTILIZATION
G. G. GAS GENERATOR

Figure 6-11. Sustainer/vernier engine hydraulic system.

pump on the accessory drive pad of the sustainer turbopump, a hydraulic tank, a hydraulic accumulator, and associated hardware. In addition, hydraulic power is supplied to the Rocketdyne sustainer engine control package for control of the sustainer head suppression, PU, and gas generator propellant valves.

6.4.3 SYSTEM OPERATION. Each hydraulic system is rated at 3000 psig operating pressure. The hydraulic pressure is supplied by variable volume pressure compensated pumps, each driven by its respective engine. The pumps supply hydraulic power at 10 to 100 percent of flow capacity to meet system demands automatically. The system is designed to operate between -30°F and +275°F. Riseoff and staging disconnects permit factory and launch site checkout of the airborne systems via a ground hydraulic power unit (HPU).

The hydraulic actuators are the end-use components. One end of the actuator is secured to the vehicle structure and the other end is attached to the thrust chamber outriggers. Applying pressure to the actuator causes a force to be applied through the outrigger to the thrust chamber, which causes it to move (gimbal). Movement of the thrust chamber changes the engine thrust vector alignment with the vehicle center of gravity and causes the launch vehicle to change direction.

The launch vehicle flight control system senses an error in vehicle attitude or receives a direction change requirement from the guidance system. Electrical signals are transmitted to the servocontrol valves which are integral with the actuator housing. In response to the electrical signals, the servocontrol valve routes hydraulic pressure to the appropriate side of the actuator piston causing the thrust chamber to move to the proper position, thereby correcting vehicle attitude or causing a change in vehicle attitude.

6.4.4 FLIGHT OPERATIONS. Inflight operation of the booster and sustainer/vernier hydraulic systems occurs only during the powered phase of Atlas flight. Nominal operating time of the booster system is 150 seconds. The sustainer system, operating from liftoff through SECO/VECO, has a nominal operating time of about 230 seconds.

Prior to liftoff, the vehicle receives hydraulic power via the ground HPU at 2000 psig. Under control of the vehicle flight control system, all engine servoactuators are maintained at a null position in preparation for engine ignition. In addition, hydraulic power from the HPU is available to the Rocketdyne sustainer engine control system. During the engine ignition sequence, the airborne hydraulic pumps are activated and generate 3000 psig pressure to override the ground HPU. At liftoff, HPU and vehicle separation occurs via the riseoff disconnects.

During the booster phase of flight, the booster engine servoactuator control is active, the sustainer engine is maintained in a null position, and the vernier servoactuators are active. At BECO, the booster engines are shutdown, including the hydraulics, and the booster section is jettisoned. The sustainer servoactuators are activated for vehicle control during the sustainer phase of flight. At SECO, both the sustainer and vernier engines are shutdown, including the hydraulic system.

6.5 ATLAS PNEUMATIC SYSTEM

The SLV-3D pneumatic system (Figure 6-12) provides pressurization for the tank, engine controls, and thrust section separation.

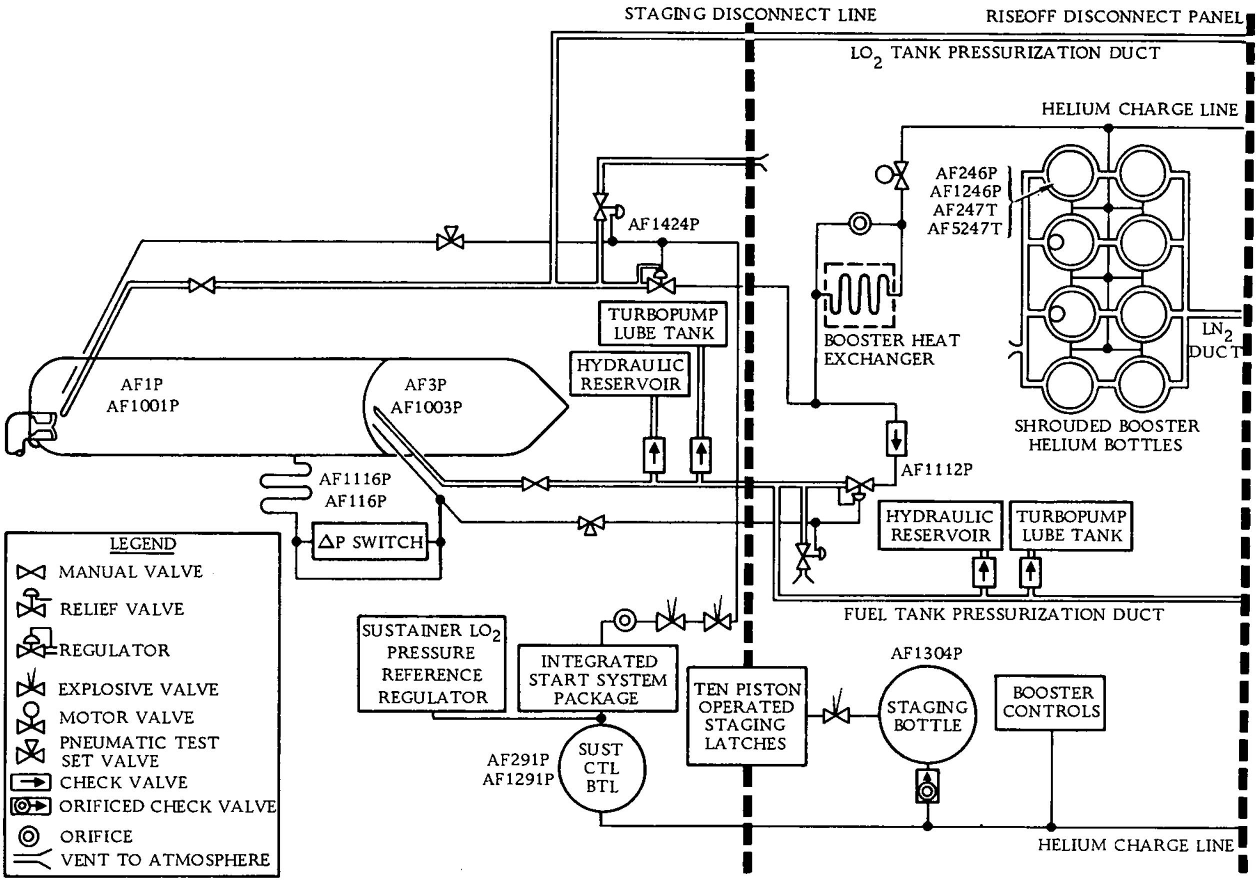


Figure 6-12. Pneumatic subsystem.

6.5.1 TANK PRESSURIZATION SUBSYSTEM. The tank pressurization subsystem maintains propellant tank pressures, for structural integrity, during the terminal period of countdown (approximately the last 60 seconds before liftoff) and during flight until booster jettison. Tank pressurization is not required during sustainer phase. Nominal tank pressures during flight are:

Fuel Tank: 64 to 67 psig

LO_2 Tank:
First 20 seconds: 27 to 31 psig
After 20 seconds: 32 to 35 psig

In addition to the primary function of pressurizing the fuel propellant tank, the fuel pressurization duct also supplies helium to the booster and sustainer hydraulic reservoirs and lube oil tank pressurizing valve. Helium is used for tank pressurization because of its light weight, chemical stability, and good thermodynamic properties.

The tank pressurization subsystem consists of four major components:

1. Helium Storage Vessels. Eight 7365-cu. in. helium vessels are mounted in the thrust section to provide inflight tank pressurization. The bottles are covered with shrouds which are filled with liquid nitrogen during countdown. The combination of high pressure and low temperature increases the helium density substantially and allows storage of a large mass of helium in a relatively small volume. Redline helium storage conditions for countdown are:

   3400 psig at -299°F
   3100 psig at -315°F

   At liftoff the helium storage ground charge line is closed by a spring loaded riseoff disconnect. LN_2 is allowed to drain out of the shrouds following liftoff.

2. Motor-Operated Shutoff Valve. A motor-operated valve is used to isolate the helium storage vessels from the tank pressure regulators during ground operations, until the last minute before liftoff. A motor operated valve, which is relatively slow-opening, is used to minimize pressure shock at the regulator inlet.

3. Heat Exchanger. During flight the helium is expanded through a heat exchanger to decrease its density and increase its pressurization efficiency. Outlet temperature is approximately 300°F. The heat exchanger is installed in the turbine exhaust duct.

4. Regulators. Remote sensing pressure regulators are used to reduce bottle pressure to the required tank pressures. The fuel regulator is mounted in the thrust section, the LO_2 regulator on the sustainer Quad IV jettison rail. The regulators are required to maintain the specified tank pressures with inlet pressure that varies from the initial bottle charge pressure to less than 150 psig, helium flow demands varying from zero to more than 0.6 lb/sec., and inlet temperatures from -65°F to 300°F.

   LO_2 tank pressure must be less than the normal regulation pressure during the first 20 seconds of flight to avoid possible damage to the intermediate bulkhead due to pressure and LO_2 head oscillations caused by release loads. This pressure reduction is accomplished by providing false sensing to the LO_2 pressure regulator during the first 20 seconds of flight, a technique generally identified as "programmed pressure". Helium from the ISS regulator flows through a calibrated orifice and through the regulator sensing line into the LO_2 tank, thereby decreasing LO_2 tank pressure by an amount equal to the pressure drop between the regulator and the LO_2 tank. At t + 20 seconds, two series mounted, normally open explosive valves are fired to eliminate the false sensing.

Other components installed in the pressurization system to facilitate ground operations and to provide emergency functions are:

Shutoff Valves. Manually operated "butterfly" shutoff valves are installed in the pressurization ducts, downstream of the regulators, to isolate the vehicle tanks from the pressurization system during fabrication, transport, and periods of ground operations.

Relief Valves. Remote sensing relief valves are installed downstream and adjacent to the regulators to relieve excessive pressure in event of system overpressurization.

Disconnects. Spring-loaded poppet disconnects are installed in staging and riseoff panels to provide a ground pressurization path in parallel to the airborne system and to provide continuity between the booster and sustainer sections.

LO_2 Relief and Shutoff Valve. A solenoid pilot operated relief and shutoff valve (boiloff valve) is installed on the LO_2 tank forward bulkhead. The primary function of this valve is to relieve excess LO_2 tank pressure during LO_2 tanking, but it also serves as a low pressure relief valve during periods of ground operations. The valve is activated to the closed or shutoff position during fabrication, transportation, and launch.

Differential Pressure Switch. A differential pressure switch, sensing LO_2 and fuel tank pressures, is installed across the intermediate bulkhead in the B-1 pod. The switch initiates warning signals if low bulkhead differential pressure occurs during transportation and ground operations.

Checkout Fittings. Checkout disconnect fittings are installed in the regulator sensing lines to allow system tests.

6.5.2 ENGINE CONTROLS SYSTEM. A 4650-cu. in. helium storage vessel is mounted on the sustainer section to provide engine controls pressurization. Spring-loaded poppet disconnects are mounted on riseoff and staging disconnect panels to provide bottle charge. The bottle is charged with ambient helium to 2900 to 3400 psig.

The engine controls' bottle supplies helium to the booster engine control regulator, booster and sustainer LO_2 reference regulators, and ISS regulator.

6.5.3 THRUST SECTION SEPARATION SYSTEM. An 886-cu. in. storage vessel mounted in the thrust section provides the pneumatic supply for the separation latches. The bottle is charged from the engine control system, but is isolated with a bleed check valve. Two parallel normally closed explosive valves are fired at booster jettison to provide helium to the separation latches.

6.6 ATLAS ASTRIONICS SYSTEMS

SLV-3D electronics are extensively integrated with the Centaur D-1A astrionics hardware and software systems. The electronics aspects of the Atlas guidance, flight control, sequencing, and telemetry systems are all provided by the Centaur. These systems are described in detail in Sections 4 and 5.

Integration of the SLV-3D electronics with D-1A Centaur is accomplished by adding an integrated electronics kit to the basic SLV-3D vehicle. This kit consists of a servo inverter unit (SIU) and a rate gyro unit for the control system, and a remote multiplexer unit (RMU) and a signal conditioner for telemetry. The electrical harnessing for the unit is provided on the basic SLV-3D Atlas.

Other electronics functions which do not lend themselves to integration are mechanized as in the SLV-3C and are described in later paragraphs.

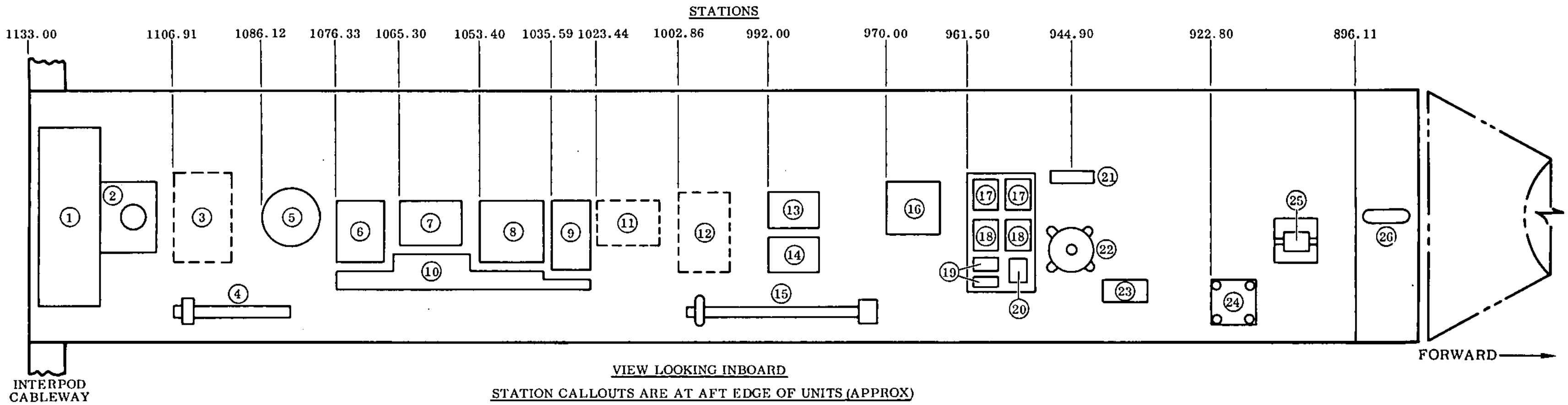
Most Atlas electronics packages are mounted in the larger side-mounted equipment pod (Number I). Equipment layout is shown in Figure 6-13. A range safety antenna and the retrorocket controllers are mounted in Pod II. (Figure 6-14). The Atlas rate gyro unit is located near the front end of the Atlas LO_2 tank.

6.6.1 GUIDANCE AND CONTROL. Steering commands from Centaur are accomplished by gimbaling the Atlas engines. The following packages were added to SLV-3D.

Servo Inverter Unit. This unit is a modified Centaur SIU. It positions the SLV-3D engines to the positions commanded by the Centaur digital computer unit (DCU). It provides a-c power for the PU system, rate gyro unit, and engine feedback transducers. The frequency of this power is synchronized by a Centaur reference signal.

Rate Gyro Unit. This package provides rate signals to the Centaur DCU for vehicle stabilization. Outputs of the rate gyro unit are also used during ground wind monitoring of the vehicle on the launch pad.

6.6.2 INSTRUMENTATION AND TELEMETRY. The instrumentation and telemetry function on Atlas SLV-3D airborne is slave to the Centaur computer and transmitter.



1 UMBILICAL PANEL
2 PNEUMATIC INLET POD COOLING
3 SIGNAL CONDITIONER
4 FUEL MANOMETER
5 ENGINE RELAY BOX
6 MAIN BATTERY
7 POWER CHANGEOVER SWITCH
8 ELECTRICAL DISTRIBUTION BOX
9 PROPELLANT UTILIZATION COMPUTER
10 TRANSDUCER AREA
11 REMOTE MULTIPLEXER UNIT
12 SERVO INVERTER UNIT
13 PROGRAMMED PRESSURE RELAY BOX (NO. 1)
14 BOOSTER SEPARATION RELAY BOX (NO. 2)
15 LO_2 MANOMETER
16 RANGE SAFETY POWER CONTROLLER
17 RANGE SAFETY BATTERIES
18 RANGE SAFETY RECEIVERS
19 FUEL DEPLETION CONTROLLERS
20 PROPELLANT DEPLETION RELAY
21 FUEL PROBE AND LO_2 LEVEL DISCONNECT
22 FUEL PROBE OUTLET
23 DESTRUCT UNIT
24 RANGE SAFETY HYBRID JUNCTION
25 DIFFERENTIAL PRESSURE SWITCH
26 RANGE SAFETY ANTENNA

Figure 6-13. Electronic package layout in equipment pod 1.

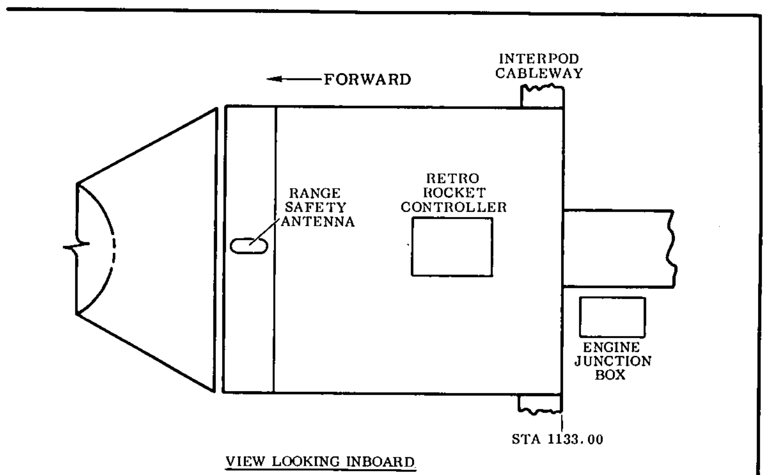


Figure 6-14. Electronic package layout in equipment pod 2.

The Atlas system (Figure 6-15) consists of transducers, a remote signal conditioner, and a remote multiplexer.

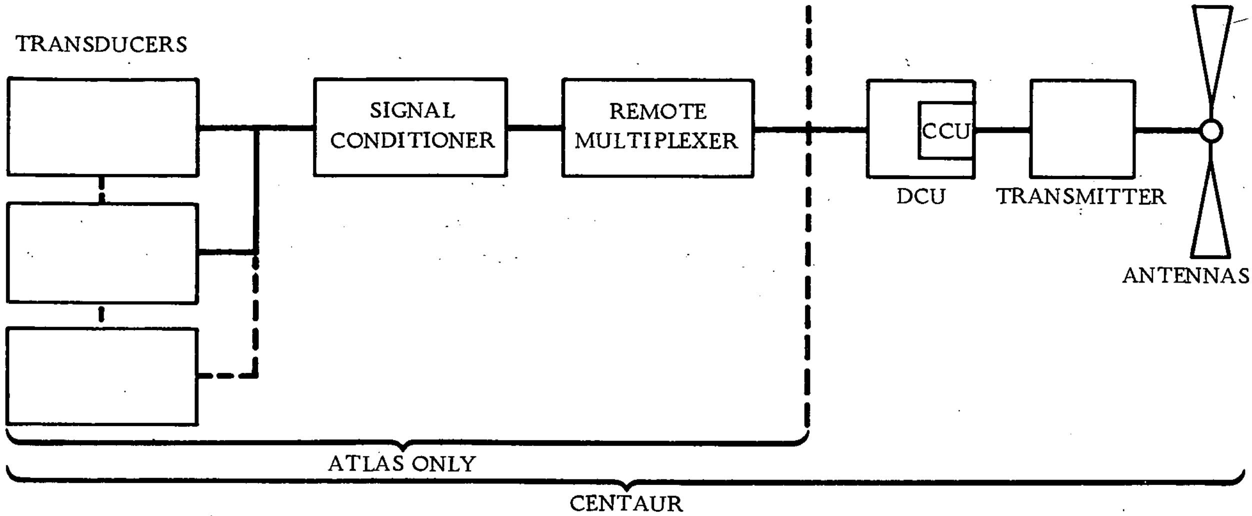


Figure 6-15. Typical Atlas/Centaur instrumentation system (airborne).

Transducer. The transducers measure the source and convert physical phenomena or parameters into electrical analogues.

Signal Conditioner. This unit provides excitation and processing for Atlas measurements.

Remote Multiplexer Unit. This unit digitizes instrumentation data from the signal conditioner and outputs it to the Centaur DCU central controller unit (CCU) on command, for transmission by the Centaur S-band telemetry.

For checkout and launch, both the PCM digital stream and an umbilical channel monitor the Atlas instrumentation system.

6.6.3 RANGE SAFETY COMMAND SYSTEM. The range safety command system enables the range safety officer to shut off the Atlas engines and destroy the vehicle if a serious flight malfunction occurs. The Atlas SLV-3D system is identical to that of the Centaur D-1A except for the antennas. Atlas antennas are blade type; tuned cavity antennas are used on Centaur.

The Atlas destructor unit is mounted within the B-1 pod at the intersection of the intermediate bulkhead and the outer tank walls. When commanded, the unit will detonate, dispersing the propellants and destroying the vehicle.

The two range safety command batteries are manually activated silver-zinc type, to which potassium hydroxide electrolyte is added prior to usage. Each battery is capable of delivering steady-state current from 60 to 180 milliamperes between 28 and 34 volts d-c. The battery has an ampere-hour capabity of 0.5. Under destruct conditions, it is required to deliver 20 amperes at 24v d-c for 1/2 second.

6.6.4 TRACKING. The Centaur D-1A system is the only tracking system used on Atlas/Centaur flights.

6.6.5 RETROROCKET SYSTEM. Eight staging retrorockets are mounted on the Atlas SLV-3D tank ring at Station 1133. Each solid propellant rocket develops a sea level thrust of 490 pounds with a total impulse of 455 pound-seconds. The orientation of the of the rockets is shown in Figure 6-16.

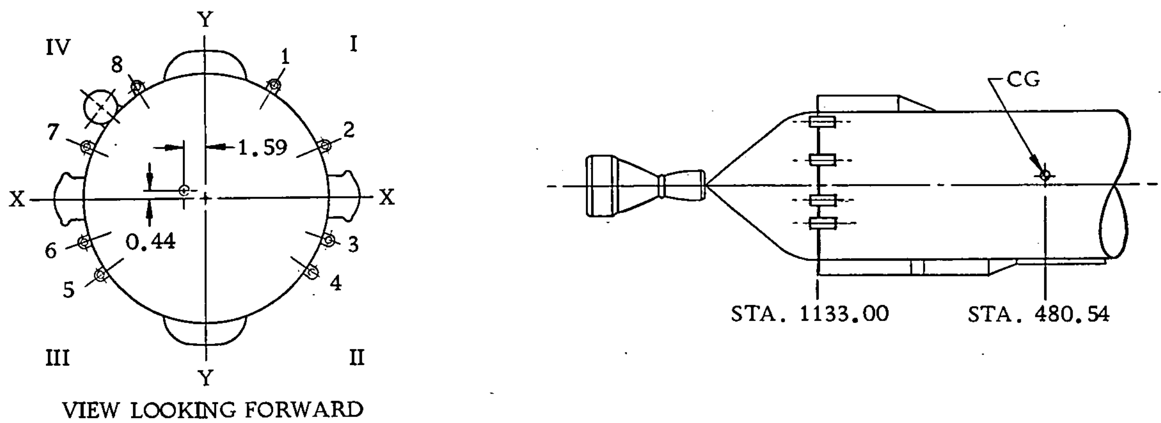


Figure 6-16. Atlas SLV-3D retrorocket orientation.

The retrorockets receive firing current from the main missile battery by way of the power changeover switch and relay assembly. They are fired upon command from the Centaur digital computer unit, two seconds after Atlas sustainer engine cutoff and 0.1 second after separation.

6.6.6 ELECTRICAL SYSTEM. The Atlas SLV-3D electrical system (Figure 6-17) supplies and distributes 28-volt d-c power to the various Atlas systems during checkout, countdown, and flight operations. It also provides remote internal/external power changeover switching capability.

Main Electrical System. The electrical system is capable of independent operation during flight and integrated operation during ground checkout and countdown. The Atlas 28v d-c electrical system is supplied by one battery and divided into four busses. Bus 1 powers the instrumentation signal conditioner and multiplexer; Bus 2 powers the servo inverter unit and the propellant utilization computer; Bus 3 provides power to Centaur Bus 4 and all Atlas electromechanical devices; and Bus 4 provides power for the Atlas pyrotechnics and Atlas/Centaur separation pyrotechnics. 115V, 400 Hz, a-c power for the Atlas propellant utilization system is provided by the servoinverter unit (SIU), part of the integrated electronics kit.

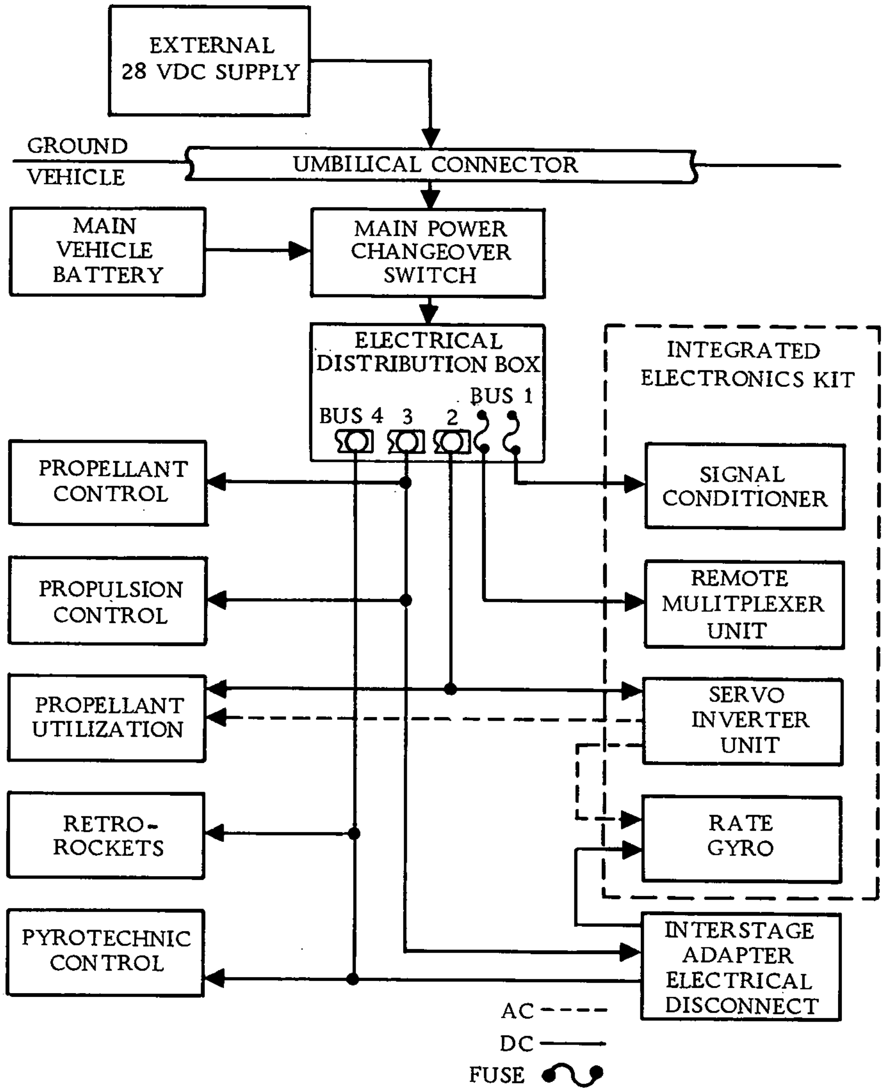


Figure 6-17. Main electrical system block diagram.

Main Missile Battery. The main missile battery is a manually activated silver-zinc type to which potassium hydroxide electrolyte is added prior to usage. The battery is capable of delivering steady-state current from 40 to 115 amperes at a voltage between 26.5 and 30 volts d-c. Total available capacity is 23 ampere hours. The SLV-3D requires a maximum inflight load of 21 amperes at 26.5 to 30 volts d-c. The battery is partially discharged before installation to remove the silver peroxide voltage and meet voltage requirements at the reduced discharge rate. Capacity remaining after discharge is 8 ampere hours. After installation and just prior to switching to internal, a load is applied to the battery to minimize the transient which occurs when transferring vehicle loads from the GSE power supply to the battery.

Main Power Changeover Switch. This distributes electrical power from either the ground (external) source, or from the main vehicle battery (internal) to the airborne systems. The switch is capable of remote operation and of transferring from the d-c ground source to the airborne source or vice versa. The a-c circuits provide break-before-make operation and the d-c circuits provide make-before-break contact connection.

The changeover switch switches and carries the following rated loads with less than a 300-millivolt voltage drop across each switch circuit:

115v a-c, 400 Hz, 5 amps (15 circuits)
25 to 30v d-c, 200 amps (1 circuit)
25 to 30v d-c, 25 amps (4 circuits)

Harnessing and Distribution. Electrical harnesses are designed with H-film wire and MIL-C-83723 connectors for enhanced reliability. Wires are of minimum gauge consistent with rated load, length of run, and voltage-drop restrictions. Voltage drop in harnesses of 115-volt systems under rated load is less than 4 volts (continuous rating) or 8 volts (intermittent rating). Harness voltage drop for 28-volt systems is less than 1 volt (continuous rating) or 2 volts (intermittent rating). Where practical, electrical harnesses are mounted independently of instrumentation harnessing. Case or equipment grounding of shock-mounted equipment is provided by a flexible braid connection across the shock mount. All other equipment packages are grounded by metal-to-metal contact of the package housing to the tank structure.

Atlas/Centaur Interface. Atlas and Centaur are electrically independent in that each has primary a-c and d-c power sources that are not interconnected. The Atlas/Centaur electrical interface routes signals between the two stages. The interstage adapter staging disconnect routes instrumentation, command, monitoring, and power signals between the adapter and Centaur. The connector is separated by lanyard pull during Atlas/Centaur staging.

Two connectors provide the electrical and two connectors provide the instrumentation interface between Atlas and the interstage adapter. These connectors are not disconnected in flight, since the interstage adapter remains attached to the Atlas sustainer section at staging. A 55-pin connector routes Atlas-to-Centaur command and monitoring functions and a 14-pin connector routes power for insulation panel jettison and Atlas/Centaur separation functions.

Atlas/AGE Interface. SLV-3D monitor, command, and instrumentation circuits interface with AGE through five umbilical disconnects which are electrically ejected at two-inch liftoff. A mechanical lanyard eject system is provided for backup. The 42-inch riseoff disconnect used by Centaur programming is disconnected mechanically by a lanyard 42 inches after riseoff. A static ground is also mechanically pulled at riseoff. Provisions for two additional umbilicals are available.

INDEX

All items refer to Centaur D-1A except where specifically identified under "Atlas".

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GENERAL DYNAMICS

Convair Aerospace Division