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TITAN IIIE/CENTAUR D-1T SYSTEMS SUMMARY

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DOCUMENT APPLICABILITY

The systems descriptions provided here are applicable to Titan IIIE and Centaur D-1T vehicles starting with vehicle number TC-1.

FOREWORD

We are pleased to present this systems summary of the Titan IIIE/Centaur D-1T launch vehicle, a key element in the National Aeronautics and Space Administration space program. This summary describes the various systems of the Titan IIIE and the Centaur D-1T.

Martin Marietta Corporation and the Convair Aerospace Division of General Dynamics are proud of their role as prime contractors for the Titan IIIE and Centaur D-1T, respectively. We have prepared this brochure for your use at the direction of the NASA Lewis Research Center, Titan/Centaur Project Manager.

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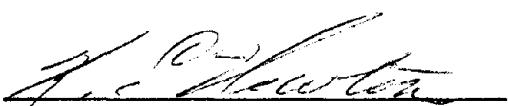
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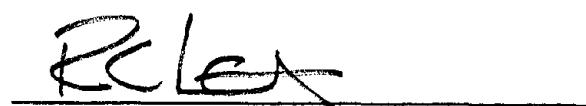
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GLOSSARY OF ACRONYMS AND ABBREVIATIONS

a-c	alternating current
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
APS	accessory power system
B/P	boost pump
B/U	backup
CCLS	computer controlled launch set
CCR	command control receiver
CCRPS	command control receiver power system
CCU	central controller unit
CCVAPS	computer controlled pressure and venting system
CDC	Control Data Corporation
CDF	confined detonating fuse
CG	center of gravity
CGS	Centaur guidance system
CIF	central instrumentation facility
CKAFS	Cape Kennedy Air Force Station
CMG	control monitor group
CRES	corrosion resistant steel
CSS	Centaur standard shroud
CTA	Centaur truss adapter
CU	converter unit
CW	continuous wave
D/A	digital to analog
DAT	design acceptance test
d-c	direct current
DCU	digital computer unit
DET	design evaluation test
DMM	data management module
DRS	data recording set
DTS	data transmission set
EED	electro-explosive device
EMV	electro-mechanical valve
ETR	Eastern Test Range
F	Fahrenheit

FAP	flight acceleration profile
FBR	forward bearing reactor
FCC	flight controls computer
FCS	flight controls system
FLSC	flexible linear shaped charge
FM	frequency modulation
FP	flight programmer
FSMG	flight safety monitor group
FS	fire switch
FS1	fire switch 1 (engine ignition)
FS2	fire switch 2 (engine shutdown)
GD/CA	General Dynamics ¹ Convair Aerospace Division
GHE	ground handling equipment
GHE	gaseous helium
GIE	ground instrumentation equipment
GLV	Gemini launch vehicle
GN ₂	gaseous nitrogen
GOAS	guidance optical alignment shelter
GPM	gallon per minute
GSE	ground support equipment
H ₂ O ₂	hydrogen peroxide
Hz	Hertz (cycles per second)
ICBM	intercontinental ballistic missile
ICT	Improved Centaur Technical (requirements)
IMG	inertial measurement group
I/O	input/output
IPS	instrumentation power system
IRU	inertial reference unit
ISA	interstage adapter
ISDS	inadvertent separation destruct system
ITL	integrate transfer and launch
LC	launch complex
LCC	launch control console
LeRC	Lewis Research Center
LHe	liquid helium
LH ₂	liquid hydrogen
LMSC	Lockheed Missiles and Space Company
LO ₂	liquid oxygen
LR	liquid rocket
LV	launch vehicle
ma	milliamperes
MDF	mild detonating fuse
MECO	main engine cutoff
MES	main engine start

MMC	Martin Marietta Corporation
MTR	mobile transfer room
N ₂ O ₄	nitrogen tetroxide
NASA	National Aeronautics and Space Administration
N.C.	normally closed
N.O.	normally open
PAM	pulse amplitude modulation
PBAN	polybutadiene acrylic acid acrylonitrile
PCM	pulse code modulation
PETN	pentaenyltetrinitrate
P/L	payload
psia, g, d	pressure, pounds per square inch absolute, gage, differential
PSV	pressure sequencing valve
PSVOR	pressure sequencing valve override
PU	propellant utilization
P&W	Pratt & Whitney Aircraft
RCS	reaction control system
RDX	cyclotrimethylenetrinitramine
RF	radio frequency
RFI	radio frequency interference
RGU	rate gyro unit
RMIS	remote multiplexer instrumentation system
RMU	remote multiplexer unit
rpm	revolutions per minute
RSC	range safety command
RTI	real time interrupt
SAMSO	Spacecraft and Missile Systems Organization (USAF)
SAPS	switched accessory power system
SC	spacecraft
SCU	sequence control unit
S/D	shutdown
sec	second
SEU	systems electronic unit
SFC	squib fire circuit
SIU	servo inverter unit
SLV	space launch vehicle
SMAB	solid motor assembly building
SOCVU	standard ordnance circuit verification unit
SRM	solid rocket motor
ST	staging timer
Sta	station
TARS	three axis reference system
TBS	to be supplied

TC	Titan/Centaur (usually followed by vehicle number)
TCPS	thrust chamber pressure switch
TCV	thrust chamber valve
TICS	Teledyne interpretive computer simulation
TLM	telemetry
TPS	transient power system
TVC	thrust vector control
T/W	thrust-to-weight ratio
UDMH	unsymmetrical dimethyl hydrazine
vac	vacuum
VECOS	vehicle checkout set
VIB	vertical integration building
VPDC	van power distribution control
VTA	Viking transition adapter
XDS	Xerox data systems

1

TITAN IIIE/CENTAUR D-1T PROGRAM SUMMARY

1.1 INTRODUCTION TO TITAN/CENTAUR

Booster: Titan IIIE (Stage 0, Stage I, and Stage II)

Third Stage: Centaur D-1T

Fourth Stage: TE-M-364-4 (Optional)
Solid Motor

Launch Site: Cape Kennedy Air Force
Station, Complex 41

Status: In development, operational
in early 1974

The Titan/Centaur launch vehicle shown in Figure 1-1 will be a key element in the nation's space program. Centaur is being integrated with Titan III initially for high-performance escape missions. In addition, this combination is particularly applicable for large spacecraft in synchronous orbits.

1.1.1 INTRODUCTION TO TITAN IIIE.

Titan IIIE (Figure 1-2) consists of a two-stage, liquid propellant core vehicle, with two 120-inch diameter five-segment solid rocket motors (SRMs) attached to form "Stage 0". Strap-on SRMs used as the initial stage of the Titan III vehicles are referred to as Stage 0 in order to retain the Stage I and II nomenclature for the common core.

The core vehicle consists of two stages evolved from the Titan II ICBM. Both stages use storable propellants which can

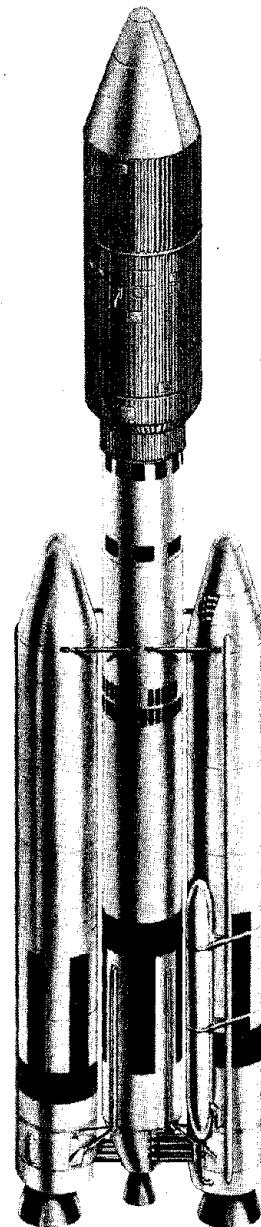


Figure 1-1. Titan/Centaur.

be left aboard the launch-ready vehicle over extended periods. The fuel is a 50/50 mixture of hydrazine and unsymmetrical dimethyl hydrazine (UDMH). The oxidizer is nitrogen tetroxide. These propellants are hypergolic, that is, they burn spontaneously when mixed, requiring no igniter.

The Titan IIIE is 98 feet tall without the Centaur upper stage. Because of the parallel Stage 0 arrangement, Titan IIIE is a broad vehicle, measuring slightly more than 30 feet across at the extreme. The vehicle weighs approximately 1,368,000 pounds when all stages are fully fueled.

Solid Rocket Motor Stage 0

Height: 85 feet, with nose fairing

Diameter, each motor: 10 feet

Weight: more than 500,000 pounds for each solid propellant motor

Titan Thrust, both Motors: 2.4 million pounds

Segment Heights:

Center: 129 inches

Aft: 63 inches

Forward: 95 inches

Burn Duration: 117 seconds

Core First Stage

Height: 63 feet

Diameter: 10 feet

Weight Fueled: 277,000 pounds

Weight Empty: 15,000 pounds

Engine Thrust: more than 520,000 pounds

Burn Duration: 146 seconds

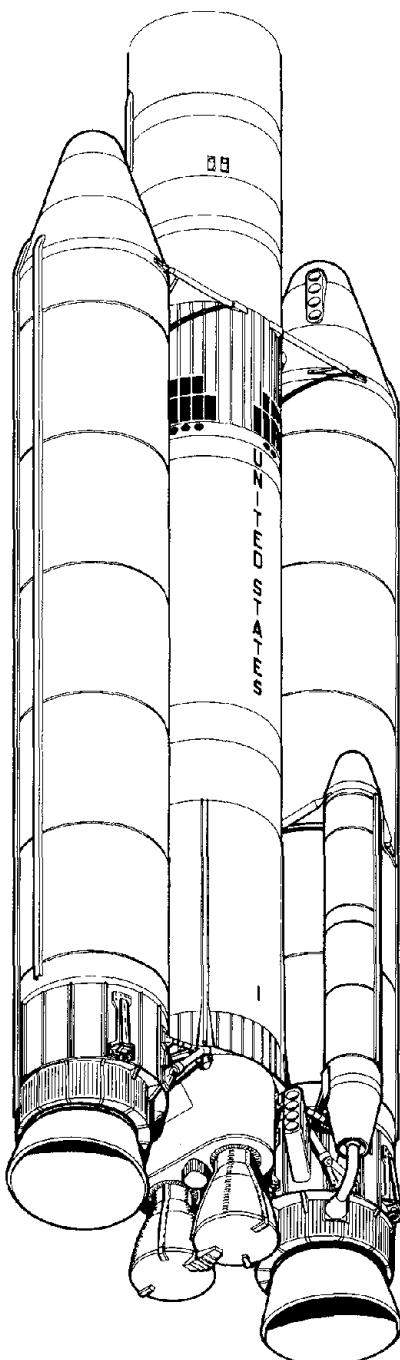


Figure 1-2. Titan IIIE.

Core Second Stage

Height: 23 feet

Diameter: 10 feet

Weight Fueled: 23,000 pounds

Weight Empty: 6,000 pounds

Engine Thrust: 101,000 pounds

Burn Duration: 210 seconds

Titan Family. Titan IIIE is the newest member of the Titan family of launch vehicles. Figure 1-3 depicts the evolution of the Titan family of missiles and space boosters. Over 400 of these have been built or are on order. This number spans the Titan I, Titan II, Gemini launch vehicle (GLV), Titan IIIA, IIIB, IIIC, IIID, and now the IIIE. Following is a brief description of each of the presently operational members of the Titan III family.

The Titan IIIB configuration, launched from Vandenberg Air Force Base for United States Air Force missions, is composed of the Titan III common core and the Agena upper stage.

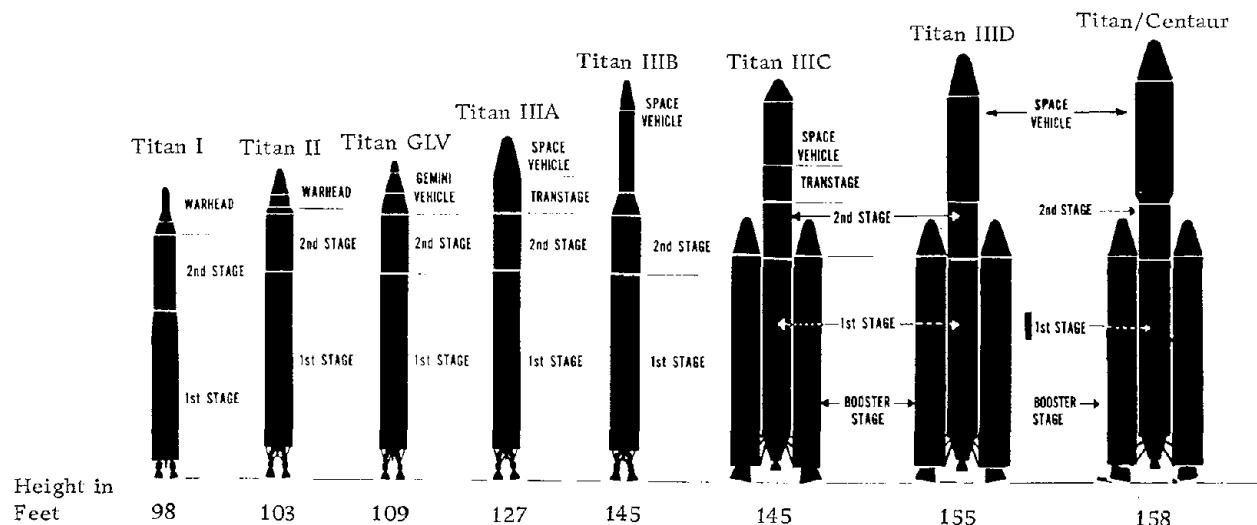


Figure 1-3. Titan family of missiles and space boosters.

Titan IIIC consists of the common core with two 120-inch diameter, five-segment, solid rocket motors for its liftoff stage and the Transtage as its upper stage. Launched from the Cape Kennedy Air Force Station, IIIC has a versatile multi-orbit, multi-payload capability due to the restartable Transtage. It is capable of launching payloads with a weight in excess of 28,000 pounds into low-earth orbit.

Titan IID is designed for United States Air Force missions. This configuration uses two five-segment, 120-inch diameter SRMs identical to those of Titan IIIC and IIIE, in conjunction with the common core. Titan IID is launched from Vandenberg Air Force Base.

Titan Reliability. The high reliability of all members of the Titan III family is greatly enhanced through the common core concept, wherein Stages I and II of all Titan III vehicles are essentially identical. The core for all configurations uses common parts, common drawings, and common manufacturing and testing processes. Modifications to meet specific mission requirements are made with discrete kits that do not degrade the reliability integrity of the basic core. The use of propellants storable at ambient temperature and pressure, eliminates the holds and delays inherent in the handling of cryogenic liquids. This feature gives Titan IIIE the capability of meeting critical launch windows within two seconds of any preestablished time.

Mission reliability of the Titan vehicle family has been demonstrated by every operational configuration of the Titan launch vehicle. As of the date of this publication, Titan has flown 55 consecutive successful missions. These missions include placement of heavy payloads into low-earth orbit and single or multiple satellites into the synchronous equatorial corridor.

1.1.2 INTRODUCTION TO CENTAUR D-1T. Centaur D-1T is a high-energy upper stage. Liquid hydrogen/oxygen propellants and pressurized stainless steel structure provide a top performance vehicle. The Centaur D-1T Proof Flight vehicle is shown in Figure 1-4.

The Centaur D-1T is 31.5 feet tall (including truss adapter) without the interstage adapter and Centaur standard shroud, and 10 feet in diameter. The vehicle weighs approximately 34,000 pounds when filled with propellants and has a jettison weight of approximately 4400 pounds, depending on the mission.

The Titan/Centaur interstage adapter (ISA) is 9.5 feet tall and 10 feet in diameter. The ISA supports the Centaur D-1T atop the Titan IIIE and also serves as the lower support structure for the Centaur standard shroud.

The Centaur standard shroud (CSS) is a large volume, biconical/cylindrical metal structure which encloses the payload, Centaur D-1T vehicle, and part of the ISA.

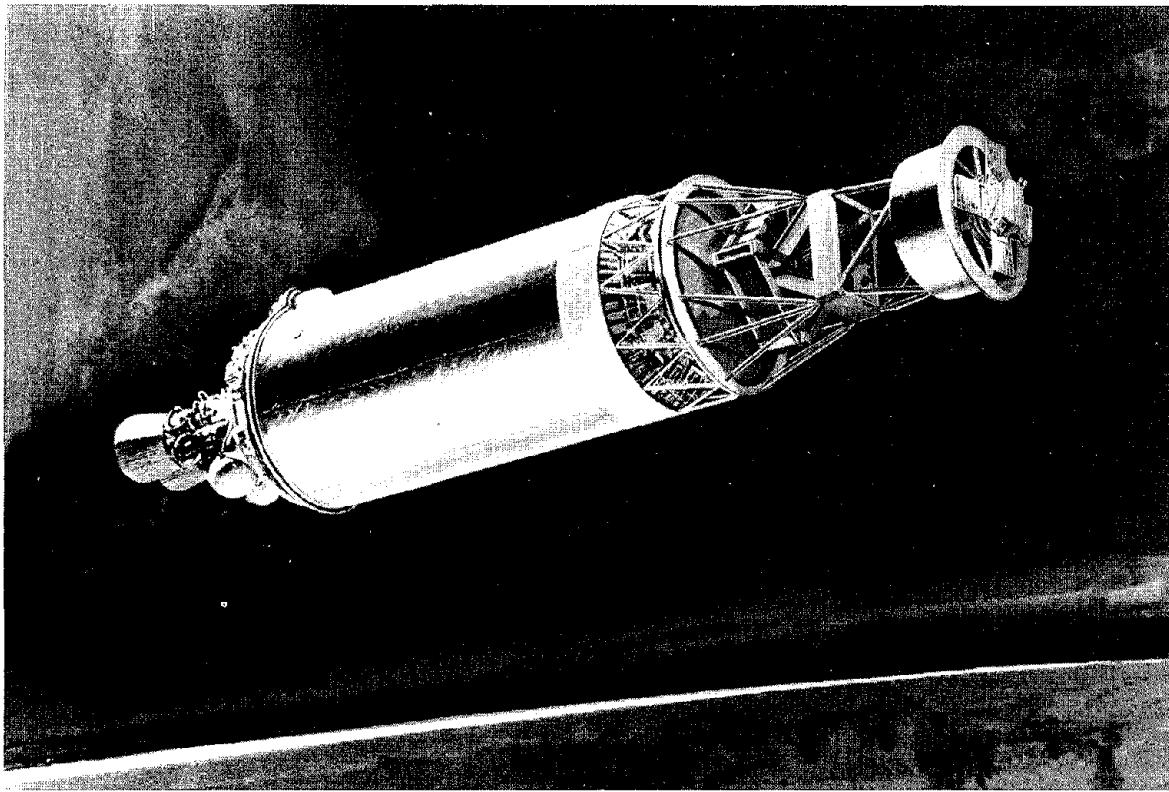


Figure 1-4. Centaur D-1T Proof Flight.

The overall CSS is 58 feet tall. The cylindrical portion of the CSS is 14 feet in diameter and 42 feet tall. The biconic nose portion of the CSS is 16 feet tall. The CSS payload envelope is generally 150 inches in diameter, and 344 inches high within the shroud structure. The exact payload envelope is mission peculiar.

Centaur Family. Centaur D-1T is the latest version of the nation's first high-energy upper stage. Major improvements in the astrionics and payload structure provide increased mission flexibility at lower cost. The Centaur D-1T payload structure is designed to accept payloads up to 12,000 pounds.

Centaur D-1T is the designation for the model that flies with Titan IIIE. The Titan/Centaur and CSS will be operational in early 1974 following a four-burn Proof Flight. The Centaur D-1T is scheduled to fly the Viking, Helios, and Mariner-Jupiter-Saturn missions.

Centaur D-1T has a sister vehicle, the Centaur D-1A which is the second stage for the Atlas SLV-3D.

The Atlas SLV-3D/Centaur D-1A became operational in early 1973 with the flawless launch of the Pioneer G.

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.

The Atlas SLV-3D/Centaur D-1A configuration is launched from Complexes 36A and 36B.

Centaur Reliability. The D-1 Centaur vehicles were designed so that maximum commonality exists between the D-1T and D-1A versions. Maximum use was made of the Centaur D designs consistent with the improvements in the avionics and payload support systems. In addition, all Centaur D-1T and D-1A vehicles, starting with AC-36, have redundant mechanical systems to enhance vehicle reliability.

1.2 MANAGEMENT RESPONSIBILITIES

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

The Titan IIIE is produced by the Martin Marietta Corporation, Denver Division, under contract to USAF Space and Missile Systems Organization (SAMSO) for LeRC. Design and specifications for the Titan IIIE are developed by MMC under contract to LeRC.

The Centaur standard shroud (CSS) is purchased by LeRC from Lockheed Missiles and Space Company (LMSC).

General Dynamics' Convair Aerospace Division is the prime contractor for Centaur D-1T. Convair Aerospace also builds the Titan/Centaur interstage adapter and is responsible for integrating the CSS into the launch vehicle.

Associate contractors for the Titan IIIE are:

Aerojet Liquid Rocket Company	Engines
United Technology Center	Solid Rocket Motors

Associate contractors for the Centaur D-1T are:

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

1.3 FACILITIES

1.3.1 TITAN ASSEMBLY FACILITIES. The core of the Titan IIIE vehicle is assembled and tested at the Denver, Colorado plant of the Martin Marietta Corporation (Figures 1-5 and 1-6). Final manufacturing testing of the core vehicle is performed in the vertical test facility (Figure 1-7). The TIIIIE core is transported by C-5A aircraft (Figure 1-8) from Denver, Colorado to the Cape Kennedy Air Force Station (CKAFS). The solid rocket motors are shipped by rail from the United Technology Center at Sunnyvale, California directly to the integrate, transfer, and launch (ITL) facility. Total vehicle assembly is performed at the ITL facility.



Figure 1-5. Martin Marietta Corporation, Denver, Colorado Plant.

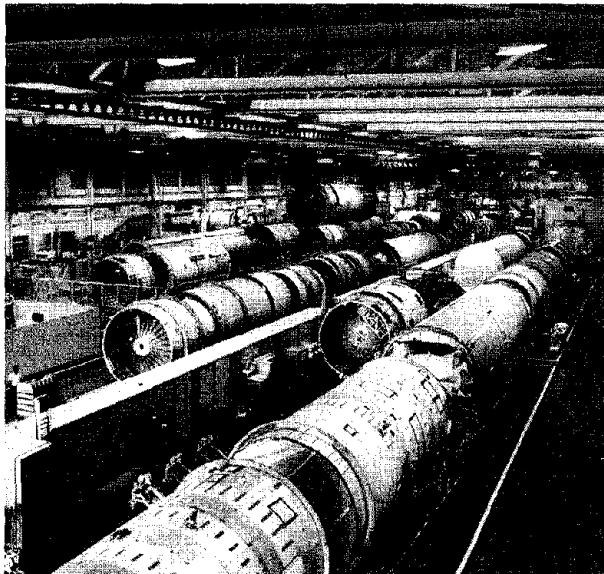


Figure 1-6. Titan III assembly line.

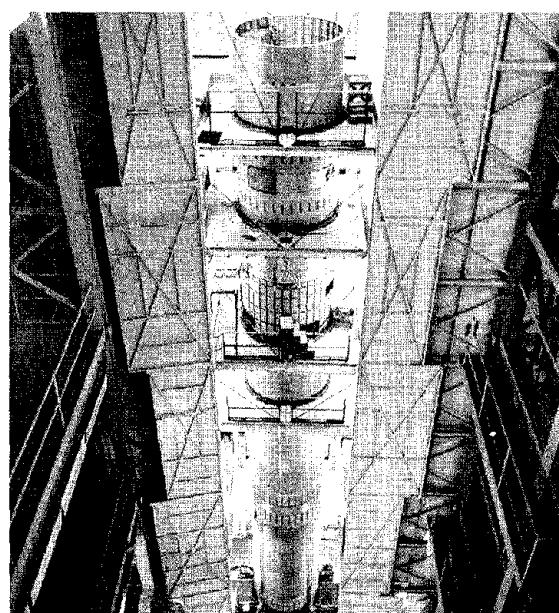


Figure 1-7. Titan IIIIE core vehicle in vertical test facility.

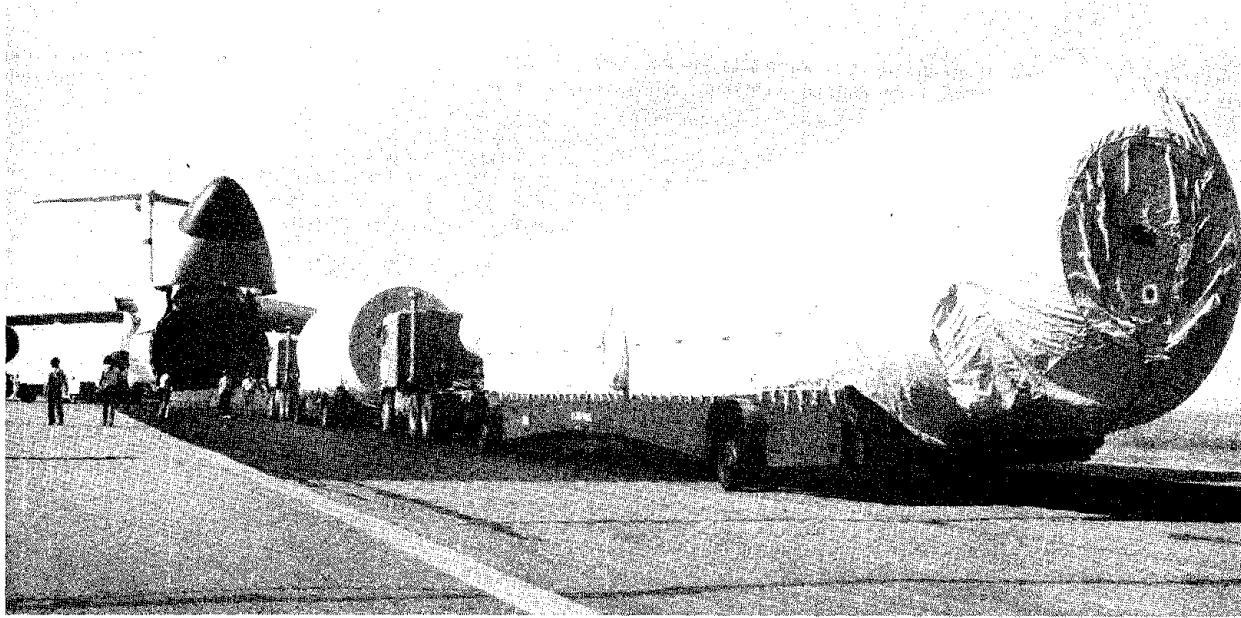


Figure 1-8. Titan IIIE transportation.

1.3.2 CENTAUR ASSEMBLY FACILITIES. Centaur is assembled by Convair Aerospace in San Diego, California. Final assembly of the vehicle is done at the Convair Aerospace Kearny Mesa facility (Figures 1-9 and 1-10).

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 Cape Kennedy Air Force Station. Figure 1-11 shows the loading of TC-1 into the C-5A.



Figure 1-9. Convair Aerospace Division of General Dynamics Kearny Mesa Plant.

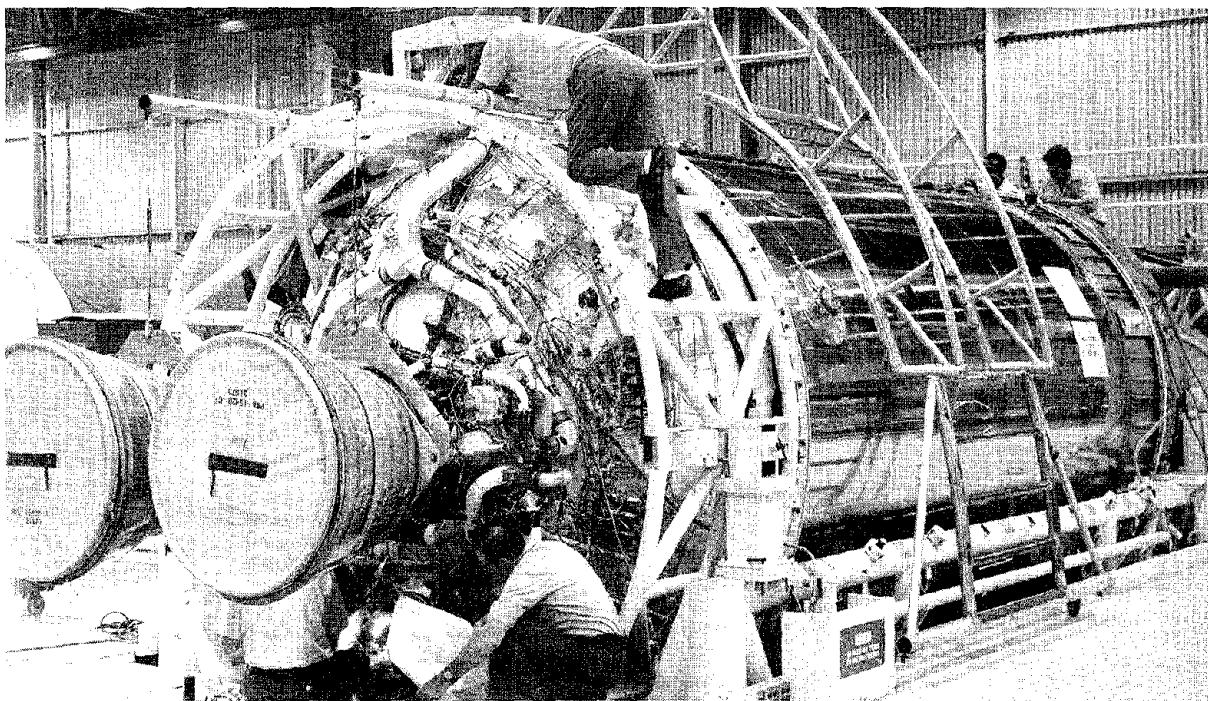


Figure 1-10. Convair Aerospace Centaur assembly area.

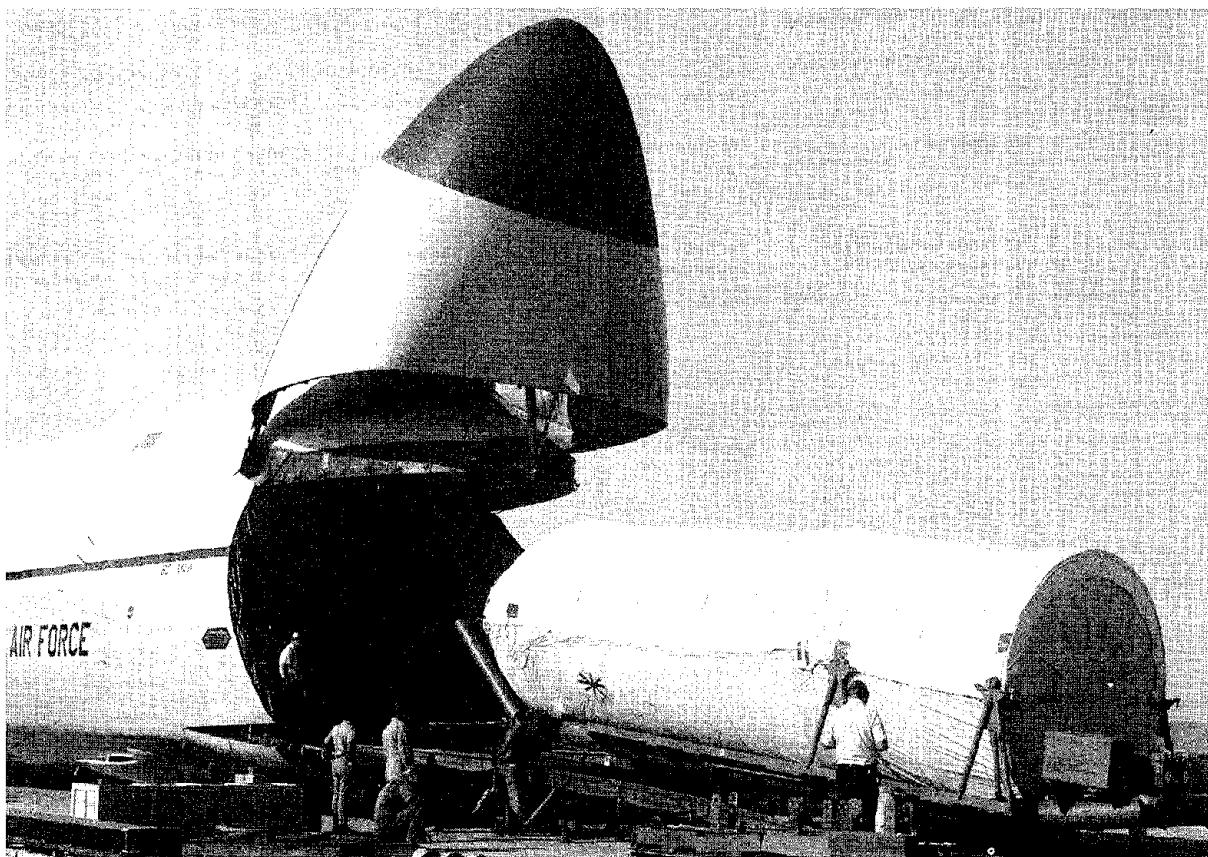


Figure 1-11. Loading D-1T for shipment.

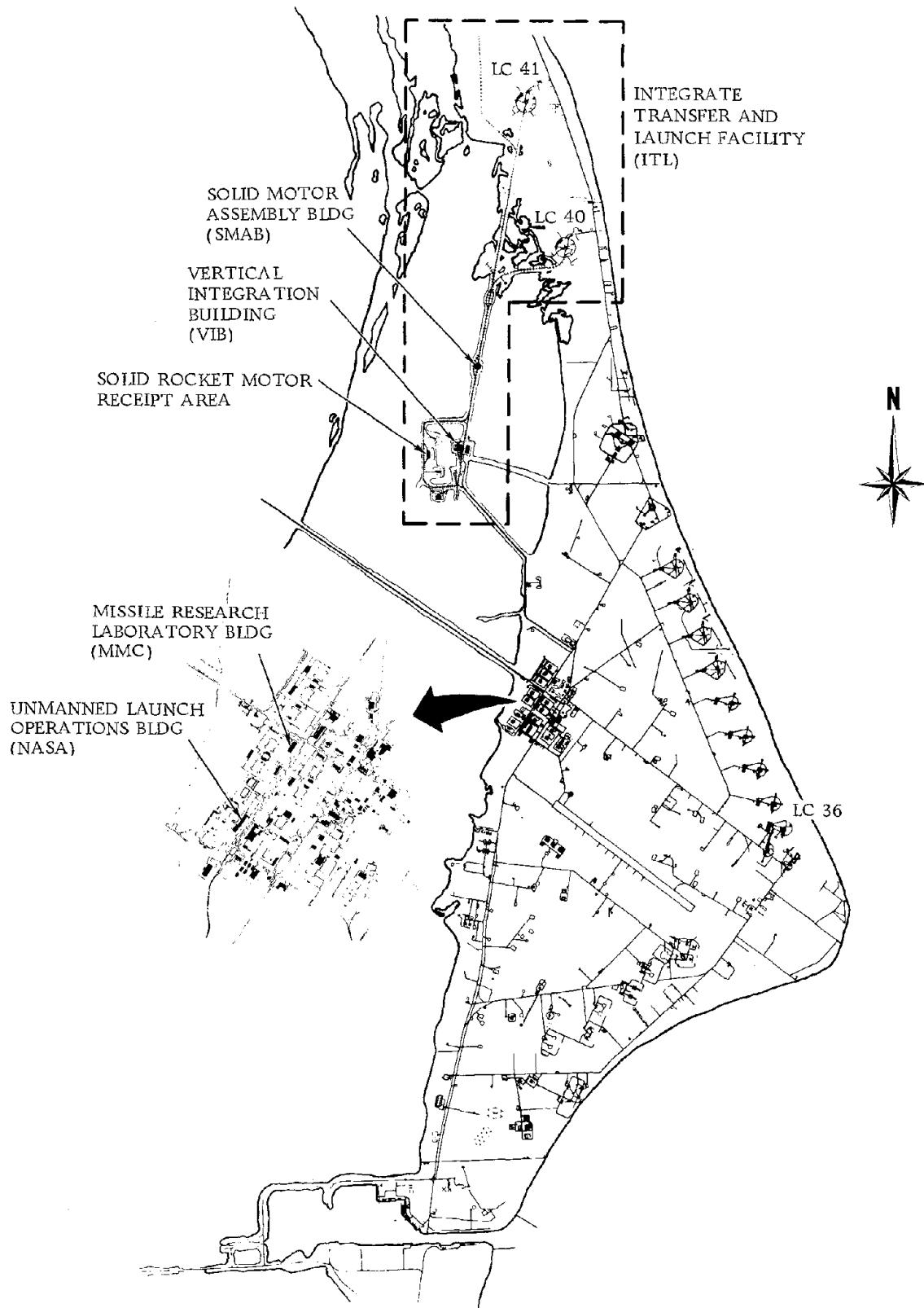


Figure 1-12. Cape Kennedy Air Force Station.

1.3.3 TITAN/CENTAUR LAUNCH FACILITY. Titan/Centaur will be launched from Launch Complex 41 which is a part of the integrate, transfer, and launch (ITL) facility at the Cape Kennedy Air Force Station (Figures 1-12 and 1-13). The Titan IIIE core vehicle and the Centaur upper stage are received, inspected, erected, and totally checked out in Cell No. 1 of the vertical integration building (VIB).

The solid rocket motor components are shipped by rail from the United Technology Center plant and received in the solid rocket motor receipt and inspection area. SRM components are transported on the ITL rail system to the solid motor assembly building (SMAB), where the two solid rocket motors are assembled.

Upon completion of the Titan/Centaur core vehicle checkout, the vehicle and its associated aerospace ground equipment (AGE) are moved by rail from the VIB to the SMAB where the solid rocket motors (Stage 0) are attached to the core vehicle. The assembled Titan/Centaur launch vehicle is then moved by rail to Launch Complex 41 (Figure 1-14) for payload mating, final checkout, and launch.

The launch countdown functions for Titan/Centaur are controlled from Control Room No. 1 in the VIB. Checkout functions on the Centaur vehicle are performed by the computer controlled launch set (CCLS) located at the Launch Complex 36 blockhouse.

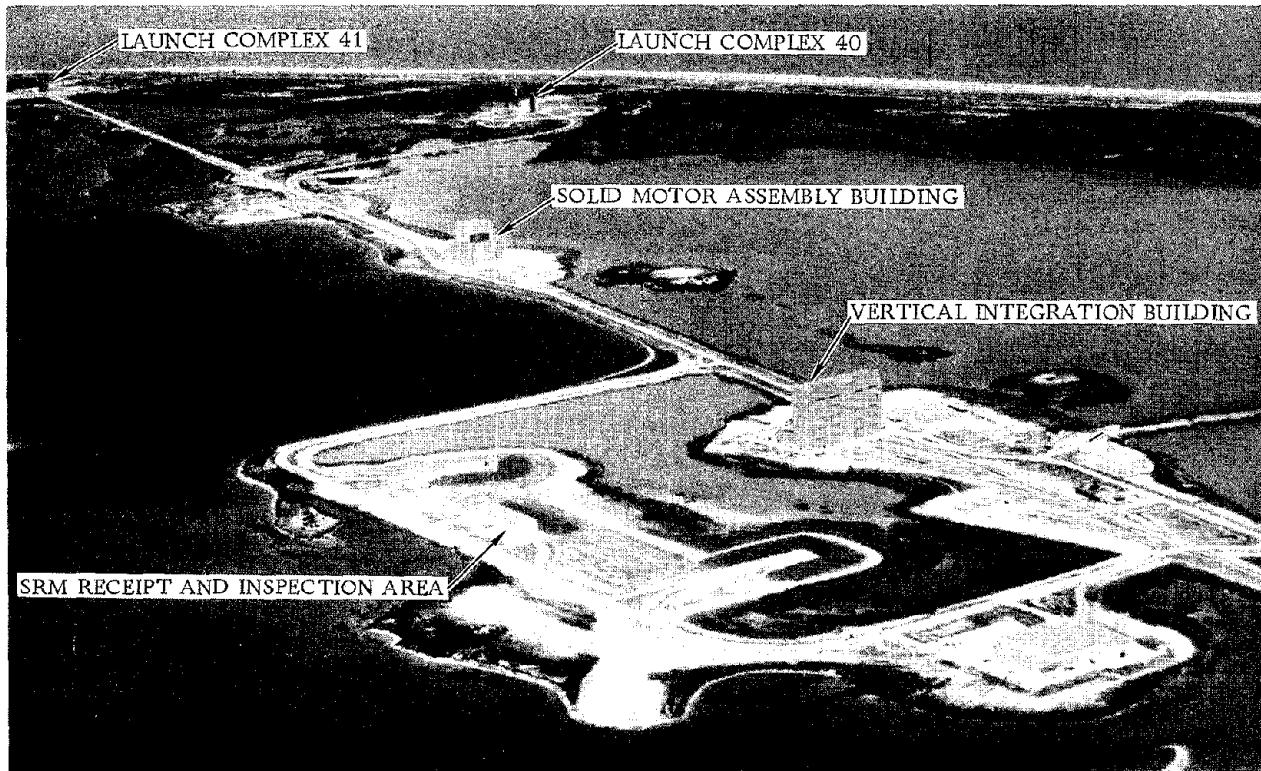


Figure 1-13. Integrate, transfer, and launch facility.

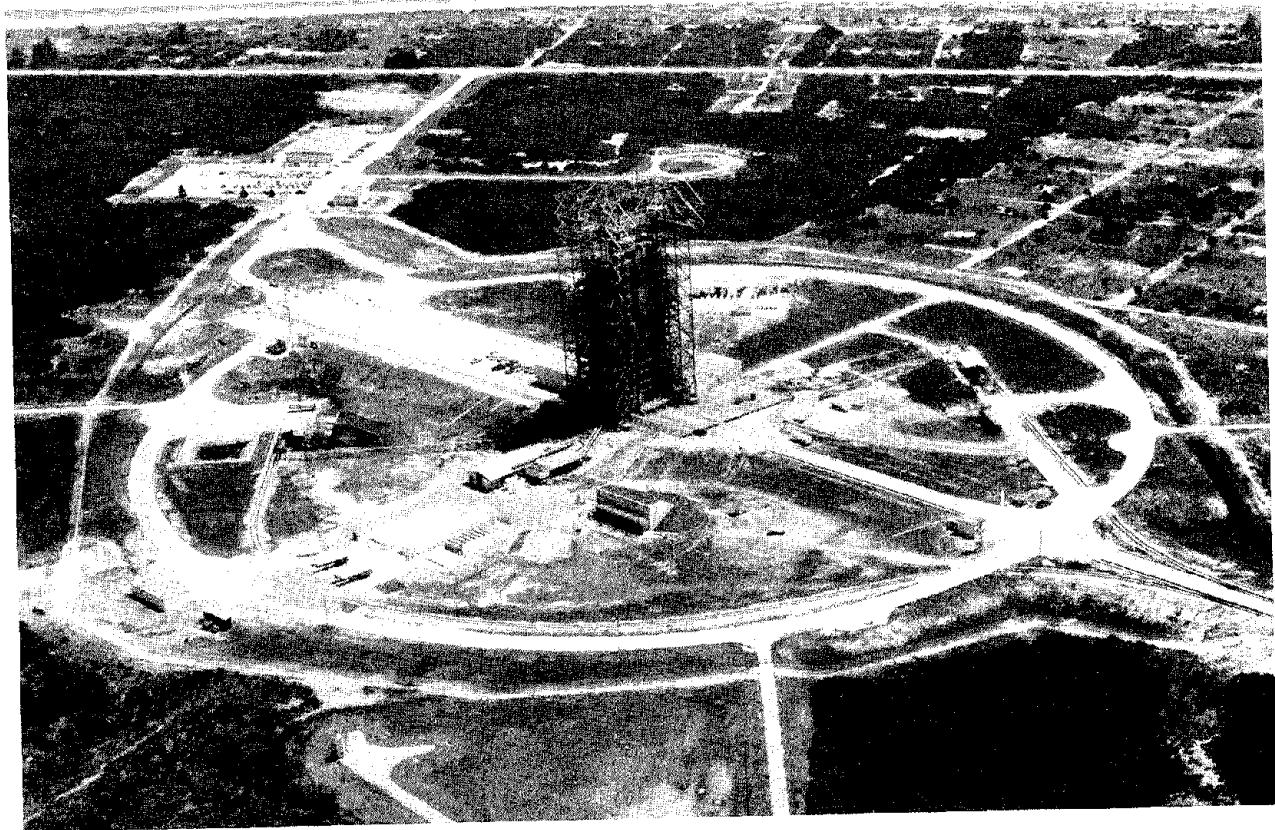


Figure 1-14. Launch Complex 41.

2

SYSTEM AND OPERATIONAL SUMMARY

2.1 TITAN/CENTAUR

The combination of Titan IIIE and Centaur D-1T is called Titan/Centaur and is illustrated in Figure 2-1. The Titan/Centaur interstage adapter supports the Centaur atop the Titan IIIE and remains with the Titan IIIE at Centaur separation. The Centaur standard shroud covers the spacecraft, which is mounted to its adapter, the Centaur vehicle, and part of the interstage adapter.

2.2 TITAN IIIE SYSTEM SUMMARY

Height: 98 feet

Diameter: 10 feet

Propulsion:

SRM:

Rated Thrust	1,200,000 lb ea.
Rated I _{sp}	266 sec
Propellants	Powdered Aluminum Ammonium Perchlorate

LR87AJ-11:

Rated Thrust	520,000 lb
Rated I _{sp}	301.1 sec (vac)
Propellants	Aerozine 50 Nitrogen Tetroxide

LR91AJ-11:

Rated Thrust	101,000 lb
Rated I _{sp}	318.7 sec (vac)
Propellants	Aerozine 50 Nitrogen Tetroxide

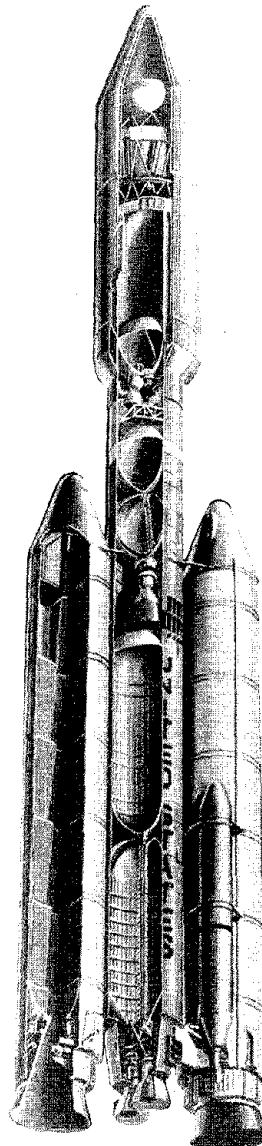


Figure 2-1. Titan/Centaur.

The Titan IIIE core vehicle is shown in Figure 2-2 in the final assembly area of the Martin Marietta Denver plant.

Airframe Structure. The structural design of the IIIE core vehicle is of frame-stabilized monocoque and conventional skin-stringer-frame construction. The primary construction material is aluminum alloy. Alloys 2014 and 7075 are used in all primary structural members. Alloy 2014 is used in the skin panel sections and alloy 7075 is used for extruded or forged frames, stringers, and longerons.

The stages are composed of major assemblies or segments within the stages. These major assemblies are provided with bolt-together interfaces called tension splices, which allow ease of installation during manufacture and facilitate replacement of major assemblies as needed.

Stage I airframe includes four major assemblies: the fuel tank assembly, the oxidizer tank assembly, the interstage structure, and the engine heat shield.

The fuel tank assembly includes an aft skirt, the fuel tank, and a forward skirt with a tension splice interface frame. The tank wall also serves as the vehicle exterior wall.

The oxidizer tank assembly includes an aft skirt with tension splice frame, the oxidizer tank, and a forward skirt with transportation splice frame.

The interstage structure is considered an extension of the oxidizer tank forward skirt. To shorten the Stage I airframe for handling and transportation purposes, the interstage structure is removed and shipped with Stage II.

The engine heat shield assembly protects the Stage I engine from the high temperatures radiated from the SRM nozzles. The engine heat shield is composed of an upper engine shroud which attaches to the aft frame of Stage I and encloses the major portion of the engine from the thrust chamber throats up.

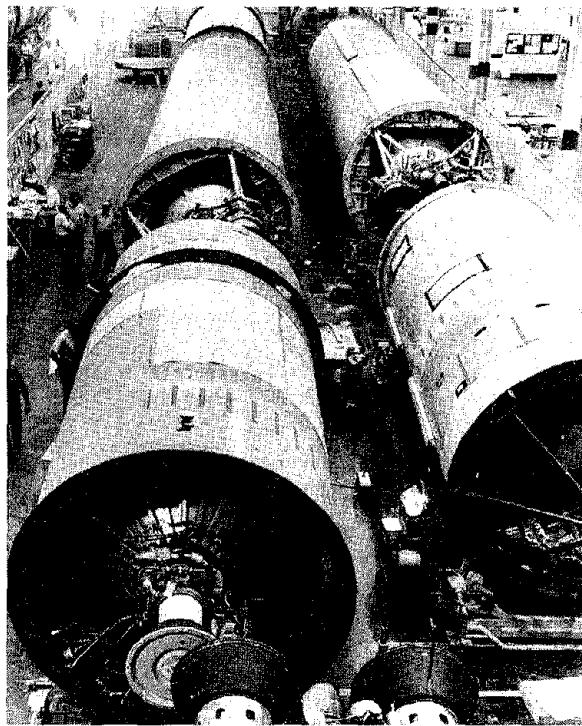


Figure 2-2. Final assembly of Titan IIIE core vehicle.

Stage II airframe includes two major assemblies: the fuel tank assembly and the oxidizer tank assembly.

The fuel tank assembly has an aft skirt with an aft staging interface, the fuel tank, and a forward skirt with tension splice interface.

The oxidizer tank assembly includes an aft skirt with tension splice frame, the oxidizer tank, and a forward skirt with a forward interface for the Centaur interstage adapter structure. An internal truss mount in the Stage II forward skirt provides mounting for flight control system components, batteries, telemetry equipment, etc.

A thermal barrier is provided at the forward end of the Stage II airframe to provide an environmental separation between the Titan and Centaur vehicles.

Tank Pressurization. The only pressurization requirements for the Titan IIIE core vehicle is within the propellant tanks. Prelaunch requirements are provided by ground-supplied GN₂ and inflight tank pressures are maintained by the autogenous pressurization system from the liquid rocket engines.

Propulsion. Primary thrust for the Titan IIIE vehicle is provided by three separate systems. The first system is the solid rocket motor propulsion system. Two identical solid rocket motors provide a total of approximately 2.4 million pounds of thrust. Burn time of this system is 117 seconds. The second system is the Stage I liquid propulsion system. The LR87AJ-11 engine provides approximately 520,000 pounds of thrust with a burn time of approximately 146 seconds. The third system is the Stage II liquid propulsion system. The LR91AJ-11 engine provides approximately 101,000 pounds of thrust and burns for 210 seconds. Both the Stage I and Stage II engines are regeneratively cooled, turbopump fed engines.

Flight Control System. The function of the flight control system is to provide vehicle attitude stability through all phases of powered flight; to control trajectory through implementation of guidance and programmed steering commands; to provide solid rocket motor thrust vector control fluid usage regulation by dumping excess on-board fluid; and to provide time and acceleration dependent discretes for implementing the flight sequence of events.

Airborne Electrical System. The airborne electrical system supplies and distributes power to the various airborne systems, provides power switches for transferring from ground to airborne power, and also contains the electrical flight sequencing system that converts command signals from the flight control system and/or guidance into event signals to perform flight functions.

Telemetry. The airborne PCM telemetry system consists of an S-band transmitter, its associated antennas, and the remote multiplexing instrumentation system

(RMIS). This system senses and transmits all measurement samplings in the vehicle. The system is capable of transmitting a maximum of 502 analog and 80 bilevel measurements.

Range Safety. The range safety system comprises two separate subsystems. The first is the command destruct subsystem. The airborne portion of this subsystem is completely redundant and consists of the hardware necessary to receive the engine shutdown and vehicle destruct signal from the Range Safety Officer.

The second subsystem is the inadvertent separation destruct subsystem. If, during flight, the vehicle should break up, a command destruct signal could not be sent from the command destruct subsystem to the stage that had broken away. To destroy the separated stage, a hot-wire circuit causes the Stage I destruct battery to power the Stage I destruct system and destroy Stage I. This same system is utilized in case of inadvertent separation of the SRMs.

2.3 CENTAUR SYSTEM SUMMARY

Length:	31.5 ft (including truss adapter - no shroud)
Diameter:	10 ft
Guidance:	Inertial
Propulsion:	P&WA RL10A-3-3
Rated Thrust:	30,000 lb (total)
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-1T vehicle is shown in the final assembly and checkout docks at Convair's Kearny Mesa plant in Figure 1-10.

Shroud System. The Centaur standard shroud (CSS) encloses both the Centaur and payload and provides environmental protection for the Centaur and spacecraft on the ground and in flight. The shroud is a biconical/cylindrical metal structure 58 feet high. The cylindrical portion is 14 feet in diameter. The CSS contains many bolt-ons such as antennas, and inflight electrical disconnects as well as the insulation system for environmental protection.

Structure and Insulation. The tank structure (Figure 2-3) cylindrical section is made from pressure-stabilized stainless steel, 0.014 inch thick. A double-walled, vacuum-insulation intermediate bulkhead separates the liquid oxygen (LO_2) from the liquid hydrogen (LH_2). The aft and intermediate bulkheads form a 1.38:1 ellipsoidal

LO_2 tank. The forward bulkhead of the LH_2 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 stub adapter is also used as the attach point for a truss type adapter when necessary to support large spacecraft. Small spacecraft (up to 4000 lb) are supported by a payload adapter attached to the forward end of the equipment module.

The entire cylindrical section of the vehicle LH_2 tank is covered by a radiation shield. The radiation shield consists of three separate layers of an aluminized Mylar-dacron net sandwich. The forward tank bulkhead and tank access door are insulated with a multilayer aluminized Mylar. The aft bulkhead is covered with a membrane which is in contact with the tank bulkhead and a rigid radiation shield supported on brackets. The membrane is a layer of dacron-reinforced aluminized Mylar. The radiation shield is made of laminated nylon fabric with aluminized Mylar on its inner surface and white polyvinyl fluoride on its outer surface.

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 is provided from two sources. With no propellant in the tank, pressure is provided by a gaseous helium system. After propellants are loaded,

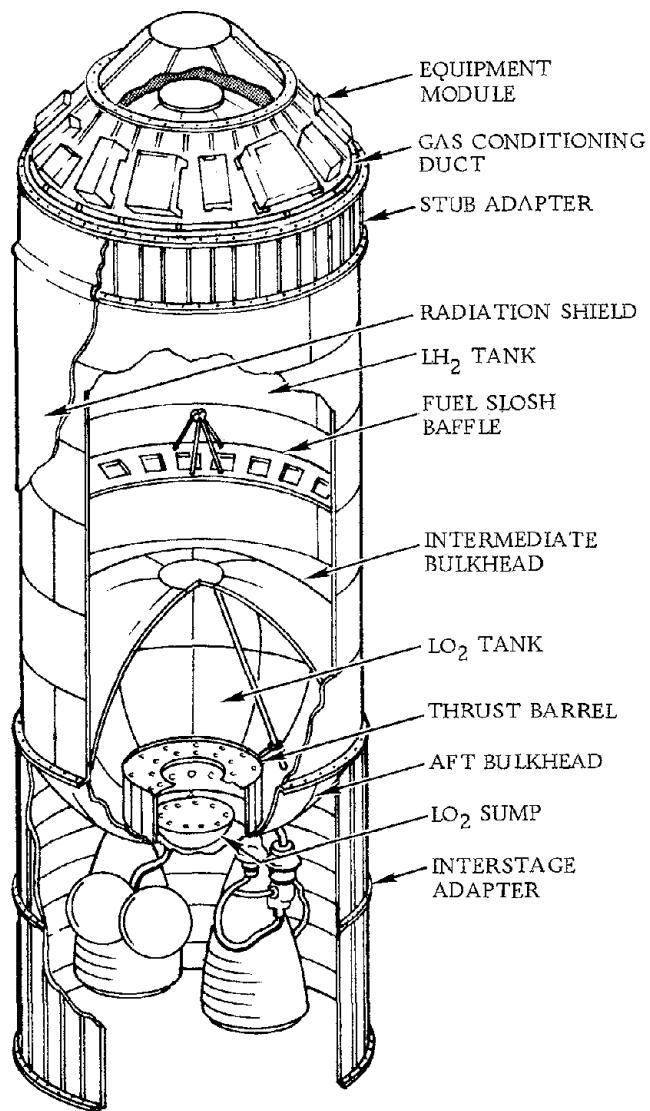


Figure 2-3. Centaur D-1T.

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.

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.

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 (6 pounds thrust) attached to the aft bulkhead. H₂O₂ is supplied to the boost pumps and the attitude control engines from two storage spheres. A 300-psi helium system pressurizes the H₂O₂ 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 2-4).

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 discrete to the sequence control unit. Engine commands go to the Centaur servoinverter unit through six digital-to-analog channels.

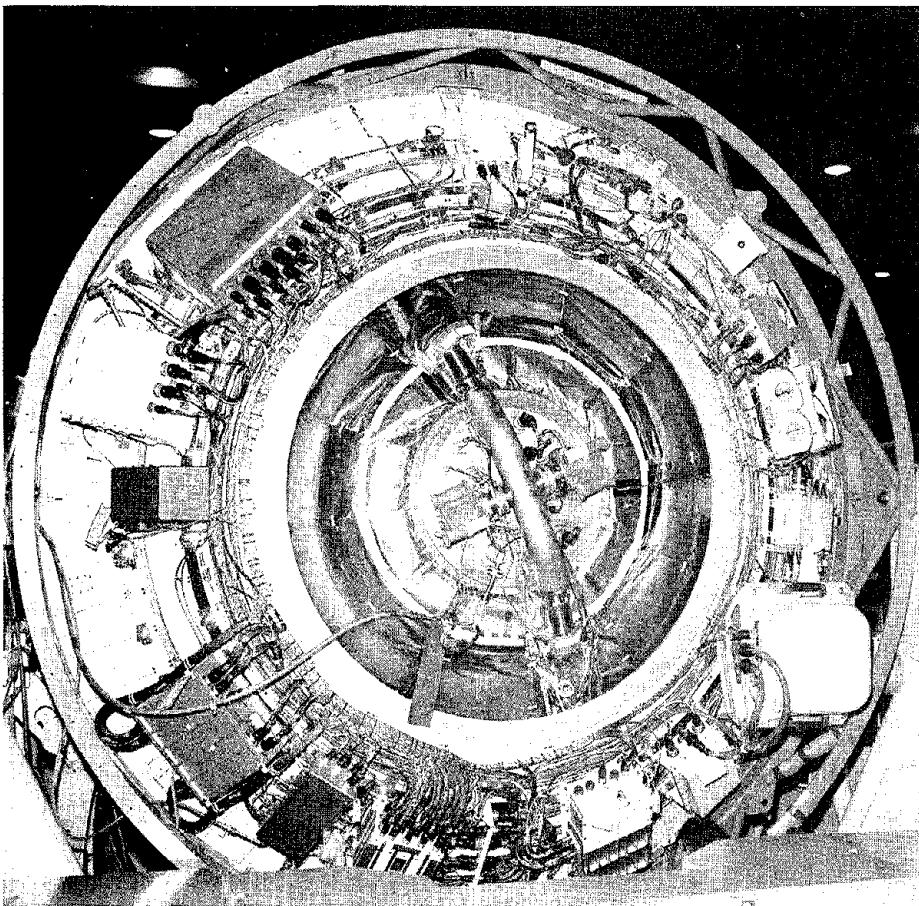


Figure 2-4. Astrionics packages mounted on Centaur equipment module.

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 and are frequency multiplexed and routed 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 (SEU) 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-1T system also provides steering commands for the Titan 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, two of which are currently used for Centaur D-1T flights.

Propellant Utilization. A propellant utilization system controls the LO₂ 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-1T is nearly completely redundant and consists of antennas, destructor and related equipment. The C-band tracking system provides ground tracking of Centaur during flight.

2.4 TITAN/CENTAUR FLIGHT SEQUENCE

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

The flight compendium illustrated in Figure 2-5 and discussed in the following paragraphs is for a Titan/Centaur mission to place a spacecraft on an escape mission.

"Liftoff" officially occurs approximately 0.2 second after solid rocket motor (SRM) ignition (T-0). Only the SRMs are thrusting at liftoff. At 6.3 seconds after liftoff, Titan starts a proprogrammed roll maneuver commanded by Centaur guidance which is complete prior to T+10 seconds. This roll maneuver and all Stage 0 attitude control is accomplished by the Titan thrust vector control (TVC) system injecting liquid nitrogen tetroxide into the SRM nozzle cone, which deflects the rocket exhaust gases.

The Titan baseline (or no wind) open-loop pitch maneuver is preprogrammed in the Titan flight control system and is initiated at liftoff +10 seconds. Wind compensation steering programs in pitch and yaw are issued from the Centaur digital computer unit (DCU) for Stage 0 flight.

Titan Stage I ignition occurs at approximately 112 seconds after liftoff. The actual time is controlled by three majority-voted acceleration switches which command ignition and initiate a staging timer when vehicle acceleration reduces to 1.5g. Approximately 12 seconds later the SRMs are jettisoned upon command from the staging timer. Titan Stage I then boosts the vehicle until propellant depletion is sensed by Stage I thrust chamber pressure switches, which occurs approximately 258 seconds after liftoff.

Titan Stage II ignition is commanded at the same time Stage I propellant exhaustion occurs (258 seconds after liftoff). Approximately one second later, Stage I/Stage II separation occurs. During Stage I and Stage II phases of flight, the vehicle attitude in pitch and yaw is controlled by the Titan flight control system with steering corrections supplied by the Centaur guidance system.

The Centaur standard shroud (CSS) is jettisoned at Stage I shutdown + 10 seconds, upon command from the Centaur guidance system.

The Centaur boost pumps are started at 181 seconds after Stage I shutdown.

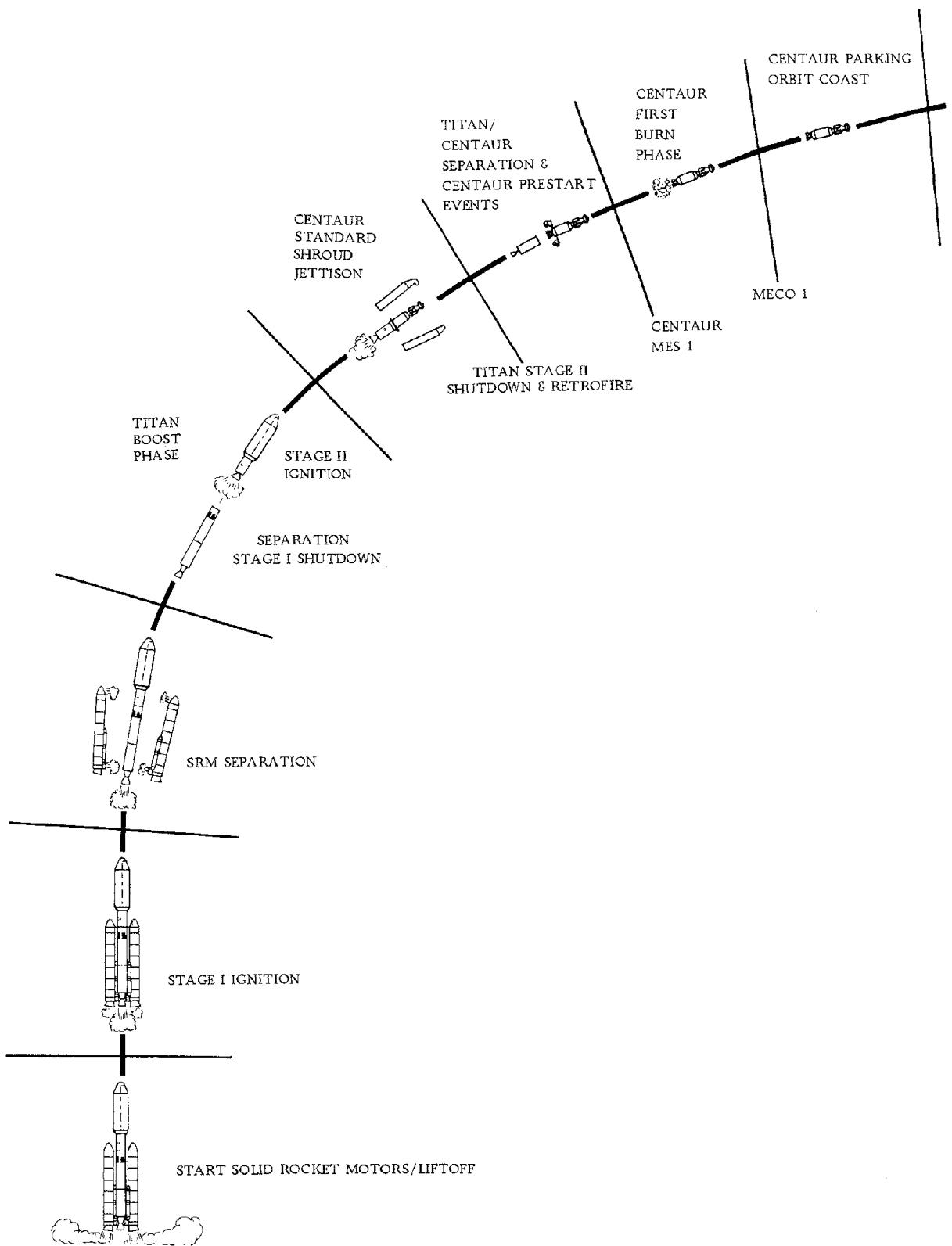
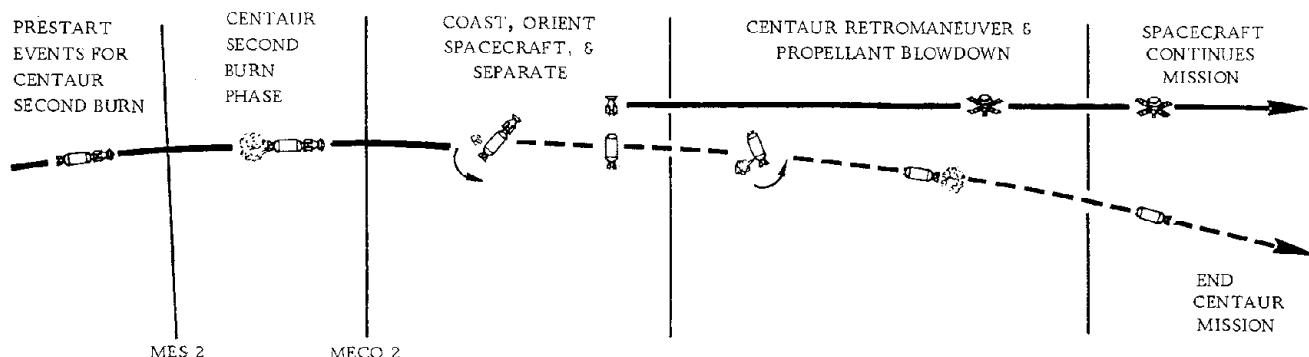


Figure 2-5. Titan/Centaur parking orbit mission.



EVENT	BASIS	APPROXIMATE TIME FROM SRM IGNITION (SEC)
SRM IGNITION (T_0)	GROUND COMMAND	0
LIFTOFF (SRMs FIRING, STAGE 0)	VEHICLE MOTION	0.2
START STAGING TIMER, T_1	1.5 g	112
CORE STAGE 1 IGNITION	$T_1 + 0.114$ SEC	112
JETTISON SRMs, T_2	$T_1 + 12$ SEC	124
STAGE 1 SHUTDOWN, T_3	PROPELLANT DEPLETION	258
CORE STAGE 2 IGNITION	$T_3 + 0$	258
SEPARATE STAGE 1 & STAGE 2, T_4	$T_3 + 1.0$ SEC	259
JETTISON SHROUD	$T_4 + 10$ SEC	269
STAGE II SHUTDOWN (ACCELERATION LEVEL = T_5)	PROPELLANT DEPLETION	468
SEPARATE TITAN/CENTAUR	$T_5 + 18$	486
MES 1 (CENTAUR MAIN ENGINE START, FIRST BURN)	$T_5 + 27.5$	496
MECO 1 (CENTAUR MAIN ENGINE CUTOFF, FIRST BURN)	PARKING ORBIT (GUID.)	612
MES 2 (CENTAUR MAIN ENGINE START, SECOND BURN)	COAST PERIOD (GUID.)	1607
MECO 2 (CENTAUR MAIN ENGINE CUTOFF, SECOND BURN)	SYNCHRONOUS TRANSFER (GUID.)	1919
SEPARATION	MECO 2 + Δt (VARIES)	2139

Titan Stage II boosts the vehicle until loss of acceleration due to propellant depletion is sensed by Titan acceleration switches; at which time Stage II shutdown is commanded (approximately 468 seconds from liftoff). When Stage II acceleration decays to approximately 0.012 g, Stage II/Centaur separation is commanded from Centaur and the pyrotechnic system releases the Centaur from the Titan/Centaur interstage adapter. Separation is achieved by simultaneously firing the retrorockets mounted on the Titan.

The Centaur main engine prestart valves are opened at 2.5 seconds after Titan/Centaur separation, providing control capability and initiating the propulsion system chilldown. In this process the fuel and oxidizer flow through the Centaur propellant feed system and engine pumps, and are vented overboard through discharge valves (LH_2) and the main engine thrust chambers (LO_2).

Centaur main engine start (MES 1) occurs about 10.5 seconds from Titan/Centaur separation. Attitude maneuvers are accomplished by gimballing the two Centaur main engines.

Centaur main engine cutoff (MECO 1) is commanded by Centaur guidance when the proper parking orbit is achieved.

During Centaur coast in the parking orbit, vehicle attitude and propellant retention is controlled by the smaller reaction control engines. All four axial engines are turned on for propellant settling at the start of coast phase. The axial engines are then turned on and off alternately in pairs during the coast period to maintain the proper axial thrust for propellant settling and to minimize the effect of a single axial engine failure. The eight lateral engines are turned on and off as necessary during the coast phase to maintain the proper vehicle attitude. A second chilldown process occurs for 17 seconds prior to MES 2.

MES 2 is initiated after the Centaur coasts for a predetermined time in the parking orbit. The second burn continues until guidance calculates that the vehicle has accelerated the spacecraft to the required velocity for injection.

After MECO 2, the lateral engines 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 like Helios, with a fourth stage, Centaur is merely backed away using a gaseous helium retrothrust system.) Then the axial engines fire to move Centaur further away from the spacecraft and continue to fire 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-1T MECHANICAL SYSTEMS

This section presents descriptions of the following Centaur D-1T mechanical systems:

SUBSECTION	SYSTEM	Page
3.1	Structural System	3-1
3.2	Insulation Systems	3-15
3.3	Shroud	3-21
3.4	Main Propulsion System	3-28
3.5	Reaction Control System	3-35
3.6	Hydraulic System	3-39
3.7	Pneumatic System	3-47

3.1 STRUCTURAL SYSTEM

ELEMENTS:	Subsection
• Tank Structure and Stub Adapter	3.1.1
• Equipment Module	3.1.2
• Interstage Adapter	3.1.3
• Titan/Centaur Staging System	3.1.4
• Payload Adapters	3.1.5
• Bolt-on Structures	3.1.6

FUNCTIONS:
• Contain Main Propellants
• Establish Primary Structural Integrity
• Provide Support for All Airborne Systems and Components
• Provide Titan Centaur Interface
• Support Payload

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

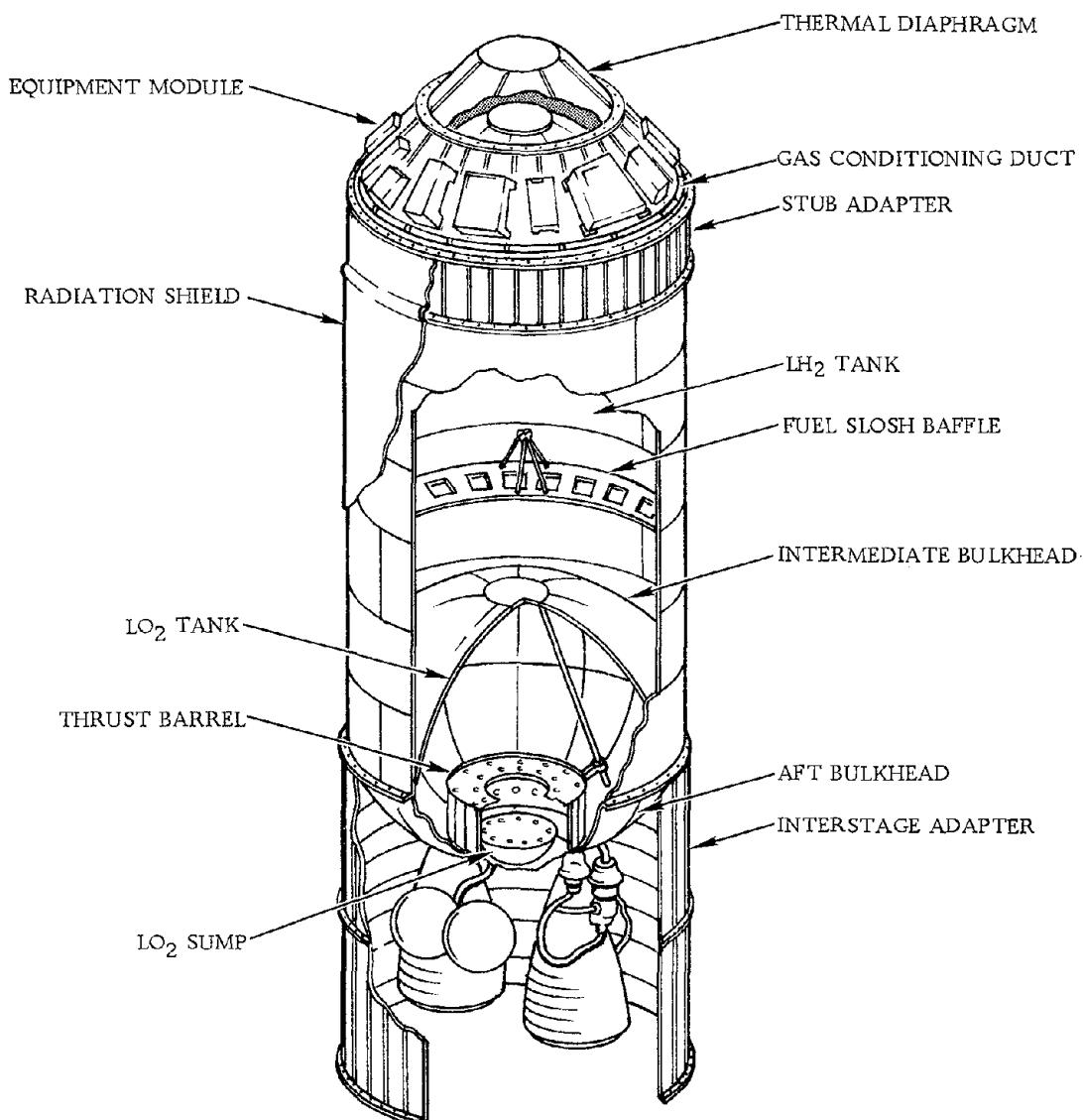


Figure 3-1. Centaur structural components.

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 (extra hard)
Dimensions (each)	0.014 in. thick × 32.0 to 34.5 in. long × 120 in. dia.

Brackets welded to the tanks walls support the various tank-mounted systems; flanged tank weldments facilitate attachment of propellant ducts. Internal brackets welded to the tank support the slosh baffle. Multiple sets of brackets are installed in each tank to allow optimum positioning of the slosh baffle for each mission. The stub adapter and the interstage adapter attach to machined CRES rings at the ends of the tank cylindrical portion.

An oxidizer sump is mounted on a flanged 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. Mechanical support is provided by applying stretch to ground handling adapters attached to the forward and aft cylindrical tank rings. After erection, structural integrity is assured by minimum 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 encapsulated payload is installed.

Intermediate Bulkhead. The intermediate bulkhead (Figure 3-3) forms the interface between the fuel and oxidizer tanks. This bulkhead is a doublewalled, ellipsoidal structure consisting of a structural bulkhead and an insulation bulkhead, separated by plastic mesh and fiberglass matt insulation. A vent tube from between the bulkheads leads to an instrumentation and emergency vent system. Both bulkheads are made

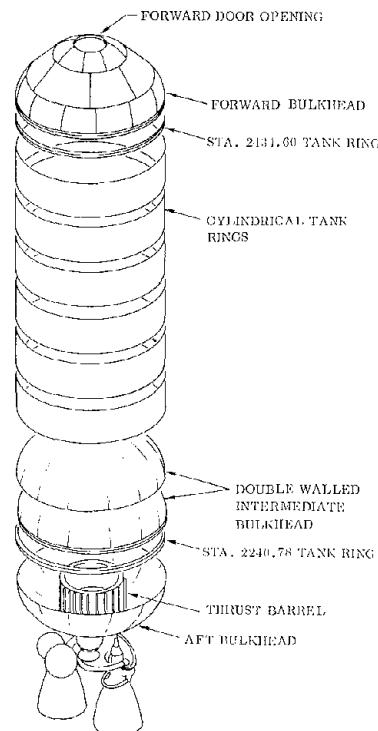


Figure 3-2. Tank fabrication concept.

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 in order to leak check the bulkhead. The volume is then back-filled with gaseous nitrogen prior to tanking. When the cryogenic fuel (LH_2) is 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 sidewall. The insulation bulkhead is welded in a similar manner to the oxidizer tank just forward of the Station 2240.78 ring. A spring-type expansion ring is located at the lower edge of the insulation bulkhead.

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). An annealed 321 CRES flanged ring is welded to the forward end.

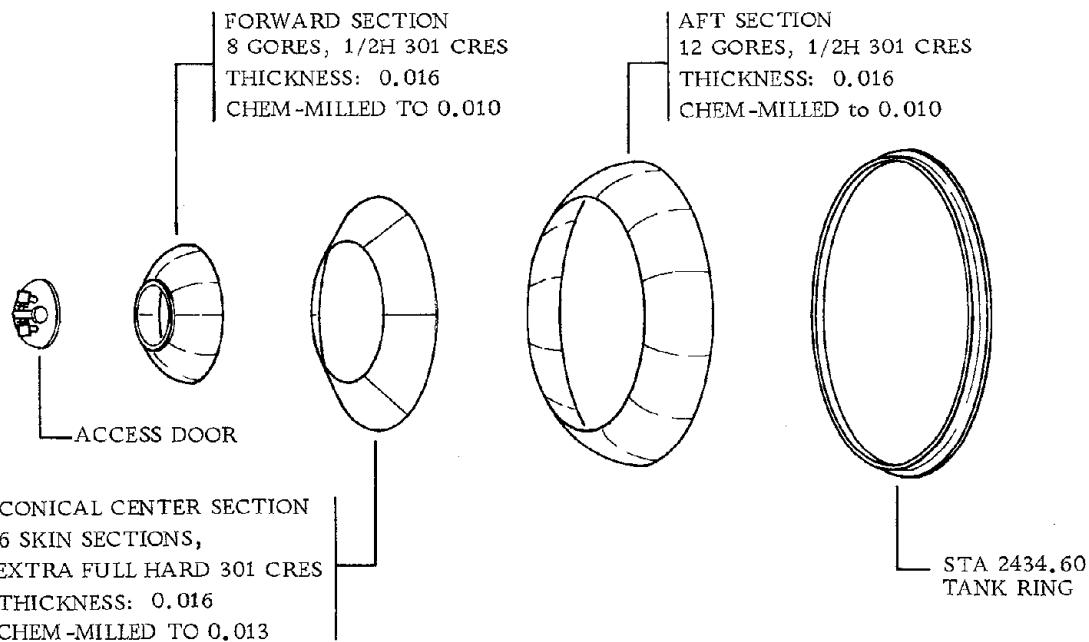


Figure 3-4. Forward bulkhead.

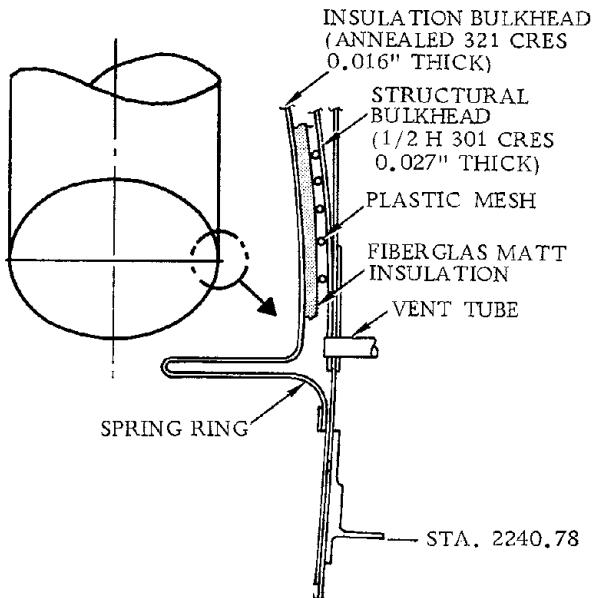


Figure 3-3. Intermediate bulkhead sectional view.

The forward door mounts to this flanged ring. The door is a shallow spherical section made of 0.032-in. 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 LH₂ vent valves, mounting brackets for the LH₂ vent ducts, a helium pressure inlet diffuser, and bosses for various LH₂ 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 301CRES gores are buttwelded and then reinforced by splice doublers, some of which are contoured to make provisions for mounting brackets. Central caps, doublers, and flanges complete the aft bulkhead.

The thrust barrel is a 50-inch-diameter cylinder of aluminum skin and stringer construction. It is riveted to a 321 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 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 provide attachment for the helium and hydrogen peroxide bottles, engine actuators, separation bumper guides, the hydrogen peroxide system, instrumentation boxes, pneumatic panels, radiation shield, LO₂ 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 inter-stage 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) mounts to the tank forward ring (Station 2434.60), it supports the equipment module and either the payload truss adapter or the payload adapter environmental shield as applicable. The adapter is approximately 120 inches in diameter and 25 inches long and is of conventional skin-stringer construction with the stringers on the outside of the skin. The adapter consists of a titanium skin, titanium stringers, forward and aft aluminum rings, an internal aluminum mid-frame, and internal and external longerons matching the truss adapter mounting points. Titanium is used for skins and stringers to limit heating into the LH₂ tank during long coast period missions.

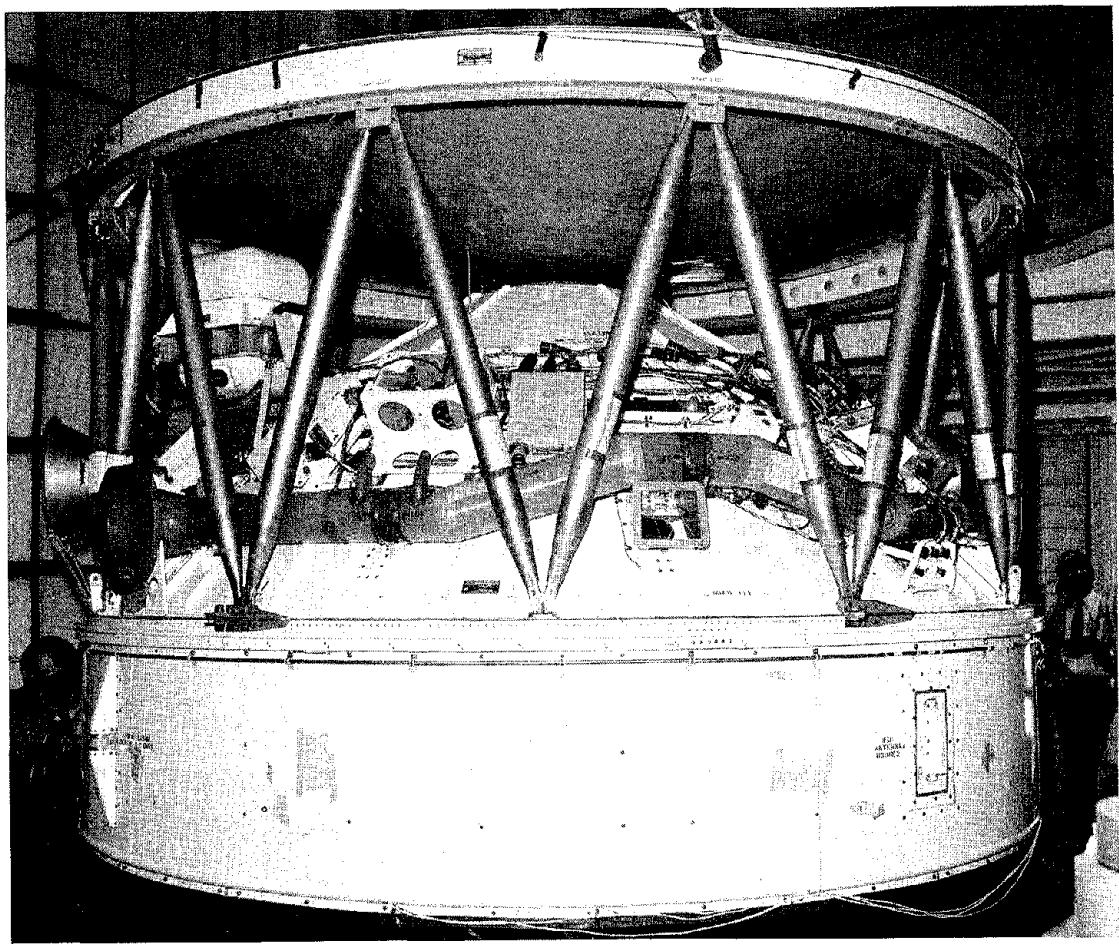


Figure 3-5. Centaur D-1T stub adapter and equipment module.

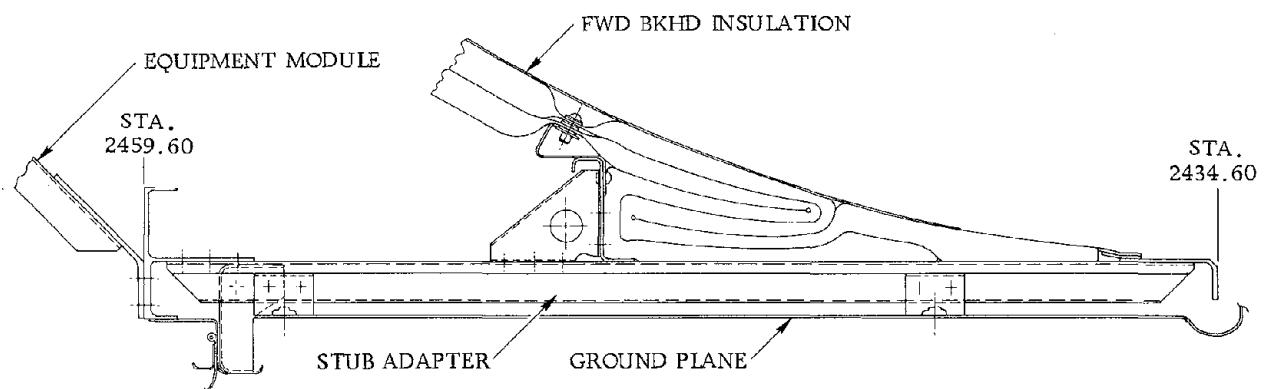


Figure 3-6. Stub adapter cross-section.

The stub adapter incorporates mounting provisions for the forward bearing reactor system struts at the 60°, 120°, 180°, 240°, 300°, and 360° vectors.

The stub adapter incorporates mounting provisions for:

- a. Two S-Band antennas.
- b. Two C-Band antennas.
- c. Two Range Safety Command (RSC) antennas.
- d. Two LH₂ tank vent system nozzles.
- e. The forward shroud seal. (This incorporates a moisture collector).
- f. The ground plane/radiation shield (see Subsection 3.2.1).

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 shroud.
- f. Provide an equipment area gas conditioning duct (see Subsection 3.7.7).

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 machined rings at each 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 hat-section 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).

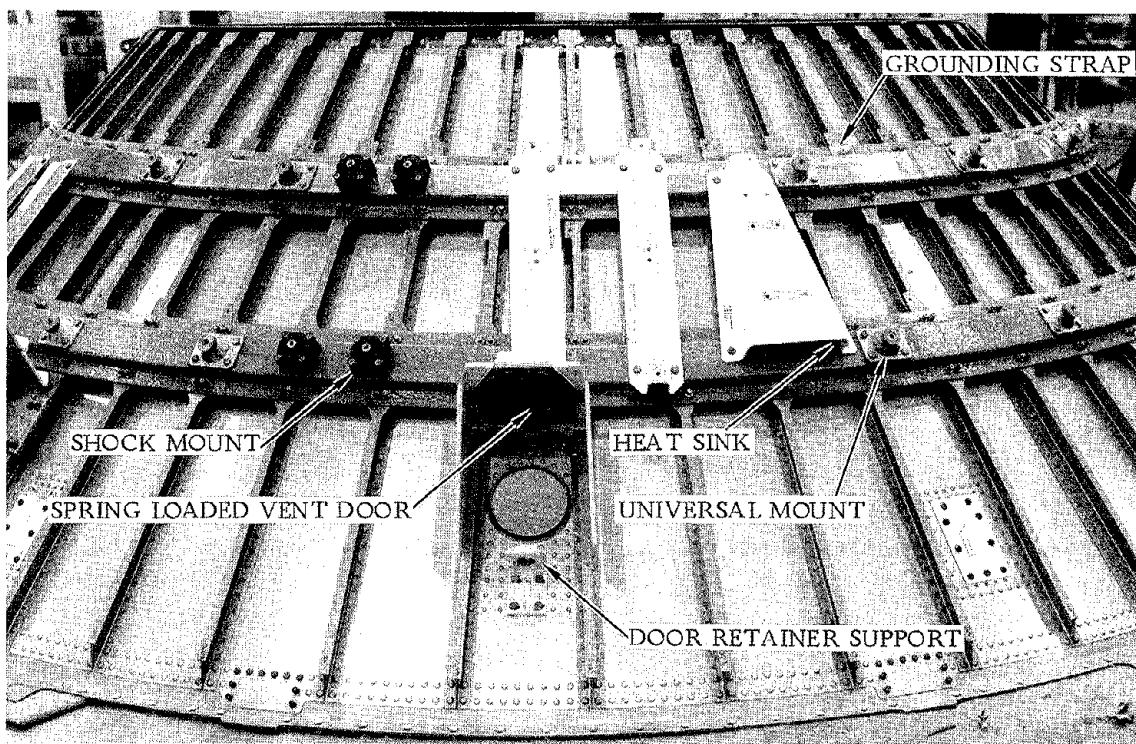


Figure 3-7. Equipment module structure.

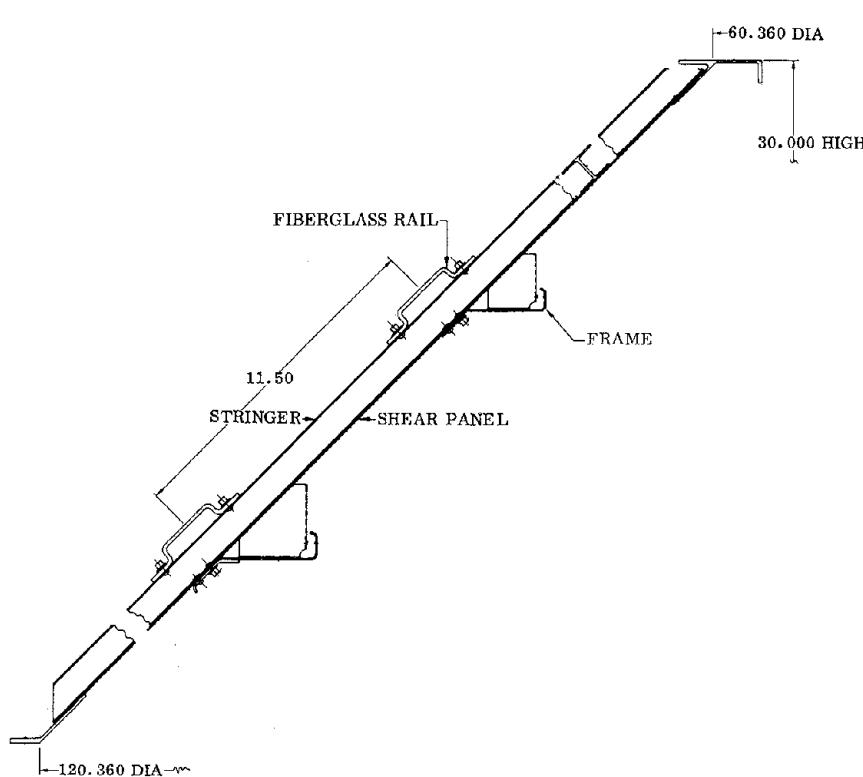


Figure 3-8. Equipment module cross-section.

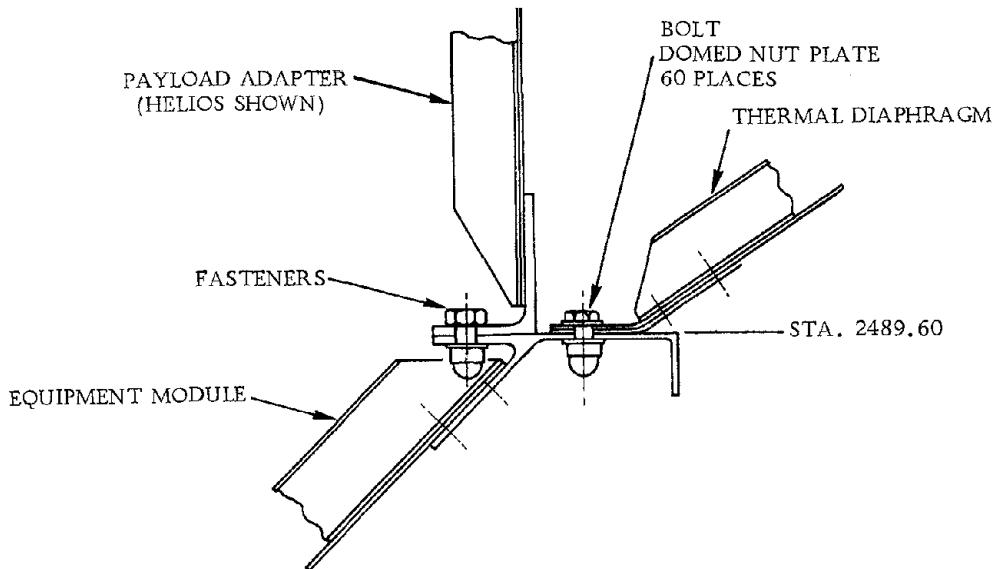


Figure 3-9. Equipment module, payload adapter, and thermal diaphragm interface.

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 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 Titan and Centaur vehicles. The adapter is an aluminum skin-stringer cylindrical section 10 feet in diameter and 113.35 inches long. The adapter bolts to both the Centaur aft ring at Station 2240.78 and to the Titan forward skirt forward ring at Station 2127.43. In addition, the adapter provides an interface ring at Station 2180.48 for bolt attachment of the Centaur Standard Shroud.

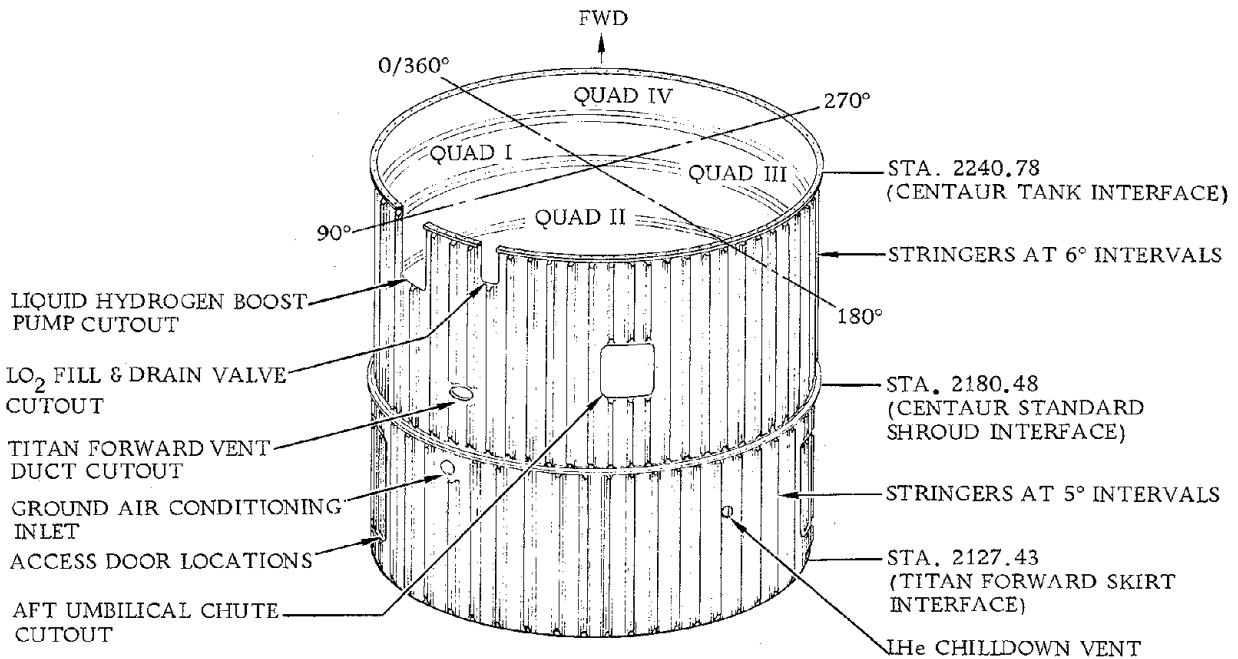


Figure 3-10. Interstage adapter.

Engine childdown vent valves are attached by flexible lines to two manifolds in the interstage adapter. Each manifold empties into a childdown vent duct. The two ducts are located 180 degrees from each other. Breakaway connections are provided for four flexible teflon tubes that connect between the engine's helium childdown tubing and the interstage adapter helium childdown collector manifold. Slip-joint disconnects are provided for the LO₂ tank vent duct and the H₂O₂ 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 pair of ducts for venting the Titan forward skirt compartment are mounted inside the adapter, routed from the aft end forward and exiting through the forward portion of the adapter to the Centaur Standard Shroud.

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, six small doors allow for reaction-control engine servicing. One door permits servicing of the H₂O₂ storage bottle.

Cutouts in the forward portion of the adapter accommodate the liquid oxygen fill and drain valve, the liquid hydrogen boost pump and propellant feed line, the aft umbilical panel chute, the Titan forward skirt vent ducts, and miscellaneous electrical harnesses and tubing.

3.1.4 TITAN/CENTAUR STAGING SYSTEM.

A flexible linear shaped charge (FLSC) system (Figure 3-11) separates the Centaur vehicle from the Titan 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 Titan 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 Titan/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 a isolation of pyrotechnic devices so that a device will not inhibit the firing of another. The pyrotechnic control unit is located on the inside surface of the interstage adapter.

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

3.1.5 PAYLOAD ADAPTERS. The payload adapters support the payload and distributes the loads imposed by the payload to the supporting Centaur structure. Two basic types of payload adapters are available for the D-1T Centaur. For payloads weighing 4000 pounds or less, a payload adapter mounted on the equipment module forward end is used. For heavier payloads a truss and transition adapter payload support structure is used which attaches to the stub adapter forward ring. The present payload capability of the truss/transition adapter combination is approximately 8000 pounds, however, both payload adapter systems have growth capability and allow for remote encapsulation of the payload. The stub adapter structure is capable of supporting payloads of approximately 12,000 pounds

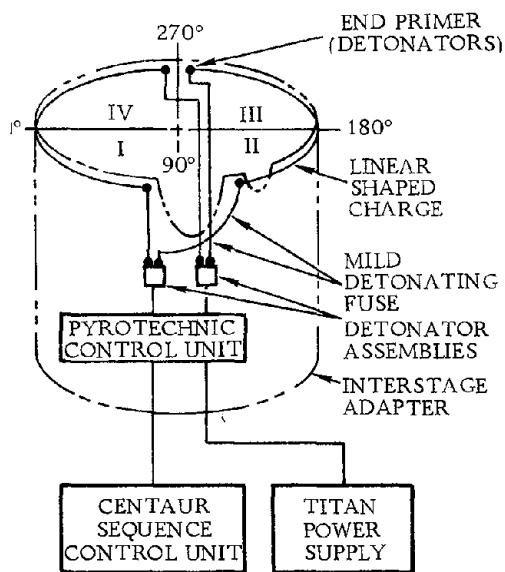


Figure 3-11. Titan/Centaur separation system arrangement.

Helios Payload Adapter. The Helios adapter, shown in Figure 3-12, is of conventional aluminum skin-stringer construction consisting of machined forward and aft rings, a skin, a stretch-formed Z-section interior midframe, and external aft section stringers. The aft ring of the adapter bolts to the equipment module forward ring (Figure 3-9).

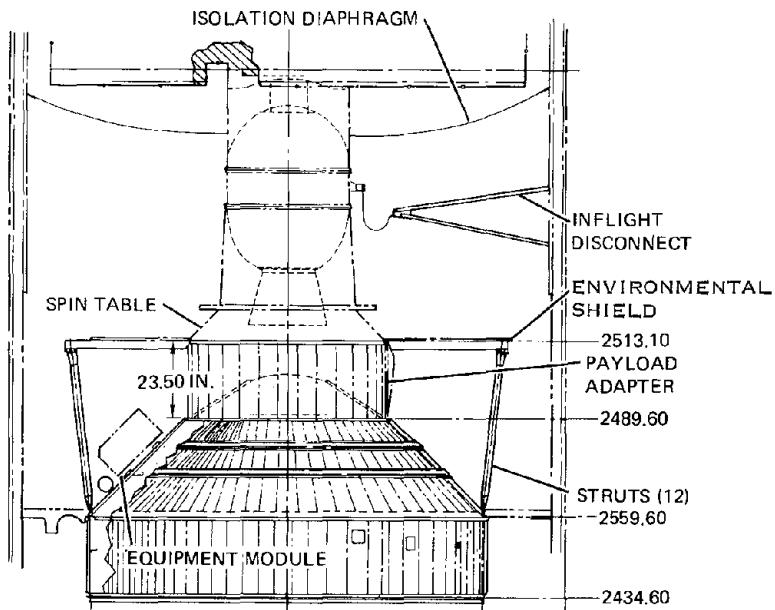


Figure 3-12. Centaur/Helios payload adapter.

An environmental shield is attached to the adapter forward ring as shown in Figure 3-12. The outer edges of the shield are supported from the equipment module aft ring by 12 tubular struts. The shield also serves as a work platform for access to the payload. The shield consists of eight segments, each made from a sandwich panel of balsa wood core with aluminum sheets.

Viking Payload Support Structure. This structure consists of two major structures, the Centaur truss adapter (CTA) and the Viking transition adapter (VTA) which are described separately (Figure 3-13).

The Centaur truss adapter consists of 24 tubular struts with each end swaged to form a clevis, 12 forward attach fittings for connecting the strut ends, and 12 aft attach fittings mounted on the equipment module.

The VTA is a built-up box section ring approximately 120 inches in diameter which mounts on the truss adapter forward fittings. The adapter consists of a forward and aft machined ring, 12 machined "hard point" fittings located between the rings, inner and outer cylindrical skins, and some additional internal stiffening ribs.

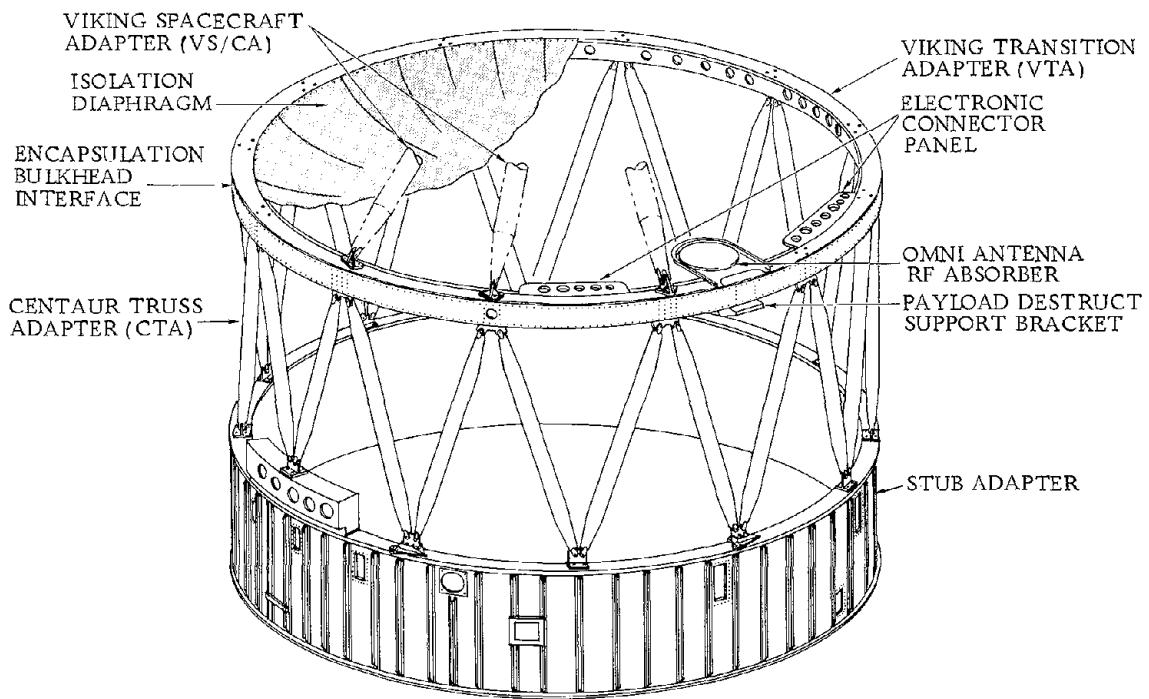


Figure 3-13. Viking payload support structure (equipment module structure omitted for clarity).

Mounting provisions (captive nuts) are provided on the forward surface for the payload mounting truss. The outer edge of the forward ring incorporates mounting provisions for the encapsulation bulkhead which extends to the inner surface of the shroud. The area encircled by the adapter is sealed by a flexible isolation diaphragm.

The adapter includes two electrical connector (to payload) mounting panels, an RF absorber under the spacecraft omniantenna, and mounting provisions for the space-craft destructor unit.

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

- An aft umbilical panel which includes provisions for mounting aft disconnects.
- 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.
- 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. Pneumatics panels mounting.
 - 4. Signal conditioner and multiplexer mountings.
 - 5. Instrumentation package mounting.
 - 6. Four separation bumpers which assist in guiding the Centaur out of the interstage adapter during vehicle separation.
- 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 and RF disconnect mountings.
- f. The aft seal plate, in conjunction with the aft wiring tunnel bulkhead is the inboard mounting point for the tank to shroud aft seal (Figure 3-14).

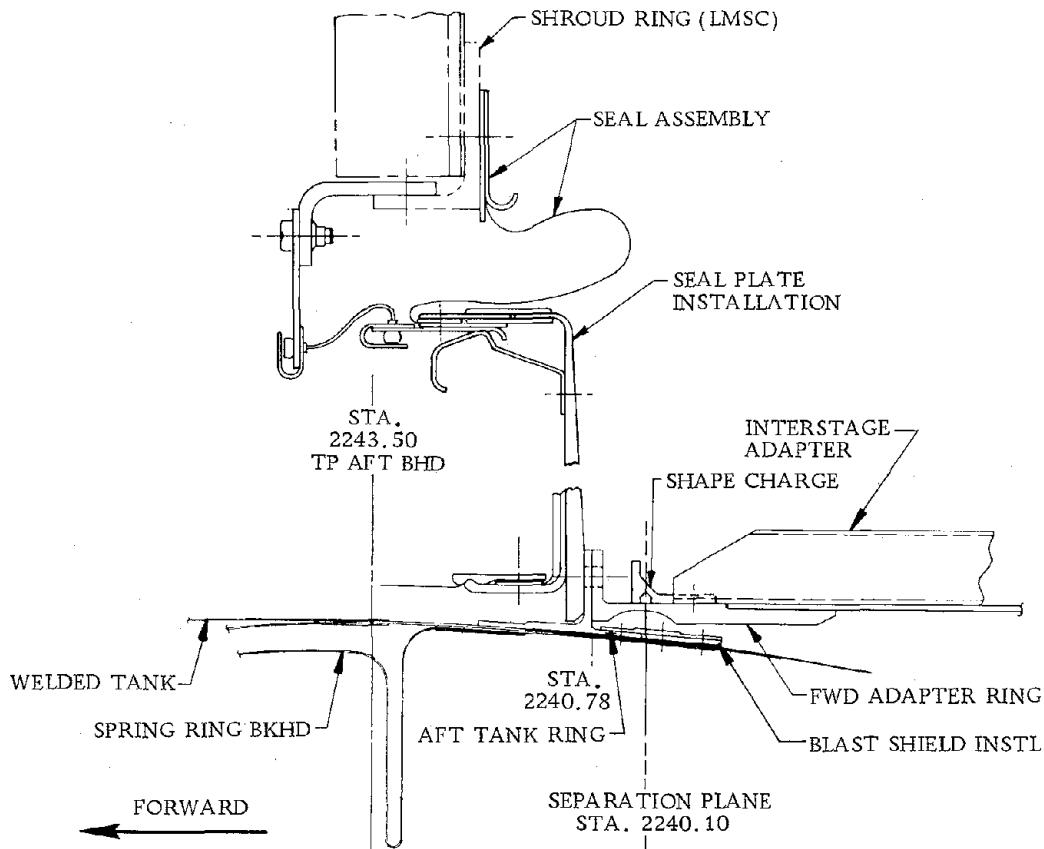


Figure 3-14. Aft seal.

- g. The tank/shroud aft seal (Figure 3-14) is located at approximately Station 2240. The aft seal extends around the full circumference of the tank and mates with a ring provided on the inside of the shroud. It provides a seal which minimizes the flow of purge gas from the forward portion of the vehicle into the aft end of the shroud and into the interstage adapter.

The flexible portion of the seal is dacron/mylar fabric. The outer edge of the seal is attached to the shroud. The inner edge of the seal is a spring-loaded slip fit to the seal plate. On shroud jettison, the inner edge pulls out of the spring-loaded joint and the seal is jettisoned with the shroud. The seal assembly is in two halves, one for each shroud half, with a separate seal joining the seal halves at the splitline.

3.2 INSULATION SYSTEMS

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

FUNCTIONS:
<ul style="list-style-type: none"> • Reduce Propellant Tank Heat Flux from Earth and Sun after Shroud Jettison • Protect Components from Excessive Cooling • Insulate Cylindrical Tank Section on Ground and During Ascent to Limit LH₂ Boiloff • Minimize Heat Transfer Between Tanks

3.2.1 FORWARD BULKHEAD AND STUB ADAPTER INSULATION. (Figures 3-15 and 3-16). This insulation system is comprised of three major portions, the forward tank bulkhead insulation, the tank access door and LH₂ vent system insulation, and the stub adapter insulation.

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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 LH₂ 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 LH₂ tank vent ducts. The stub adapter is surrounded by a continuous radiation shield/antenna ground plane made of fiberglass with aluminum foil bonded to its inner surface. The shield is attached to the stub adapter with fiberglass brackets so that it does not touch the adapter except as required for grounding. The aft end of the shield serves as the attachment point for the forward end of the tank radiation shield.

3.2.2 TANK SIDEWALL RADIATION SHIELD. The radiation shield, (Figure 3-17), covers the entire cylindrical section of the vehicle LH₂ tank. It extends from the tank/stub adapter joint (approximately Station 2438.70) aft to the tank/interstage adapter joint (approximately Station 2240.78). The function of the radiation shield is to reduce the amount of heat flux entering the LH₂ tank as a result of solar albedo and earth thermal radiation.

The shield is a single assembly, rectangular in shape when laid flat, which covers the full length required and the total circumference of the tank less the width of the wiring tray. The shield is comprised of three layers of an aluminized Mylar-Dacron net (scrim) sandwich. The inner and middle layers are identical in construction, an aluminized Mylar sheet, a Dacron scrim on top of it, and another aluminized Mylar sheet on top of the scrim, all bonded together. The outer layer is of similar construction except that one of the Mylar sheets is plain, not aluminized. The three layers are laid one on top of the other to form a total shield. The plain Mylar surface of the outer layer is the outside of the shield. The three-layer shield is in panels approximately 48 inches wide. Each panel is permanently bonded to the next with one-inch aluminized Mylar tape. The outer and middle layers have 1/4-inch diameter vent holes arranged in a staggered pattern so that the patterns in the two layers do not match. The inner layer has no holes. The shield edges are clamped between two pieces of fiberglass edging riveted together. The edging strips, forward, aft, and longitudinal, are short lengths (approximately 18 inches) with a small gap between adjacent strips. This arrangement allows for both tank and radiation shield expansion and contraction. The short lengths forward and aft permit the shield to be folded, accordion style, for ease in handling and installation and also provide flexibility for ease of installation. Shielding of similar construction to fit under and over the wiring tray and around the fuel tank elbow is provided.

All layers of the shield are electrically bonded together and to the stub adapter ground plane. An aluminum foil strip interleaved with the shield layers at the forward end makes contact with the aluminized tape on the stub adapter ground plane.

A protective cover for the entire radiation shield protects the outer surface during storage, transportation, and after the shield is installed on the vehicle. The cover is removed just prior to shroud installation.

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

3.2.4 AFT BULKHEAD RADIATION SHIELD AND MEMBRANE. 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 (Figures 3-18 and 3-19). It is made up of 12 gores that form a complete ellipsoidal half that covers 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 LO₂ 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.

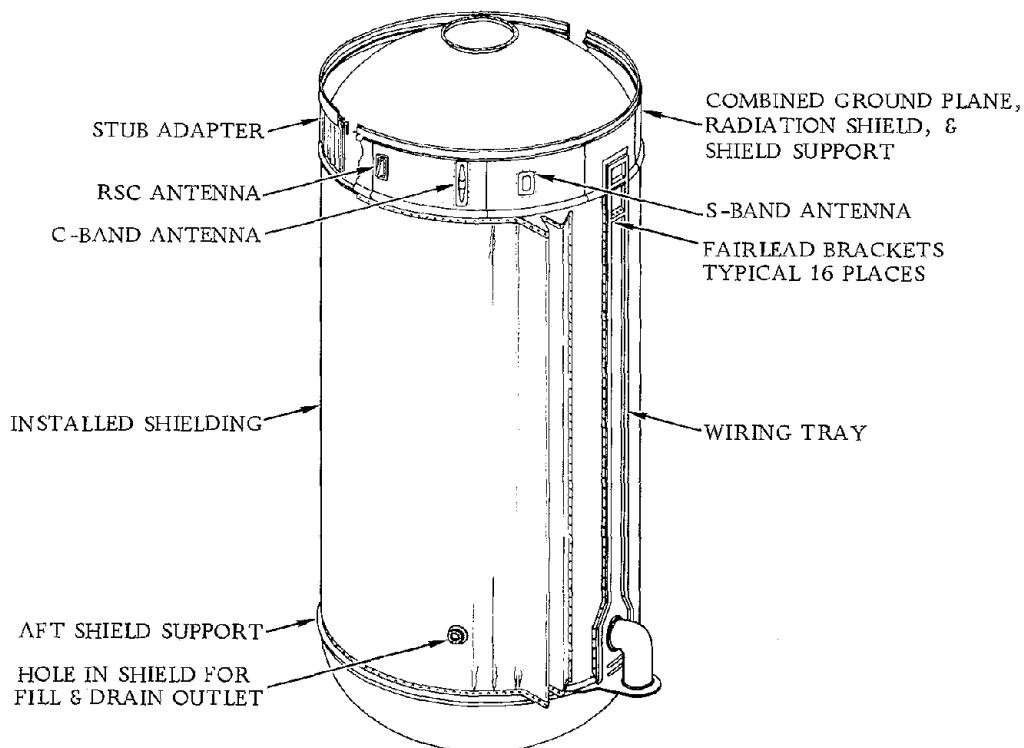


Figure 3-17. Tank sidewall radiation shield.

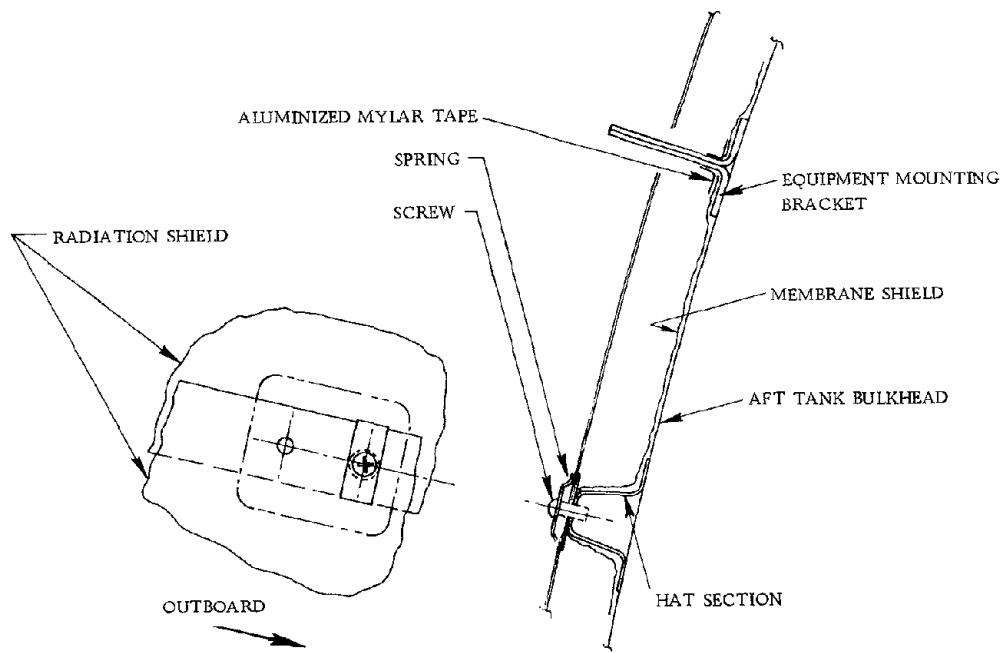


Figure 3-18. Aft bulkhead radiation shield and membrane.

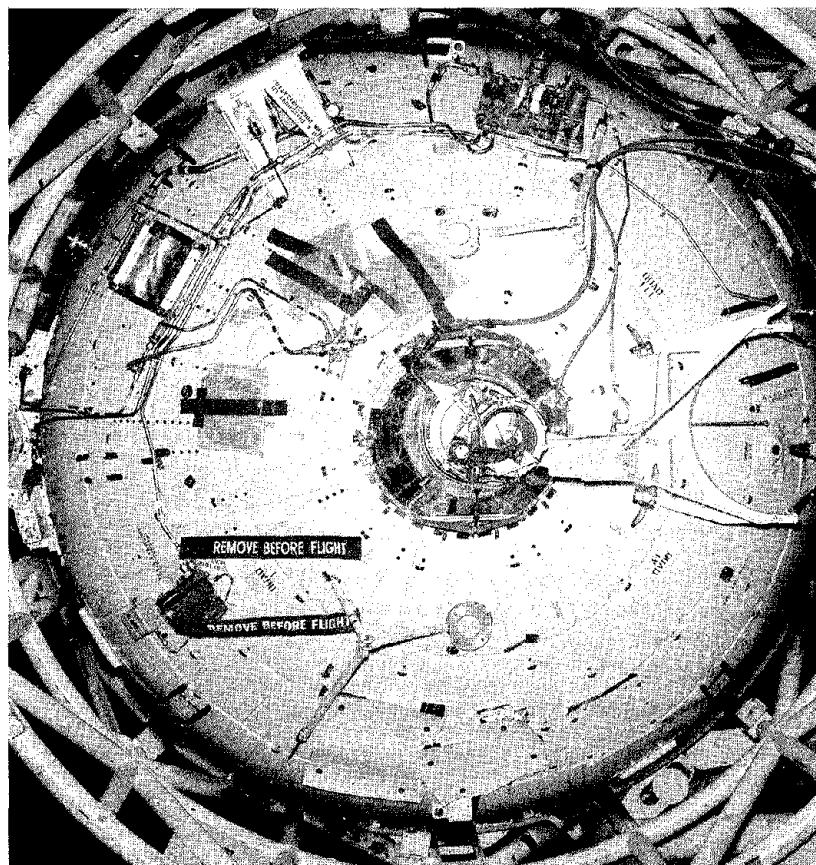


Figure 3-19. Aft bulkhead radiation shield.

3.3 CENTAUR D-1T STANDARD SHROUD SYSTEM

SECTIONS:

- | | Subsection |
|--|------------|
| • Shroud System
Conical/cylindrical metal structure | 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 SHROUD SYSTEM. The Centaur standard shroud (CSS) encloses both the Centaur and the payload and provides environmental protection for the Centaur and spacecraft on the ground and in flight (Figure 3-20). The cylindrical portion of the shroud is 14 feet in diameter. Total shroud length is 58 feet.

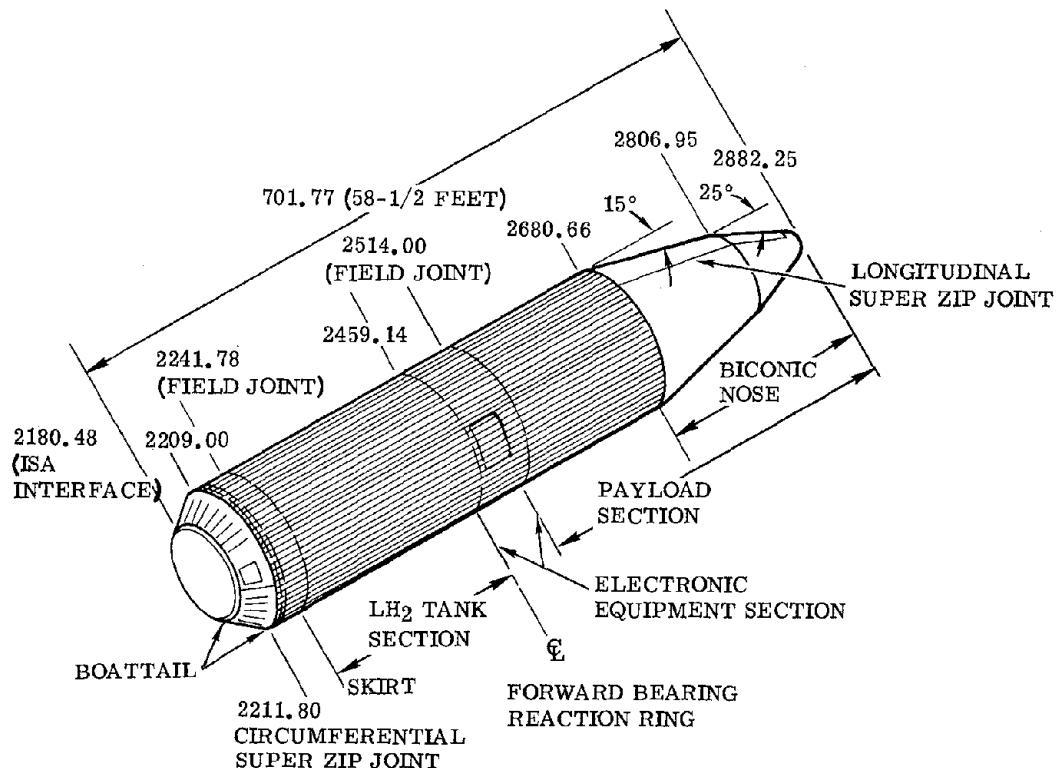


Figure 3-20. Centaur D-1T standard shroud.

The CSS payload section (forward of the field joint at Station 2514.0) is nearly 31 feet long. The nose cap is made from corrosion-resistant steel. The two conical sections are of magnesium semimonocoque construction. The cylindrical section is of corrugated aluminum semimonocoque. Figure 3-21 shows the CSS structural configuration.

A seal between the shroud field joint and the spacecraft truss adapter ensures a clean and thermally controlled gas environment in the payload area. During ground operation and ascent, the gas is vented into the aft equipment area.

The remaining section extends aft around the Centaur and bolts to the interstage adapter. Construction to the aft field joint at Station 2241.78 is also of corrugated aluminum. Two sections from the field joint to the interstage adapter complete the shroud. The first section (from Station 2209 to 2241.78) contains the shroud circumferential separation mechanism and separation springs. This section is also constructed of corrugated aluminum and is connected with separation hinges to the final aluminum skin-stringer conical section, which bolts to the interstage adapter.

A forward bearing reactor system from the shroud to the Centaur stub adapter reduces relative deflections between the shroud and payload during launch and ascent (Figures 3-22 and 3-23). The reactor path is released after maximum dynamic loading, when the six bearing struts retract to the shroud.

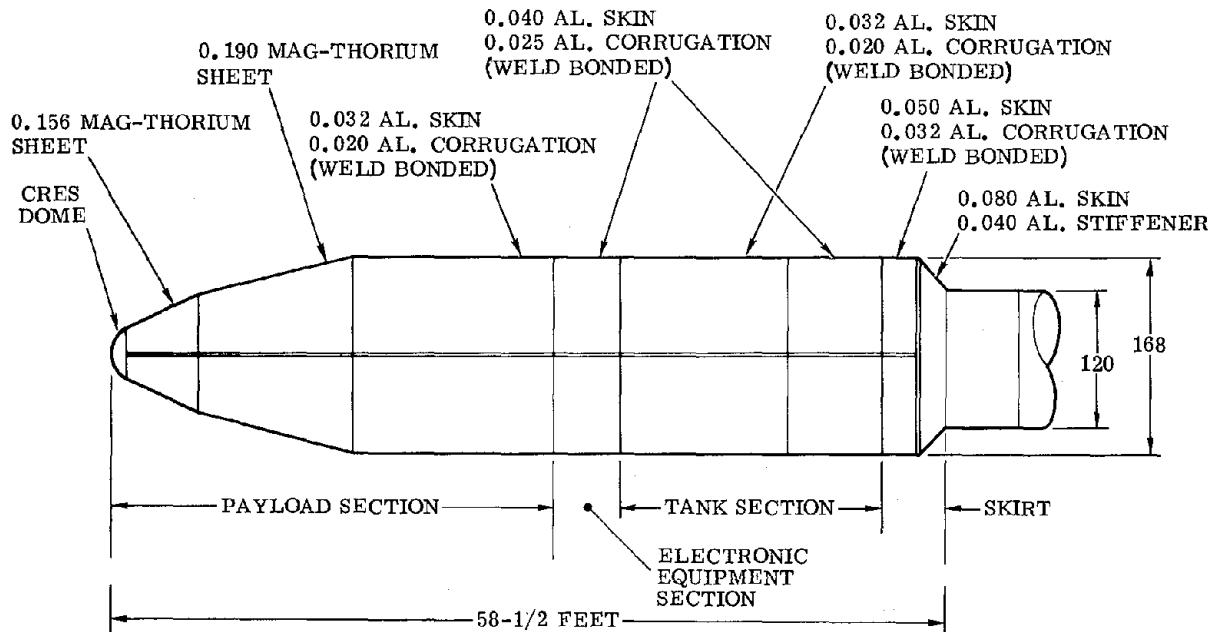


Figure 3-21. CSS structural configuration.

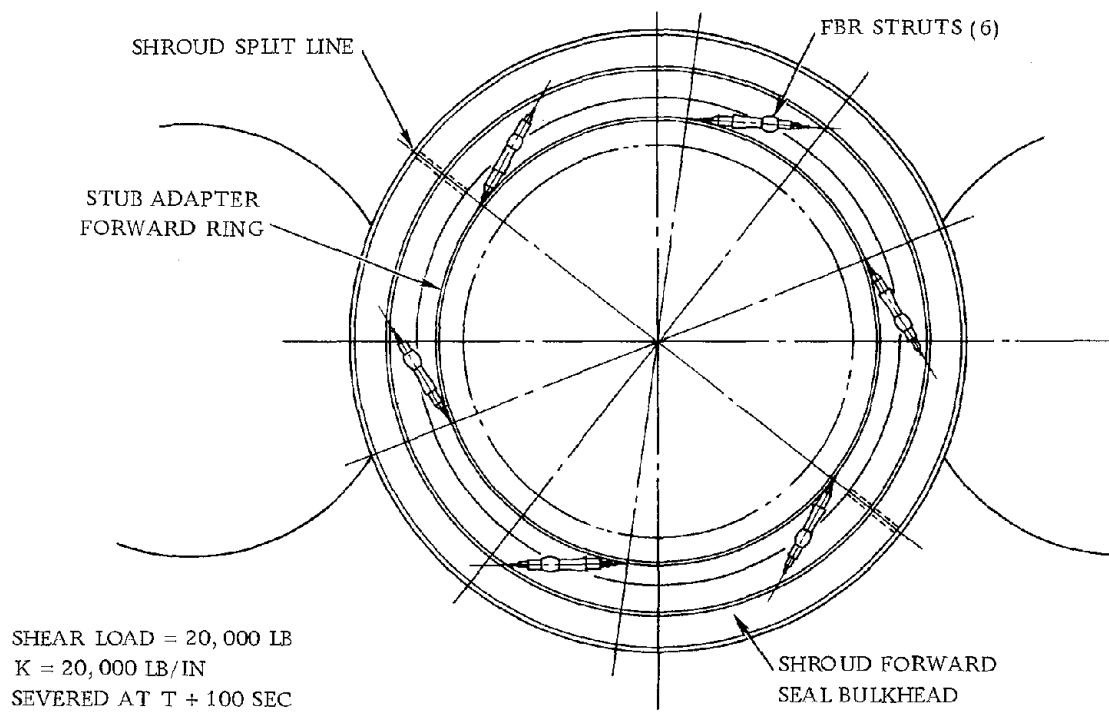


Figure 3-22. Forward bearing reactor system.

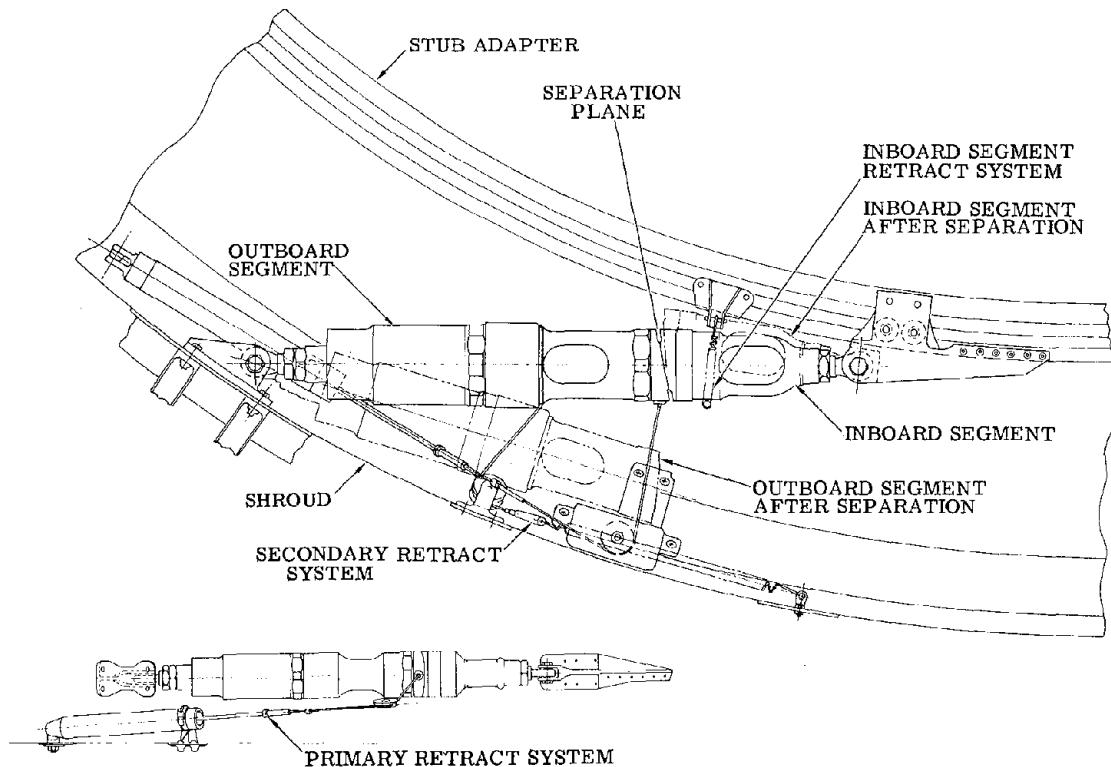


Figure 3-23. Forward bearing reactor strut installation.

At Station 2459.16 there is a bulkhead and purge seal. The seal separates the helium-purged environment around the Centaur tank from the equipment area environment. The environment seal is maintained until just prior to Stage I shutdown. An aft seal is located at Station 2241.

The shroud insulation system consists of two aluminum radiation shields attached to the inside of the nose cap, 1.1-inch thick fiberglass blankets attached to the internal rings in the payload section, and an aluminum radiation shield in the equipment module section. Shroud insulation in the Centaur tank area consists of fiberglass batting packed in the cavities between the skin and the inner cap of the frames plus 3.3-inch fiberglass blankets mounted on the inner cap of the frames. Figure 3-24 is a schematic of the insulation system.

The CSS contains many other bolt-ons including six antennas; two C-band, two S-band, and two range safety; compartment vent provisions; a liquid hydrogen vent fin for venting the Centaur liquid hydrogen tank; inflight electrical disconnect/lanyard provisions; and an encapsulation bulkhead which separates the payload and electronic compartments until liftoff. The CSS also contains chutes between the CSS and Centaur to guide the withdrawal of umbilicals including the forward electrical umbilical, the electronic compartment air conditioning umbilical, the liquid hydrogen fill and drain umbilical and valve, the liquid oxygen fill and drain umbilical and valve, and the aft

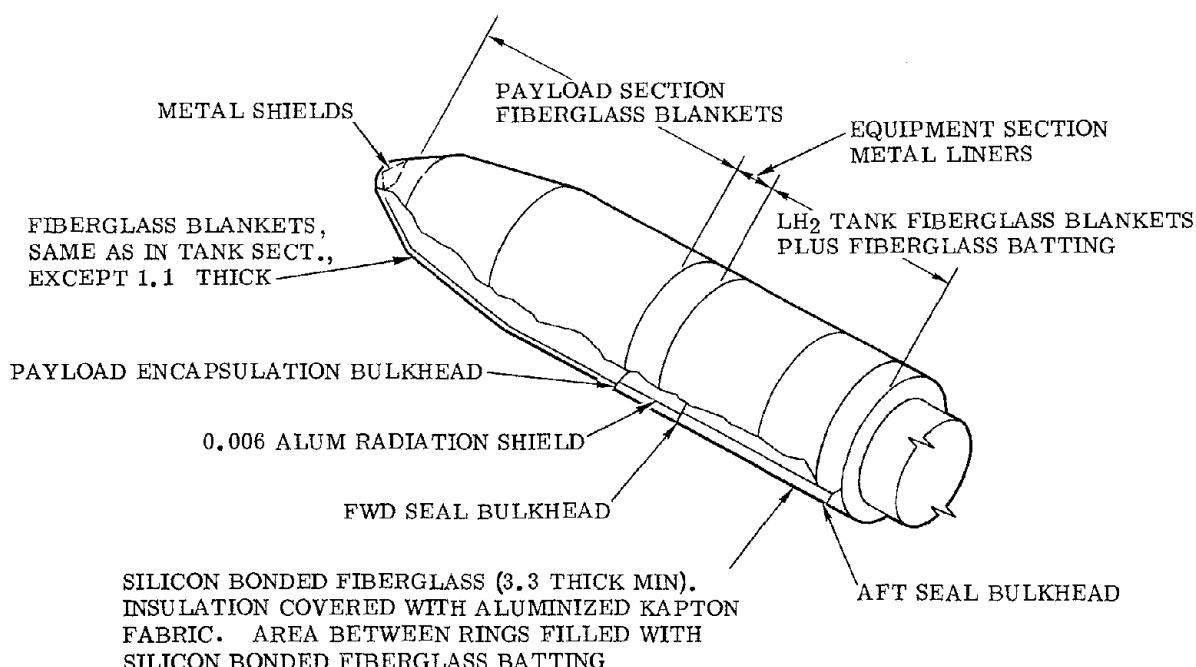


Figure 3-24. Centaur standard shroud insulation system.

umbilical panel. Each of these chutes has a door that closes off the chute at the shroud skin line after the umbilical is withdrawn. Finally, the CSS contains many access doors used during ground operations to gain access to various items on the vehicle. Several mission-peculiar bolt-ons are provided for the various payloads, including air conditioning provisions, inflight electrical disconnect provisions, and access provisions.

3.3.2 SPACECRAFT ENVELOPE. The spacecraft envelope is shown in Figure 3-25 with the 10-foot diameter truss adapter for heavy payloads. Light payloads can be mounted on the Equipment Module, which provides a 5-foot mounting ring at Station 2489.60.

The envelope shown is for the maximum permissible dimensions of the spacecraft after accounting for dynamic motion and manufacturing tolerances.

3.3.3 SEPARATION SYSTEM. The two halves of the Centaur standard shroud join along a longitudinal separation plane. Each half also joins to the fixed aft part of the shroud along a circumferential plane.

At jettison (about 25 seconds after Titan Stage 2 ignition), all split lines are severed by a noncontaminating pyrotechnic system. The aft conical boattail is bolted to the interstage adapter and jettisoned with the Titan stage.

The pyrotechnic separation system, called Super Zip, is shown schematically in Figure 3-26.

The longitudinal and circumferential pyrotechnic systems are both redundant; each system contains a primary and secondary system. The detonators are ignited electrically using the Titan transient power supply routed through pyrotechnic control units mounted on Titan. The signal for initiation of the separation system is supplied by the Centaur Digital Computer Unit.

The system cross-section consists of two (one redundant) mild detonating cords housed in a flattened steel tube which is in turn mounted between two frangible (notched) plates. When either mild detonating cord is ignited, the gas expansion causes the flattened steel tube to become cylindrical, thereby causing the notched

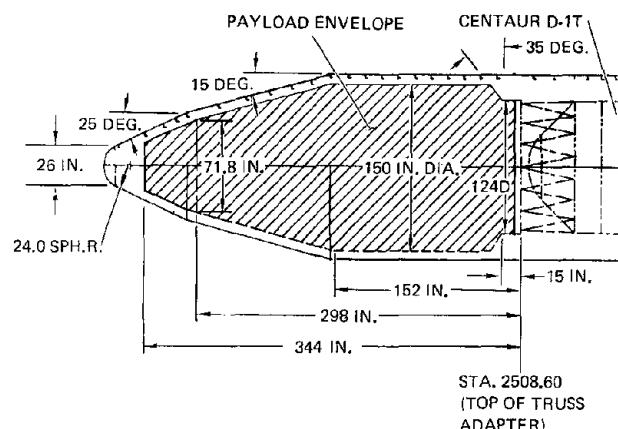


Figure 3-25. Spacecraft envelope with Centaur standard shroud.

plates to fracture. The steel tube remains sealed which makes the system truly non-contaminating. A typical cross-section of the system is shown in Figure 3-27.

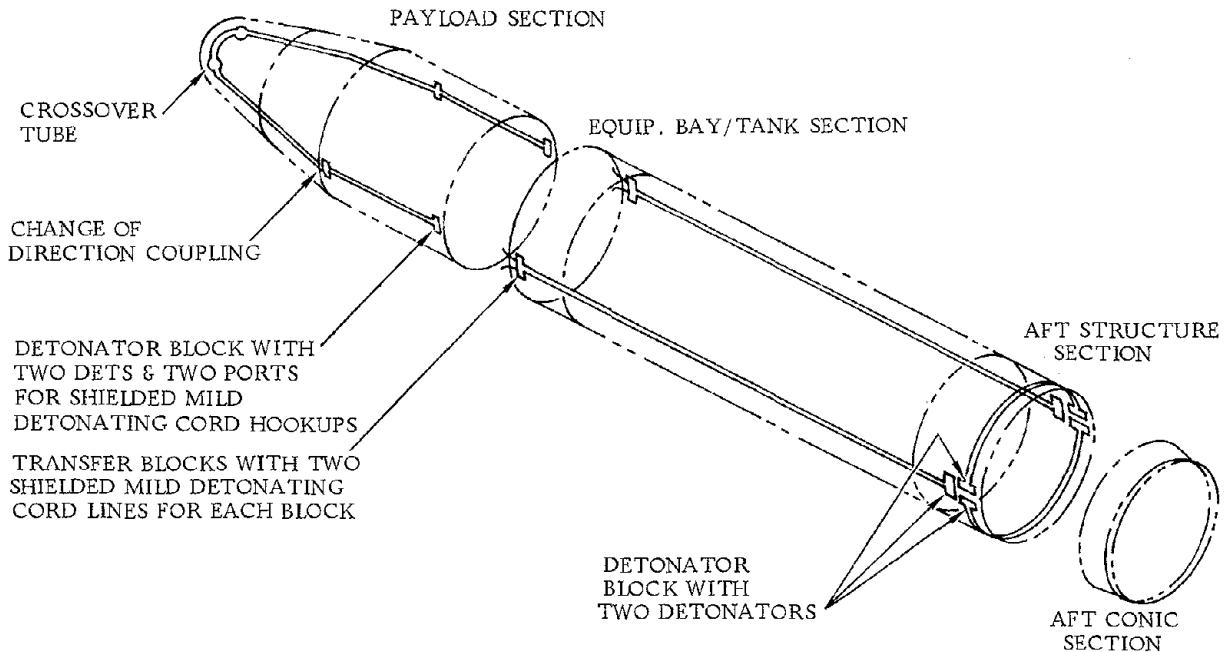


Figure 3-26. Pyro/Super-Zip joint arrangement.

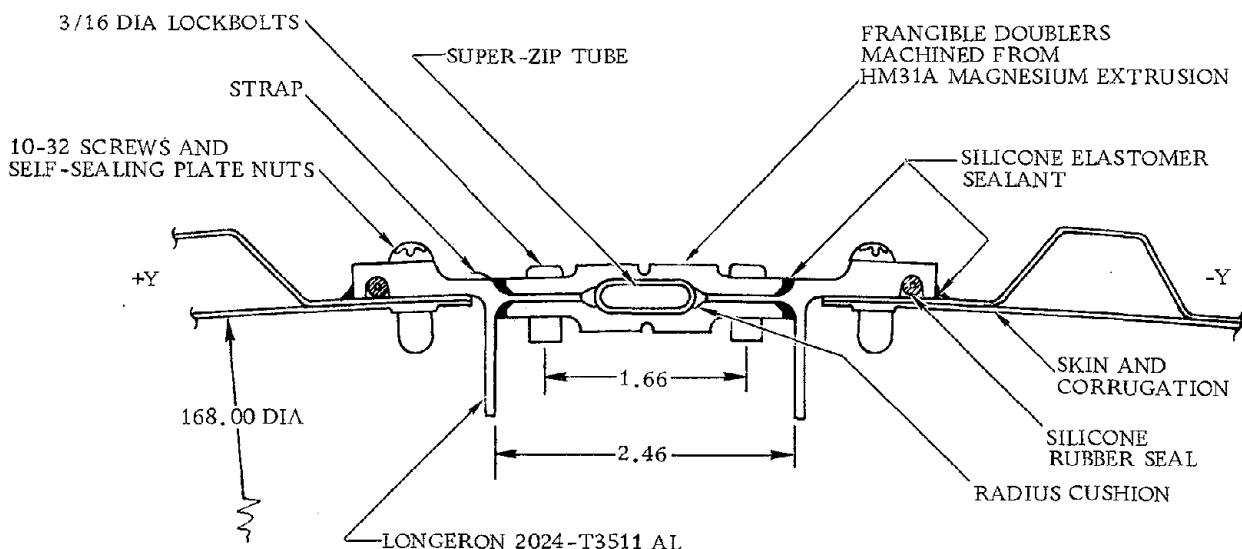


Figure 3-27. Typical cross-section thru Super-Zip joint.

After the longitudinal and circumferential Super Zip joints are severed, the CSS begins the jettison sequence. Three sets of springs assure successful shroud jettison. First, a pair of springs located near the nose cap ensure that the nose joint will initially "pop" free. Second, a pair of lateral thrusters located near the inflight electrical disconnects at Station 2460 provide energy in addition to the main thruster during the critical period of disconnecting the electrical connectors; and finally, eight main thrusters, four per CSS half, continue to provide energy until each CSS half center of gravity has rotated past top dead center. The hinges are designed to retain each half on the vehicle until about 55 degrees of rotation, at which time the halves are free to translate laterally away from the vehicle. A schematic of the jettison sequence and spring thrust locations is shown in Figure 3-28.

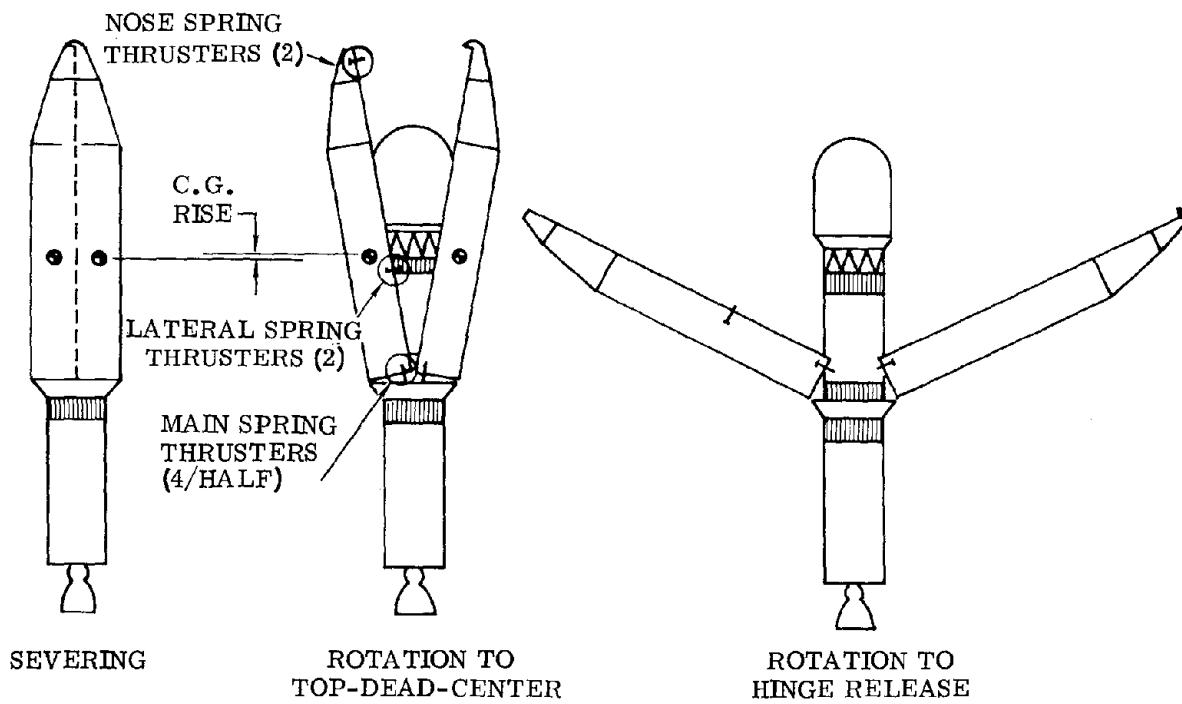


Figure 3-28. CSS jettison sequence.

3.4 MAIN PROPULSION SYSTEM

ELEMENTS:

- | | |
|---|------------|
| <ul style="list-style-type: none">● Two Main Engines● Propellant Feed System (LO₂ and LH₂)● Engine Childdown System | Subsection |
| | 3.4.1 |
| | 3.4.2 |
| | 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 3-29). 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 3-30.

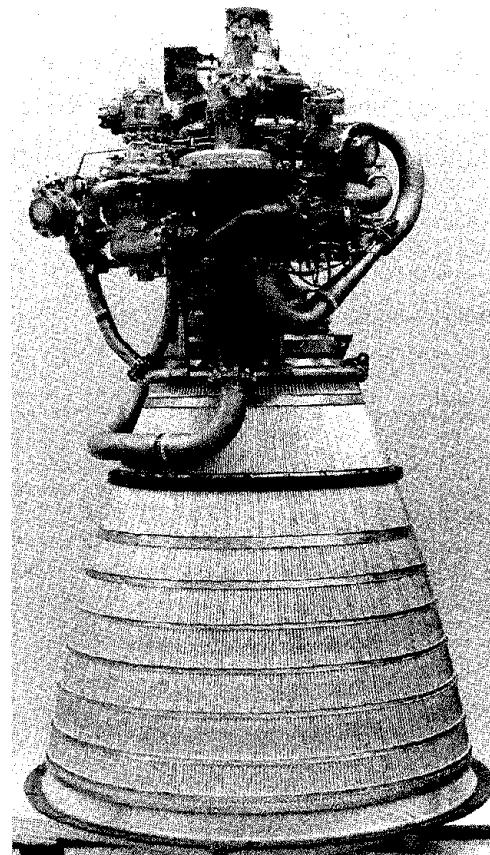


Figure 3-29. RL10A-3-3 engine.

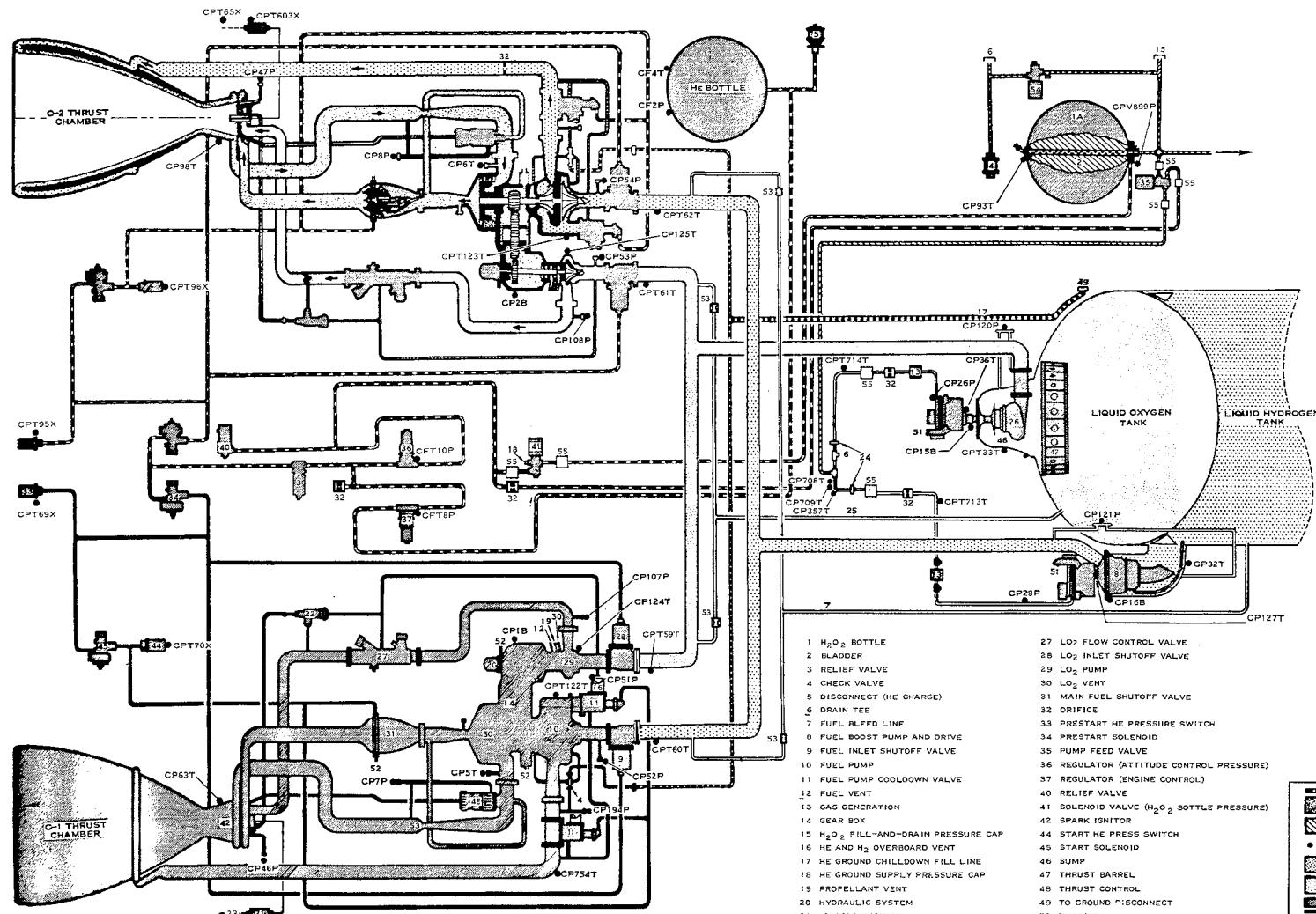


Figure 3-30. Centaur propulsion system.

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 vantage 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 (steady state) 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) (Minimum at 5.7 mixture ratio)	12,100 (+600, -400) rpm 11,500 rpm
Accessory Drive Pad Running Torque (maximum)	20 lb-in.
Permissible Engine Gimbal 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.

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
- LH₂ 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 3-31).

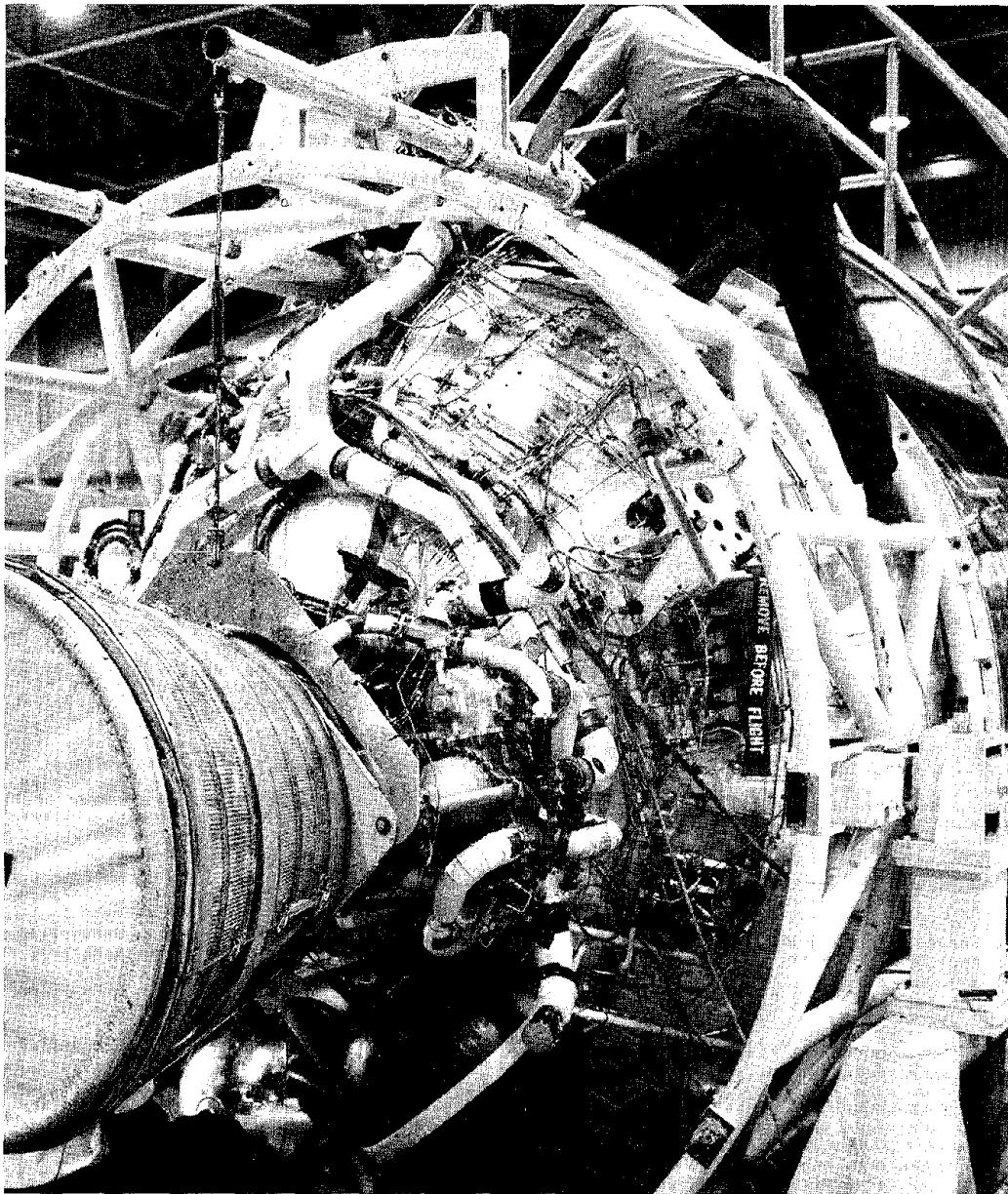


Figure 3-31. Centaur propellant feed system.

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 3-32.

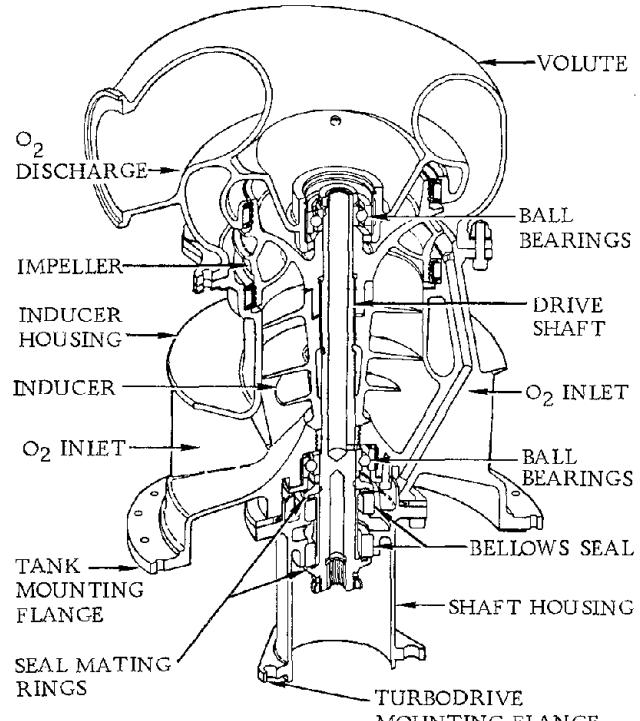


Figure 3-32. Oxidizer boost pump cutaway.

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 LO₂ tank.

Liquid Hydrogen System. The turbine-driven centrifugal liquid hydrogen boost pump assembly operates while completely submerged in liquid hydrogen. A diagram of the LH₂ boost pump system is shown in Figure 3-33. The inducer bell is vaned to prevent fluid pre-rotation.

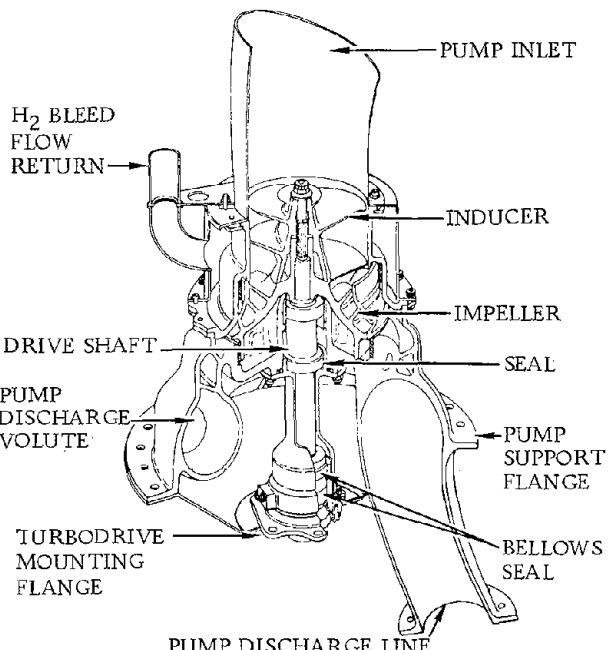


Figure 3-33. Fuel boost pump cutaway.

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 different routing. A hydrogen bleed line serves the same function as the corresponding oxygen bleed line.

3.4.3 ENGINE CHILDDOWN 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 LH₂ and LO₂ through the engine. To reduce the amount of flight propellant loss for this childdown, 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 childdown check valve on the fuel turbopump at a temperature below -390°F. The ground childdown requires maintenance of fuel turbopump housing temperatures at or below -310°F for at least 15 minutes prior to launch (until approximately 30 seconds before liftoff).

The vehicleborne portion of the ground helium childdown system consists of transfer lines, supply tubes, connecting fittings, vent ducts, collector manifold, vent fin, and helium childdown 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 childdown check valve. Vantage flows out the gearbox vent and the childdown valves through staging ducts to the interstage adapter collector manifold, where it is exhausted to atmosphere.

Inflight childdown of the fuel system is accomplished following Titan/Centaur separation during the prestart cycle prior to each engine ignition. The cycle is initiated (eight seconds for first burn and either seventeen or 24 seconds for restarts, depending upon the coast time duration) 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 childdown valve where a portion of fuel is vented overboard through engine vent ducts. The remaining fuel passes through the second stage of the turbopump and into the discharge childdown 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, LO₂ 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 drives the fuel pump and oxidizer pump (through a gearbox). Increasing fuel pump discharge pressure is fed back into the actuator of the turbopump interstage chilldown valve, causing it to close completely. 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-1T REACTION CONTROL SYSTEM

ELEMENTS:	Subsection
• Two 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
- Provides Thrust for Propellant Settling during Coast
- Make Post-Injection Separation and Orientation Maneuvers

3.5.1 REACTION CONTROL ENGINE SYSTEM. The system consists of:

- a. Four 6-pound-thrust propellant settling axial engines (S2A, S2B, S4A, S4B).
- b. Eight 6-pound-thrust lateral engines, consisting of:
 1. Four 6-pound-thrust pitch/roll engines (P1 through P4)
 2. Four 6-pound-thrust yaw/roll engines (Y1 through Y4)

The axial engines provide thrust for propellant settling during the coast period and retrothrust during a portion of the retromaneuvers. Two thrust levels are obtainable for these functions by using either one or both pairs of the settling engines. The higher thrust level is used to expedite propellant settling following first main engine cutoff, to provide an elevated acceleration level during prestart events preceding second main engine start, and to increase the distance between vehicle and payload after payload separation. The lower thrust level is used for propellant settling/retention during the coast phase. The effect of a single axial engine failing to fire (no valve actuation) is minimized by switching from the A-pair to the B-pair of engines midway through any two-engine on phase.

For missions where their functions are not needed, the settling engines can be readily removed.

The eight lateral engines provide thrust for pitch, yaw, and roll control during the settled or zero-g coast periods, during vehicle precision pointing prior to payload separation, and during reorientation and retromaneuver after payload separation. Redundancy for the fail-to-fire mode is provided in pitch, yaw, and roll by the eight engine attitude control configuration and associated basic control logic (without failure detection).

The fixed axis, constant thrust, hydrogen peroxide (H_2O_2) engines are clustered in two groups of two and two groups of four on separate manifolds attached to the Centaur aft bulkhead on the 45/225 degree and 135/315-degree axes, respectively, (Figure 3-34). The axial engine thrust vectors are aligned parallel to the vehicle longitudinal axis. The lateral engine thrust vectors are aligned parallel to the vehicle 0/180-degree and 90/270-degree axes. The H_2O_2 feed line configuration provides a parallel redundant flow path to each engine cluster manifold.

3.5.2 HYDROGEN PEROXIDE MONOPROPELLANT SUPPLY SYSTEM. The hydrogen peroxide monopropellant supply system (Figure 3-35) feeds the boost pump turbodrives and the reaction control system. The system consists of: two manifolded positive expulsion supply bottles pressurized from an airborne helium supply; parallel redundant boost pump feed valves and a redundant purge valve; series redundant H_2O_2 vent valves; a pressurization valve with a redundant pneumatic vent valve; and tubing for pressurizing, venting, and transporting hydrogen peroxide to the boost pump turbodrives and reaction control supply system.

All feed system metallic parts contacting the hydrogen peroxide are manufactured from 300 series corrosion resistant stainless steel. Stainless steel encased constant-power line heaters, silver-brazed to the hydrogen peroxide supply line sections, maintain the propellant temperature between 40 and 120 degrees F.

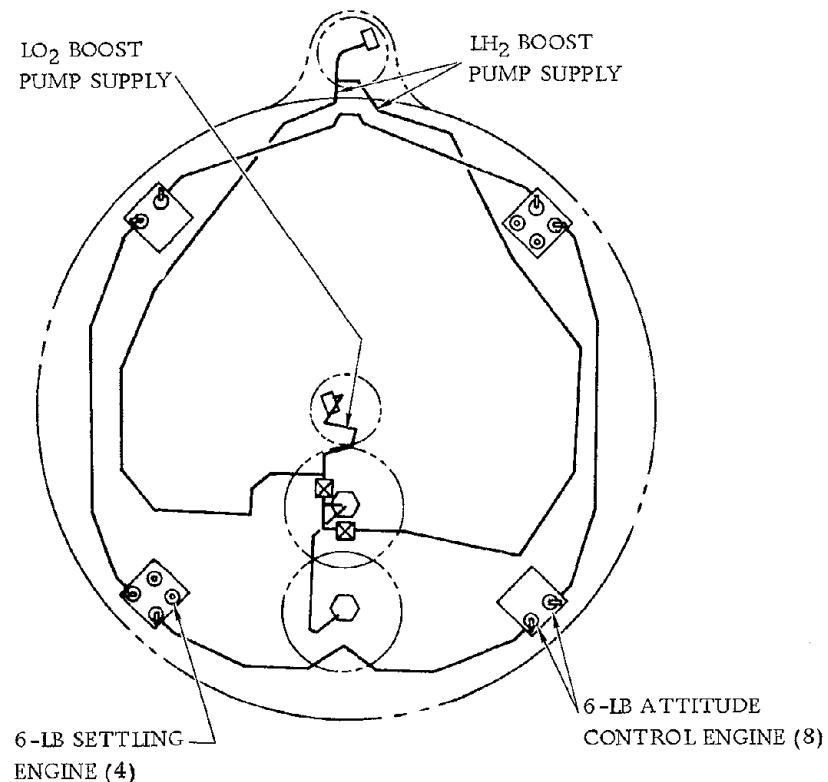


Figure 3-34. H_2O_2 reaction control system arrangement.

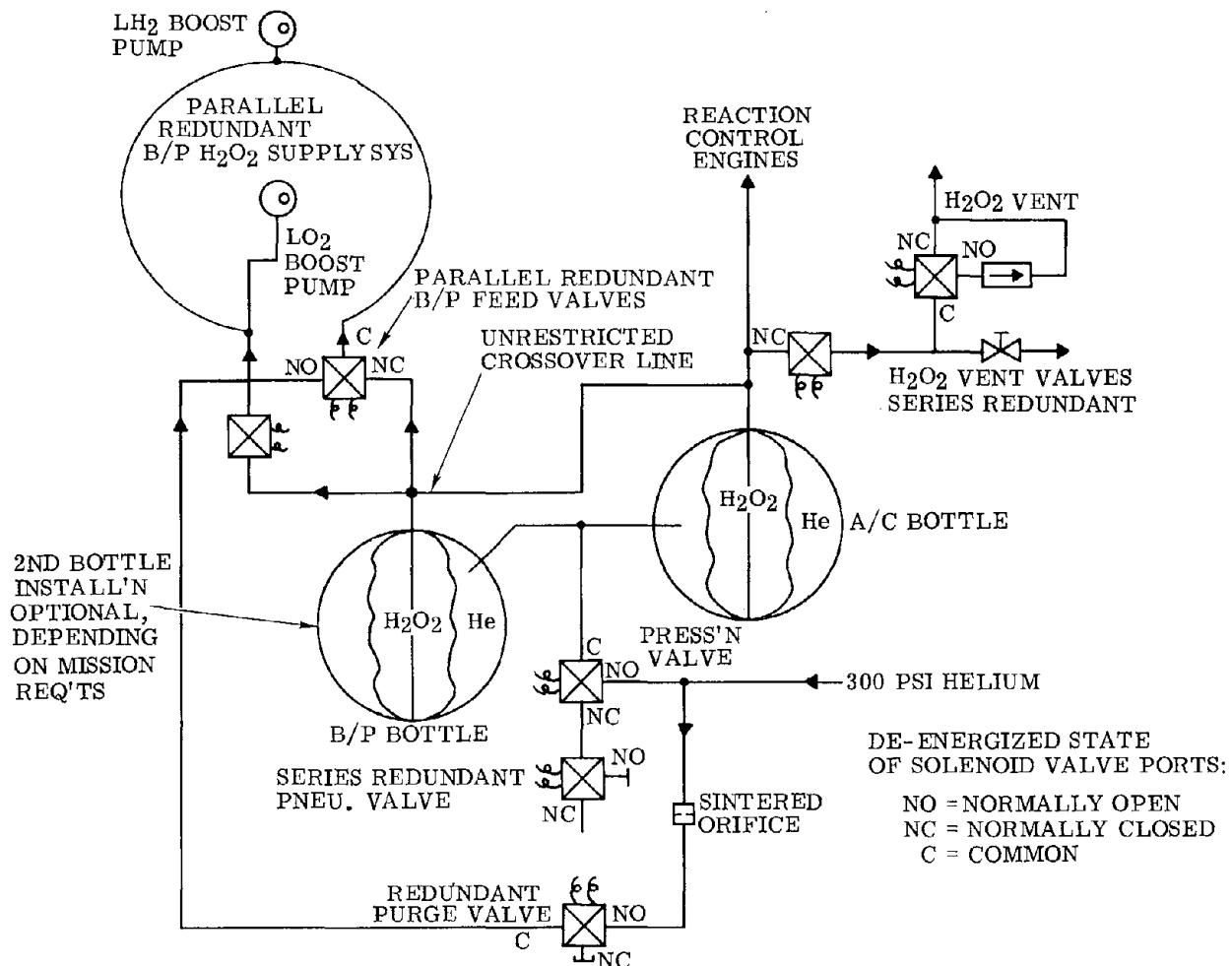


Figure 3-35. D-1T H₂O₂ system.

The positive expulsion supply bottles are pressurized to a nominal 306 psia. The hydrogen peroxide is separated from the helium pressurizing gas by a silicone rubber bladder. Nominal usable capacity of each supply bottle is 242 pounds. Total unusable residual propellant weight of the reaction control and hydrogen peroxide supply systems is eight pounds.

The three-way, two-position boost pump feed solenoid valve, when energized, shuts off the low pressure helium purge and permits hydrogen peroxide to flow from the supply bottles to the boost pump turbodrives. The three-way, two-position purge solenoid valve is simultaneously actuated to provide redundant shutoff of the low pressure purge. The two-way, two-position boost pump feed valve also is actuated simultaneously to provide a redundant hydrogen peroxide supply to the boost pump turbodrives.

During the Titan boost phase and subsequent to second main engine start, the reaction control engines are operated to expel gas that otherwise would pass through the boost pump turbodrives.

Hydrogen peroxide system venting and emergency dump capability are provided for by a relief system consisting of two ground controlled, series redundant vent solenoid valves and a relief valve. The two-way, two-position vent solenoid valve is followed by a three-way, two-position vent solenoid valve which provides a normally open port to accommodate a 350-psig relief valve. This valve will relieve any pressure increase due to hydrogen peroxide decomposition in the line between the valves when the valves are in their normally closed (deenergized) flight position.

The supply bottles are pressurized or vented by a ground controlled three-way, two-position solenoid valve. In addition, a ground controlled three-way, two-position solenoid valve and interconnecting line are attached to the pressurizing valve normally closed port to provide a redundant seal when the valves are in their normally closed (deenergized) flight position.

The two-bottle hydrogen peroxide supply system can be readily converted to a one-bottle supply system for missions requiring 242 pounds or less of hydrogen peroxide propellant.

3.6 CENTAUR HYDRAULIC SYSTEM

ELEMENTS:	Subsection
• Hydraulic Power Package (2)	3.6.1
• Servoactuator assemblies (4)	3.6.2
• Manifold, Insulation Block, and Thermostat	3.6.3

FUNCTIONS:

- Provides Power 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 Titan 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 fluid temperatures ranging from -30°F to +275°F. System weight, excluding mounting provisions, is 70 pounds (for both engines). Figures 3-36 and 3-37 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 a bootstrap reservoir; instrumentation; and the airborne half of the ground disconnects. Figure 3-38 illustrates the power package.

The high-pressure pump (Figure 3-39) is driven by the propellant turbopump gearbox accessory output drive on the RL10A-3-3 engine. A fiberglass resin block insulates the hydraulic power package from the cryogenic temperature of the propellant turbopump.

The low-pressure pump, shown in Figure 3-40, is driven by an electric motor (Figure 3-41) mounted on 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:

- a. Check out the hydraulic system on the ground.
- b. Equalize hydraulic system temperature by circulating hydraulic fluid when reduced temperatures are sensed by the thermostats in the manifold or actuator.
- c. Pre-position engines (on command from the guidance system prior to main engine start so that the thrust vector will be aligned at main engine start.
- d. Gimbal engines after payload separation so that thrust from residual propellants flowing through main engines aids in retromaneuver.

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, reservoir pressure is controlled by a bootstrap piston. The reservoir assembly consists of a piston assembly spring-loaded, and precharged with air at one atmosphere to provide nominal reservoir

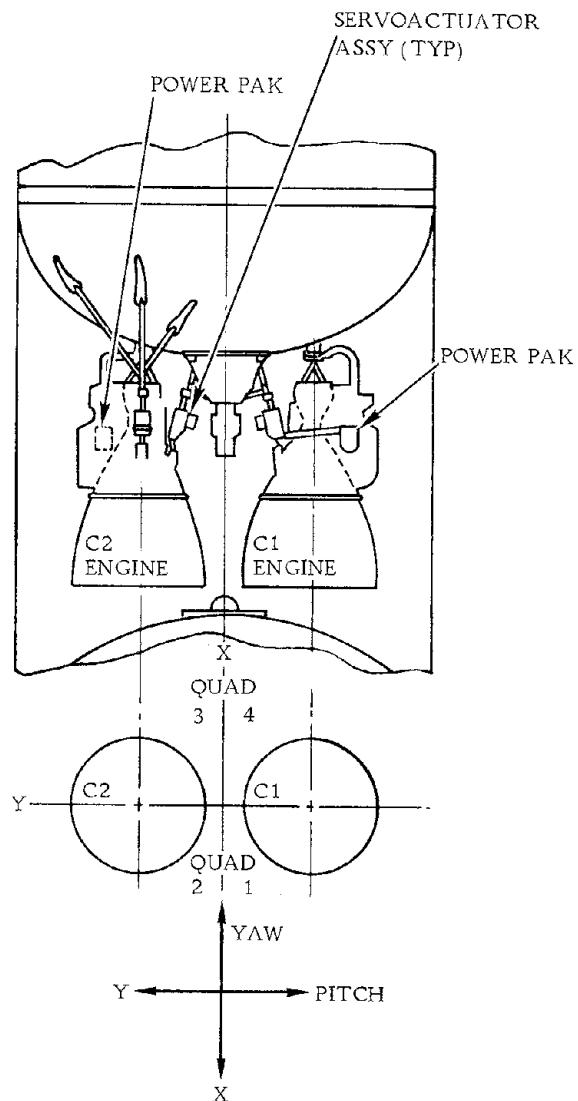


Figure 3-36. Centaur hydraulic system orientation.

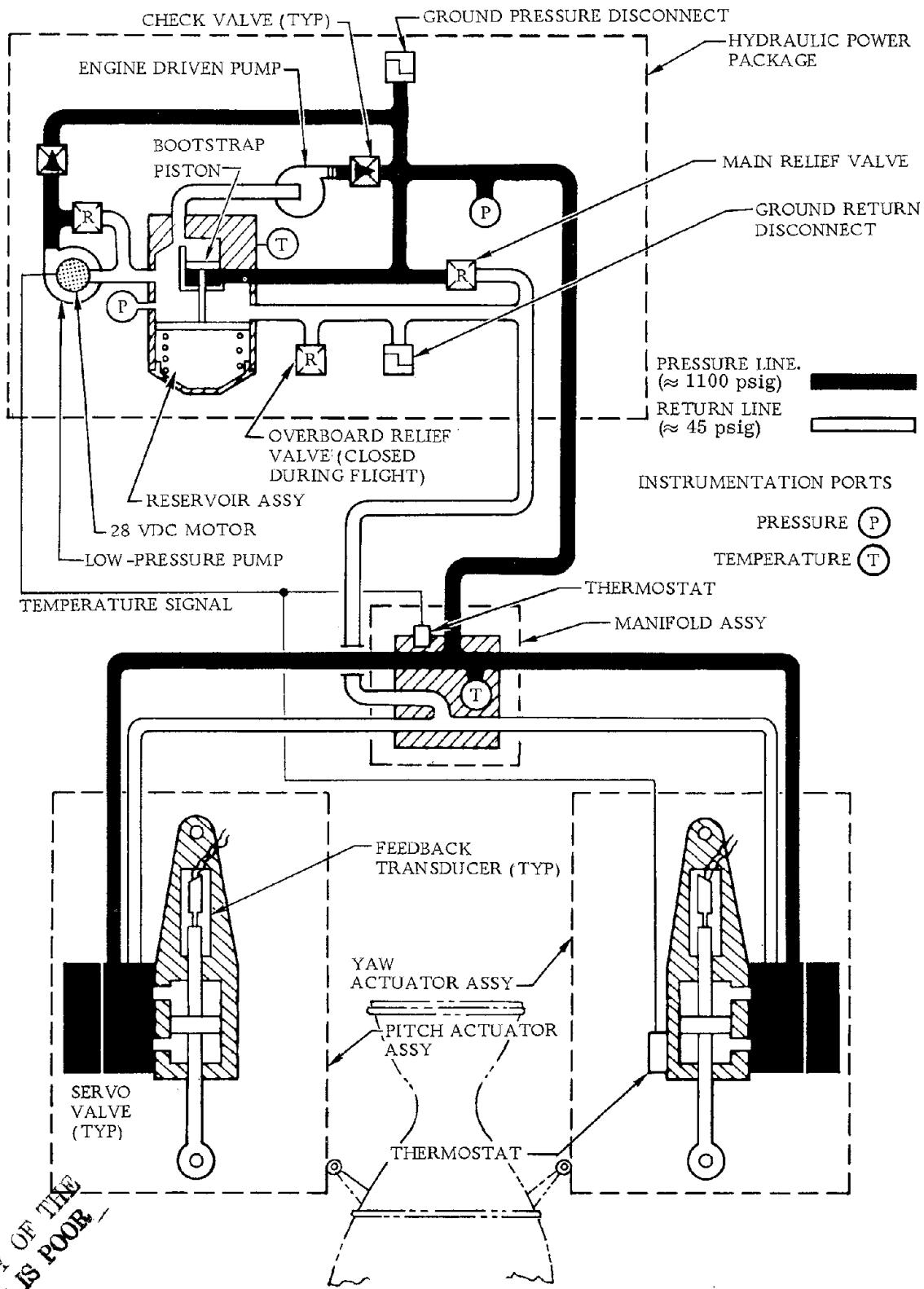


Figure 3-37. Hydraulic system schematic.

volume and pressure. This also prevents overpressurization. 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 SERVOACTUATOR ASSEMBLY.

Servoactuator assemblies convert the hydraulic pressure from the hydraulic power package into mechanical force which positions the engines. The servoactuator assembly (Figure 3-42) consists of the actuator assembly, the servovalve (Figure 3-43), and the actuator position feedback transducer (Figure 3-42). There are two servoactuator assemblies on each main engine: one to control yaw/roll movement, and one to control pitch movement of the engines. The servoactuators are controlled by electrical signals from the vehicle flight control system.

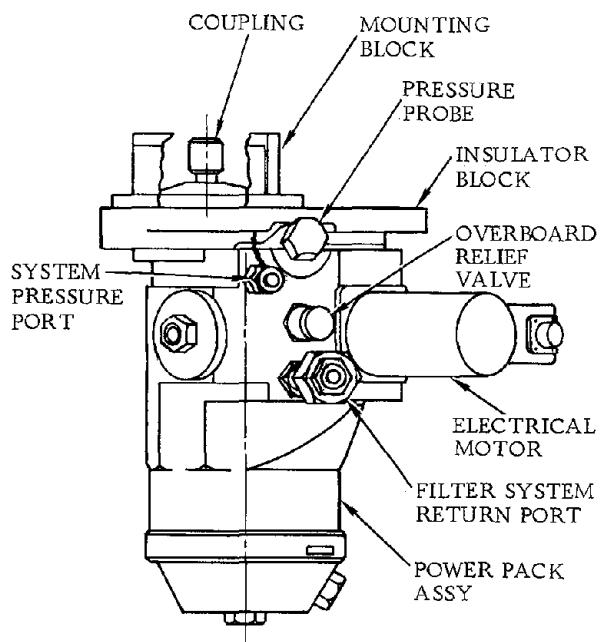


Figure 3-38. Centaur hydraulic power package.

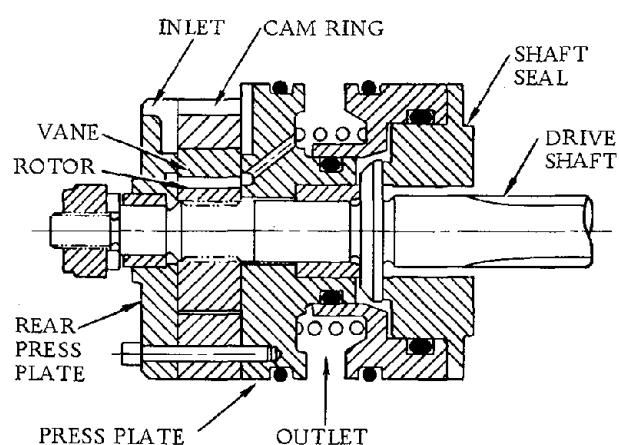


Figure 3-39. High-pressure pump.

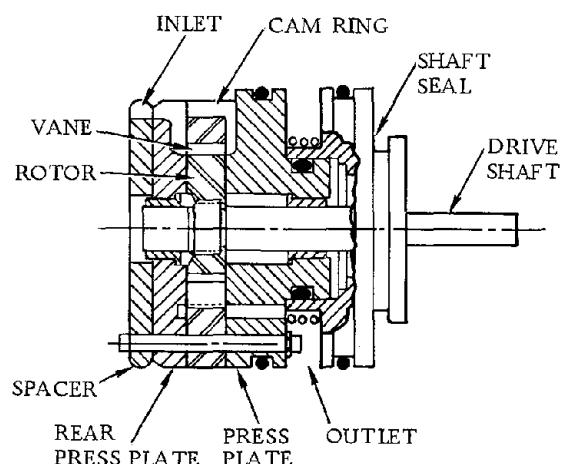


Figure 3-40. Low-pressure recirculating pump.

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.

The servovalve 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, a first stage amplifier, a second stage slide valve, and integral filter.

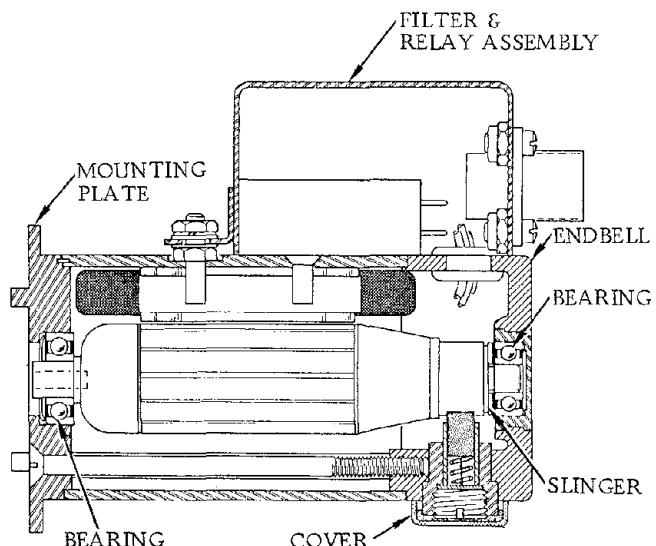


Figure 3-41. Electric motor.

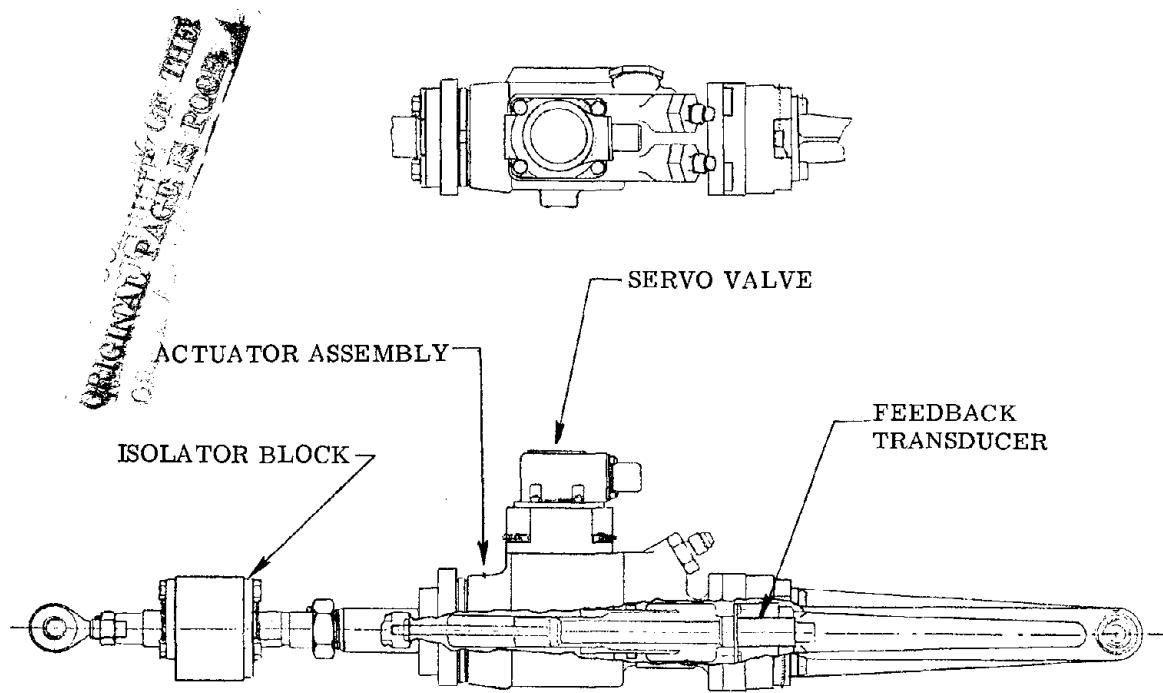


Figure 3-42. Hydraulic servoactuator assembly.

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 -inch limits the engine gimbal angle to ± 3.15 degrees from the neutral position. The force output of the actuator varies as a function of load, velocity, temperature, etc. Maximum stall force is 1678 pounds. The nominal static gain relationships for the servosystem components in the actuator assembly are as follows:

1. Actuator: 9.55 deg/sec/gpm with 1000 pounds external load.
2. 0.201 gpm/ma with 1000 pounds external load.
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.

An insulation block insulates the hydraulic power package from the engine accessory drive pad mounting; the temperature of the drive can approach LH₂ 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.

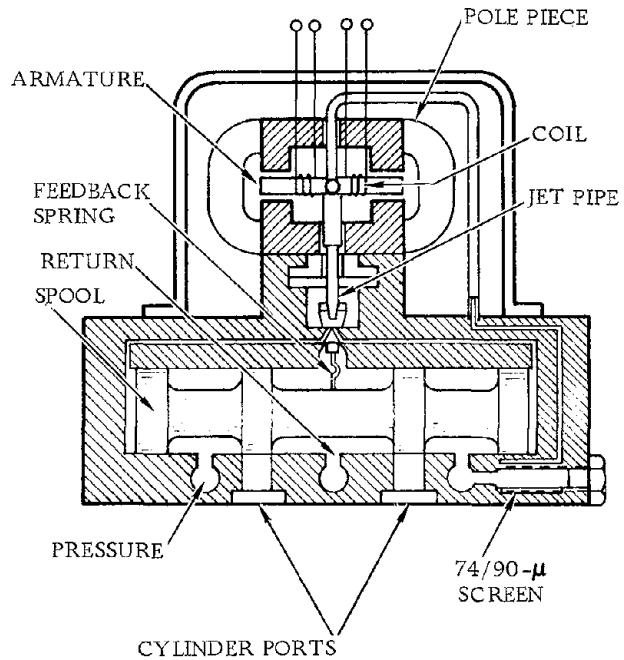


Figure 3-43. Servovalve schematic diagram.

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:	5750 rpm
Type:	Fixed volume displacement, vane

Electric Motor:

Voltage:	28v d-c
Speed:	5750 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:	98 psid

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-1T PNEUMATIC SYSTEM

ELEMENTS:	Subsection
• LH ₂ Tank and LO ₂ Pressurization System	3.7.1
• Tank Vent Systems	3.7.2
• Computer Controlled Vent and Pressurization System	3.7.3
• Engine Controls and H ₂ O ₂ Bottle Pressurization System	3.7.4
• Purge Systems	3.7.5
• Intermediate Bulkhead Vacuum System	3.7.6
• Gas Conditioning Systems	3.7.7

FUNCTIONS:
<ul style="list-style-type: none">• Provide Gaseous Helium for Propellant Tank, Engine Controls, and H₂O₂ System Pressurization• Maintain proper Propellant Tank Pressures• Provide Purge Gas for Warming certain critical areas, to prevent moisture accumulating in cryogenic areas and causing ice formation, and to control the amount of liquid in the LO₂ tank bubbler prior to pressurization• Prevent Inflow of Air to Intermediate Bulkhead to ensure desired Vacuum is achieved• Provide Conditioned Gas to maintain the proper environment in the Payload, Equipment Module, and Interstage Adapter areas prior to Launch

3.7.1 LH₂ TANK AND LO₂ TANK PRESSURIZATION SYSTEM. The pressurization system (Figure 3-44) provides the helium necessary to pressurize the LH₂ and LO₂ tanks to the levels required during various phases of flight. The system consists of one helium storage bottle, one flow control valve, four pressurization solenoid valves, two check valves, two pyrotechnic shutoff valves, a helium charge solenoid valve, pressure and temperature transducers, flow restrictors, a helium energy dissipator, a bubbler, and associated tubing. The pressurization system for the Proof Flight

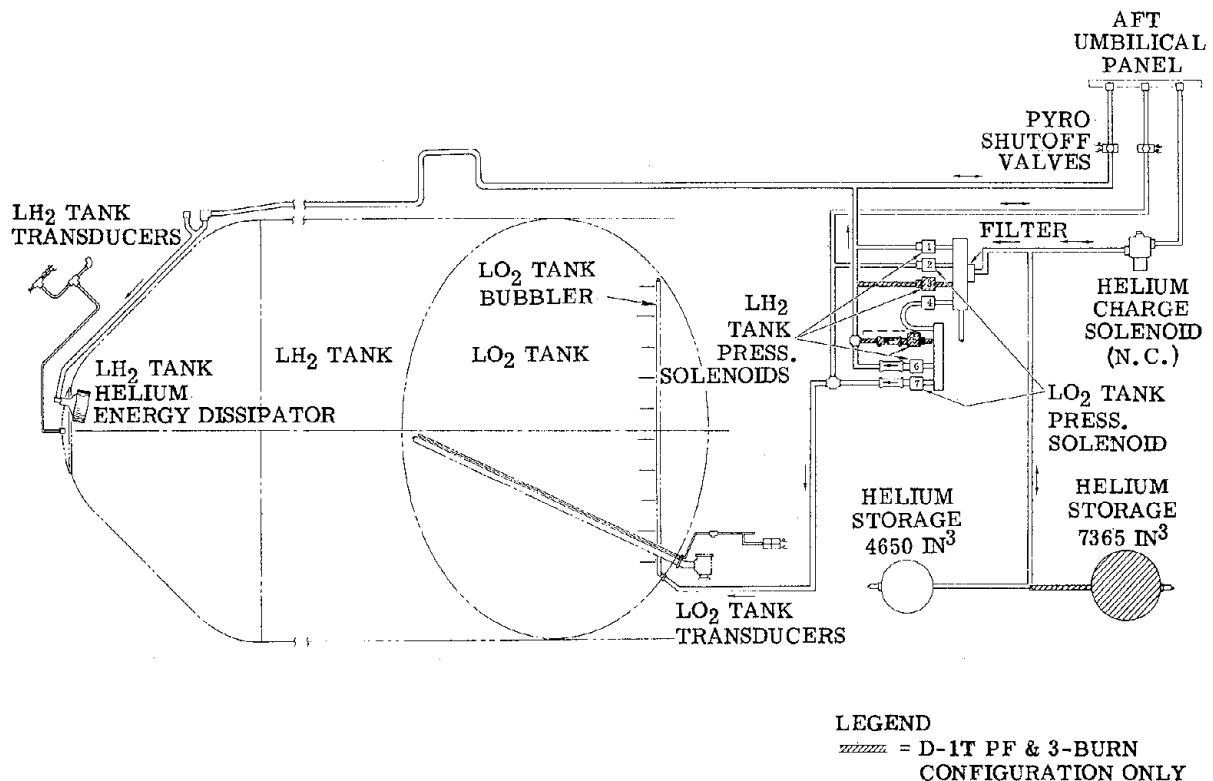


Figure 3-44. D-1T tank pressurization system.

vehicle (and all three-burn vehicles) contains an additional helium storage sphere and two additional pressurization valves for the LH₂ tank.

Prelaunch ground pressurization and helium storage bottle charging are accomplished through disconnects mounted on the aft umbilical (T-4) panel. When the T-4 panel is ejected, the airborne halves of the disconnects close to prevent loss of vehicle pressure. A normally open pyrotechnic shutoff valve is installed in each of the tank pressurization lines downstream of the LO₂ and LH₂ tank pressurization disconnects on the T-4 panel. The valves are closed just before liftoff to back up the closure of the self-sealing disconnects. Actuation of these valves is triggered automatically by a relay on the ground when a signal is received indicating that the shroud door, through which the T-4 panel was jettisoned, closed.

During countdown, ambient temperature helium is stored in the bottle at 3300 psi. Prior to main engine start, the helium is used to increase the pressures of the LH₂ and LO₂ tanks. The amount of pressure increase is controlled by the Computer-Controlled Vent and Pressurization System (CCVAPS - Subsection 3.7.3). The pressure increase is required so that the boost pumps will not cavitate during the main engine start sequence.

Two pressurization solenoid valves (one for each tank) and the flow control valve make up the primary system. The other two pressurization solenoid valves are backups. In the event of a failure of one or both of the primary valves, the flow control valve is closed and the backup valves are used for pressurization. Control of the solenoid valves is handled by CCVAPS.

The LH₂ tank pressurization line terminates at the helium energy dissipator mounted inside the forward door. The LO₂ tank pressurization line terminates at the bubbler mounted inside the LO₂ tank.

3.7.2 TANK VENT SYSTEMS. Venting of the propellant tanks is accomplished as required during ground operations and throughout flight to maintain the desired tank pressures. The hydrogen vent system (Figure 3-45) consists of two solenoid-operated vent valves (primary and secondary), two balanced thrust space vent nozzles with mating inflight disconnects, associated plenum chamber, and ducting. The LO₂ vent system consists of a single solenoid-operated vent valve, a space vent nozzle with mating inflight disconnect, and associated ducting. The solenoid-operated vent valves operate in either a relief or shutoff mode. When in the relief mode, the vent valves maintain a preset tank pressure. The vent valves are normally in the relief mode. Energizing the solenoid, positions the vent valves to the shutoff mode. The valves are positioned to maintain the required ullage pressure by the flight control system using either the programmed flight sequence or the Computer Controlled Vent and Pressurization System (CCVAPS).

Fuel Tank Vent System.
LH₂ tank venting is controlled by the primary vent valve during ground tanking and during flight after T + 90 seconds. The secondary vent valve controls venting during ascent prior to T + 90 seconds.

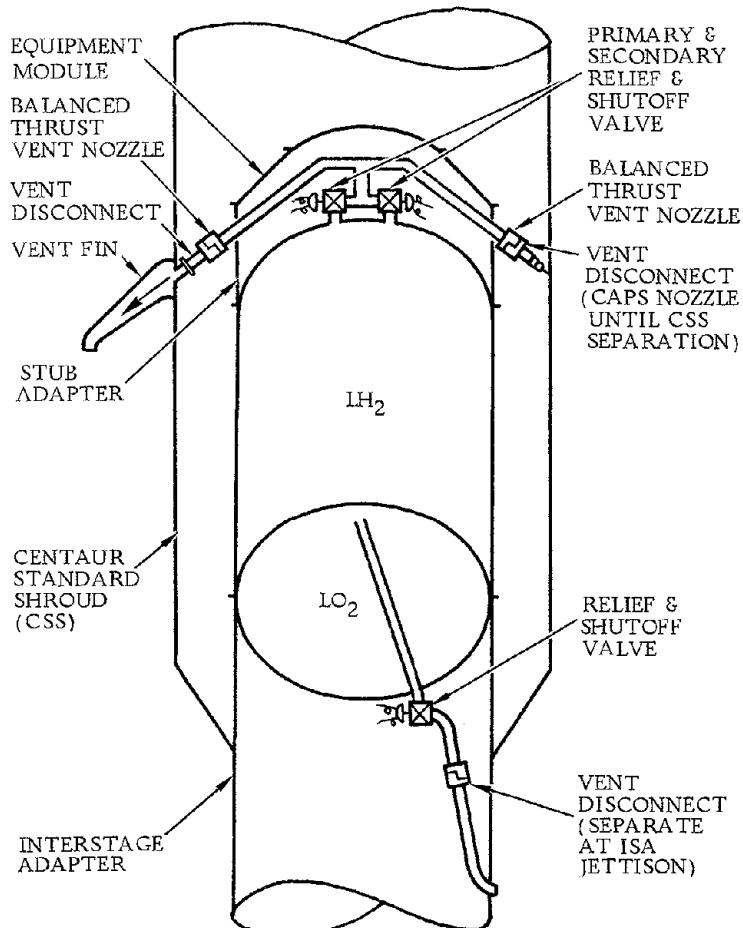
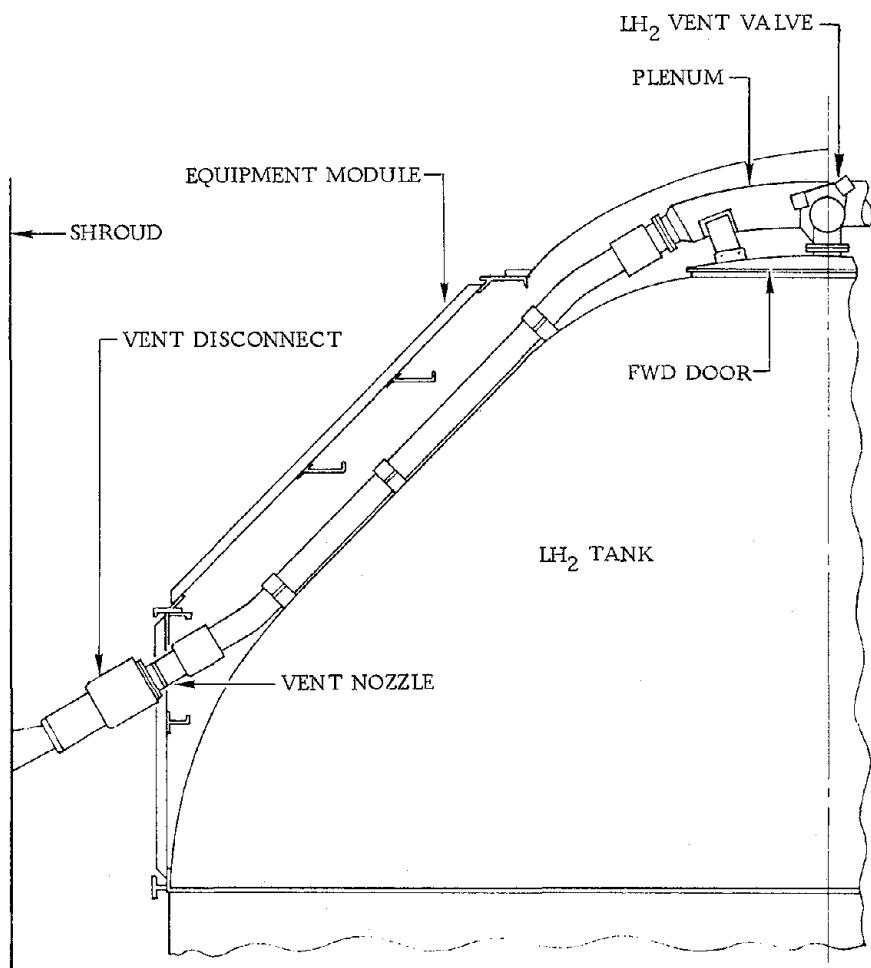


Figure 3-45. Hydrogen vent system.

The two vent valves are mounted opposite of each other on the LH₂ tank access door (Figure 3-46). A common plenum chamber is located between the two valves and connects the valve outlets to the two diametrically opposite vent duct legs which are mounted to and follow the basic contour of the LH₂ tank forward bulkhead and penetrate the stub adapter. The two opposite converging-diverging space vent nozzles are mounted on the exterior surface of the stub adapter and interface with the two vent duct legs at the stub adapter penetration points. The nozzles point aft 30 degrees from the vehicle station plane. All venting prior to Centaur Standard Shroud (CSS) jettison is through the nozzle which interfaces with the overboard vent fin through an inflight disconnect.



NOTE: Vent fin side (112.5°) shown — opposite side is same except disconnect caps nozzle until shroud jettison.

Figure 3-46. Centaur D-1T LH₂ vent system.

A similar opposite disconnect prevents flow from the other nozzle until shroud jettison. Both disconnects utilize ball joints with interconnecting telescoping sections to allow relative deflection between the vent nozzles and the Centaur Standard Shroud. Initial outward motion of the shroud, during jettison, exceeds the telescoping capability of the disconnects, causing the disconnects to separate from the nozzles by either of two redundant modes of release.

Both disconnect nozzle interfaces are covered with a fabric purge bag that is supplied with dry gaseous helium from the forward shroud seal purge ring. The purge bag is attached to the nozzles with a garter spring arrangement which allows it to separate with the disconnects. The total vent system is located within the helium environment of the forward bulkhead/equipment module cavity and the annular shroud/tank cavity. The plenum chamber and the two outlet ducts are under a multilayer insulation blanket. The two nozzles and the vent fin side disconnects are insulated with sheet foam and aluminized Mylar tape.

Oxidizer Tank Vent System. The LO₂ vent valve (Figure 3-47) is mounted on the standpipe flange on the aft bulkhead. A short section of aluminum alloy ducting connects the valve outlet to the space vent converging nozzle which has a slotted flange connection that allows the nozzle to be aligned to minimize vehicle disturbances during space venting. Nozzle alignment is made at installation. The outlet vector of the nozzle can be aligned from parallel to the vehicle centerline axis to outward 15 degrees from the vehicle centerline axis. The centerline axis of the nozzle is in the 90°/270° plane.

A section of ducting is mounted to the interstage adapter with the forward end connected to a ball-joint type disconnect with a sliding seal interface that mates with the space vent nozzle. The aft end of the ducting penetrates the interstage adapter (ISA) skin to allow venting overboard until ISA separation. The ISA mounted ducting is positioned at installation to align with the required nozzle angle. The disconnect/nozzle interface

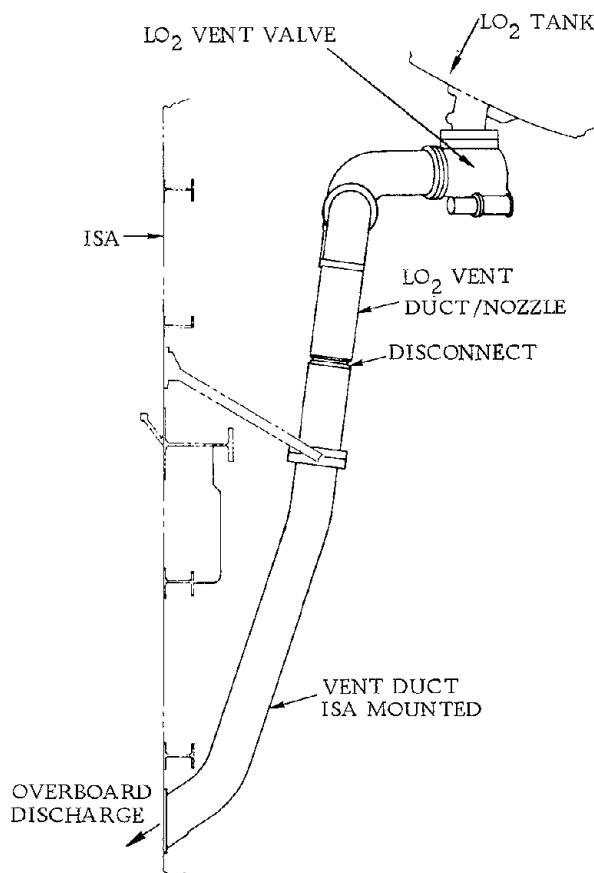


Figure 3-47. Centaur D-1T LO₂ vent system.

is covered with a fabric purge bag which is supplied with dry gaseous helium from the ISA purge system. The purge cover is attached to the nozzle with a garter spring arrangement which allows it to separate with the disconnect.

All of the vent system, except the vent valve, is insulated with 1/2-inch thick sheet foam with either aluminized Mylar tape or white PVC tape covering. Leak deflectors are installed around the valve inlet and outlet flanges to prevent leak impingement on temperature sensitive components.

3.7.3 COMPUTER CONTROLLED VENT AND PRESSURIZATION SYSTEM. The computer controlled vent and pressurization system (CCVAPS), when activated, maintains the LO₂ tank and LH₂ tank pressures at required levels during various phases of flight. CCVAPS (Figure 3-48) consists of the pressurization system, the vent system, the ullage pressure transducers, the digital computer unit (DCU), and the sequence control unit (SCU).

The DCU monitors the tank pressures during selected periods of the flight and issues appropriate commands to the SCU relays which activate the appropriate pressurization solenoid valves and solenoid-operated vent valves. The DCU is programmed

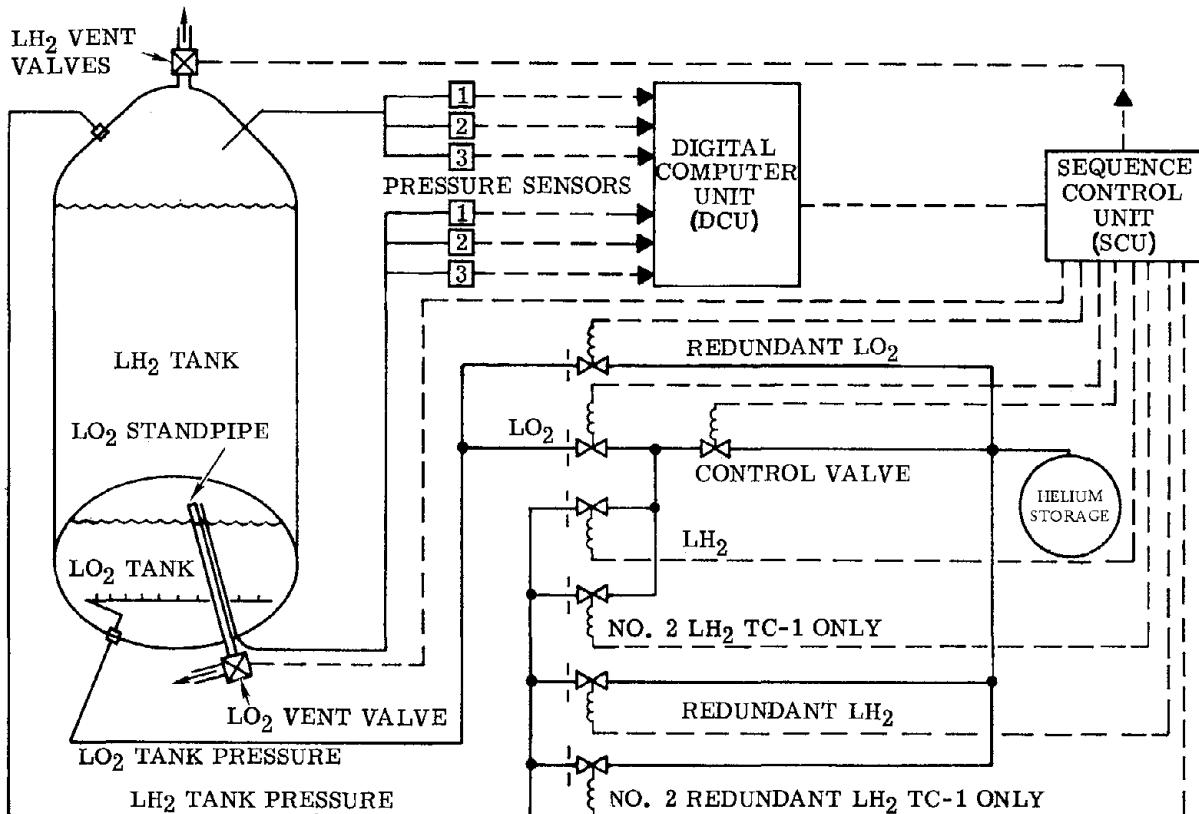


Figure 3-48. CCVAPS schematic.

with maximum and minimum allowable tank pressures and pressure increases prior to main engine start. In addition, the DCU is able to detect ullage pressure transducer failures and pressurization solenoid valve failures.

The DCU monitors each tank pressure through three transducers, a primary, a reference, and a backup. In the event of a discrepancy between the outputs of the primary and reference transducers, the DCU uses the output of the backup transducer to determine tank pressure.

During a pressurization sequence, if the DCU monitors an insufficient or excessive increase in pressure (due to a pressurization solenoid valve failure), the DCU commands the flow control valve closed and uses the redundant pressurization solenoid valves (both tanks) for the remaining portion of the flight.

Prior to main engine start, CCVAPS is activated, the pressurization solenoid valves and the solenoid-operated vent valves are activated, and both tanks' pressures are increased by predetermined amounts and maintained at those levels until main engine start. CCVAPS is deactivated at main engine start.

During the coast phase, CCVAPS is activated to prevent the LO₂ tank and LH₂ tank pressures from exceeding maximum allowables. The appropriate H₂O₂ engines are fired to settle the propellants, then the solenoid-operated vent valves are deenergized to allow venting.

CCVAPS is also activated after the final main engine cutoff. The LO₂ tank is pressurized to prevent an adverse intermediate bulkhead ΔP condition.

3.7.4 ENGINE CONTROLS AND H₂O₂ BOTTLE PRESSURIZATION SYSTEM. The engine controls and H₂O₂ bottle pressurization system (Figure 3-49) consists of high- and low-pressure regulators, relief valves, filters, checkout fittings, and associated tubing. System pressure is supplied by the helium storage bottle (common to the LO₂ and LH₂ tanks' pressurization systems).

The high pressure regulator reduces the storage bottle pressure to 450 psi. This helium is used to control the engine system propellant valves. A constant bleed orifice downstream of the regulator prevents the buildup of pressure due to minor leakage through the regulator.

The low pressure regulator reduces the engine control regulator output pressure to 300 psi. This helium is used to expel the H₂O₂ from its bottle. A constant bleed orifice downstream of the regulator purges the H₂O₂ lines to the boost pumps when they are not operating. This eliminates residual H₂O₂ and prevents freeze up during long coast periods.

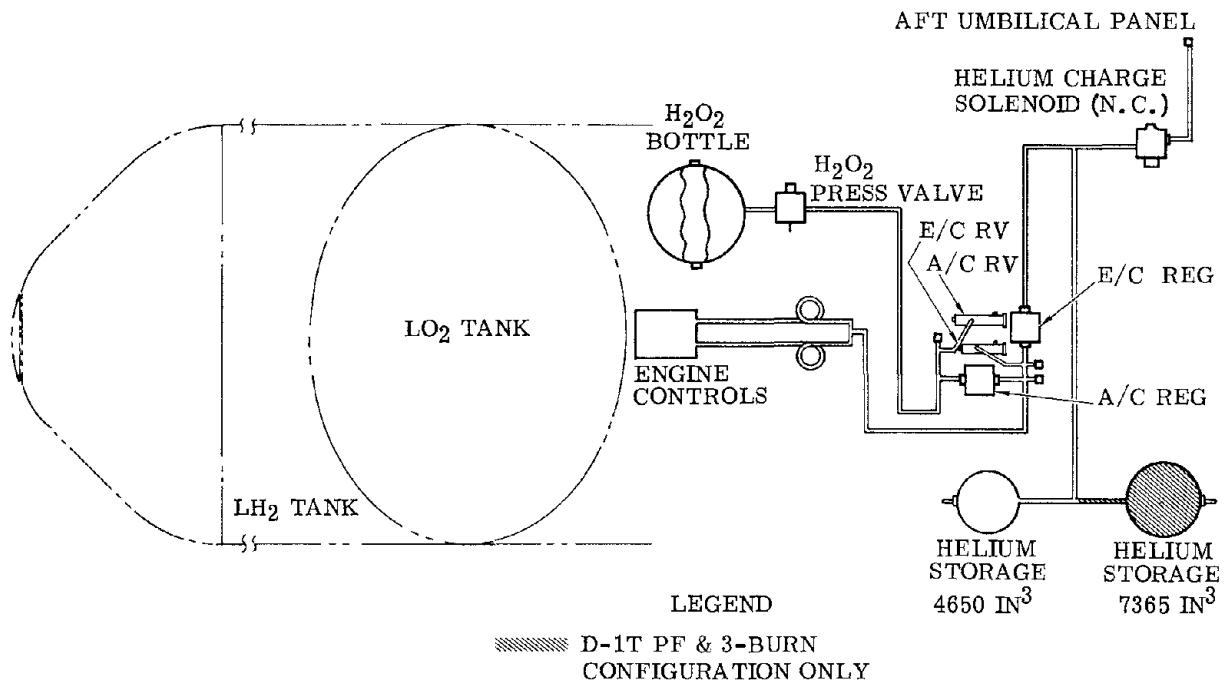


Figure 3-49. Pressurization for H_2O_2 bottle and engine controls.

Checkout fittings for the engine controls and H_2O_2 bottle pressurization system contain filters to prevent contamination of the system during checkout.

3.7.5 PURGE SYSTEMS. The purge systems (Figure 3-50) are comprised of the forward purge system, the aft purge system, and the bubbler purge system. These purge systems:

- Prevent cryo pumping in and/or provide warming to certain critical areas during tanking, detanking, and abort operations.
- Prevent moisture accumulation in the forward bulkhead and LH_2 tank radiation shielding areas during standby periods.
- Control the amount of liquid in the bubbler prior to pressurization.

The Proof Flight vehicle (and any zero-g coast mission) also uses a zero-g purge system. This system provides a low flow of helium from the engine controls regulator on the pneumatic panel to the LO_2 tank sensing line, the LO_2 tank standpipe, and the LH_2 tank pressurization line and dissipator. These purges are required to maintain the lines free of propellant during zero-g coasting periods.

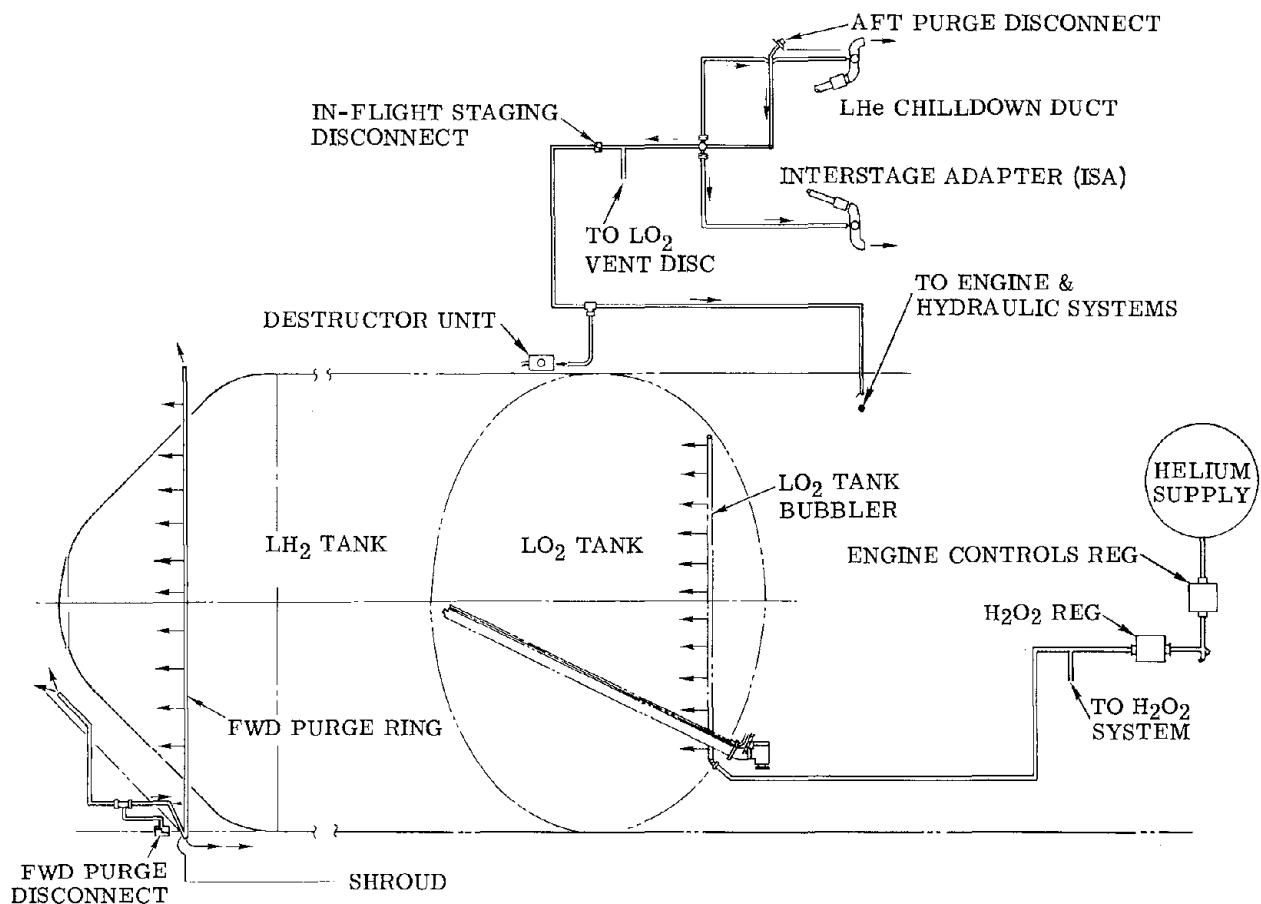


Figure 3-50. Purge systems schematic.

The forward purge system consists of a ground/airborne disconnect, flow restrictors, and associated tubing. This system provides nitrogen during standby conditions and helium prior to and during tanking operations to the equipment module/forward bulkhead cavity, the forward seal area, the LH₂ tank radiation shielding, and the LH₂ tank vent system disconnects. The purge disconnect has a secondary mode of separation if the primary mode fails to operate properly.

The aft purge system consists of a ground/airborne disconnect, and inflight staging disconnect, manifolds, flow restrictors, and associated tubing. This system provides helium prior to and during tanking operations to the helium chilldown ducts, the LO₂ vent system disconnect, the destructor unit, various propulsion system areas, various hydraulic system areas, and the fuel duct installation. The ground/airborne disconnect is identical to the one used in the forward purge system.

The bubbler purge system provides a helium purge flow to the LO₂ tank bubbler. Helium is provided through a sintered orifice located downstream of the H₂O₂ pressure regulator.

3.7.6 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 propellant tanking test and the desired vacuum cannot be attained, the airborne system, in conjunction with a ground vacuum pump, prevents a pressure buildup within the cavity.

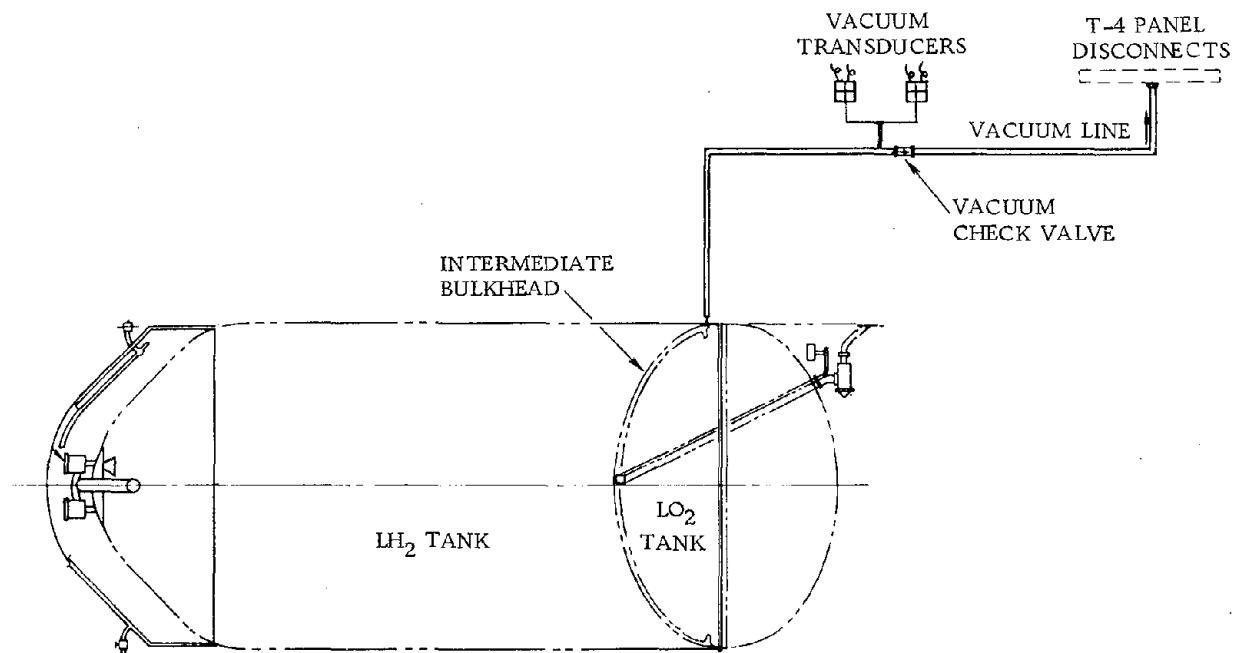


Figure 3-51. Centaur D-1T bulkhead vacuum system.

3.7.7 GAS CONDITIONING SYSTEMS

Payload Gas Conditioning System. The payload gas conditioning system (Figure 3-52) provides dry air or GN₂ to the spacecraft at the required temperature and flow-rate. The gas is provided through a ground/airborne disconnect on the shroud. A diffuser at the shroud inlet directs the gas upward to prevent impingement on the spacecraft. The gas is then vented through the encapsulation bulkhead at four redundant vent valves into the equipment module, then out to the atmosphere through the shroud. The encapsulation bulkhead is a mission peculiar item.

Equipment Module Gas Conditioning

System. This system (Figure 3-53) provides dry air or GN₂ to the equipment module area. The gas conditions the avionics, the electrical disconnect panels, the forward seal, and the forward bearing reaction struts pyrotechnics. The gas is distributed by an annular duct mounted on the equipment module. The distribution duct has numerous holes, nozzles, and bubbles or hemispheric nozzles that direct the gas to the various components. The equipment module disconnect consists of a fiberglass bellows interfacing with the Y-duct inlet.

Interstage Adapter (ISA) Gas Conditioning System. This system conditions the H₂O₂ system, various electrical and RF components, pneumatic panels, pyrotechnic shutoff valves, the signal conditioner, and the remote multiplexer unit. The gas (dry air or GN₂) is distributed by an annular duct that is part of the ISA. The distribution duct has numerous holes, nozzles, and riser ducts that direct the gas to the various components. The gas is vented through openings in the ISA skin, then overboard through the shroud vents.

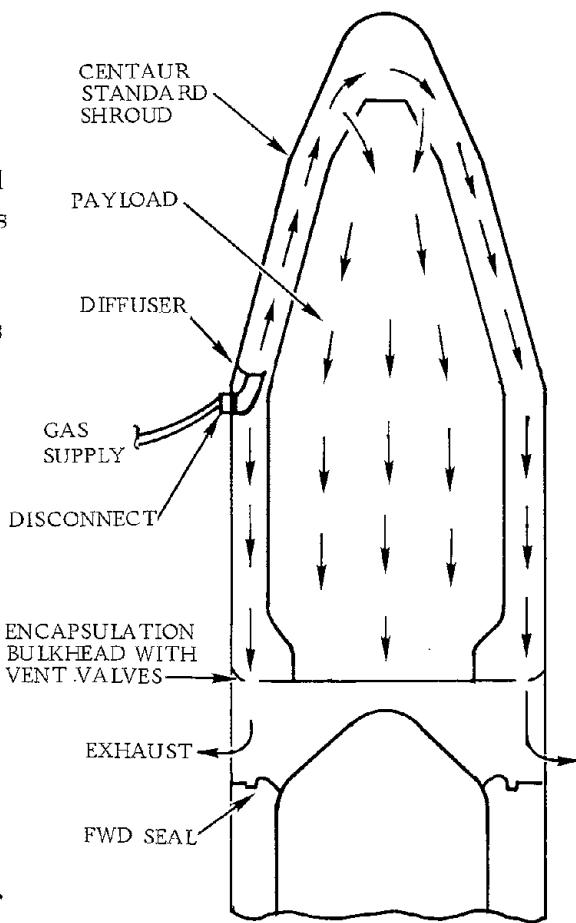


Figure 3-52. Payload gas conditioning system.

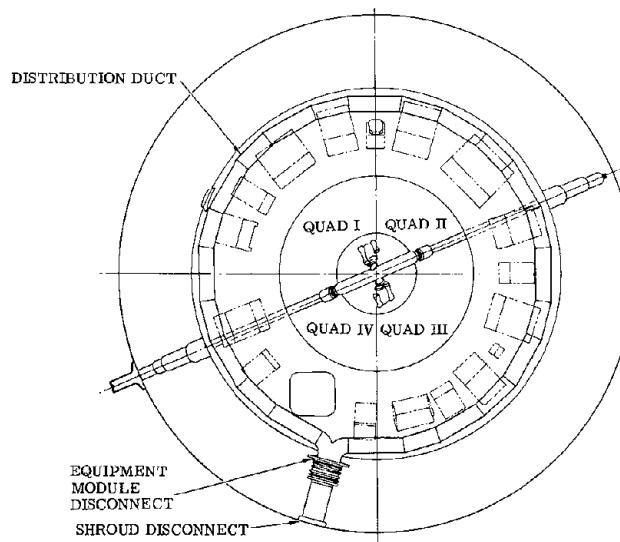


Figure 3-53. Equipment gas conditioning system.

4

CENTAUR D-1T ASTRIONICS SYSTEMS

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

SUBSECTION	SYSTEM	Page
4.1	Digital Computer Unit	4-3
4.2	Instrumentation and Telemetry	4-7
4.3	Computer Controlled Launch Set	4-14
4.4	Navigation	4-18
4.5	Guidance Function	4-23
4.6	Control System	4-24
4.7	Vehicle Sequencing	4-28
4.8	Propellant Utilization	4-34
4.9	C-Band Tracking System	4-39
4.10	Range Safety Command System	4-42
4.11	Electrical Power System	4-48

The airborne software stored 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-1T are mounted on the equipment module. In Figure 4-1 the digital computer unit is on the left and the signal conditioner is in the center. Layout of all packages is shown in Figure 4-2.

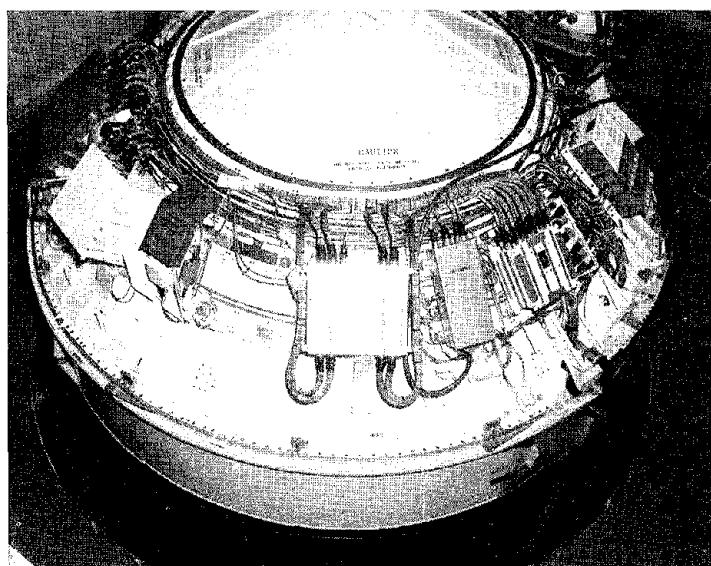
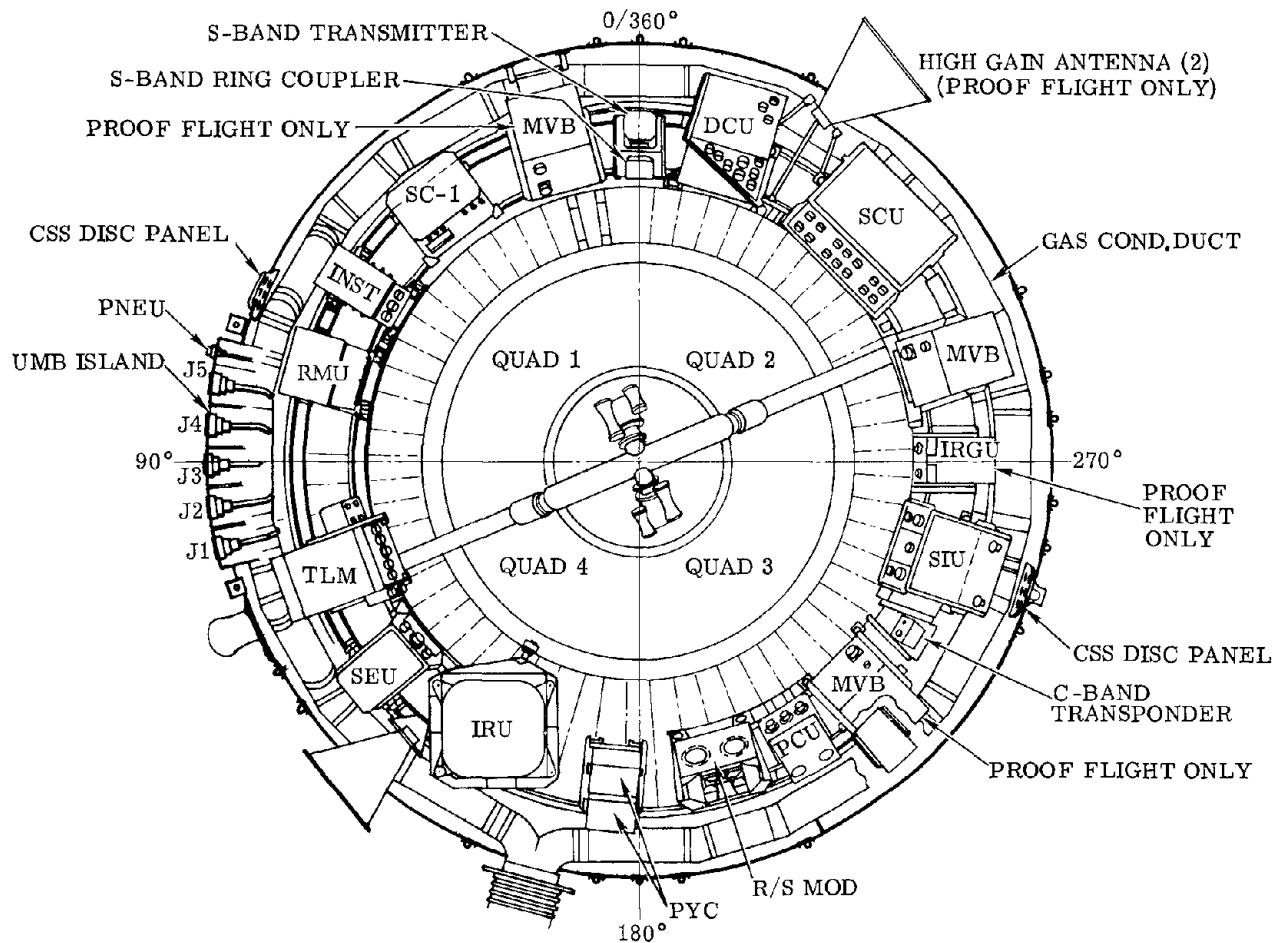


Figure 4-1. Astrionics packages mounted on equipment module.



ABBREVIATIONS

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

FORWARD UMBILICAL PANEL DISCONNECTS

J-1 CENTAUR]
 J-2 CENTAUR]
 J-3 SPACECRAFT] ————— T-4 EJECT

J-4 SPACECRAFT]
 J-5 CENTAUR]
 PNEU PNEUMATIC] ————— T-0 EJECT

Figure 4-2. Centaur D-1T 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, fast, flexible 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 digital computer has permitted many functions to be done by software, which were formerly done by other hardware. In particular, mission-peculiar requirements are handled by software where practical, permitting changes to be incorporated by programming rather than hardware modifications.

4.1.1 TASKS. The task of the DCU is to receive and process 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 desired orientation of the vehicle, and output it in terms of pitch and roll axes vectors. 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 H₂O₂ engines to orient the vehicle in the proper attitude, by using attitude error inputs and digitally derived rates.

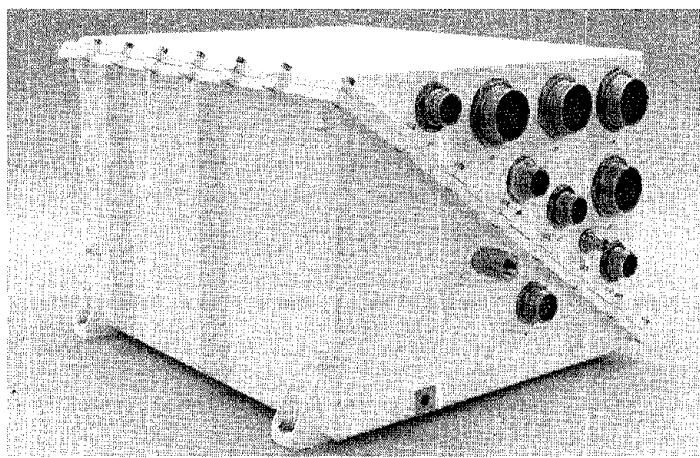
Sequencing. Determine when discrete events should occur, and command them to occur.

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 according to a selected stored format 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" x 14" x 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	
Input/Output	6 μ sec	
Multiply	22.5 μ sec	
Divide	40.5 μ sec	
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	Δ V's and Real Time
Analog - 12 Channels	11 Bits plus Sign, ± 12v d-c
Telemetry -	
Remote Multiplexer Units - 4 Channels	2 Spares with Titan 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 a Titan 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 Proof Flight 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 Proof Flight configuration includes seven SKD discretes and the twelve-bit parallel input channel. Fourteen additional output discrete bits are also available on this mission, as well as three digital to analog (D/A) output channels.

4.1.4 CHECKOUT AND LAUNCH. The DCU is checked out, with its flight program, in a Flight Acceleration Profile (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)
- FM/FM Telepak
- Ring Coupler and Antennas
- Coaxial Switches

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 RMUs.

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.

An optional FM/FM telemetry system is also used on the D-1 Centaur to accommodate certain environmental and payload peculiar measurements.

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 PCM 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 down-link 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.

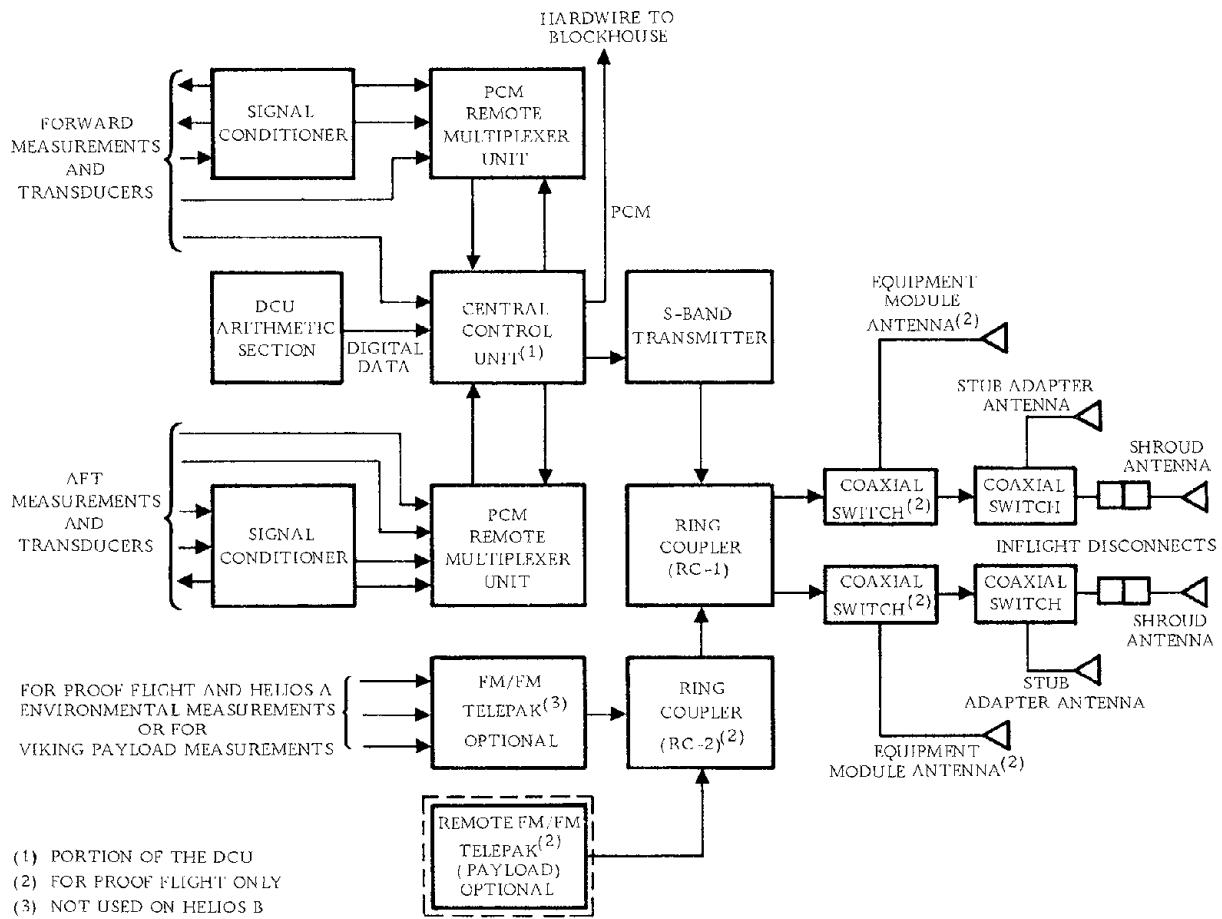


Figure 4-4. Instrumentation and telemetry system.

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. The RMUs identify signals by addresses, and when addressed by a central controller unit (CCU) send these measurements to a central controller unit.

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 a ring coupler and feeds the antennas.

The FM/FM telemeter package (Telepak) (Figure 4-4) provides excitation for and accepts data signals from transducers, and conditions the signals into proper voltages compatible and proportional to the assigned subcarrier channel oscillator input. The conditioned signals are composited and used to frequency modulate an S-band transmitter carrier frequency. The resultant RF signal is fed from the S-band transmitter (mounted on the FM/FM Telepak) to a ring coupler which splits the signal and feeds the antennas.

Two S-band antennas are installed on the Centaur standard shroud for data transmission until the shroud is separated from the vehicle. After shroud separation, data transmission is accomplished with two antennas mounted on the stub adapter. In addition, for the Proof Flight only, two high-gain S-band antennas are mounted on the equipment module for use in deep space. Switching from one pair of antennas to the other is accomplished with coaxial switches (Figure 4-4), the same as in the C-band tracking system. Two antennas are used to ensure good ground coverage.

4.2.2 SYSTEM CAPABILITY. The PCM 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 normalize them as required.

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.

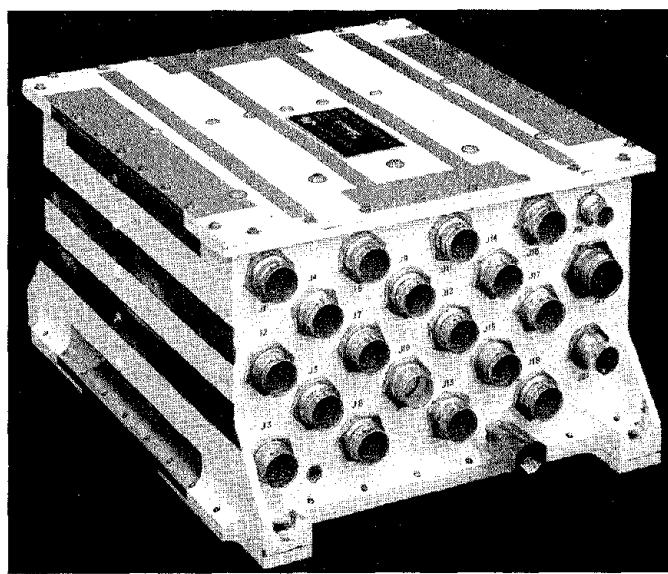
The FM/FM instrumentation and telemetry function can be added to the vehicle to satisfy mission requirements.

The FM/FM measurements are analog. The telepak has 18 channels available for monitoring data signals. These channels may be used for continuous monitoring of signals or some channels could be commutated to sample signals at various rates. Commutating some channels would require the addition of a commutator to the telemeter. The number of measurements may vary considerably depending on the mix of continuous and commutated channels.

PCM Growth. Over 1000 measurements are available for use in a format that can accommodate an additional 15,000 measurement samples per second. These can be accommodated by adding circuit boards or RMUs and extra signal conditioners, and by including these measurements in the PCM format.

4.2.3 COMPONENT DESCRIPTIONS. The PCM instrumentation and telemetry function uses the DCU (Subsection 4.1), up to four remote multiplexer units and signal conditioners, a telemetry transmitter, a ring coupler, and four antennas. The FM/FM instrumentation and telemetry function requires the addition of an FM/FM Telepak because it uses the ring coupler and antennas provided for the PCM function. The following paragraphs describe each of these components.

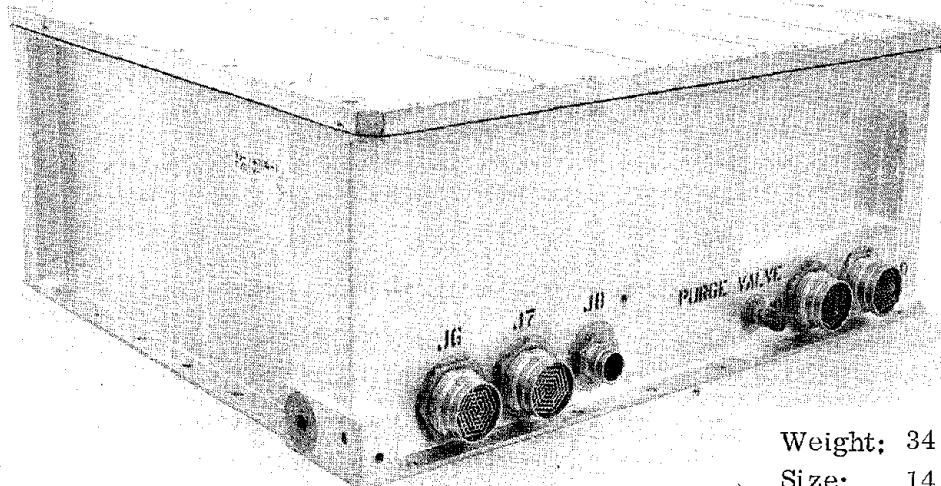
Remote Multiplexer Unit (RMU). Each RMU (Figure 4-5) is tailored to its requirements by the inclusion or omission of circuit boards that service the various types of signals. The RMU contains A/D converters, registers, logic, and power supplies. Its power consumption is rated at 25°C, below which a 26-watt heater cycles on.



Weight: 19-1/2 lb
Size: 12" x 10" x 6"
Power: 22 watts (28v d-c)

Figure 4-5. Remote multiplexer.

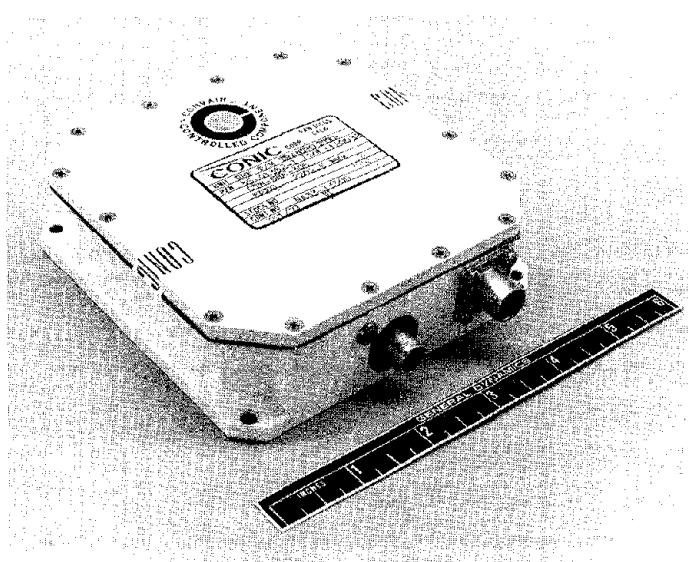
Signal Conditioners. Like the RMU, each signal conditioner (Figure 4-6) is tailored to its specific tasks. It contains circuitry to convert measurements to signal voltages compatible with the analog ranges available in the RMU. Transducers are selected from a standard list to convert physical measurements,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



Weight: 34 lb
Size: 14.6" x 13.3" x 6.5"
Power: 56 watts (28v d-c)

Figure 4-6. Signal conditioner.

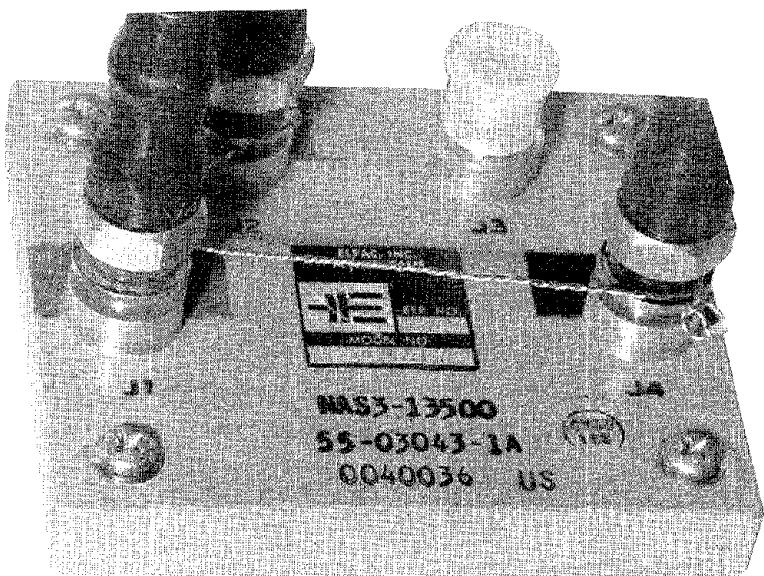
Telemetry Transmitter. The telemetry transmitter is shown in Figure 4-7.



Weight: 2 lb
Size: 4.6" x 4.6" x 1.3"
Power: 45 watts (28v d-c)
Output: 5 watts minimum
at S-Band

Figure 4-7. Telemetry transmitter.

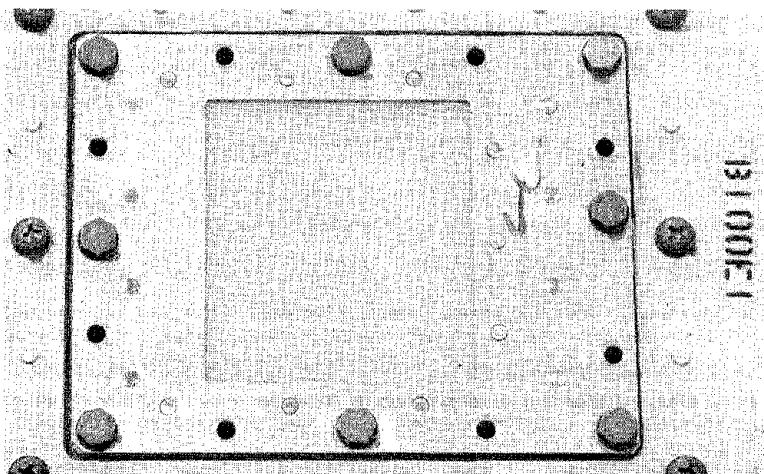
Ring Coupler. The ring coupler (Figure 4-8) divides the transmitter output power and directs half to each antenna.



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

Figure 4-8. Ring coupler.

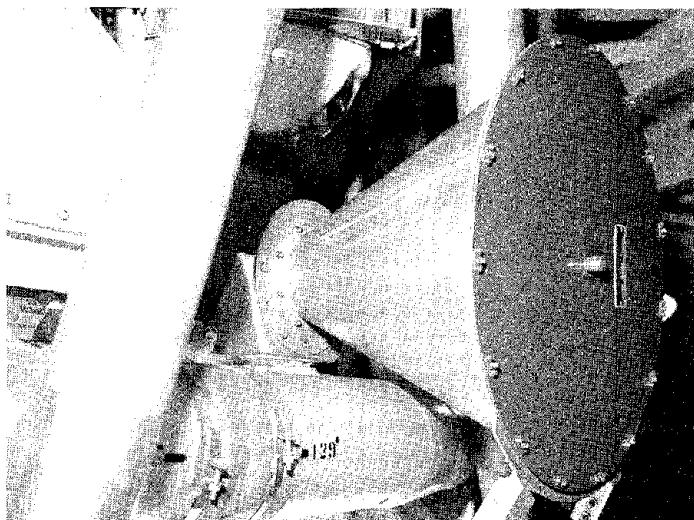
Antennas. The telemetry system has four antennas. Two antennas are located on the shroud and two on the stub adapter. The transmitter output power is switched from the shroud antennas to the stub adapter antennas by the coaxial switches at shroud separation. The radiation pattern of each antenna (Figure 4-9) is such that two antennas give continuously adequate coverage of the telemetry ground stations during all flight phases when transmission is desired on normal missions. For missions such as the Proof Flight, where the vehicle is operating in deep space, two high-gain S-band



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

Figure 4-9. S-band antenna.

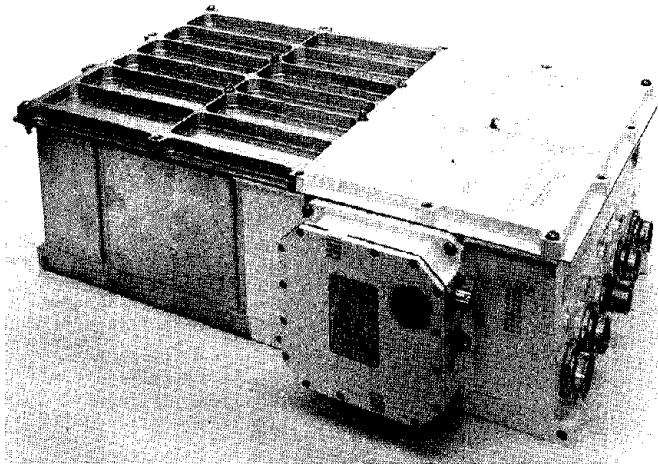
antennas (Figure 4-10) are mounted on the equipment module. The transmitter output power is switched from the stub adapter antennas to these high-gain antennas by coaxial switches.



Weight: 3 lb
Size: 16" dia. × 12" deep horn

Figure 4-10. High-gain S-band antenna.

FM/FM Telepak. The Telepak provides excitation for and accepts data signals from transducers, and conditions the signals into voltages compatible with the subcarrier channel oscillators input characteristics for transmission to the telemetry ground station. The FM/FM Telepak is shown in Figure 4-11.



Weight: 24 lb
Size: 6" × 10" × 17"
Power: 5 watts minimum

Figure 4-11. FM/FM Telepak.

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 4-12). 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. The CCLS operates in conjunction with the telemetry ground station.

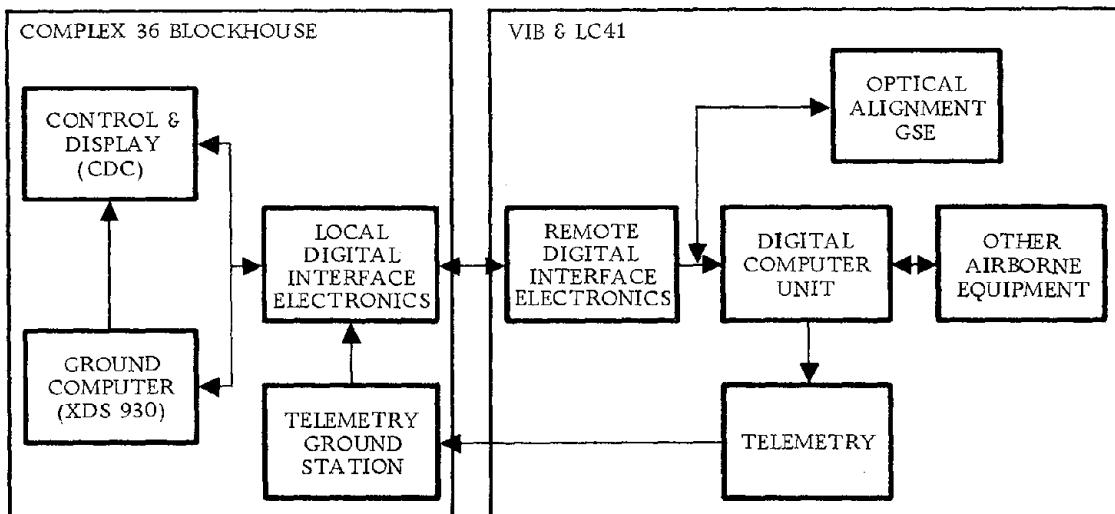


Figure 4-12. Computer controlled launch set layout.

The use of CCLS permits: (1) rigorous checkout, repeatable for every vehicle; (2) bit-for-bit verification of DCU memory contents, (3) error detection and correction for the data transmission links, (4) filtering and correlation processes for analog data, and (5) 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 Centaur SIU to ensure that: (1) the engine servoamplifiers are operating correctly, (2) the propellant utilization error detection circuitry is operating correctly, and (3) the inverter is generating 400-Hz power correctly. The CCLS also stimulates the SIU via the DCU during hydraulics checkout and monitors status prior to liftoff.

Computer Controlled Vent and Pressurization System (CCVAPS). The CCLS calibrates and checks out the CCVAPS to verify proper system response to simulated and actual tank pressures.

Combined Systems Testing. The CCLS initializes and monitors the entire astriionics 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 a teletype. A block diagram of the CCLS computer functions is presented in Figure 4-13.

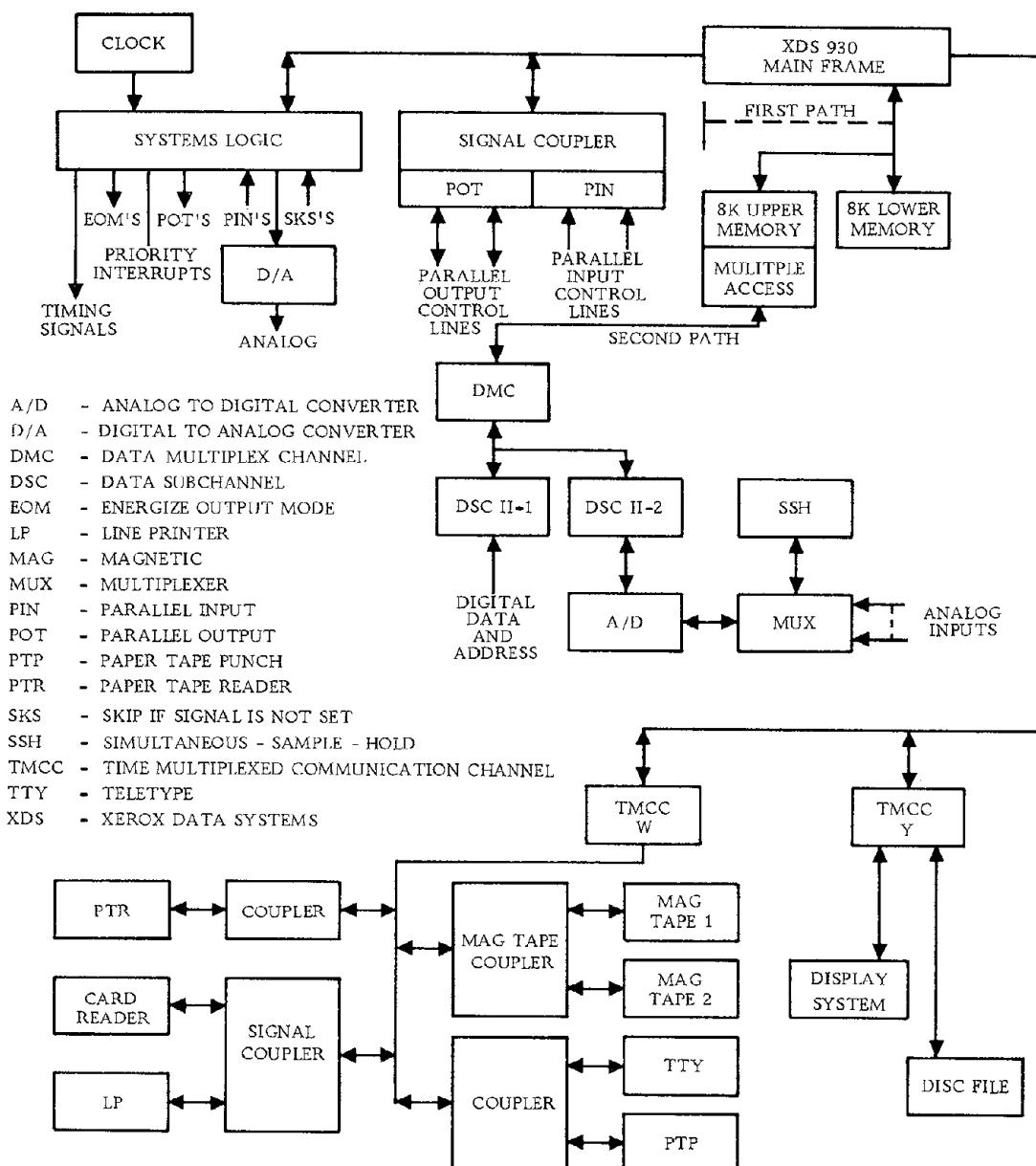


Figure 4-13. CCLS block diagram.

A Control Data Corporation CRT display (Figure 4-14) 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.

Other computer-related characteristics 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 .

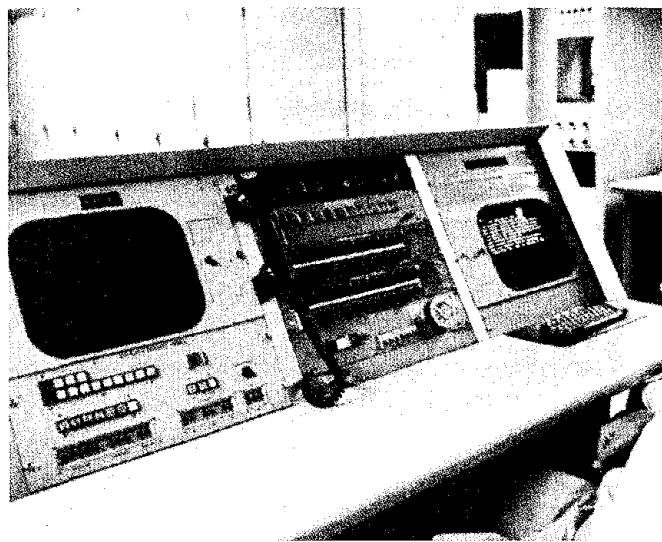


Figure 4-14. CCLS display.

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 routine: up to three different test programs can run at one time in time-share, 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:

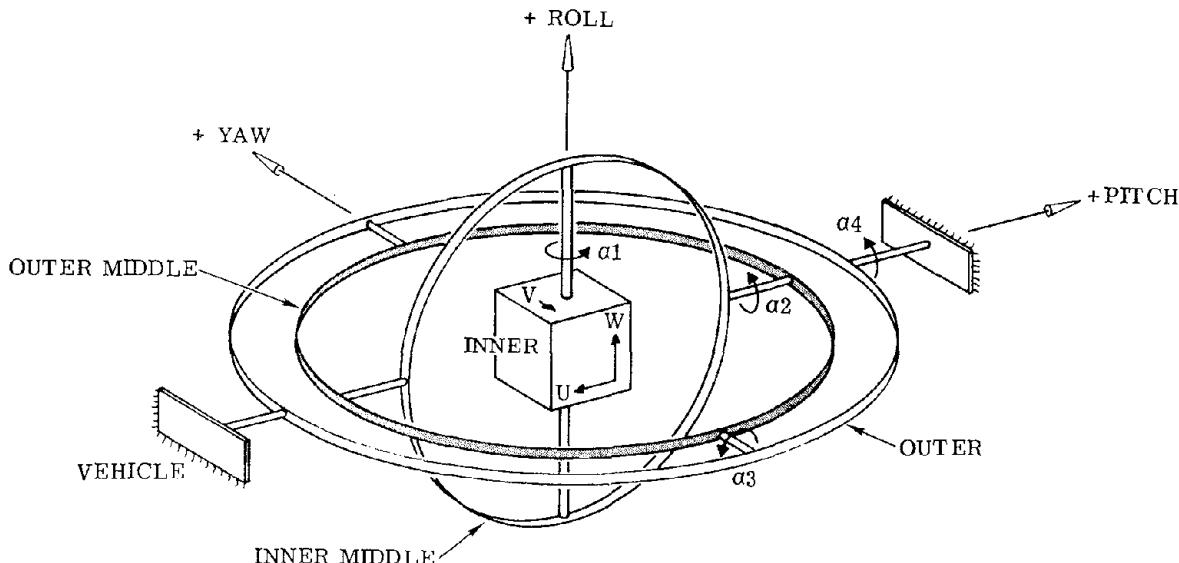
- Determine Vehicle Position and Velocity for Use by Guidance and Control

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 measurements 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 4-15). 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.



$$\alpha_1 = \alpha_2 = \alpha_3 = \alpha_4 = 0 \text{ AS SHOWN}$$

Figure 4-15. 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 4-16) 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.

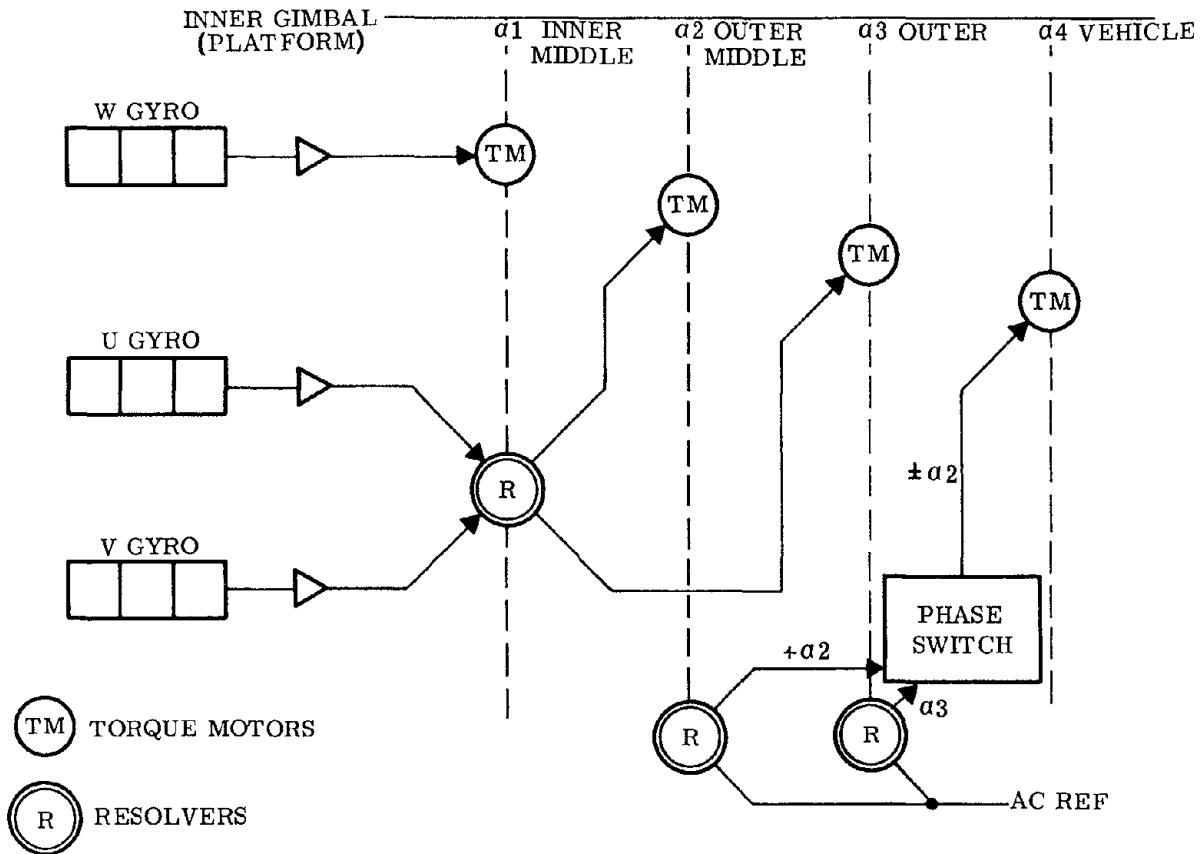


Figure 4-16. Platform stabilization mechanization.

Velocity. Three orthogonal pendulous 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 4-17). 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.

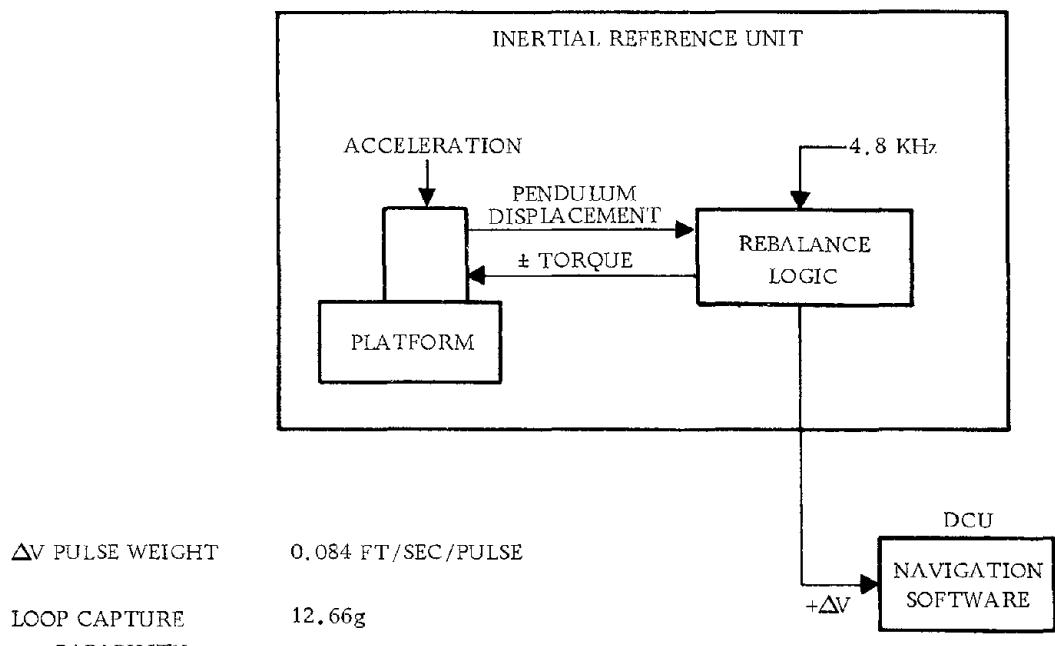


Figure 4-17. Velocity measurement.

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 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)
Pioneer G	1	MCR (m/sec)	112	40

4.4.3 COMPONENT DESCRIPTION

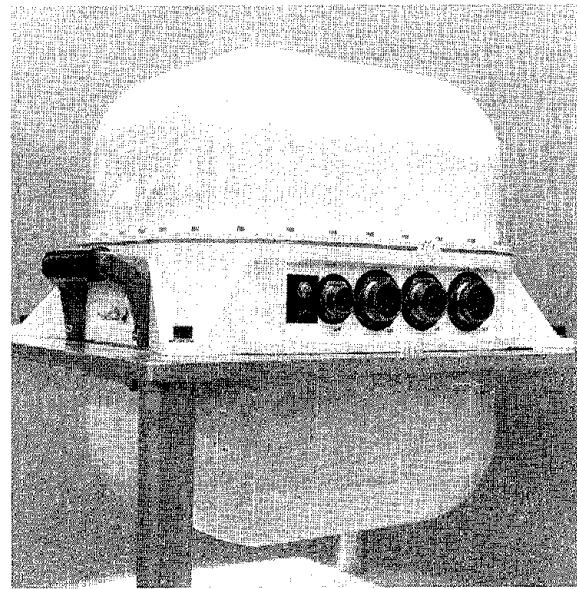
Digital Computer Unit (DCU)

(Refer to Subsection 4.1.)

Inertial Reference Unit (IRU). All power required by the IRU (Figure 4-18) 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 4-19) contains filters, power supplies, and mode control relays for the IRU, and supplies all power required by the IRU.

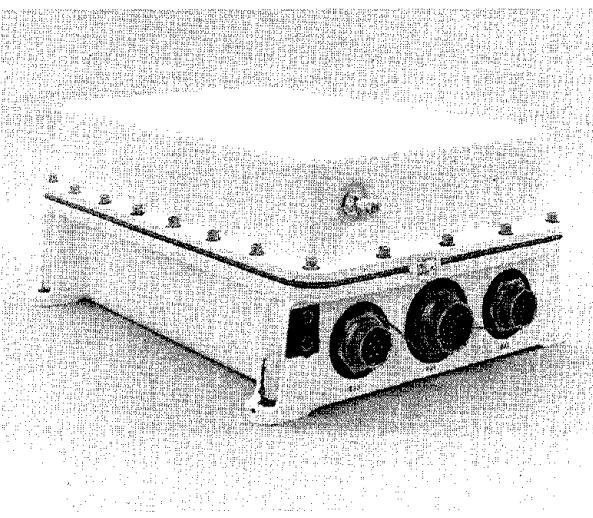
4.4.4 CHECKOUT, CALIBRATION, AND LAUNCH. Elements of the navigation function are prepared for flight by the computer controlled launch set (CCLS). Programs loaded into the CCLS ground computer, and by the CCLS into the DCU, perform checkout and calibration of the IRU, and orientation and final alignment 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.



Weight: 63 lb

Size: 18 1/2" x 18" x 13 3/4"

Figure 4-18. Inertial reference unit (IRU).



Weight: 25 lb

Size: 13 3/4" x 8 3/4" x 5 7/8"

Power: 185 watts (28v d-c)

Figure 4-19. Systems electronic unit (SEU).

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 4-20 is a block diagram of the guidance system.

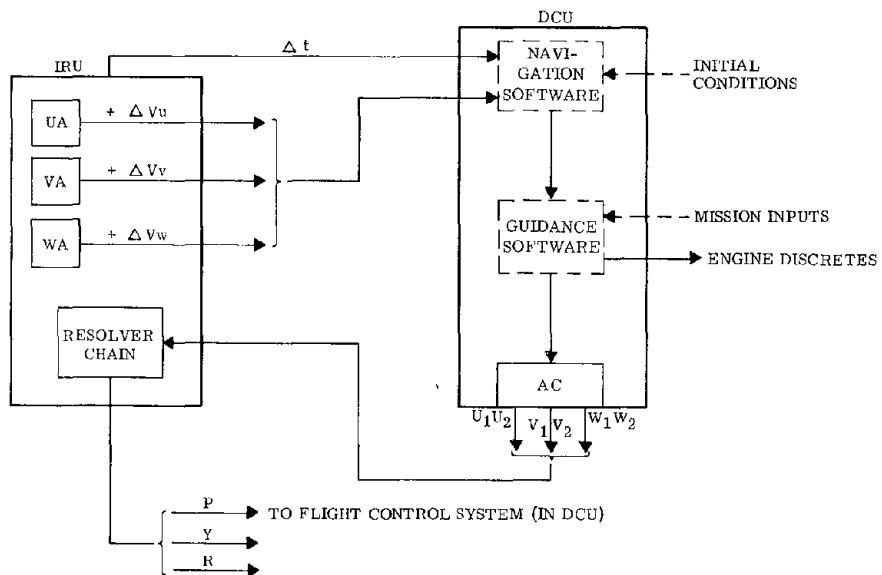


Figure 4-20. Guidance system block diagram.

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 5-33) 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 to digital commands for the coast phase control system.

4.6.1 SYSTEM OPERATION. A block diagram of the control system is presented in Figure 4-21.

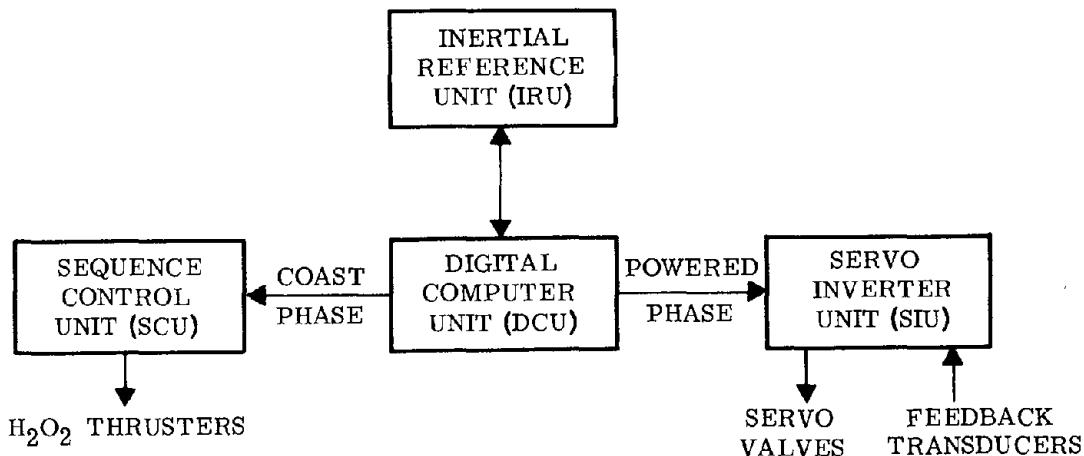


Figure 4-21. Control system.

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 servo valve. An electronic input from the SIU to the servo valve causes hydraulic fluid flow and engine movement. An engine position signal is provided to the SIU by a feedback transducer.

Each of the two Centaur engines has two actuators, one for pitch movement, and one for 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 motions in the pitch direction.

The computer pitch command drives two-position servoamplifiers which control the pitch servo valves 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 displacement errors 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 12 monopropellant (H_2O_2) thrusters mounted in an orthogonal arrangement at the perimeter of the Centaur thrust section. The thrusters 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 servo-amplifier. Each servoamplifier operates in conjunction with a servo valve, 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 4-22 is a block diagram of this configuration.

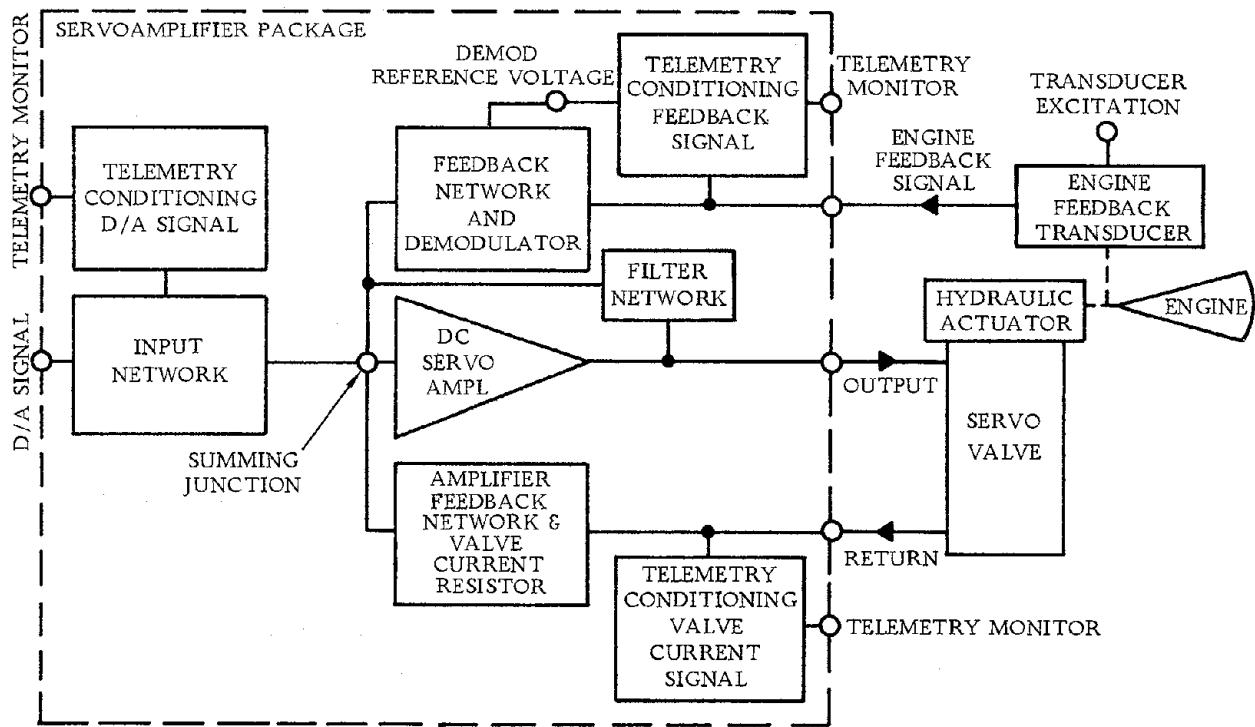


Figure 4-22. Centaur D-1T signal flow block diagram of a typical servo package channel.

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:	\pm 3.56 ma d-c (minimum linear range) \pm 6.0 ma d-c (typical linear range)
Command Gain:	2.68 ma d-c/v d-c \pm 5.0%
Static Gain:	0.737 deg/v d-c \pm 3.0%
Servoamplifier Gain (feedback):	1.78 ma/v a-c \pm 5% (assuming a transducer gain of 2.04 v/deg)
Null Offset:	\pm 0.028 deg maximum (engine position) \pm 0.013 deg typical (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 4-23.

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

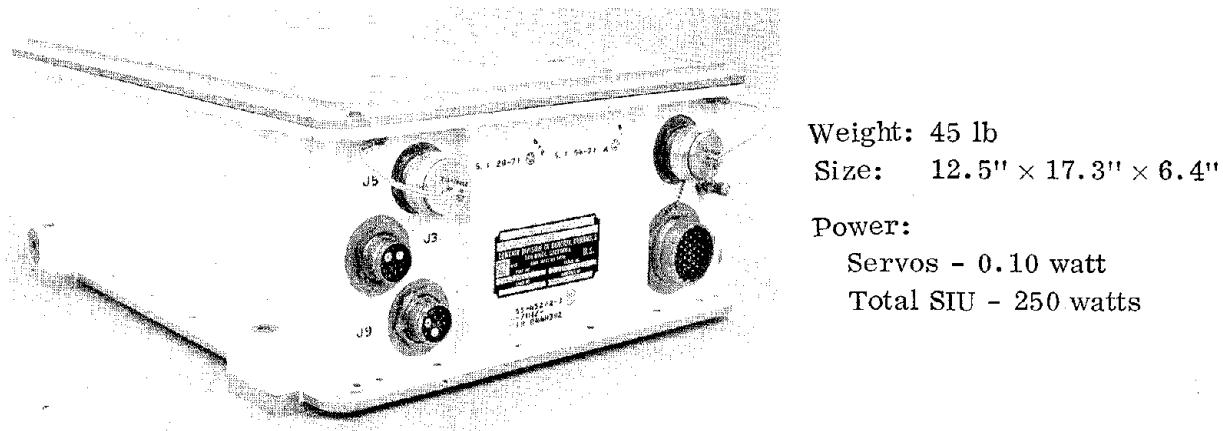


Figure 4-23. Servo inverter unit (SIU).

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. Units are exposed to normal flight environments (temperature and vibration) during tests. 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 sequence control unit (SCU) is the interface between the DCU output registers and the vehicle systems that require switched and/or timed commands.

ELEMENT:

- Sequence Control Unit (SCU)

FUNCTIONS:

- Converts and Decodes DCU Outputs into Commands Usable by Vehicle System
- Power Distribution

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). SCU 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 4-24. The operation of one switch (Switch 1 in Group 1) is shown in Figure 4-25.

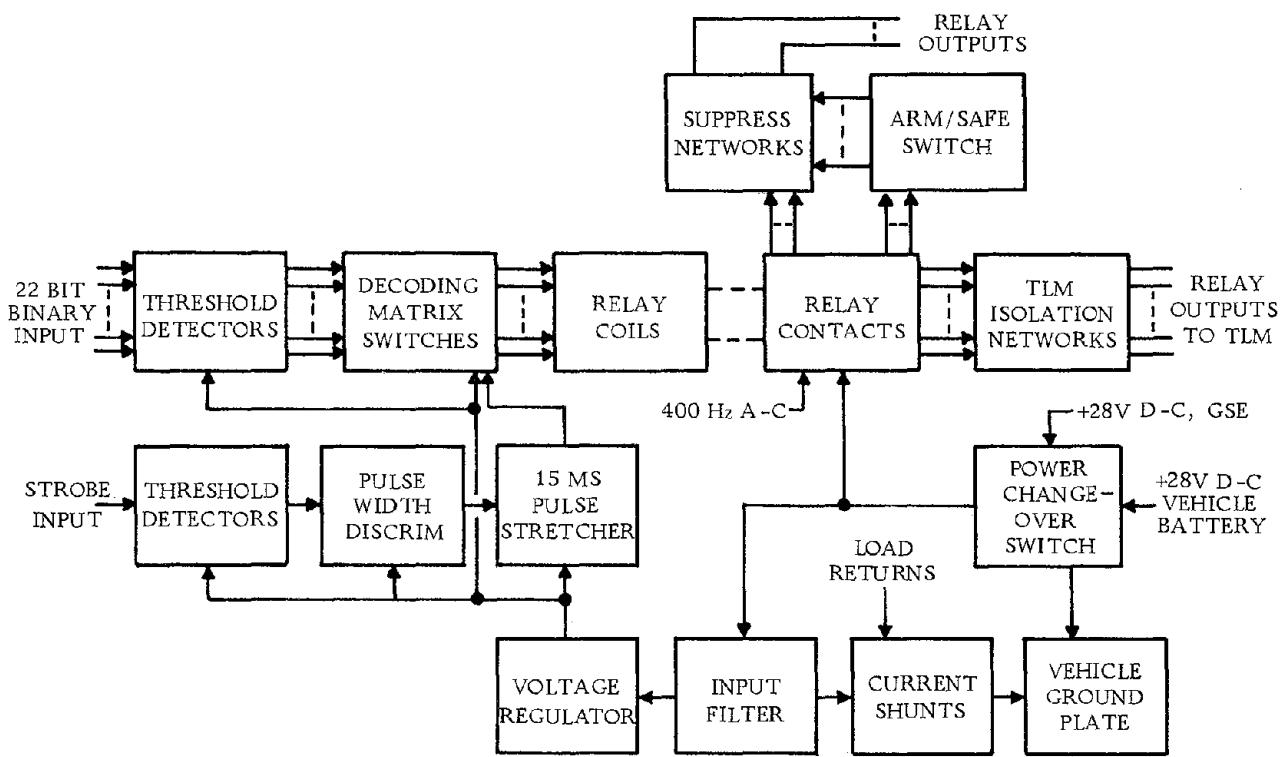


Figure 4-24. Sequence control unit block diagram.

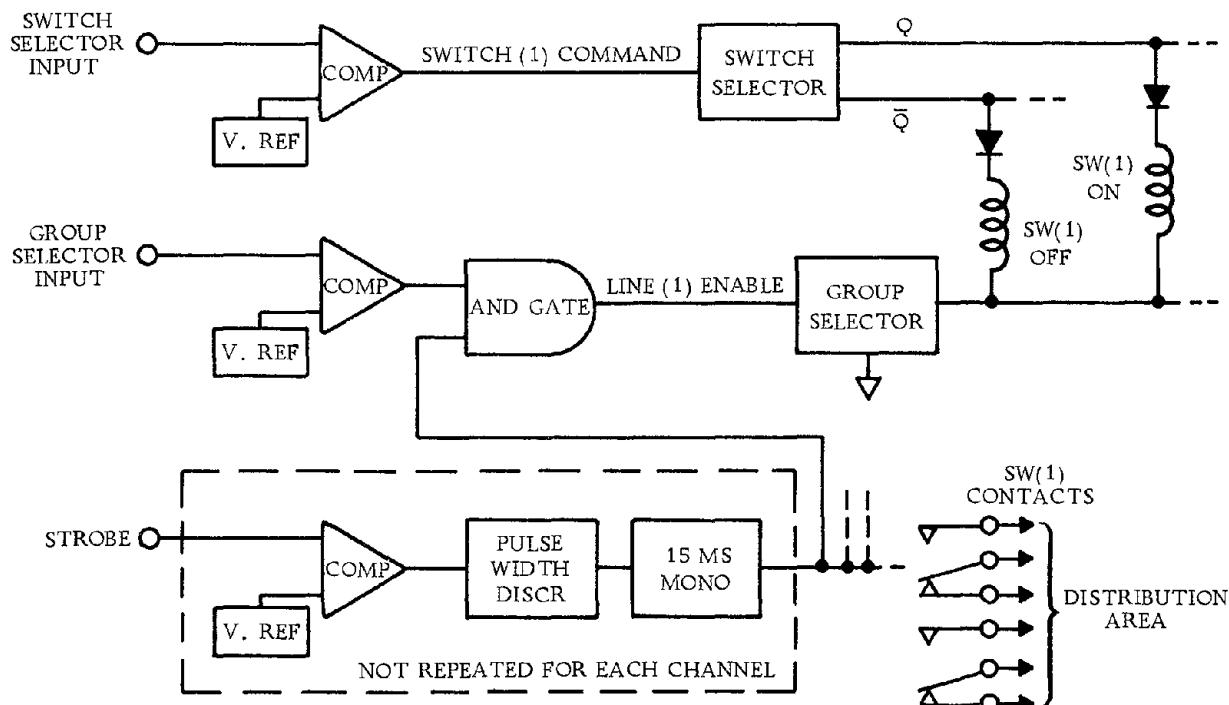
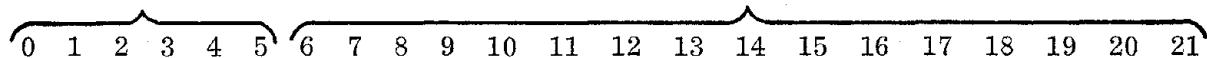


Figure 4-25. Functional channel description power matrix.

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. The relays 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.

GROUP SELECTOR

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
Overtoltage Protection:	-1.5v d-c to +10.0v d-c
Noise Rejection:	no state change for 1.0μ 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 6.0μ s recurring at 8.0K Hz maximum (any waveform)
Must Accept	4.5v d-c minimum for 12.0μ s minimum

Overtoltage 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. (Accomplished with isolated switch contacts.)
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. Non-flight-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 4-26.

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.

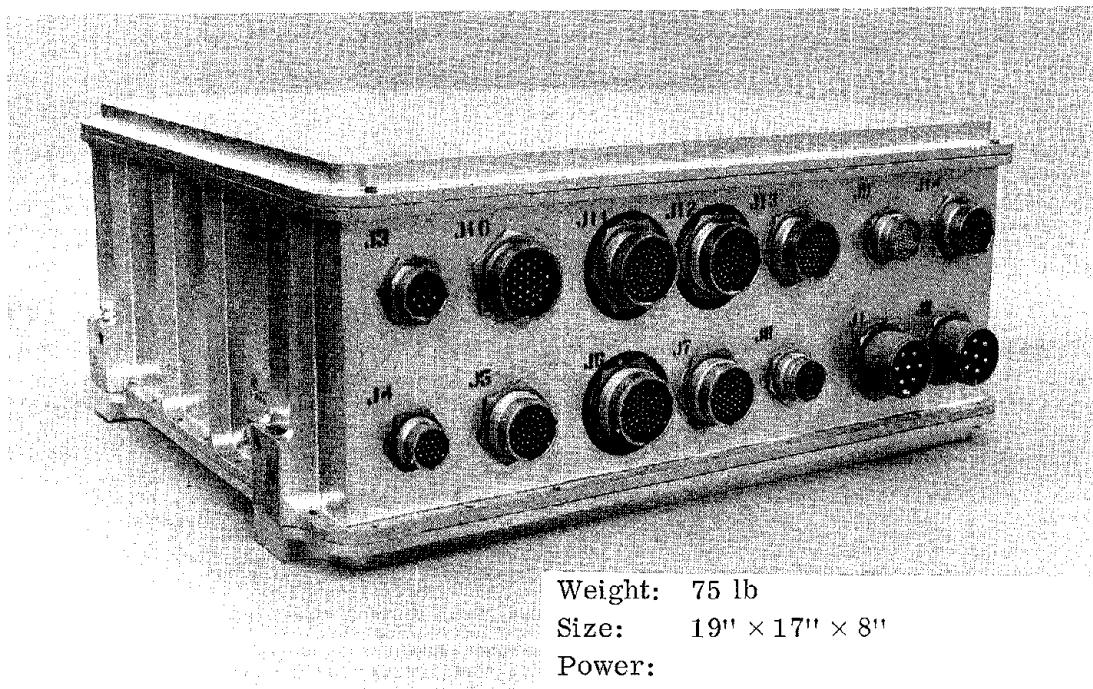


Figure 4-26. Sequence control unit (SCU).

4.8 PROPELLANT UTILIZATION

ELEMENTS:

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

FUNCTIONS:

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

To realize optimum performance in a liquid-fueled bipropellant space vehicle, it is advantageous 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 inaccuracies in calibration of engine mixture ratios, flow rates, and total thrust under flight conditions. This results in an ability 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 4-27) for proper propellant utilization (PU) is to accurately measure the ratio of propellants in the vehicle tanks during the 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 air-dielectric 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 the probe is excited from an a-c source, an output current or voltage proportional to the empty (dry) capacitance plus the increase in capacitance (Δ) due to propellant mass is obtained.

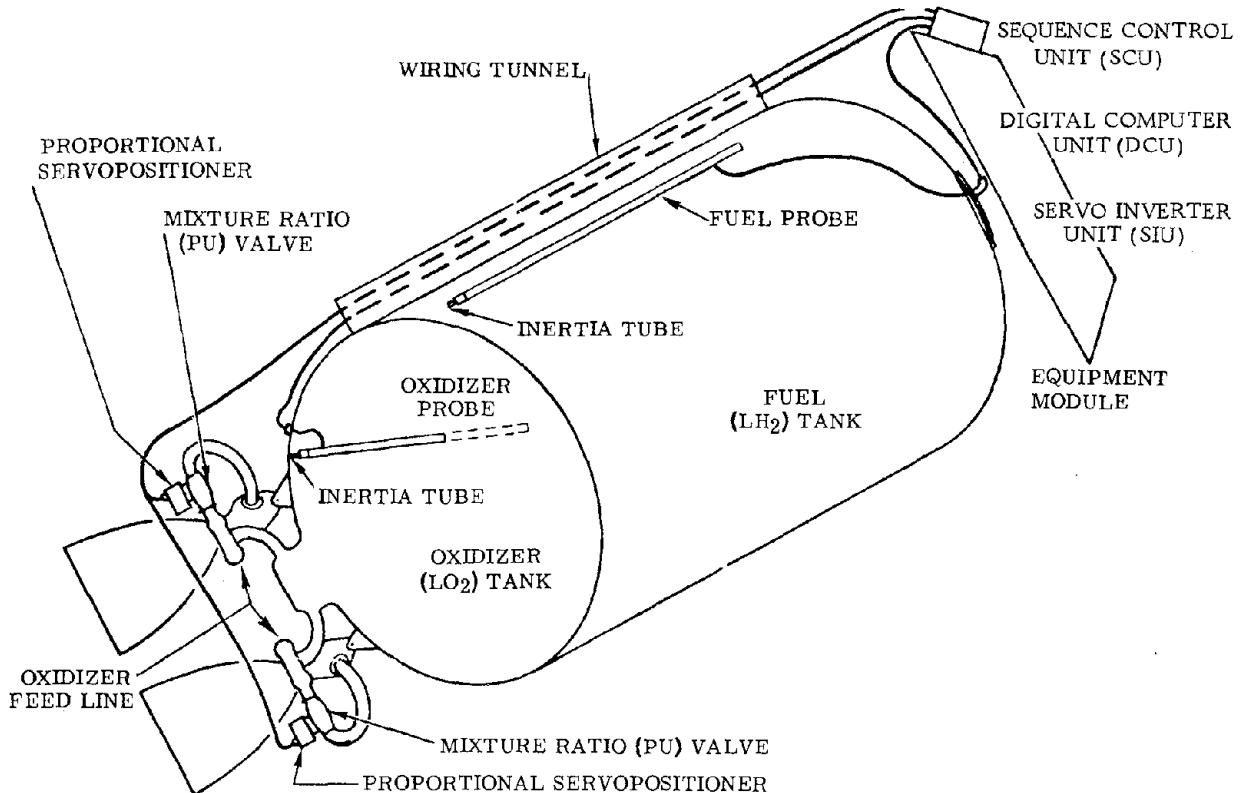


Figure 4-27. Locations of propellant utilization system components.

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 LO₂ and LH₂ equal and in the same ratio as the "full" currents. The system excitation voltages, and therefore the error detector amplifier input currently nominally provides 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 empty 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 a tank ratio change is necessary, the DCU commands a change in the positions of the PU servopositioners on the engines by operating SCU switches through the normal DCU-SCU interface. The servopositioners 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 new servopositioner angles computed five times a second and the position feedback loop is monitored 25 times a second.

Amplifiers similar to the error detector amplifier provide independent instrumentation outputs that are indicators of the total amount of LO₂ or LH₂ in each tank. A block diagram of the PU system is shown in Figure 4-28.

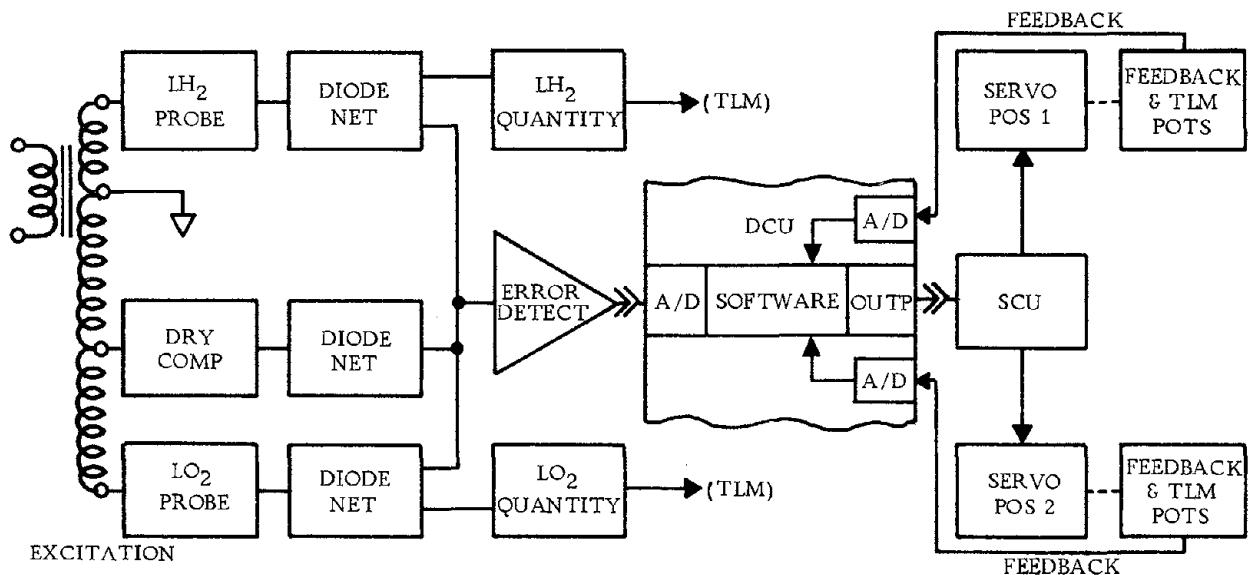


Figure 4-28. Propellant utilization system.

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

- a. Detection of propellant masses.
- b. Calculation of mass ratio errors.
- c. Propellant flow control.
- d. Failure detection and compensation.

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:

$$\begin{array}{ll} \text{LH}_2 \text{ (dry)} = 503 \text{ pf} & \text{LO}_2 \text{ (dry)} = 170 \text{ pf} \\ \text{LH}_2 (\Delta C) = 115 \text{ pf} & \text{LO}_2 (\Delta C) = 84 \text{ pf} \end{array}$$

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 limit system accuracy to ± 32 pounds of LH₂ for a typical two-burn mission and ± 29 pounds of LH₂ for a one-burn mission.

Propellant Flow Control. The engine LO₂/LH₂ mass ratio is controlled within a ± 11.5 percent of nominal. To correct mixture ratio errors, PU valves in the engine LO₂ feed lines are controlled to change the rate of LO₂ usage. The engine performance requirements therefore limit the maximum tank mass error which can be corrected. If the PU valves were commanded to one limit for the entire closed-loop portion of the flight, the resulting correction in mass error would be approximately 2100 pounds of LO₂. When considering other errors and dispersions inherent in the system, the guaranteed total correction capability is approximately 1400 pounds of LO₂.

Failure Detection and Compensation. Throughout flight, the DCU checks the reasonableness of the three inputs it receives and determines whether or not particular failure modes have occurred. The specific failure modes detected and resulting compensating action taken are:

- a. Excessively large PU error signal — PU system is operated open-loop by driving servopositioners to fixed angles.
- b. Servopositioner stuck at LH₂ rich limit — the other servopositioner is driven to the LO₂ rich limit, only during the first 110 seconds of the Centaur burn.

- c. Servopositioner feedback failure at LO₂ rich limit — the failed unit is operated open-loop to position the PU valve near its nominal angle, while the good unit continues closed-loop control.

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 electronics 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. Calibration constants are determined at this time.

During tanking for launch, gross system operation is verified by monitoring error signal and LO₂ and LH₂ 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-1T real-time position and velocity. The system is compatible with the AFETR ground system and provides data to support the range safety requirements. Figure 4-29 is a schematic diagram of the C-band tracking system.

ELEMENTS:

- Transponder
- Power Divider
- Coaxial Switches
- Antennas

FUNCTIONS:

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

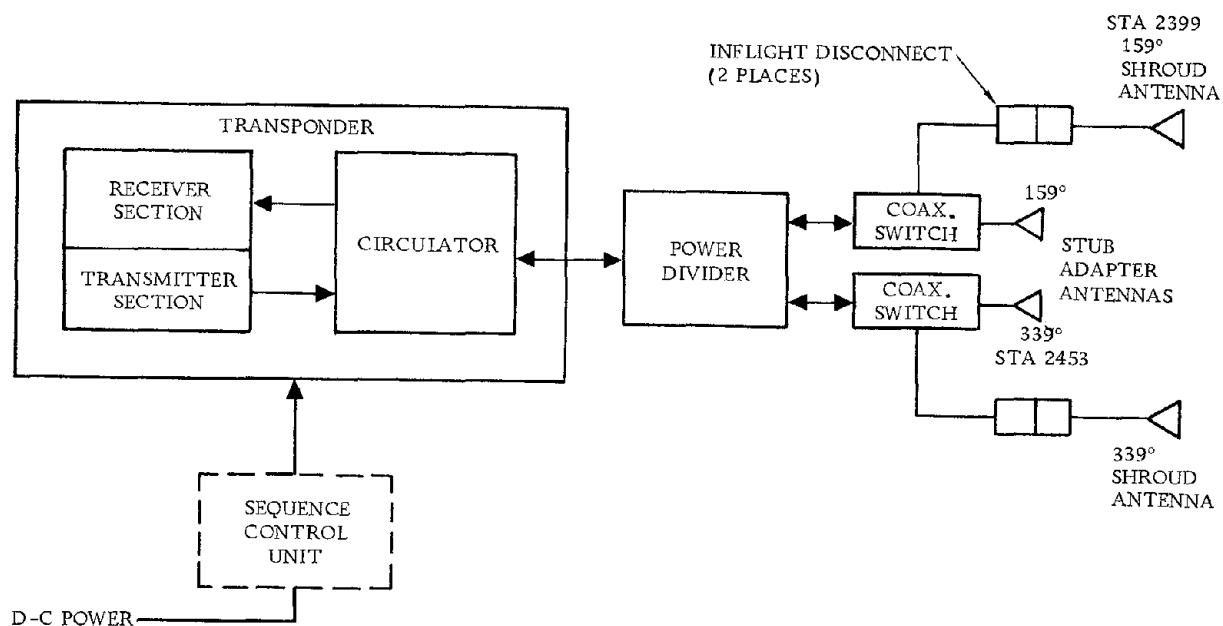


Figure 4-29. C-band tracking system block diagram.

4.9.1 SYSTEM OPERATION. The C-band tracking system ground transmitters transmit (through 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. Two antennas (Figure 4-30) are mounted on the Centaur standard shroud and operate until shroud separation. At shroud separation the two antennas mounted on the stub adapter are selected by coaxial switches.

The signal is conducted from the antenna through RF cables, and coaxial switches (Figure 4-31) to the power divider (Figure 4-32), a tee device that connects the signals together and provides interconnection to the transponder (Figure 4-33). 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.

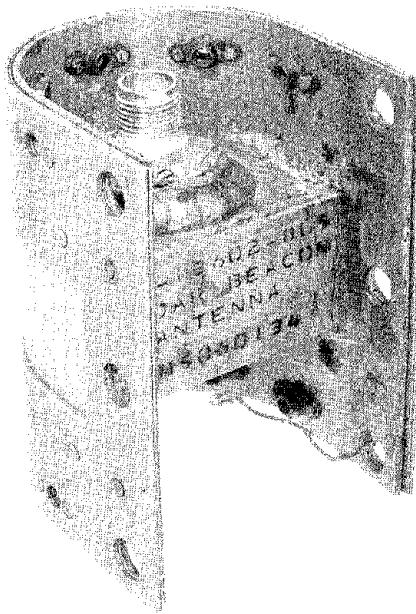
All radar ground stations within range of the vehicle can interrogate the vehicle transponder continuously using a synchronized rapid sequential time-sharing technique. The round-trip transit time of the pulsed signal (compensated for beacon delay) is measured and used by the radar computer to determine 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.

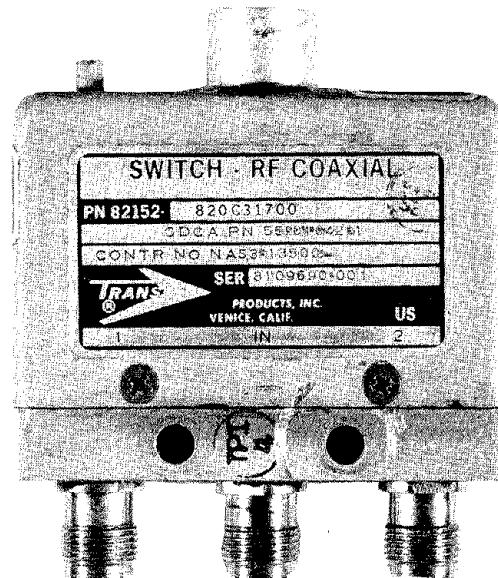
4.9.3 COMPONENT DESCRIPTION. The C-band tracking system transponder, power divider, antennas, and coaxial switch are shown in Figures 4-30 through 4-33. Weight and overall dimensions of each component are included in the figures. The coaxial switch is also used in the telemetry system.

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.



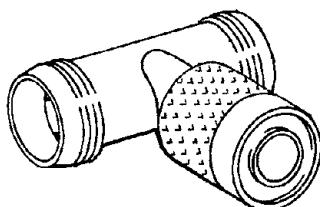
Weight: 0.2 lb (each)
Size: 3.0" x 2.2" x 2.1"

Figure 4-30. C-band antenna.



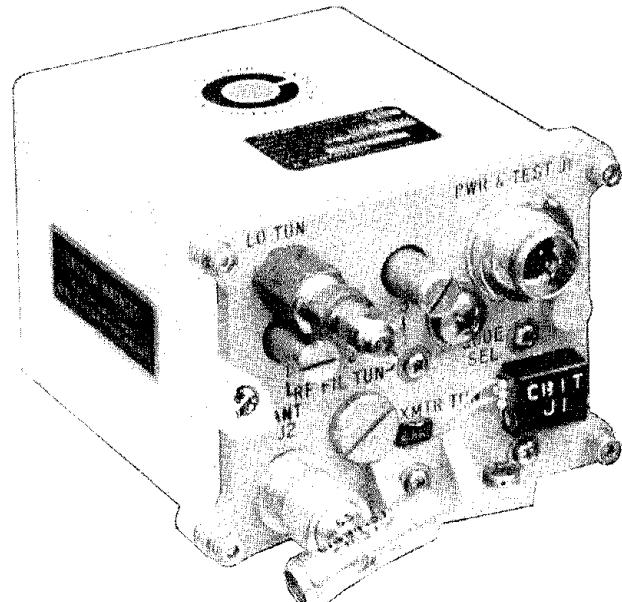
Weight: 0.6 lb
Size: 2.7" x 2.8" x 0.6 lb

Figure 4-31. Coaxial switch.



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

Figure 4-32. C-band power divider.



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

Figure 4-33. C-band transponder.

4.10 RANGE SAFETY COMMAND SYSTEM

The range safety command system terminates the flight of the Centaur D-1T 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
- RF Circulators
- 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-1T Range Safety Command System are as follows:

- a. Cut off the Centaur D-1T main engines in response to an RF command (MECO) thus imposing a condition of zero thrust.
- b. Destroy the LH₂ and LO₂ 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. Before Centaur shroud jettison, the RF carrier from the ground (the same RF carrier is used by the Titan stage) is received by one or both of the antennas that are mounted on opposite sides of the shroud. After shroud jettison, the RF carrier is received by one or both of the antennas that are mounted on opposite sides of the Centaur stub adapter. A block diagram of the RSC system is shown in Figure 4-34.

The signal is conducted from the antennas through RF cables to input ports on two RF circulators and from output ports of the circulators to input ports on a hybrid junction. The two output ports of the hybrid junction are connected to two RSC receivers by RF cables.

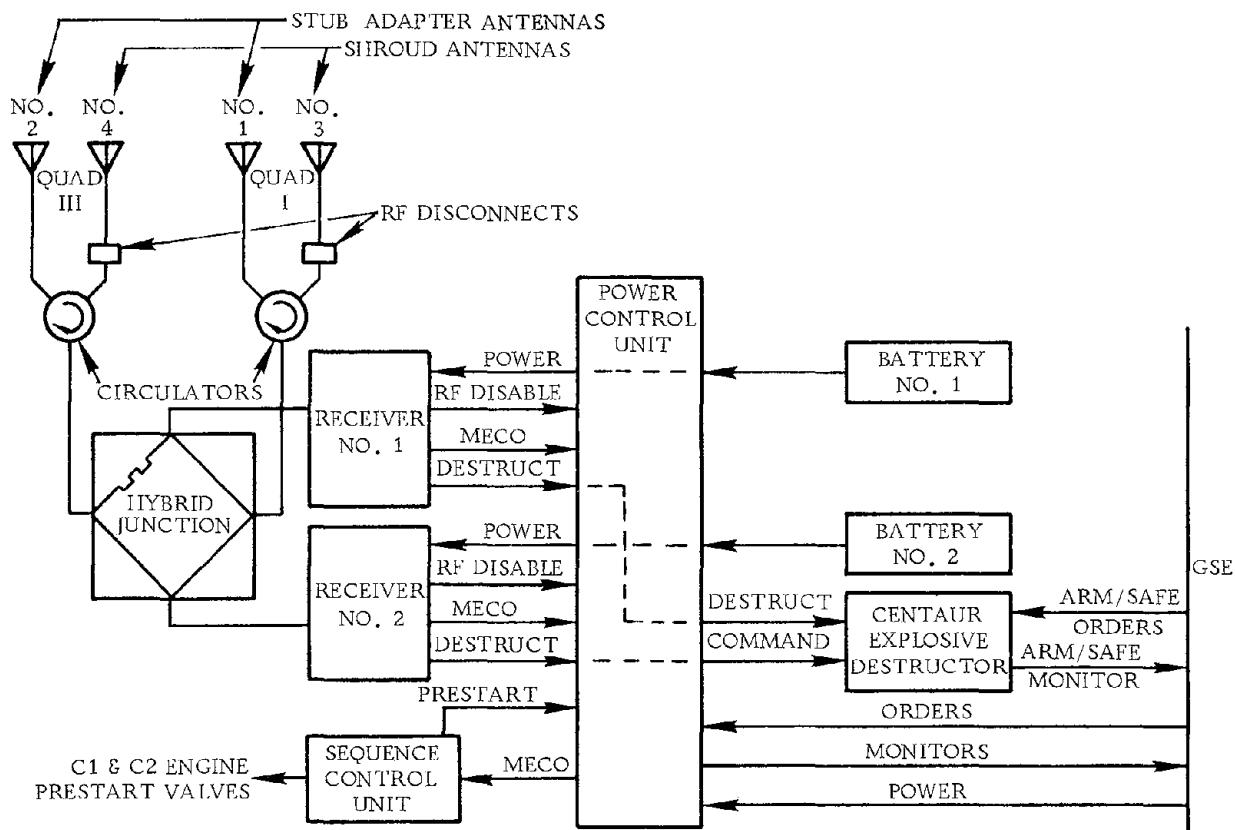


Figure 4-34. Centaur range safety command systems block diagram.

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. The destruct charge may be conical shaped type supplied by Convair Aerospace or it may be furnished by the payload contractor.

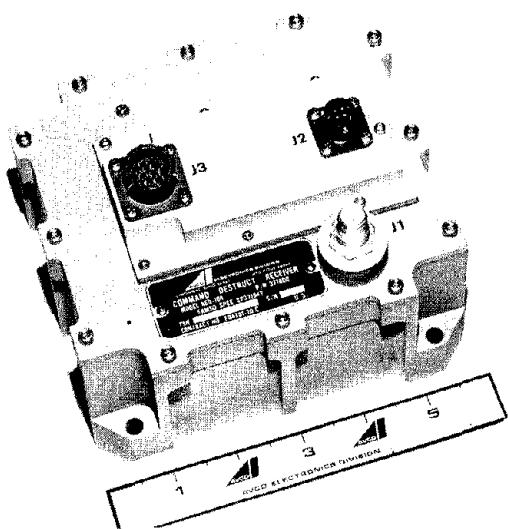
4.10.3 COMPONENT DESCRIPTION. The range safety command system is made up of four antennas, two RF circulators, a hybrid junction, two command receivers, a power control unit, a destructor, and two batteries.

Command Receiver (2). The command receiver receives the modulated RF carrier, demodulates the command tones, and converts them into 28v d-c commands for main engine cutoff (MECO), destruct, or RF disable functions. The command receiver is shown in Figure 4-35.

Power Control Unit. The power control unit performs the following functions:

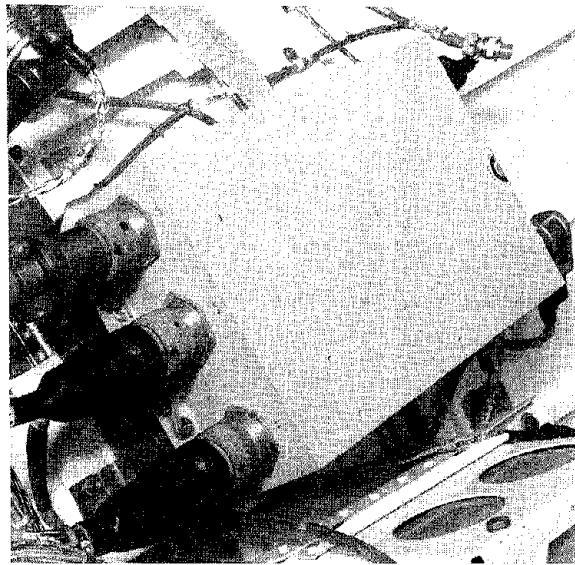
- a. Provides individual internal/external control of the receiver 28v d-c power.
- b. Conveys system commands to their required destinations by relay logic.
- c. Interconnects RSC system units.
- d. Provides a common source of telemetry and landline instrumentation (except for the Centaur destructor arm/safe monitors).

The power control unit is shown in Figure 4-36.



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

Figure 4-35. Command receiver.



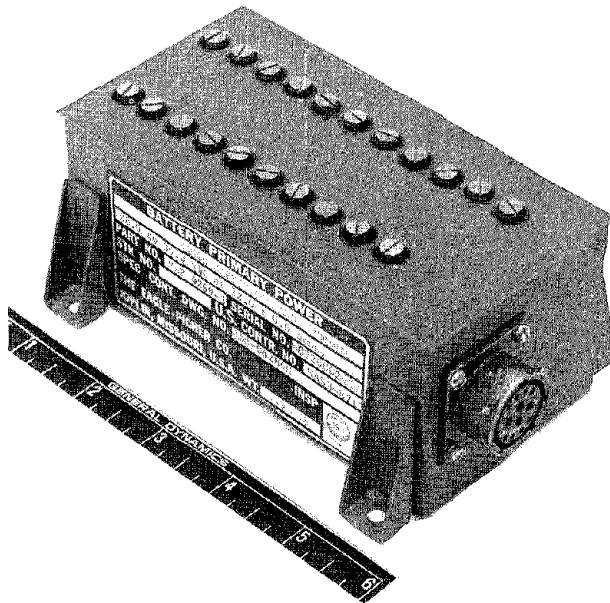
Weight: 9.8 lb
Size: 8.3" x 12.3" x 3.5"

Figure 4-36. Power control unit.

Batteries (2). The RSC batteries furnish power for the Centaur RSC system. Two identical batteries are provided, one for each redundant half of the RSC system. Each redundant half of the power supply system is isolated from the other so that failure of either battery or its associated components or circuitry will not affect the other. The battery is shown in Figure 4-37.

Hybrid Junction. The hybrid junction electrically connects the antenna system outputs from the circulators to the receivers. This arrangement provides isotropic radiation coverage about the vehicle and provides isolation to ensure that failure of either receiver or either antenna of the operating pair will not result in total system failure. The hybrid junction is shown in Figure 4-38.

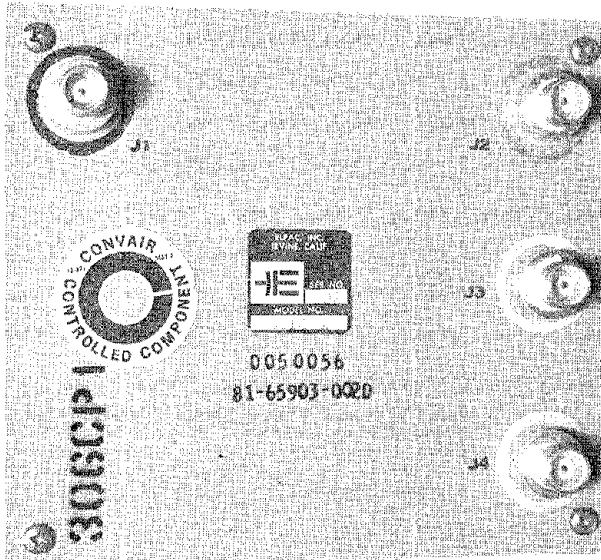
RF Circulator (2). Two identical three-port circulators, mounted on the equipment module, channel the RF energy from either pair of antennas to the hybrid junction. The circulators are passive RF



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

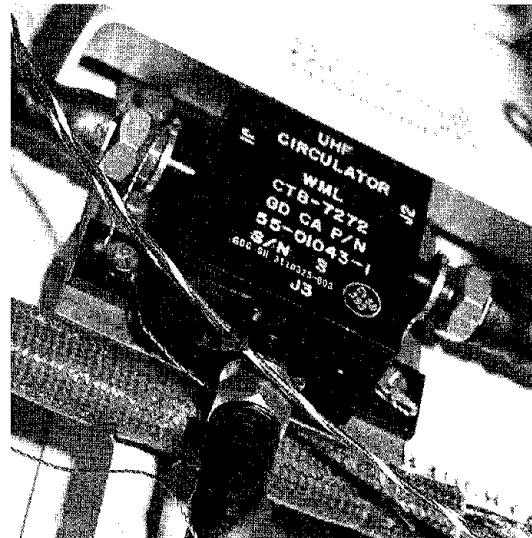
Figure 4-37. Battery.

devices, designed so that RF energy entering into one port flows in one direction, and exits from the first properly terminated port. The circulator is shown in Figure 4-39.



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

Figure 4-38. Hybrid junction.



Weight: 0.6 lb
Size: 3.3" × 2.7" × 1.3"

Figure 4-39. RF circulator.

Antennas (4). The Centaur vehicle employs four identical stub-type RSC antennas. Two units are mounted on the Centaur standard shroud opposite each other at Station 2399 (Figure 4-40). The other two antennas are located on opposite sides of the stub adapter at Station 2447 (Figure 4-41).

Destructor. The explosive destructor accepts the destruct signal, switches it to the detonator (when in the armed configuration), initiates an explosive train, and ruptures the Centaur LO₂ and LH₂ tanks to cause fuel dispersion. The destructor is shown in Figure 4-42.

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.

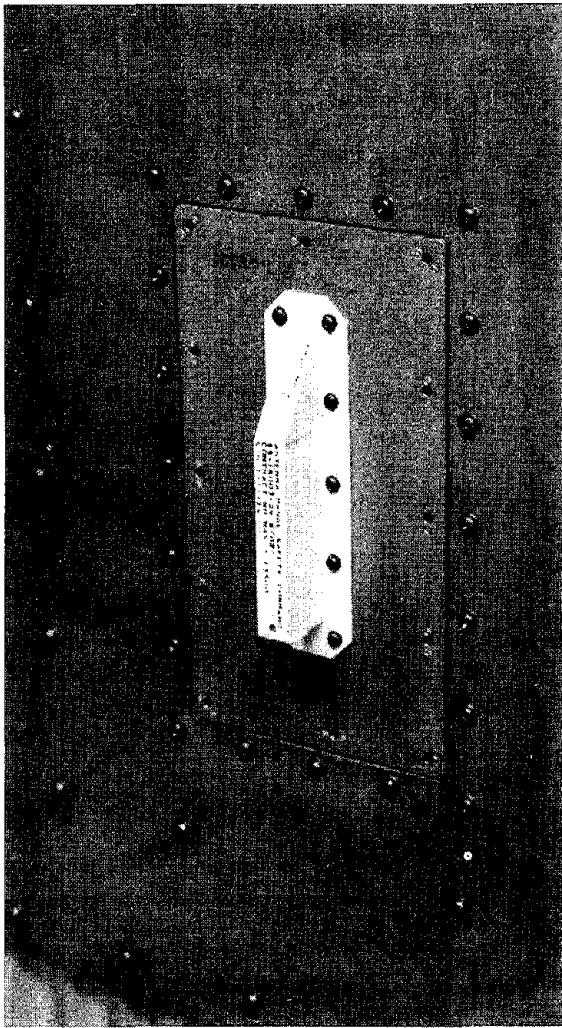


Figure 4-40. RSC antenna mounted on the shroud.

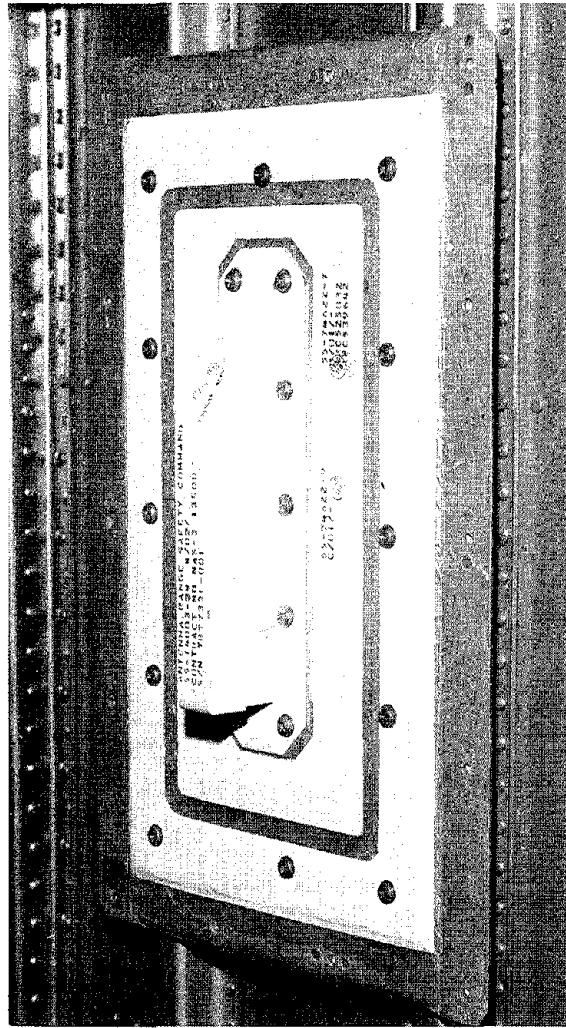


Figure 4-41. RSC antenna mounted on the stub adapter.

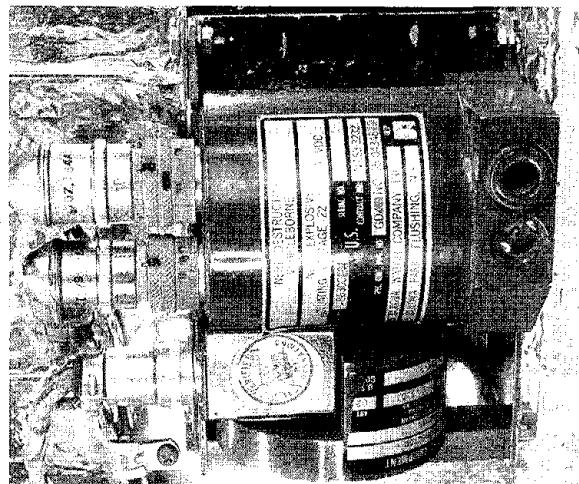


Figure 4-42. Destructor.

4.11 ELECTRICAL POWER SYSTEM

The Centaur D-1T vehicle electrical power system for one- and two-burn missions 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 change-over switch distributes power from a ground power supply prior to flight.

A three-battery system is also available for three-burn, extended-coast missions. The three batteries are arranged to feed one bus each, the batteries are interconnected via an isolation diode assembly to allow for load sharing between batteries but to eliminate the possibility of large circulating currents due to initial battery variations.

The system description which follows is for single battery operation but is generally applicable to the three-battery configuration also.

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 uses the main vehicle battery/batteries as indicated in Figure 4-43A (single battery) and Figure 4-38B (three-battery configuration) to supply d-c power to the vehicle electronic systems. The batteries supply power to three separate buses in order to isolate equipments which tend to be generators of electromagnetic interference from equipment which may be sensitive to electromagnetic interference.

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.

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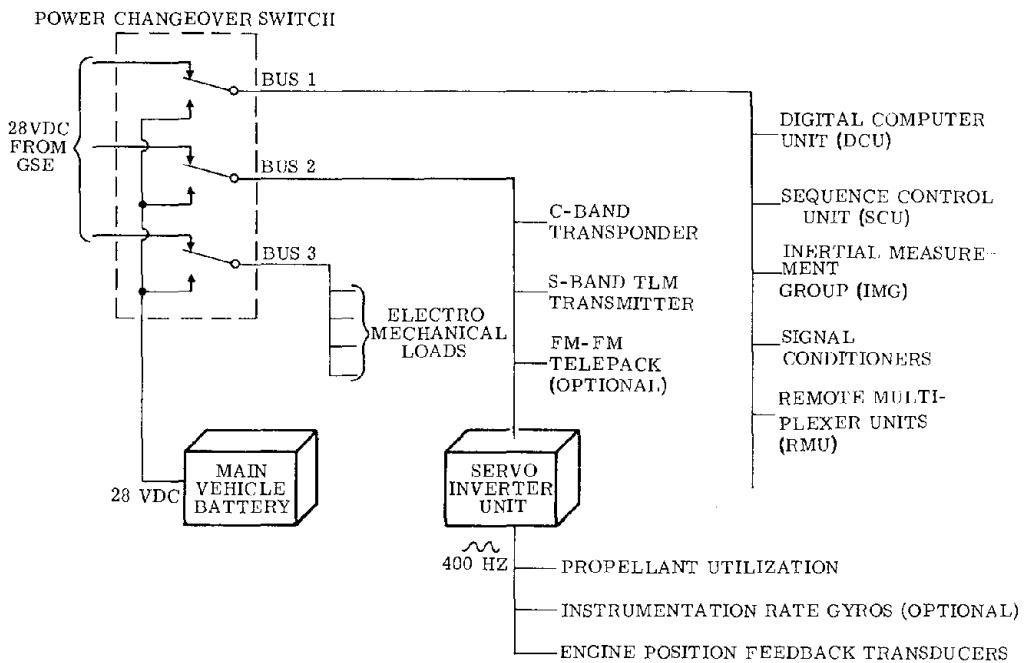


Figure 4-43A. Electrical power system (single-battery configuration).

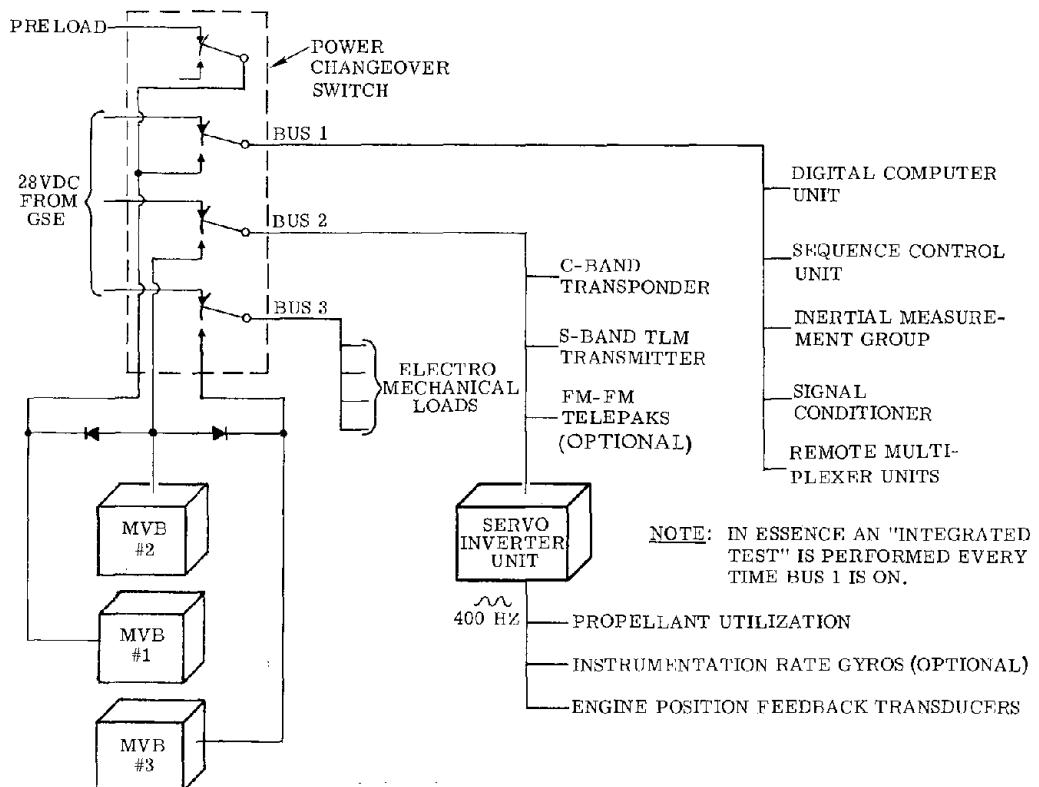


Figure 4-43B. Electrical power system (three-battery configuration).

Pyrotechnics are fired directly off the battery (battery No. 3 of the three-battery configuration) in order to get maximum isolation between systems.

Only those systems not critical for vehicle flight, such as the C-band transmitter and FM/FM telemetry, are fused.

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 available in either 100-ampere-hour or 150-ampere-hour capacity to suit mission requirements. Battery voltage is 26 to 30v d-c with loads from 50 to 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.

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.

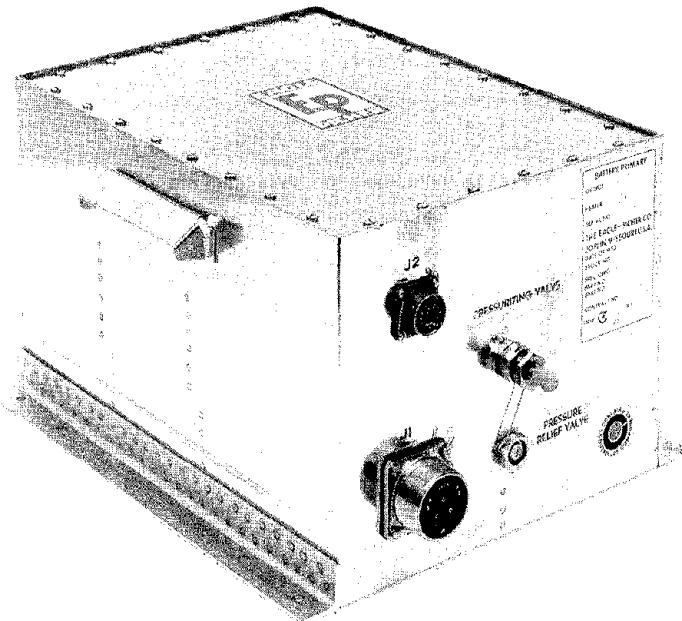
4.11.3 COMPONENT DESCRIPTIONS

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 4-44.

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 electroless 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.

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.



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

Figure 4-44. Battery.

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-1T astrionics system. The software is an integral part of the DCU. The software design and operational characteristics are described in this section.

SUBSECTION	SYSTEM	Page
5.1	Introduction	5-1
5.2	Software Objectives	5-3
5.3	Software Design Concepts	5-3
5.4	Executive System	5-7
5.5	Functional Tasks	5-11
5.6	Software Capability	5-26
5.7	Preflight System	5-28
5.8	Flight Program Validation	5-30
5.9	Software Management	5-35

5.1 INTRODUCTION

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

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ELEMENTS:

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

FUNCTION:

- Provide Command and Control for Many Vehicle Systems

The Centaur D-1T software incorporates many functions done by hardware on Centaur D. The Centaur D-1T 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.

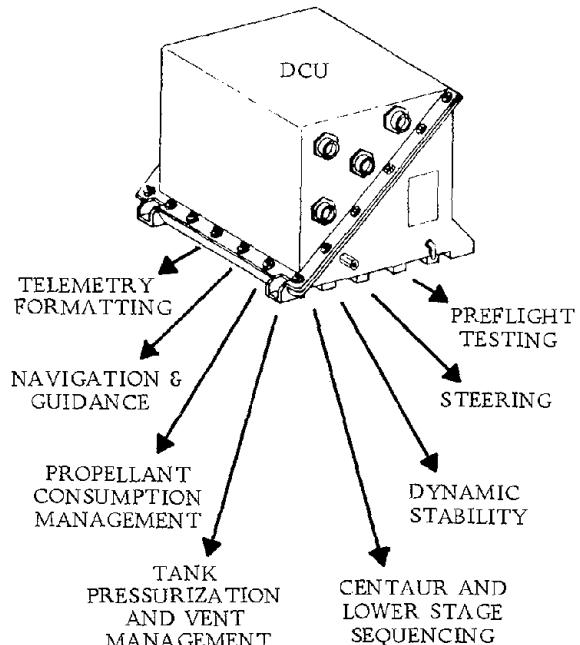


Figure 5-1. Software provides total vehicle command and control.

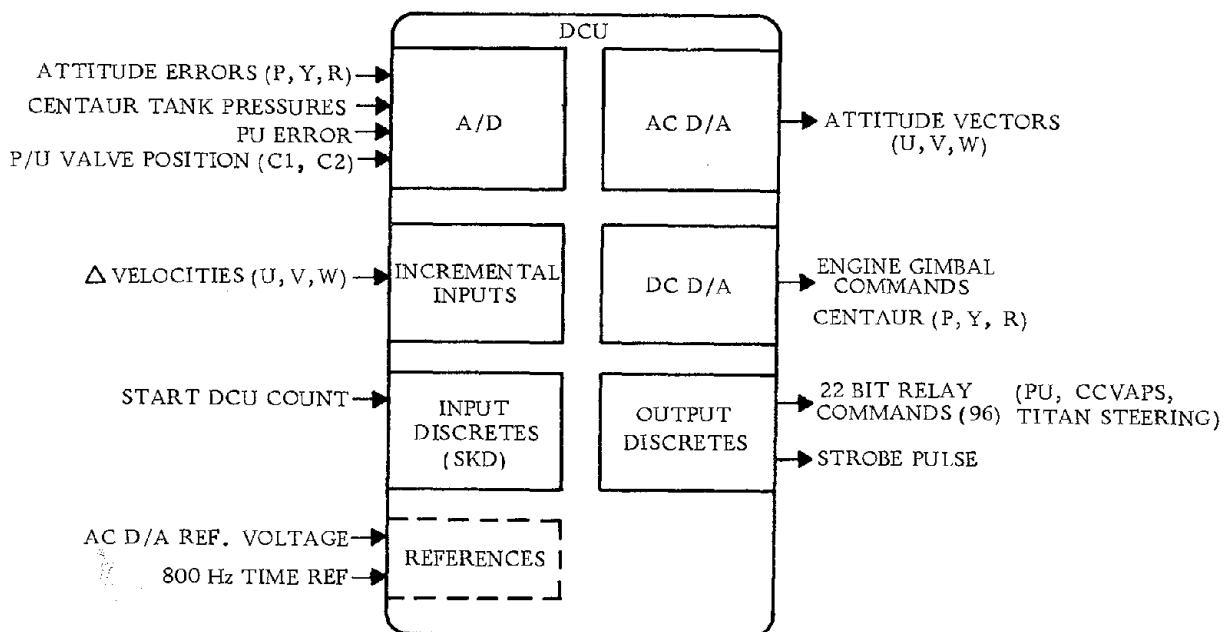


Figure 5-2. DCU input/output to serve D-1 systems.

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 missions, 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.

Independence from Interrupts. 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 any combination of two or more modules can be in a simultaneous state of interrupt; furthermore, the interrupt is allowed at any 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.

Decentralized, Parallel, Design and Checkout. 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.

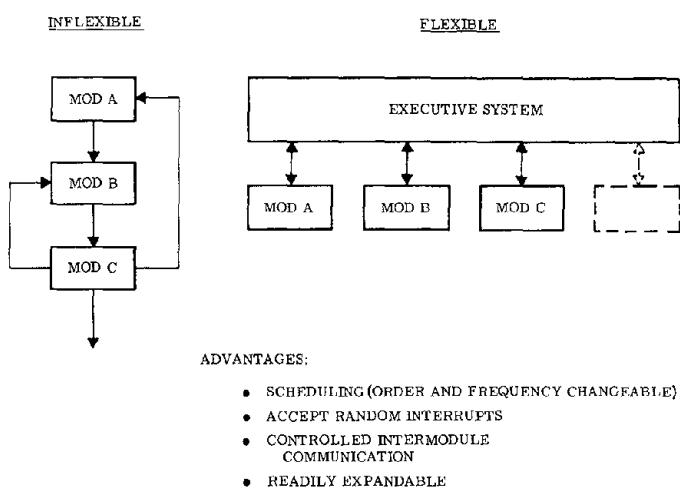


Figure 5-3. A flexible kind of modularity.

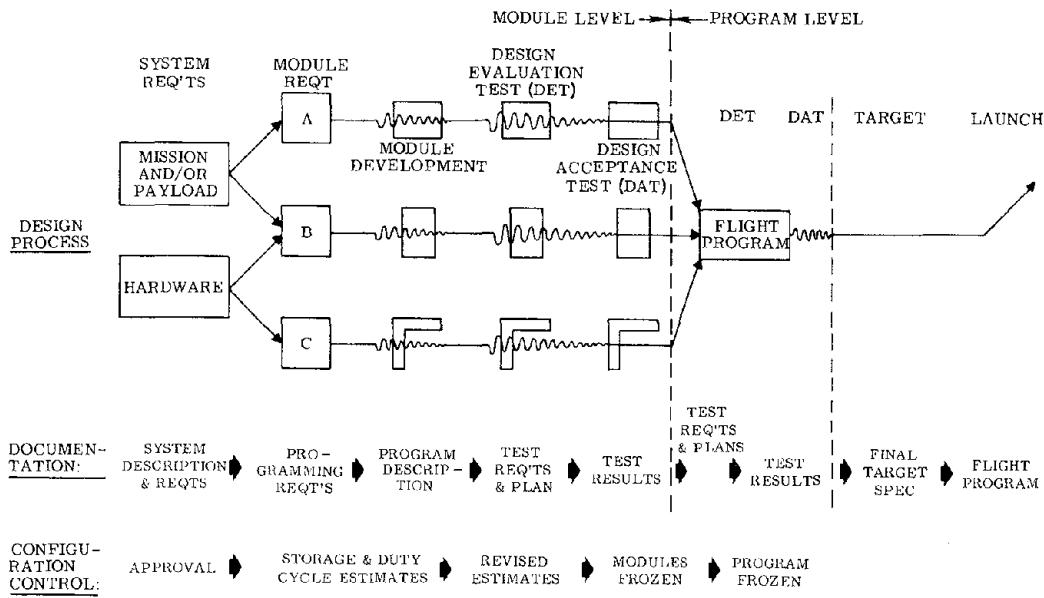


Figure 5-4. Software development — parallel module design and checkout.

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

- Attempts to Restart Engines if 1st MES Fails
- Accommodates Any One H₂O₂ Engine Out
- Forgives Transient Accelerometer Failure
- Advances Flight Phase for Early Engine Shutdown if Feasible
- 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 5-5 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 are resumed at the point of interruption as soon as higher priority tasks are completed.

<u>Entry From</u>	<u>Action</u>	<u>Result</u>
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.

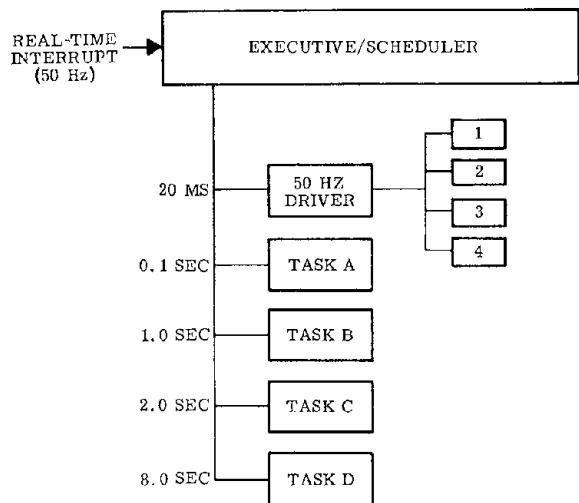
Task Organization. A "tree" structure is used to organize the tasks (Figure 5-6). The tasks are grouped according to execution priority requirements. Each task on the "tree" must appear as an entry in the task table (except for the 50-Hz tasks).

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.

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.



- 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.

Table 5-1. Adaptation at real time.

Real-Time Reference:	50-Hz Clock Interrupt
Engine Cutoff:	Precise 800-Hz Clock Countdown Register
Task Programs:	<ul style="list-style-type: none"> { Ininsensitive To Being Interrupted Ininsensitive To Time Changes
Task Scheduler:	<ul style="list-style-type: none"> { 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; 1120 24-bit DCU words are telemetered per second.

The DCU telemetry consists of 21 data words per 50-Hz frame plus data or "dummy" words to take up the slack in the relative frequencies of the DCU and PCM interrupt. Formatting of the "fast" portion (14 words) is performed as part of the real-time interrupt service subroutine (I/O Service) or, in special cases, by functional tasks. The "fast formatter" supplements the I/O service by assigning words to the

remaining seven slots. To do this it selects words from lower frequency buffers as submultiplex data in words 16 through 21 and forms the frame marker (word 1) by stepping the frame marker portion of it.

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.

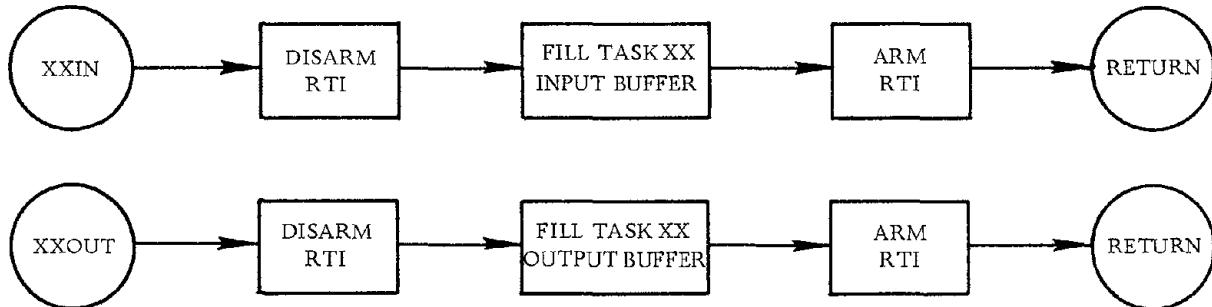


Figure 5-8. Entry points in DMM used by Task XX.

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 Stage 0 thrust decay. 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

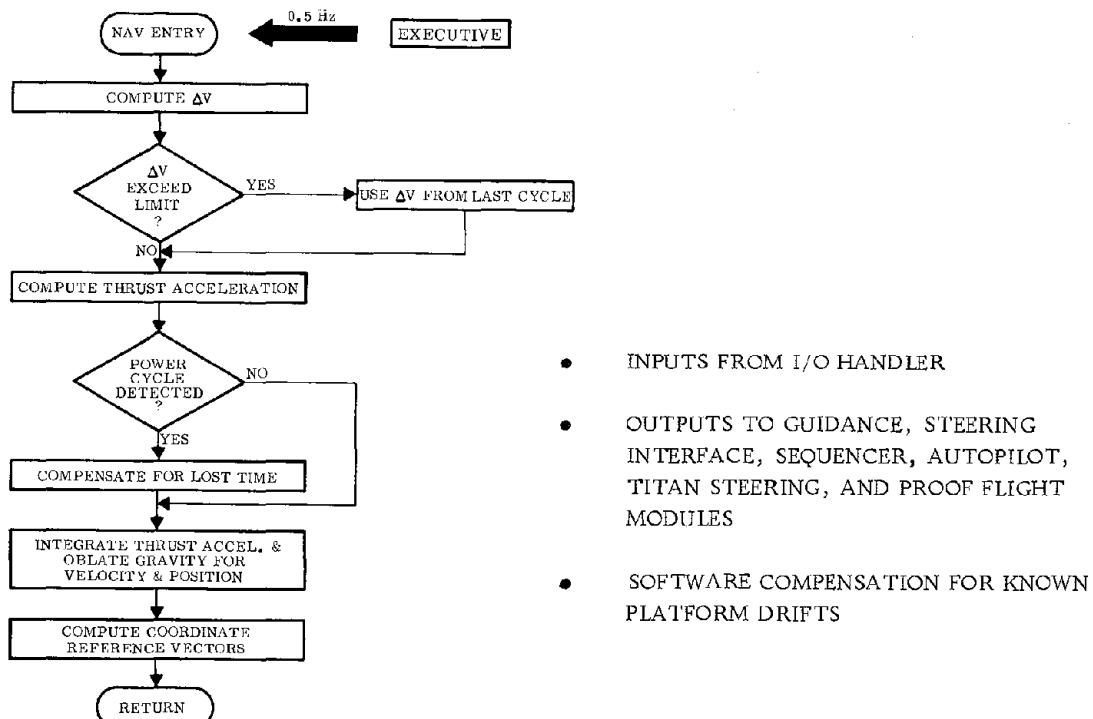
The functional tasks performed by the DCU are coded in program modules. These modules are listed in the task table from which they are called by the Executive for operation at the proper time to ensure correct frequency. Some of the modules, e.g., Navigation, operate throughout flight, while others are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

A brief description of each module follows:

NAVIGATION MODULE (NAV)

FUNCTION: Furnish position, velocity, and acceleration data to guidance. Compute t, n, and r coordinate reference vectors.

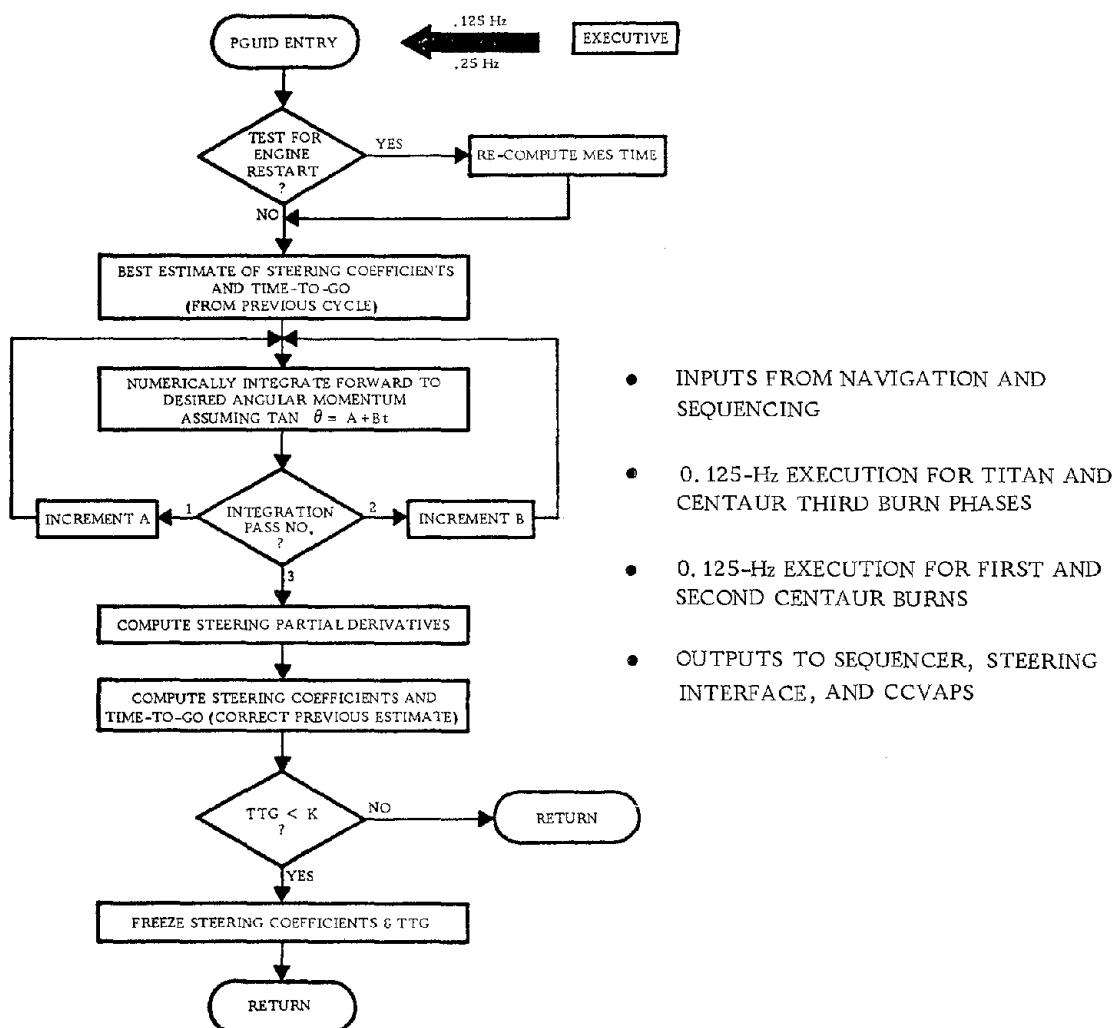
METHOD: Integrate in true inertial coordinate system, having previously converted for known platform drifts.



POWERED GUIDANCE (PGUID)

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

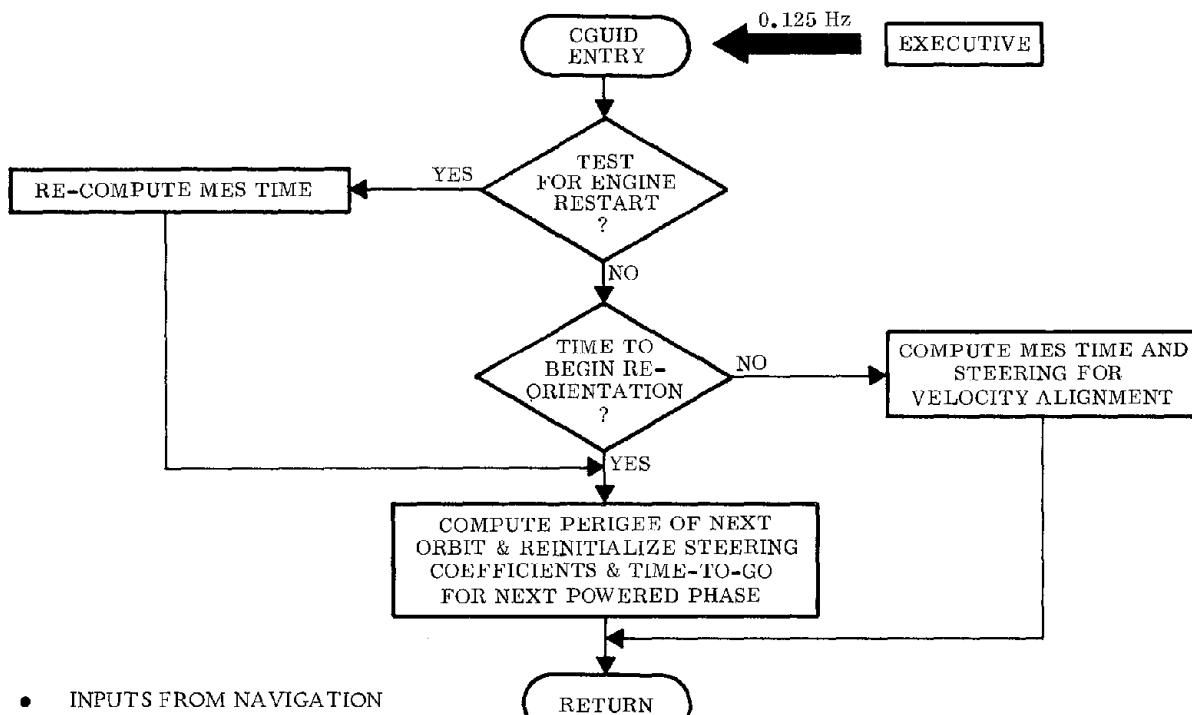
METHOD: Assumes a near-optimum linear tangent steering law in pitch and a calculus of variation steering law in yaw. Integrate to end conditions to establish partial derivation of steering coefficients.



COAST GUIDANCE (CGUID)

FUNCTION: Furnish next main engine start time to sequencer and provide reorientation steering coefficients for next burn phase.

METHOD: Compute MES time and steering for velocity alignment. Compute perigee of next orbit and reinitialize steering coefficients for next burn phase.

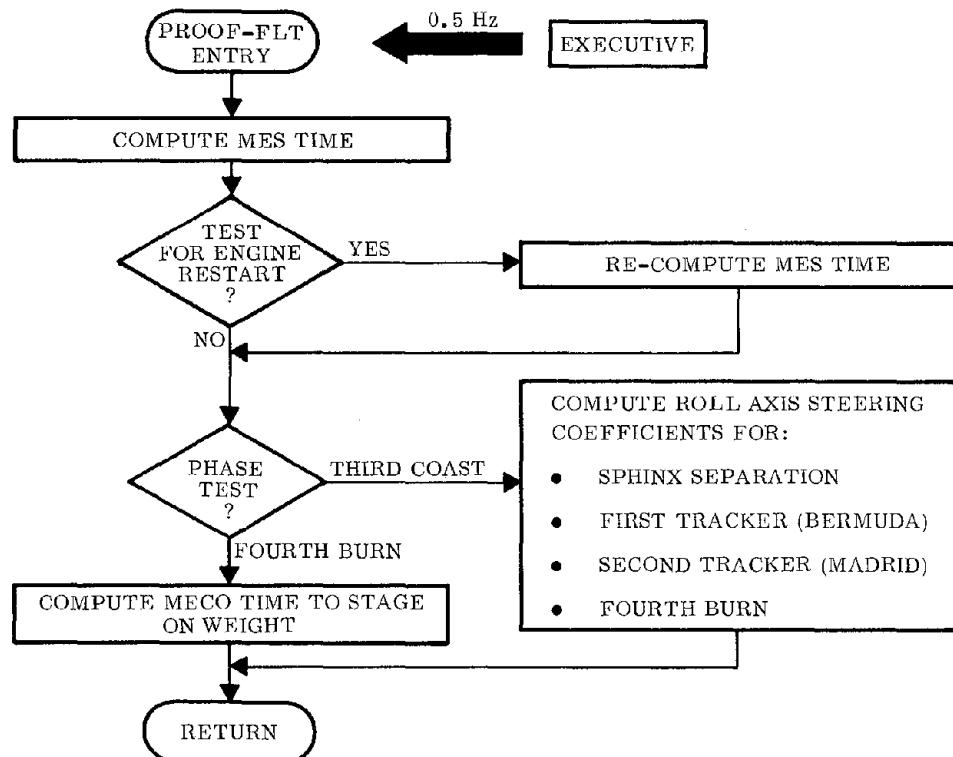


- INPUTS FROM NAVIGATION AND SEQUENCER
- OUTPUTS TO SEQUENCER, STEERING INTERFACE, AND CCVAPS
- ACTIVE DURING FIRST AND SECOND COAST PHASES ONLY

PROOF FLIGHT MODULE (PRFLT)

FUNCTION: Provide guidance after MECO3 including roll and pitch axis pointing, time for MES4, and cutoff on weight for MECO4.

METHOD: Compute MES4 time, compute steering coefficients for SPHINX separation, Bermuda and Madrid trackers, and fourth burn. Compute MECO4 time to stage on specified weight of propellants remaining.

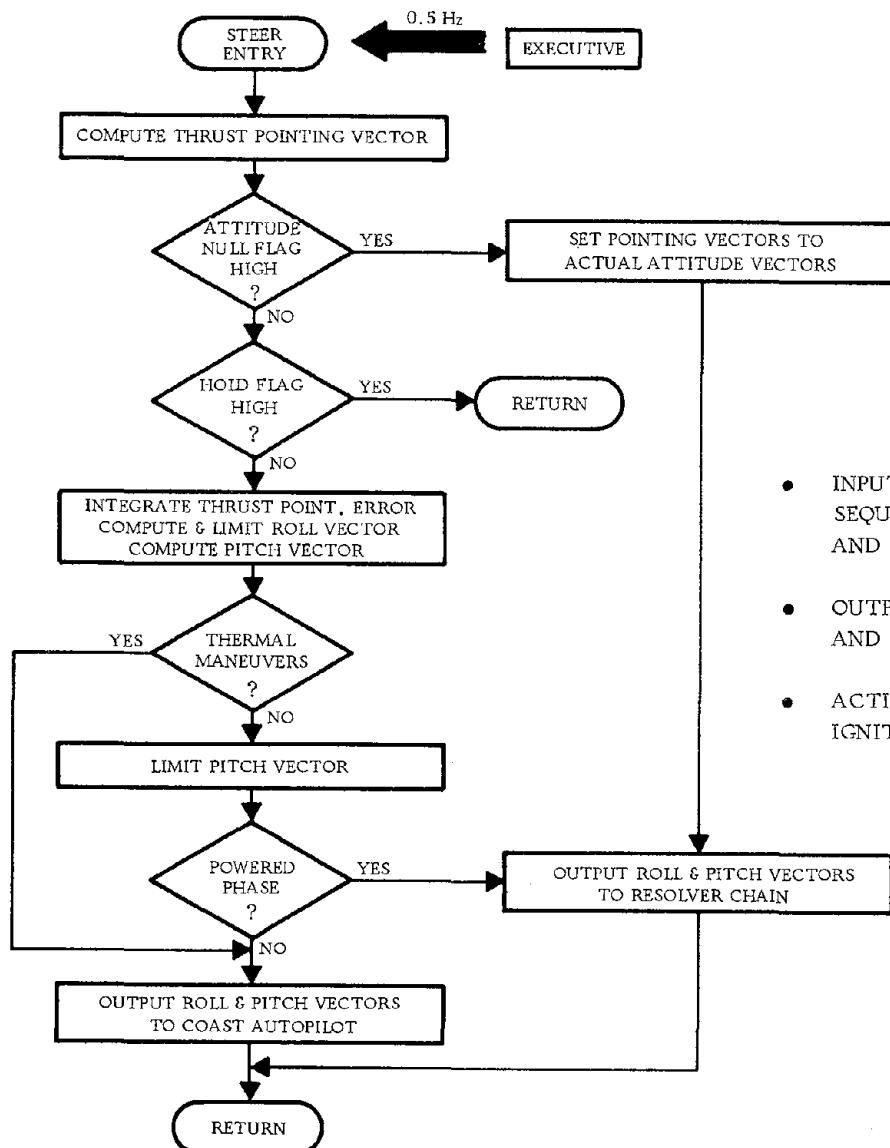


- INPUTS FROM NAVIGATION, SEQUENCER, AND PU
- OUTPUTS TO SEQUENCER, STEERING INTERFACE, AND CCVAPS
- ACTIVE DURING THIRD COAST AND FOURTH BURN

STEERING INTERFACE MODULE (STEER)

FUNCTION: Furnish the desired vehicle attitude to the platform resolver chain from Stage 1 ignition to end of flight.

METHOD: Computes desired roll pitch axis vectors and outputs them to the resolver chain.



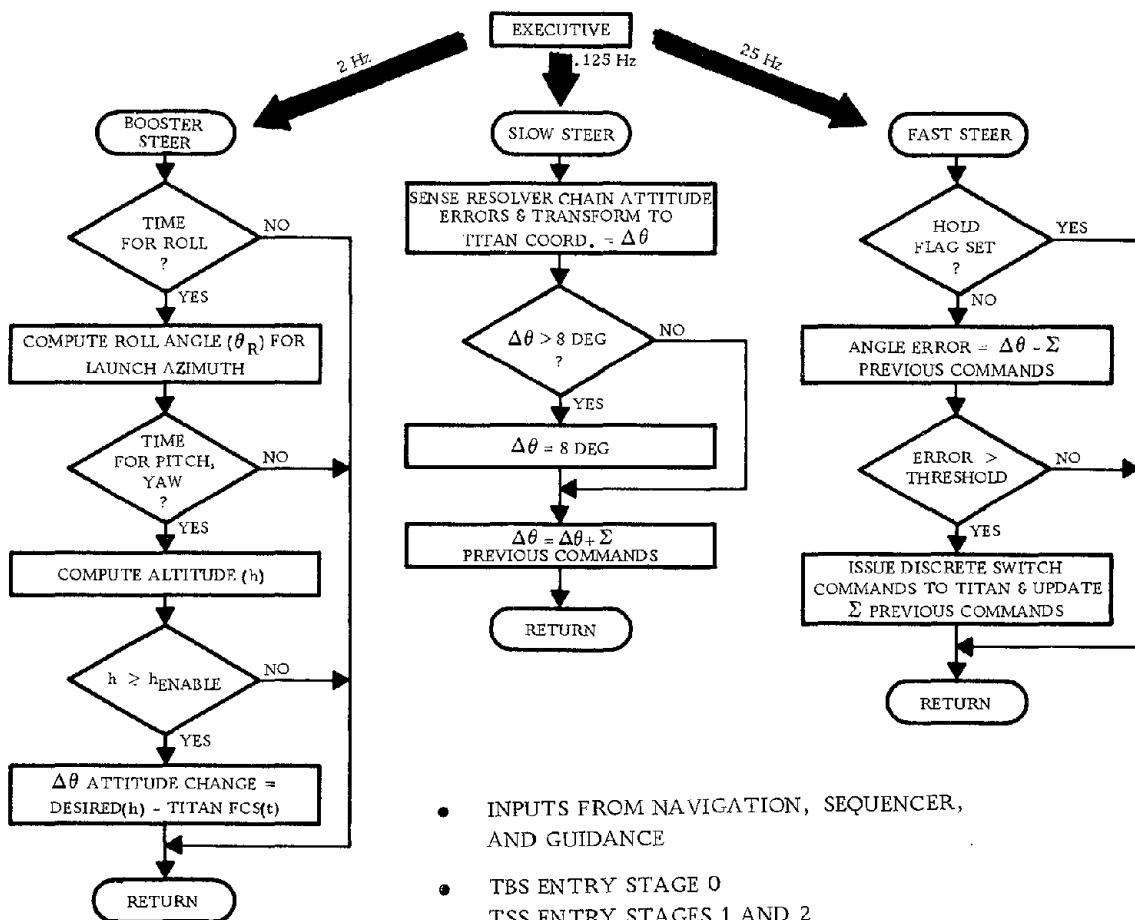
- INPUTS FROM NAVIGATION, SEQUENCER, GUIDANCE, AND I/O
- OUTPUTS TO I/O HANDLER AND COAST AUTOPILOT
- ACTIVE FROM STAGE 1 IGNITION AND ON

TITAN STEERING MODULES

(Titan Booster Steer-TBS)
 (Titan Slow Steer-TSS)
 (Titan Fast Steer-TFS)

FUNCTION: Roll to proper azimuth angle from ignition + 6.5 to 10 seconds. Pitch and yaw open-loop during Stage 0 to achieve desired attitude versus altitude. During Stages 1 and 2, pulse Titan gyros in pitch and yaw attitude according to the guidance steering laws.

METHOD: Compute roll angle for launch azimuth and execute with Titan roll gyro. Compute attitude changes by differencing desired attitude versus altitude with integral of Titan baseline pitch program for Stage 0. In Stages 1 and 2, sense attitude errors from the resolver chain and drive toward zero by pulsing Titan pitch and yaw gyros.

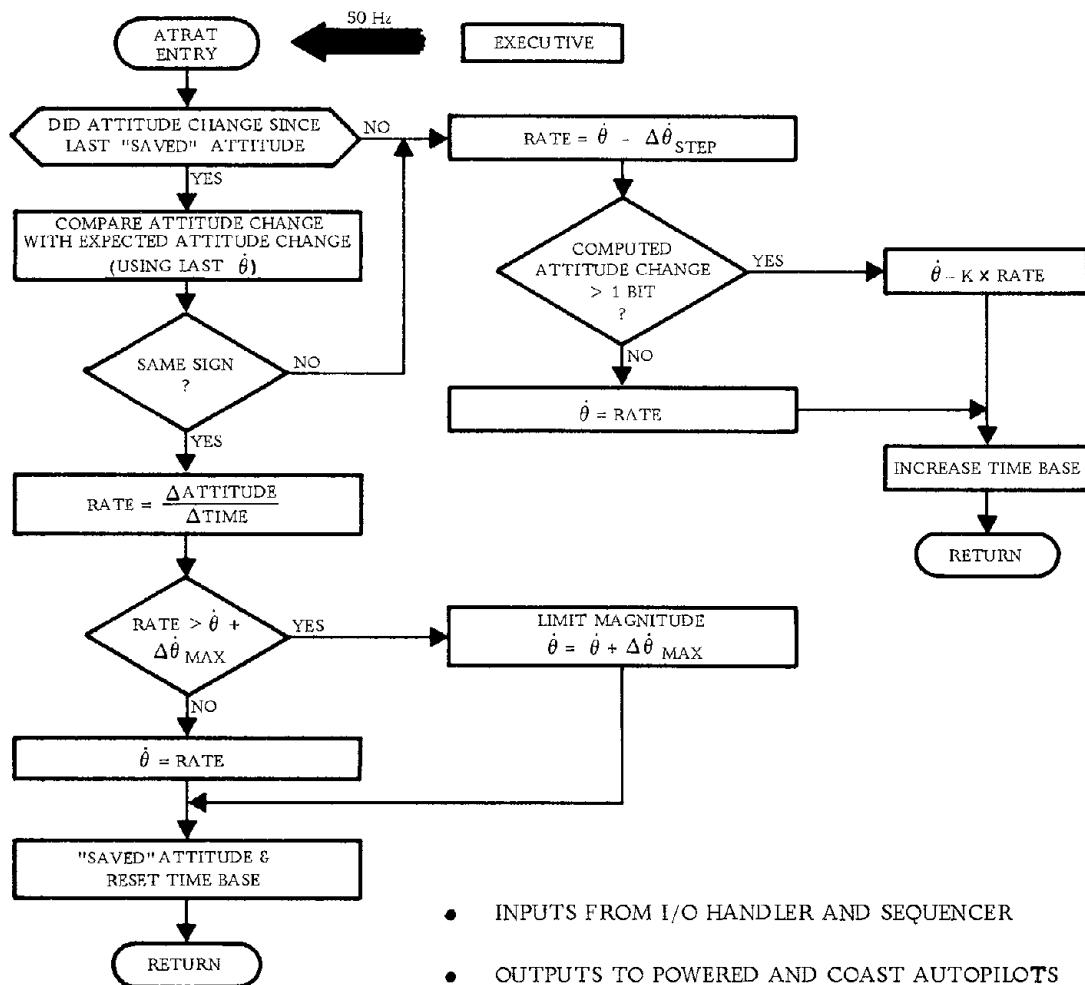


- INPUTS FROM NAVIGATION, SEQUENCER, AND GUIDANCE
- TBS ENTRY STAGE 0
TSS ENTRY STAGES 1 AND 2
TFS ENTRY STAGES 0, 1, AND 2
- OUTPUTS TO QUEUE

ATTITUDE RATE MODULE (ATRAT)

FUNCTION: Furnish rate information to powered and coast phase autopilot.

METHOD: Computes the time derivative of the attitude error signal.

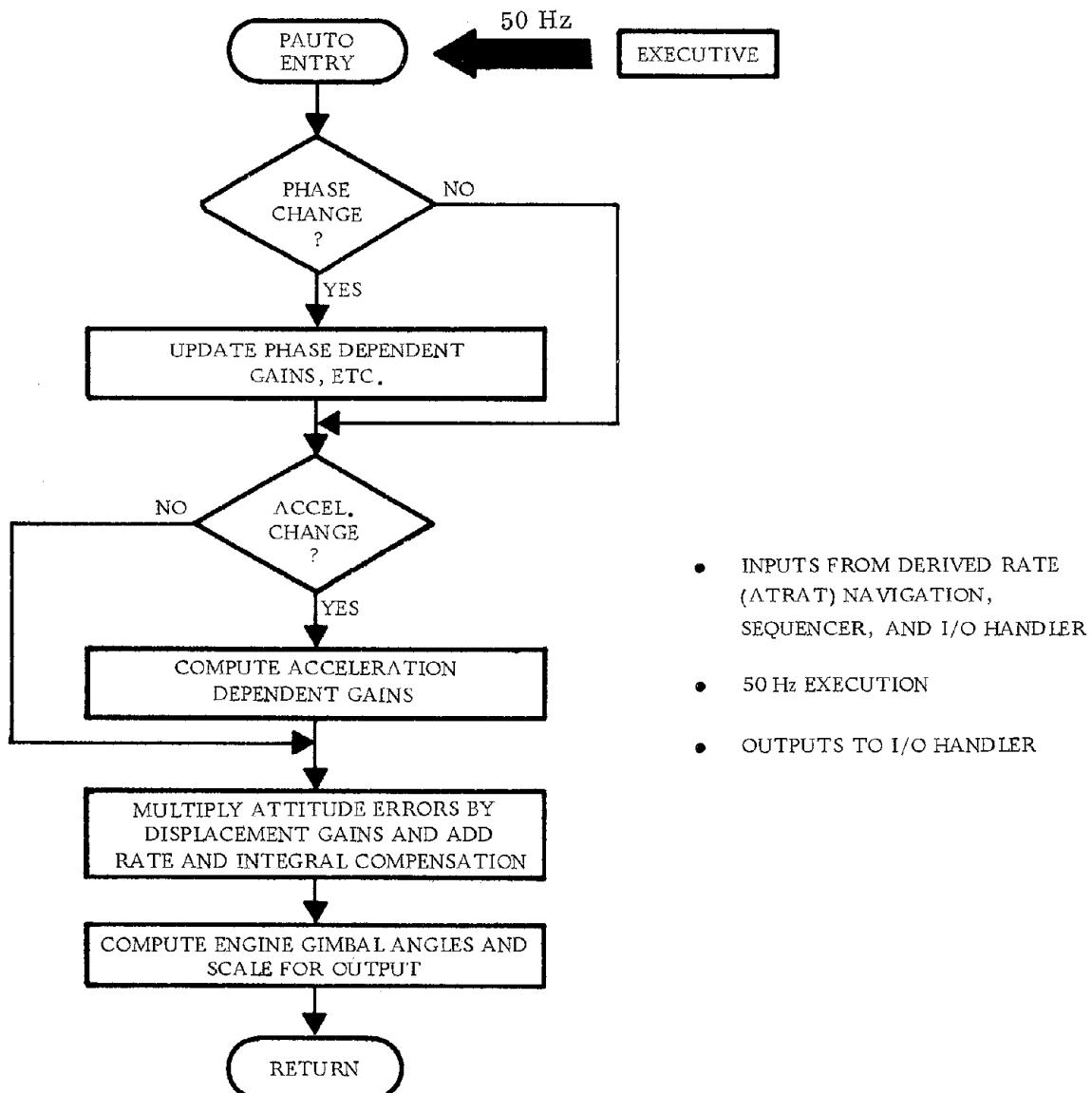


- INPUTS FROM I/O HANDLER AND SEQUENCER
- OUTPUTS TO POWERED AND COAST AUTOPILOTS
- DESIGNED FOR NOISE REJECTION

POWERED AUTOPILOT (PAUTO)

FUNCTION: Maintain control stability during Centaur main engine firings and control the vehicle axes to the desired attitudes.

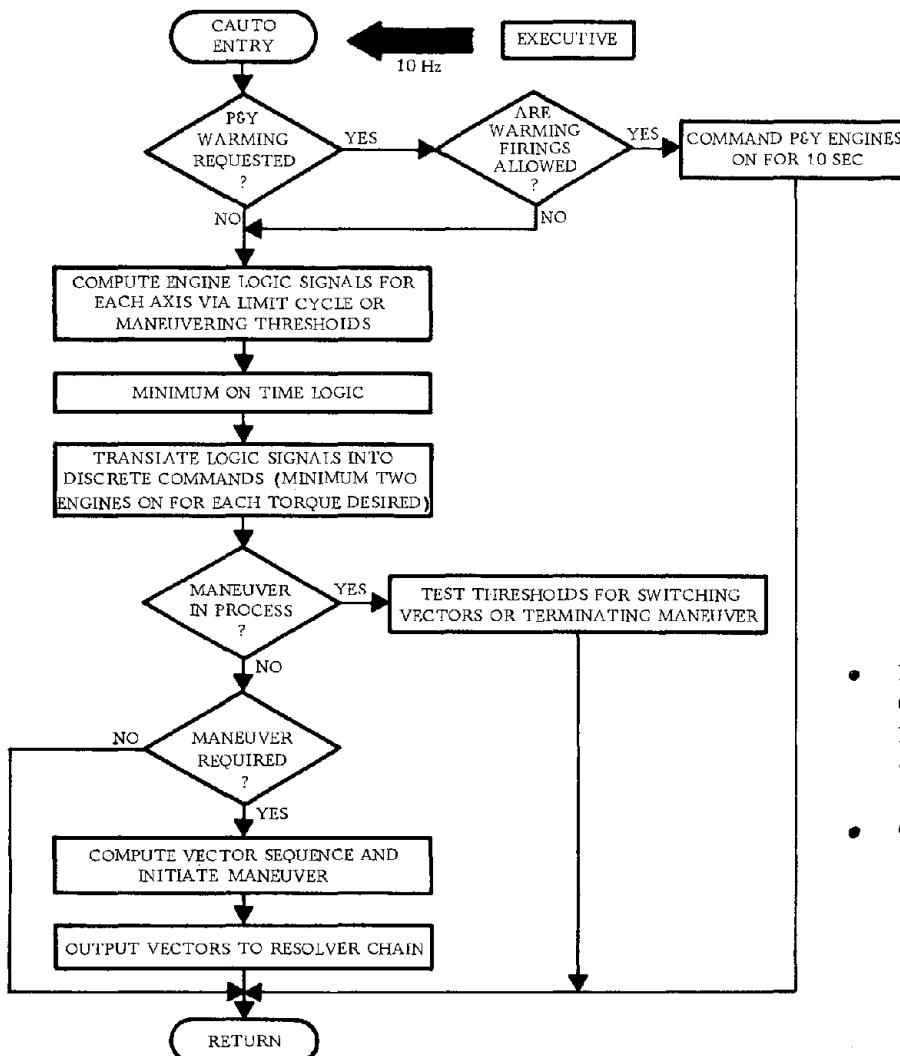
METHOD: Engine gimbal angle output commands are computed using attitude errors and error rates as inputs to control laws.



COAST PHASE AUTOPILOT (CAUTO)

FUNCTION: Control the vehicle attitude during coast phase maneuvers.

METHOD: Command H₂O₂ attitude control engines in on/off mode.

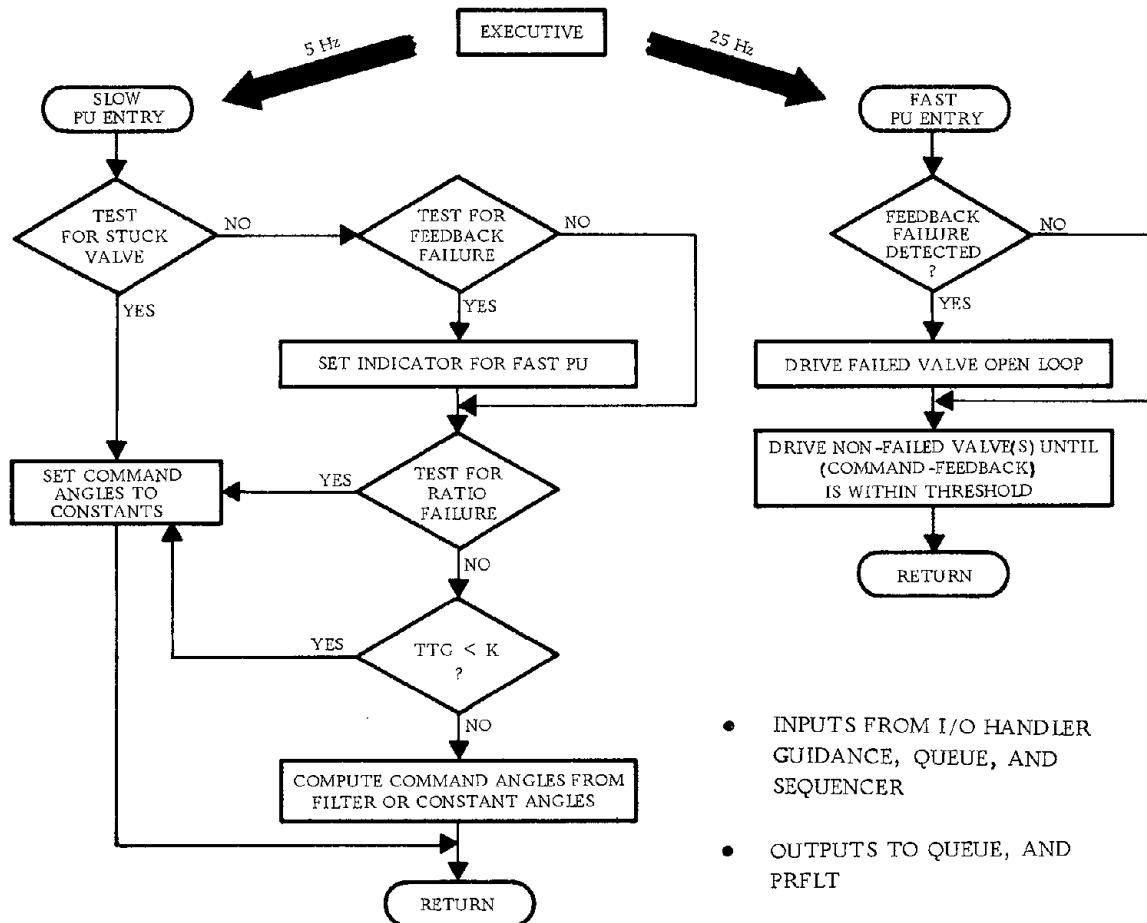


- INPUTS FROM I/O HANDLER, QUEUE, ATRAT, STEERING INTERFACE, AND SEQUENCER
- OUTPUTS TO QUEUE AND I/O HANDLER

PROPELLANT UTILIZATION (PU)

FUNCTION: Maintain a proper ratio of LH₂ and LO₂ in the tanks to preclude a premature depletion of one or the other. Test for component failures and take compensating action if they occur.

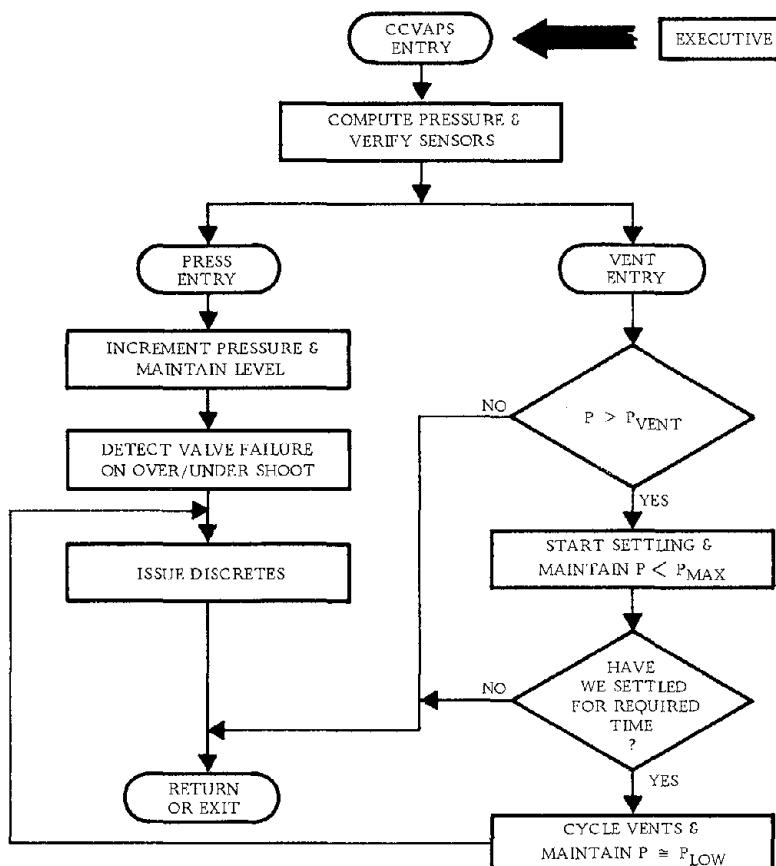
METHOD: Monitor the propellant ratio error signal and command the PU values to a position which will null the error.



COMPUTER CONTROLLED VENT & PRESSURIZATION SYSTEM (CCVAPS)

FUNCTION: Maintain proper LO₂ and LH₂ tank pressures.

METHOD: Monitor tank pressures and command venting and pressurization valves to function via SCU switch command.

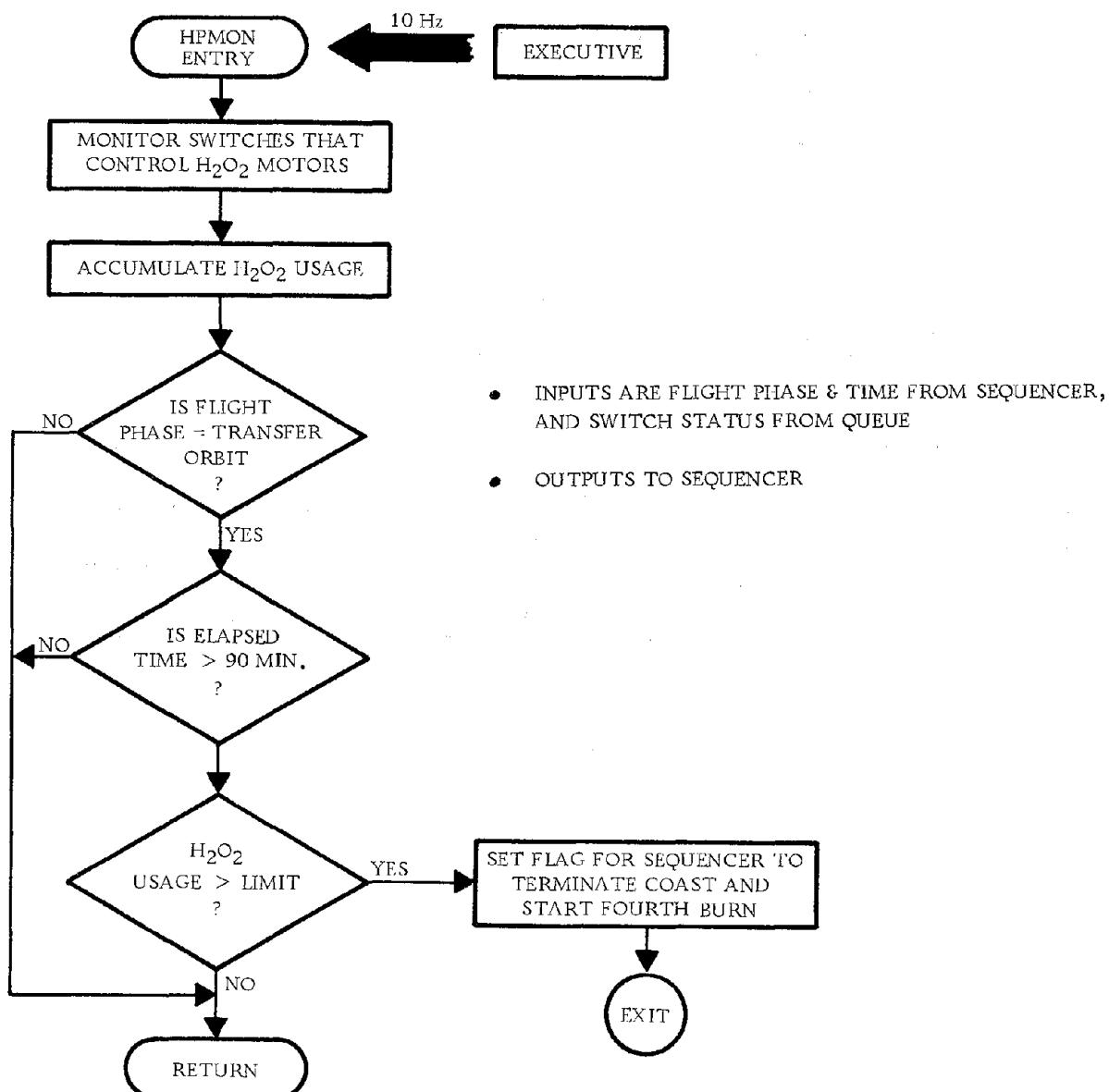


- INPUTS FROM SEQUENCER AND GUIDANCE
- 25 Hz FOR PRESSURIZATION
5 Hz FOR PROP. SETTLING
0.5 Hz FOR VENTING
- OUTPUTS TO QUEUE

HYDROGEN PEROXIDE MONITOR (HPMON)

FUNCTION: Monitor H₂O₂ usage and initiate fourth burn when a set volume has been consumed.

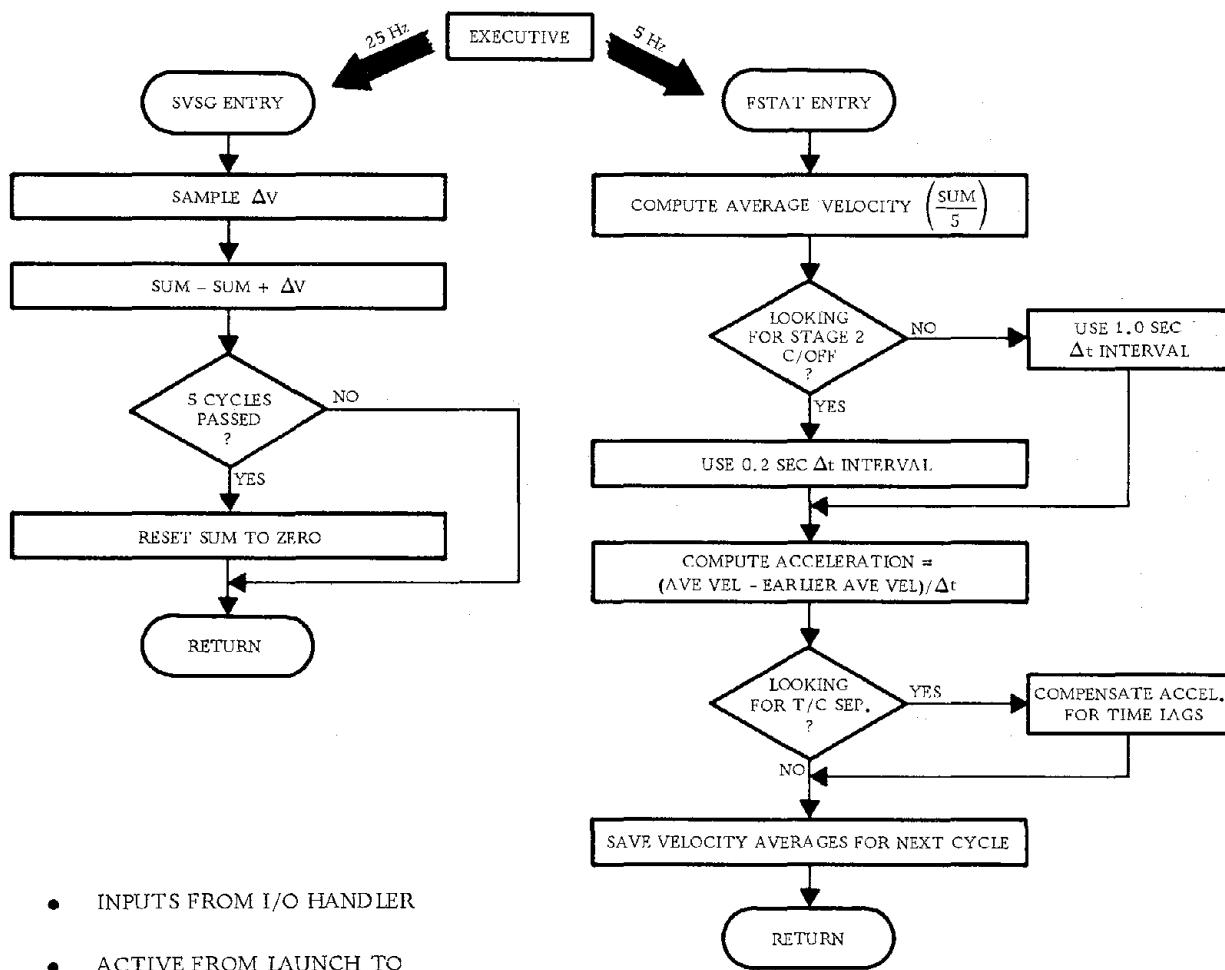
METHOD: Sum the switch on times used to control H₂O₂ motors and compare H₂O₂ usage to specified figure. Start MES4 sequencer when test is passed.



FAST ACCELERATION MODULES (SVSG, FSTAT)

FUNCTION: Provide Stage 2 cutoff backup and time for Titan/Centaur separation.

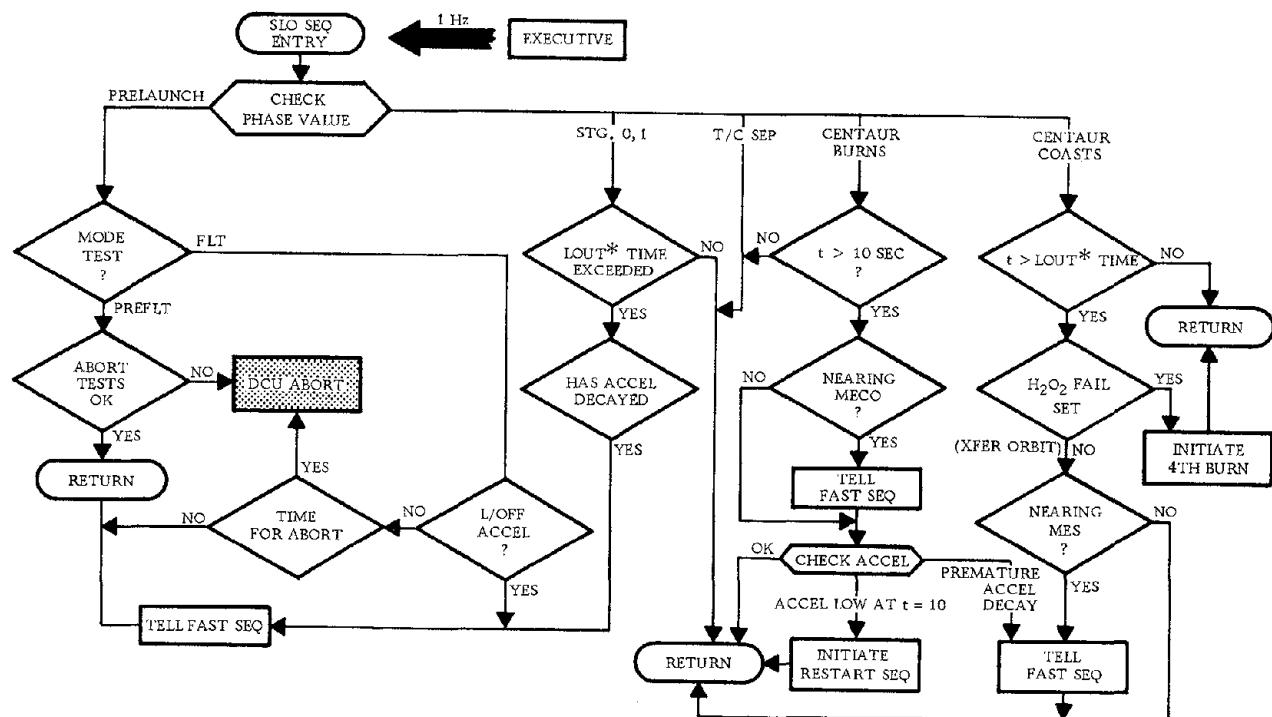
METHOD: Average velocity pulses at a high rate in order to detect acceleration levels with minimum time lags.



SLOW SEQUENCER (SLSEQ)

FUNCTION: Perform abort, backup, and other non-time critical tests required for vehicle sequencing.

METHOD: Monitor acceleration to determine vehicle state and alert fast sequencer when state changes. Use guidance generated MECO and MES times to tell fast sequencer when critical functional event times are near.



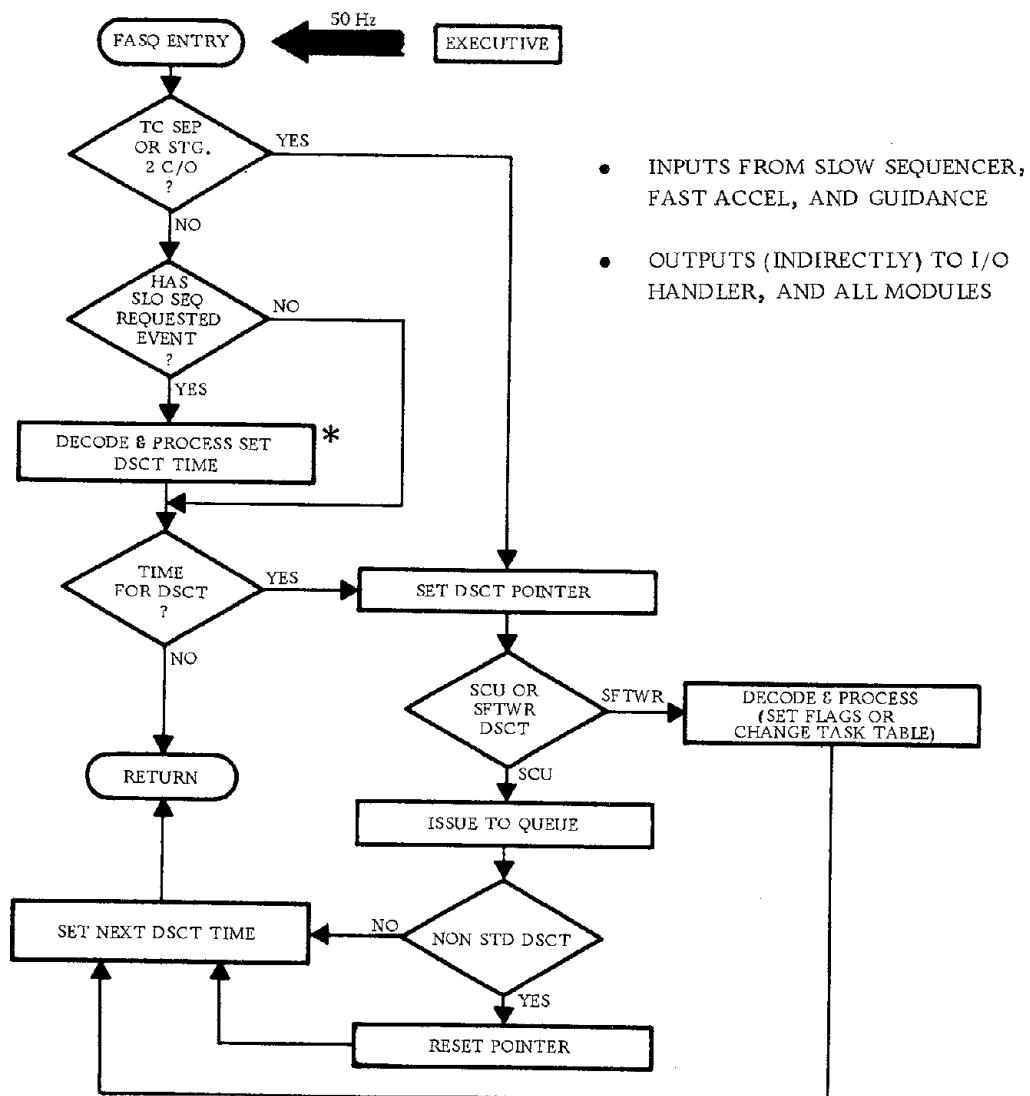
*LOUT = LOCKOUT

- INPUTS FROM NAVIGATION, FAST ACCEL, GUIDANCE, AND ABORT TESTS
- OUTPUTS TO GUIDANCE P/A PILOT, AND FAST SEQUENCER

FAST SEQUENCER (FASQ)

FUNCTION: Generate discrete events for sequencing of all required events during flight.

METHOD: Interrogate table of fixed time discrete to determine time to issue event discrete or accept output from selected modules for module dependent discrete.



*SLO SEQ "EVENTS" INCLUDE NORMAL
OR PREMATURE FLIGHT PHASE CHANGE,
OR ENGINE RESTARTS.

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. Centaur 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 ($>10^\circ$) maneuvers are required. All three modes can be required in one mission.

Another function of coast autopilot is to command the axial thrusting H₂O₂ engines for propellant settling and holding. These engines can also be used for pitch and yaw attitude control either for backup of the other H₂O₂ 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.

Another example occurs if the Centaur main engines fail to start as scheduled. The DCU software waits for 120 seconds and then issues a restart command. If this fails to start the engines, the rest of the sequence is executed at 60-millisecond intervals until the phase is completed.

5.6.5 EXPANSION CAPABILITY. Considerable expansion capability exists in the D-1 Centaur Teledyne computer. For the coded flight program (Proof Flight), 47% of temporary memory and 27% of permanent memory are unused (Figure 5-9). In the time domain (for the Proof Flight), the computer is sitting idle one-fourth of the time in the slack time module.

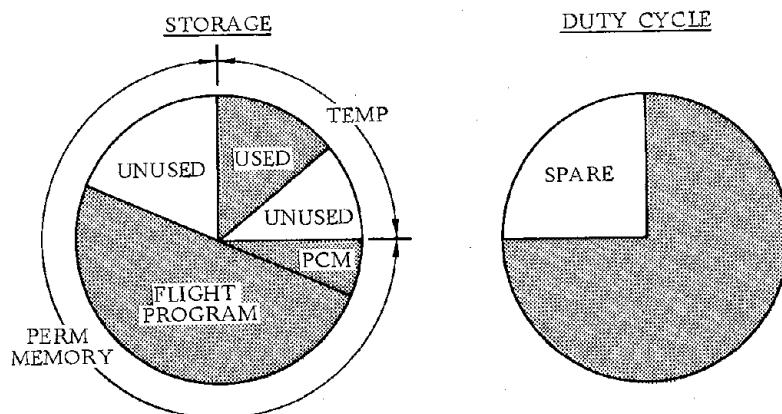


Figure 5-9. Expansion capability exists for D-1 software in Teledyne computer.

This spare capacity has allowed the addition of new functions such as the computer controlled venting and pressurization system (CCVAPS) and the consideration of using the DCU for monitoring and controlling of redundant vehicle hardware systems or components as is done on the Proof Flight in the case of redundant propellant utilization sensors.

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.

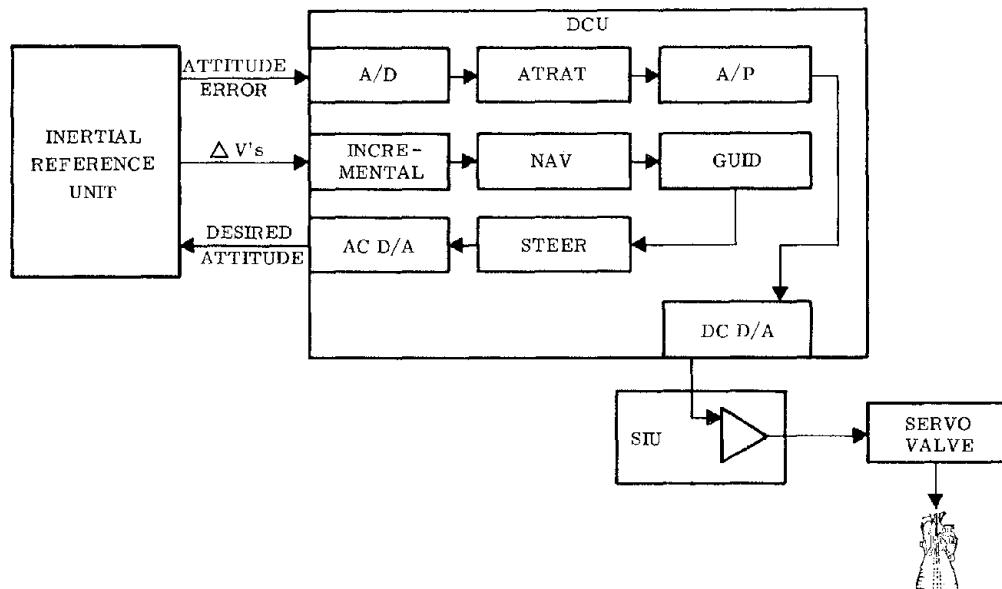


Figure 5-10. Module combination example.

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.

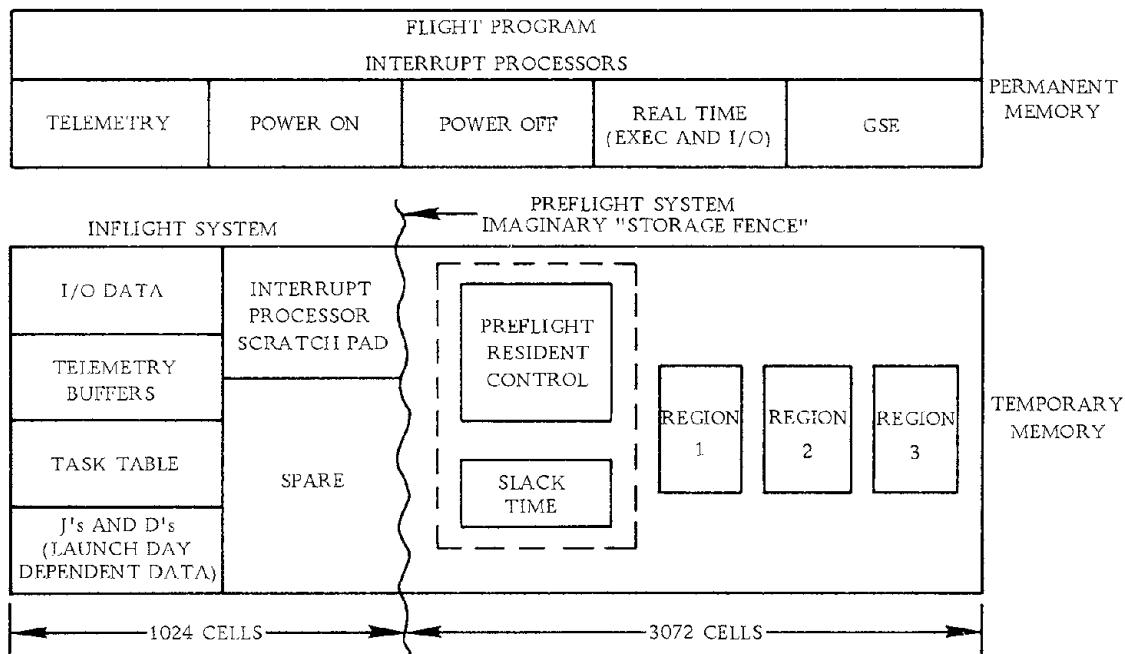


Figure 5-11. Memory configuration in the preflight mode shares the flight system software.

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 inputting data representing failed or severely dispersed hardware systems into the software. Figure 5-12 illustrates the DET.

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.

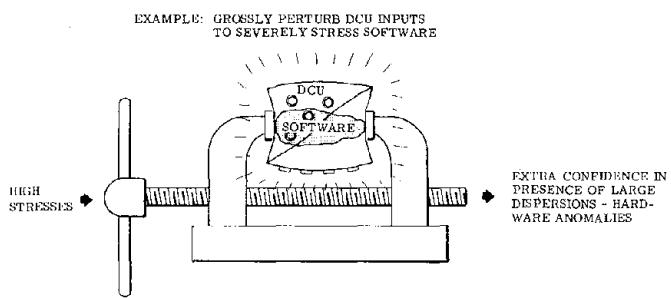


Figure 5-12. Design Evaluation Test - search for weakness.

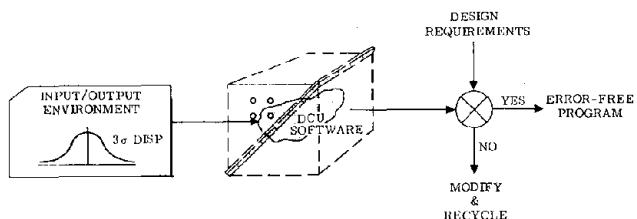


Figure 5-13. Design Acceptance Test - formal validation.

The validation of the DCU software is a multilevel function (Figure 5-14). 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.

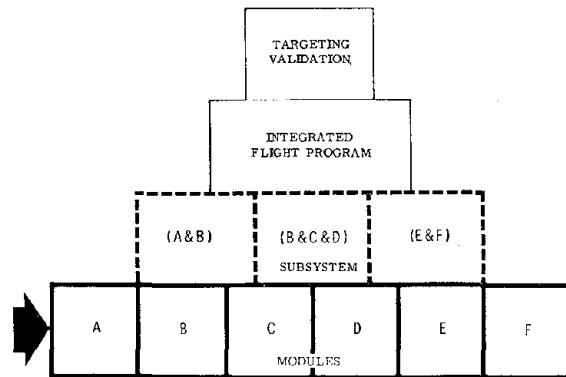


Figure 5-14. DCU software validation takes place at several program levels.

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.

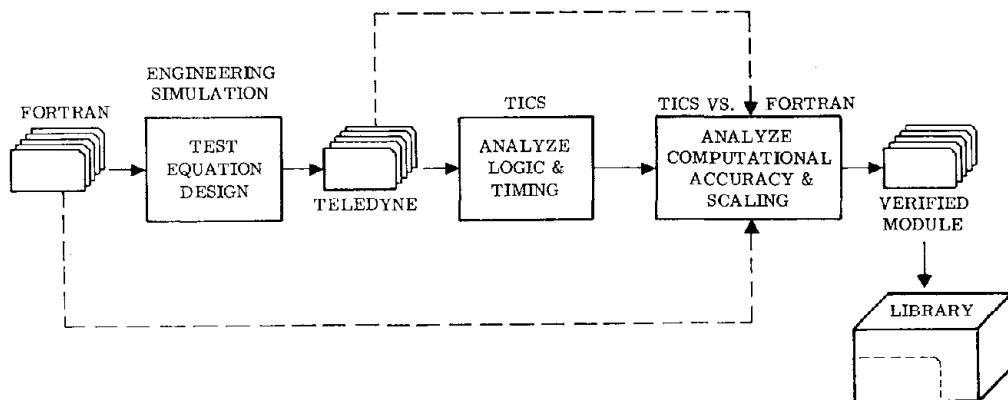


Figure 5-15. Module-level validation.

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.

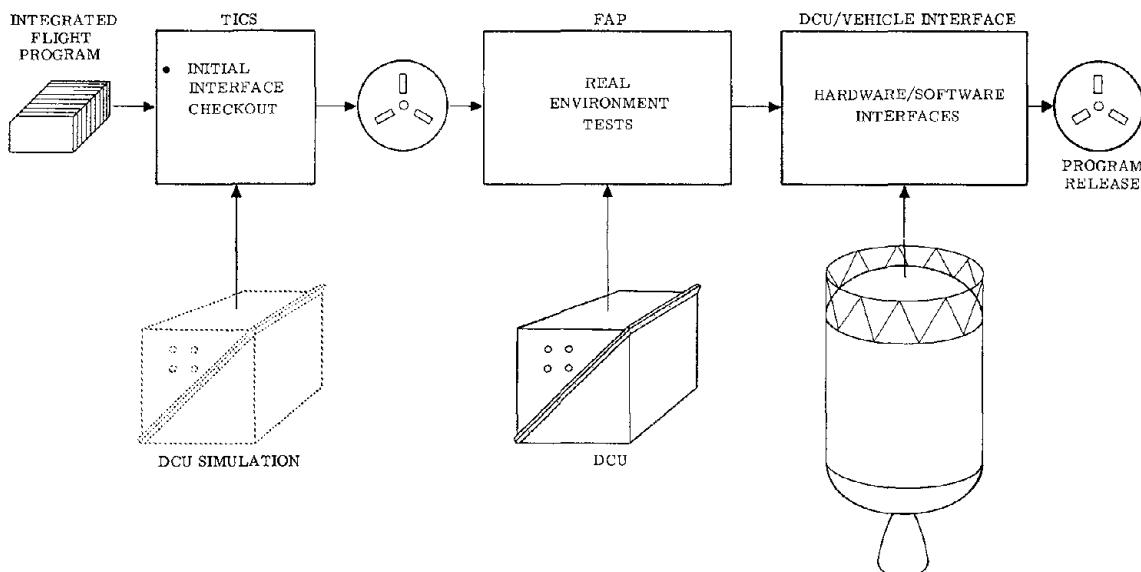


Figure 5-16. Integrated flight program validation uses many tools.

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.

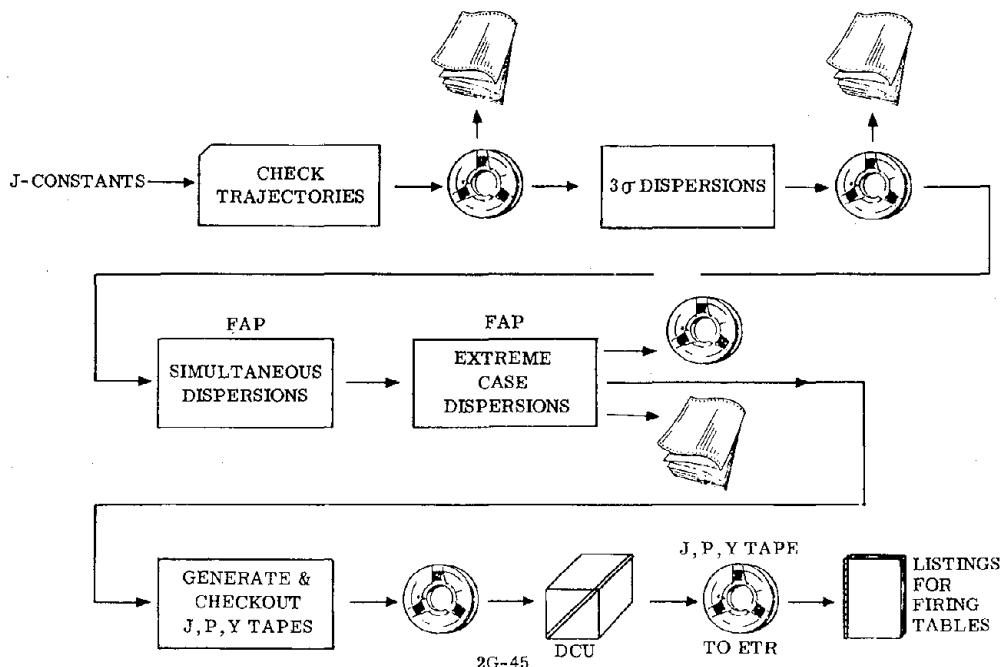


Figure 5-17. Targeting verification.

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 on the CYBER-70 engineering simulation. The results are tabulated and the unexpected data resolved and presented to the review boards for approval.

Simultaneously, sets of large 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 plus PU and CCVAPS constants. 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 SOFTWARE 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. A block diagram of the review board is presented in Figure 5-18.

There are four working group level boards:

- a. The "Check Out Equation Review Board" evaluates changes in the interfacing software between the DCU and CCLS. It is primarily a vehicle preflight test oriented review.
- b. The "Support and Interface Panel" evaluates changes in the DCU flight program related to system software, and all other functions (such as CCVAPS and PU) not covered in the following two boards.
- c. The "Digital Autopilot Review Board" evaluates changes in the Powered and Coast phase autopilot functions.
- d. The "Guidance Equation Review Board" does the same for the guidance equations.

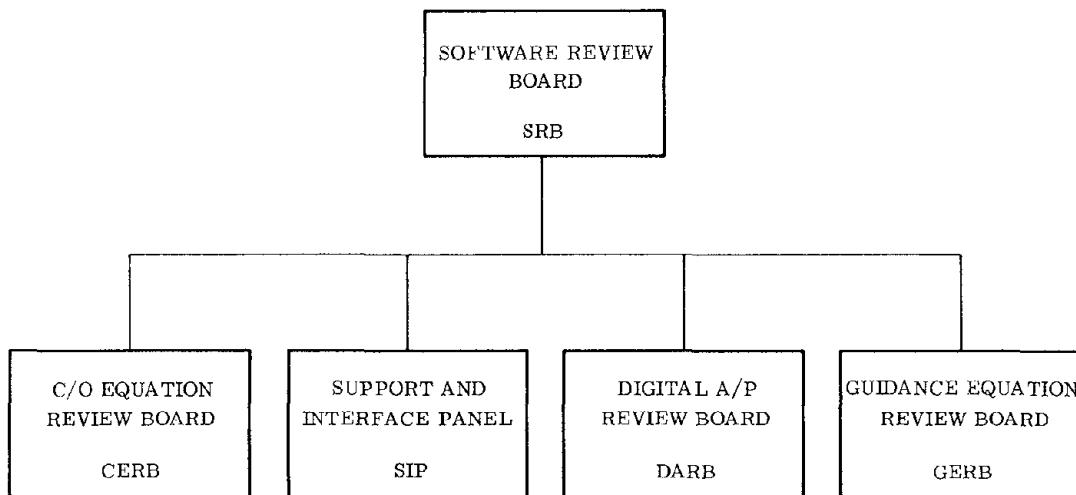


Figure 5-18. Software review board.

The working group boards provide overviews, summaries, and recommendations for each change to the Software Review Board, which renders final decisions on all changes. All boards have NASA/LeRC chairmen with members from GD/CA. The GERB also has members from Honeywell and TRW.

The DCU software control process (Figure 5-19) includes control from the documentation level through identification techniques and final release procedure. The control process starts with the ICT, which defines the software to be changed. This document includes the engineering requirements and coding level programming documentation. Next, the changes and/or new requirements are subjected to a resource control analysis. This analysis weighs the potential gains of the change against the resource consumption of storage, duty cycle, and cost.

If warranted by the above tradeoff study, the change is formally documented in a DCU Software Change Sheet (DSCS). This is a convenient form for summarizing the significant aspects of the change for subsequent GD/CA management and LeRC management review. This DSCS is then reviewed by the working groups.

When implemented into actual software, the module has a unique check figure assigned to it. This is analogous to a hardware serial number. It is unique in that it will change if any change is made to the software. At this stage, the software is ready for production use, and can be entered into the module library. The library is used to store all flight software production modules and serves as a baseline reference from which future integrated programs are evolved. Final approval and release is through a standardized procedure which is similar to that used for production hardware end-items.

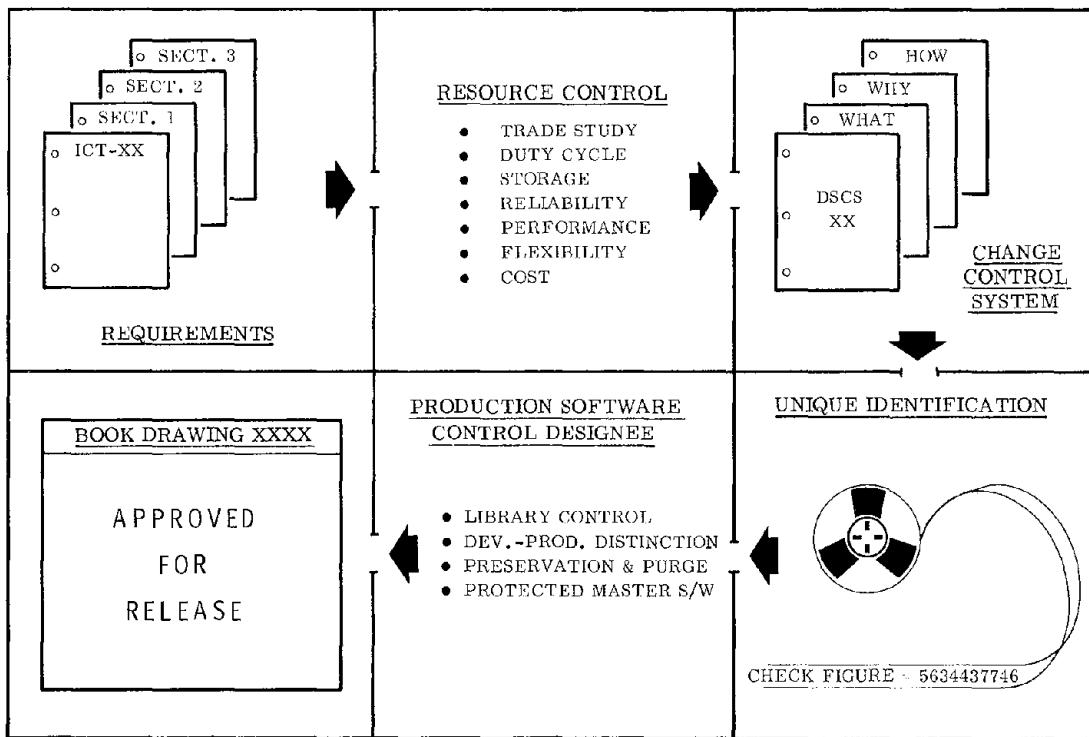


Figure 5-19. DCU software control process.

The ICT is the baseline from which all changes are referenced. Each functional software module has an ICT documentation packet as shown in Figure 5-20. The major sections of the ICT are:

- Section 1 Engineering Design Requirements, consisting of functional design constraints and performance requirements.
- Section 2 Programming Design Requirements, consisting of inputs, outputs, computation accuracies, etc., and timing constraints.
- Section 3 Program Description, consisting of flow charts, verbal description, etc., assembly listing, and check numbers.
- Section 4 Test and Validation Requirements, consisting of tests to be run, test environment, and test tools required.
- Section 5 Test Results Record, consisting of numerical data, tab listings, plots, etc., and conclusions drawn from tests.
- Section 6 Change History, consisting of statement of baseline configuration ICT, and change sheets (DSCS).

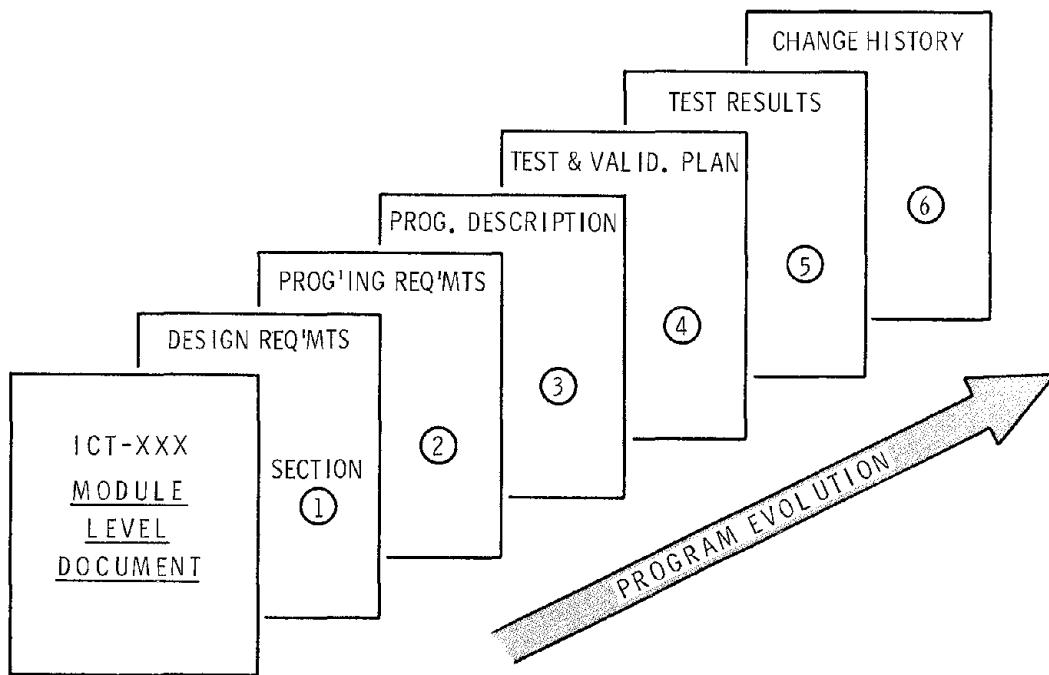


Figure 5-20. Software documentation.

Software changes are proposed and reviewed on a standard documentation form, entitled "DCU Software Change Sheet". The intent of these forms is to explain the change and make its impact on interfacing software visible to management. Storage and duty cycle resource consumption are also indicated as part of the cost. The change sheet is shown in Figure 5-21.

A unique software "part number", referred to subsequently as a check number, aids in management and identification of the configuration. There are two levels of control: module level, and overall flight program level (Figure 5-22).

The module level control allows program changes to be isolated and controlled at that level. The module check number is formed from the assembled module code and consists of a summation of the octal form of the coded instructions and data. The summation is unique in that it is sensitive to the order of summing, i.e., if two instructions or data values are listed in reverse order, the check number will change.

The flight tape check number is a similar order-sensitive sum of the octal binary core image of the complete program loaded into the DCU. The flight program simply consists of a collection of modules which have been assigned absolute core locations by the loading process. Thus, if one module changes, the flight tape check number will correspondingly change.

DCU SOFTWARE CHANGE SHEET (DSCS)		
Page 1 of 2		
1.	DSCS NO. _____ DATE _____	INITIATED BY _____ REF. ICI 74-
2.	BRIEF DESCRIPTION OF CHANGE (ATTACH FLOW CHART IF APPROPRIATE) 	
3.	REASON FOR CHANGE <input type="checkbox"/> MISSION PECCULAR REQUIREMENT <input type="checkbox"/> REDUCE STORAGE <input type="checkbox"/> VEHICLE OR HARDWARE CHANGE <input type="checkbox"/> REDUCE DUTY CYCLE <input type="checkbox"/> MODIFIED REQUIREMENTS <input type="checkbox"/> RELIABILITY <input type="checkbox"/> MODIFIED SUPPORT SOFTWARE <input type="checkbox"/> FLEXIBILITY <input type="checkbox"/> ENGINEERING ERROR <input type="checkbox"/> BOOKKEEPING <input type="checkbox"/> PROGRAMMING ERROR	
4.	EXPLAIN PROBLEM OR SITUATION THAT INITIATED CHANGE ACTION 	
5.	CHANGE IS: <input type="checkbox"/> MANDATORY <input type="checkbox"/> IMPROVEMENT	
6.	WHAT HAPPENS IF CHANGE IS NOT MADE: 	
7.	IDENTIFY <input type="checkbox"/> YES <input type="checkbox"/> NO MODULE I/O AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO TELEMETRY AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO OTHER MODULES AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO CCLS AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO FACTORY AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO DCU TENANTS AFFECTED <input type="checkbox"/> YES <input type="checkbox"/> NO TRAP AFFECTED <input checked="" type="checkbox"/> YES <input type="checkbox"/> NO ADJUST AFFECTED	
8.	WHERE ARE WE IN S/W CYCLE: <input type="checkbox"/> MODULE DEVELOPMENT <input type="checkbox"/> INTEG. PROG. DEBUG <input type="checkbox"/> FORMAL TESTING <input type="checkbox"/> POST RELEASE	
9.	ESTIMATED DCU RESOURCE IMPACT: PERM. WORDS _____ 50 Hz DUTY CYCLE _____ TEMP. WORDS _____ AVG. DUTY CYCLE _____	
DSCS NO. _____ Page 2 of 2		
10.	TEST PLAN TO IMPLEMENT AND VALIDATE (DESCRIBE BELOW OR ATTACH) 	
11.	ESTIMATED SCHEDULE IMPACT: ITEM _____ NO. OF WKS. _____	
12.	ESTIMATED MANPOWER AND COMPUTER COST TO DEVELOP, IMPLEMENT AND TEST THIS CHANGE ENGR. MHRS _____ CYBER-79 HRS _____ PROG. MHRS _____ FAB/CCLS HRS. _____	
13.	THIS ITEM HAS BEEN DISCUSSED/REVIEWED AT FOLLOWING MEETINGS AND DATES: DATE _____ MEETING _____ DATE _____ MEETING _____	
14.	CHANGE EFFECTIVITY: FLIGHT _____ FACTORY _____ UTR _____	
15.	IS THIS PART OF A BLOCK CHANGE? <input type="checkbox"/> YES <input type="checkbox"/> NO FLIGHT TAPE BLOCK NO. ASSIGNED	
16.	CD/CA APPROVALS: (STANDARD) (IF AFFECTED) ORIGINATOR _____ CCLS _____ DCU S/W PRIME _____ PC/CCVAPS _____ DCU S/W GROUP _____ SYSTEMS _____ FLIGHT MECH _____ TELEMETRY _____ CHF. ENGINEER _____ PROGRAMMING _____ PROG. OFFICE _____ FUT. CONTROL _____	
17.	LdRC APPROVALS: SRP, DARB, GRKB _____ APPROVED OR CERB CHAIRMAN _____ DISAPPROVED SRB CHAIRMAN _____ APPROVED _____ DISAPPROVED	
18.	DISTRIBUTION: D. FLOWERS, B. STAUDER, L. PEAGAN OTHERS: _____ _____ _____	

Figure 5-21. DCU software change sheet (DSCS).

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ORIGINAL PAGE IS POOR

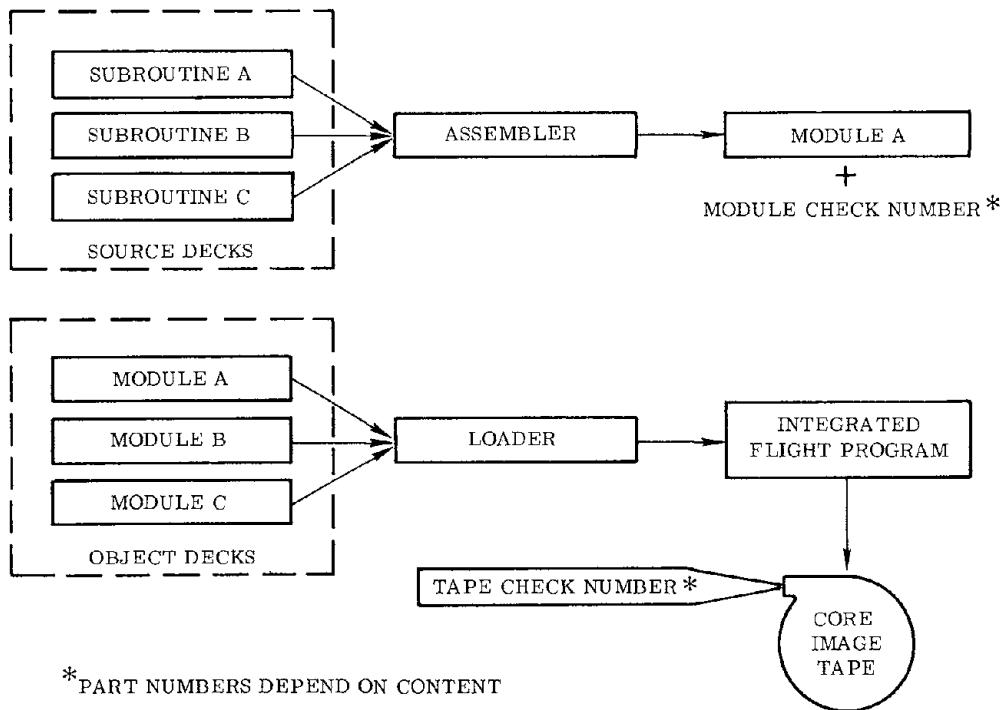


Figure 5-22. Software part number control.

6

TITAN IIIE MECHANICAL SYSTEMS

This section presents descriptions of the following Titan IIIE mechanical systems:

SUBSECTION	Page
6.1 Airframe Structure	6-1
6.2 Propulsion Systems	6-14
6.3 Thrust Vector Control Systems	6-27
6.4 Ordnance Items	6-39

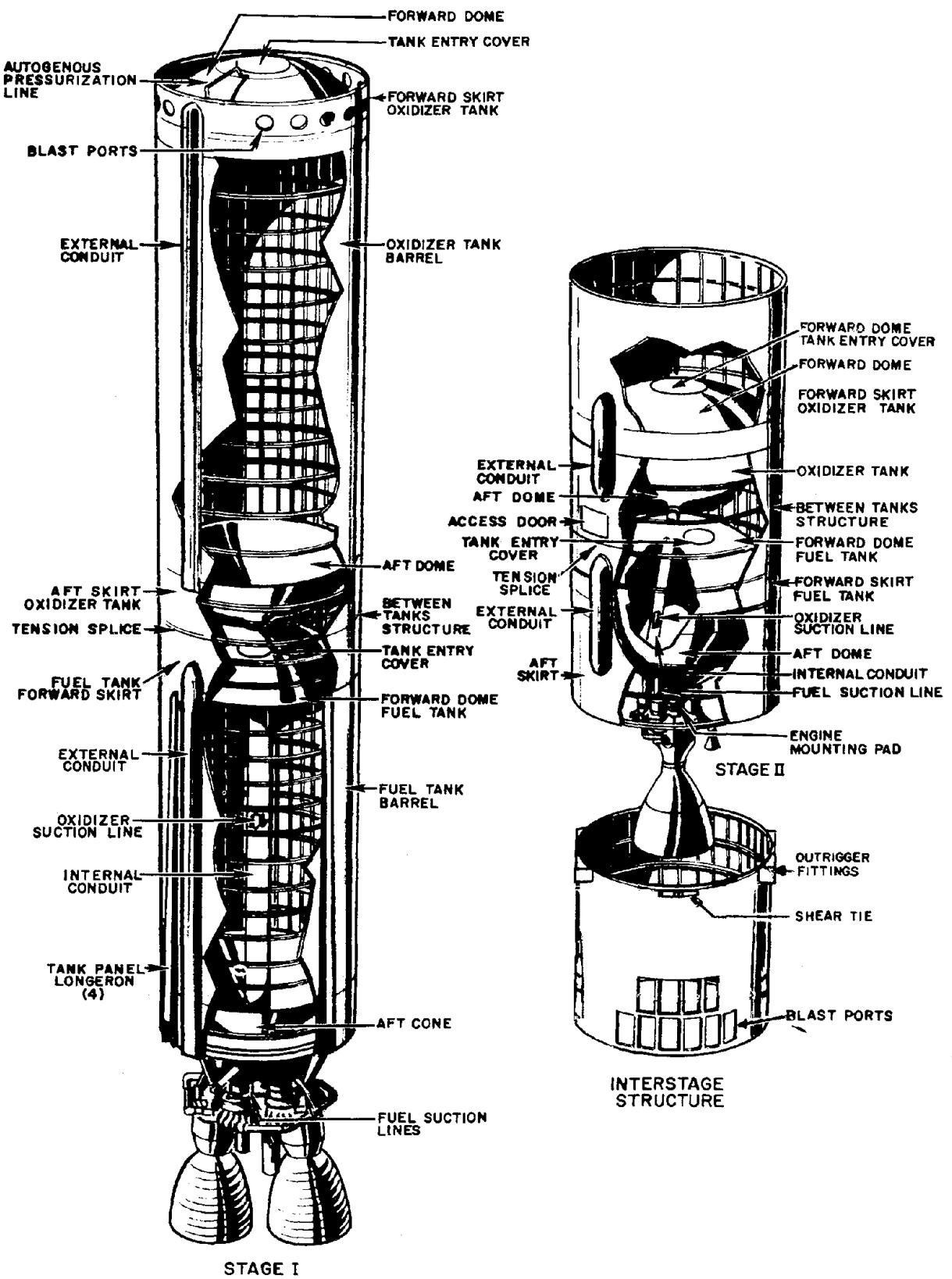
6.1 AIRFRAME STRUCTURE

ELEMENTS:	Subsection
• Stage I Airframe	6.1.1
• Interstage Structure	6.1.2
• Stage II Airframe	6.1.3
• Airframe Attachments	6.1.4

FUNCTIONS:

- Contain Propellants
- Protect Stage I Engine
- Provide Support for Airborne Systems & Components
- Provide Titan/Centaur Interface
- Support Upper Stage(s)

The Titan IIIE core airframe (Figure 6-1) is an all-aluminum structure designed to contain propellants, provide support for all add-on systems on the core, and support the upper stages. It also contains attachments for the solid rocket motors and distributes thrust loads from all propulsion systems.



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Figure 6-1. Titan IIIE core airframe structure.

6.1.1 STAGE I AIRFRAME. The Stage I airframe structure consists of the fuel tank with its longeron and skirt section, and the oxidizer tank and its skirt sections, and the Stage I heat shield.

Fuel Tank Assembly. The fuel tank assembly includes the fuel tank with forward and aft skirt assemblies and an engine mount truss at the apex of the tank conical bottom. The fuel tank wall also serves as vehicle exterior skin. The tank proper is an all-welded aluminum unit with stringer and frame reinforced walls, conical bottom, and an elliptical dome top. Special extruded frames join the tank walls to the top and bottom. These frames also provide weld attachment of the skirt assemblies. A central 13-inch-diameter conduit allows passage of the oxidizer feed line to the engine.

The tank barrel is assembled from eight taper-machined skins. Four of these skins are called longeron panels (Figure 6-2). The panels have two flanges extending at right angles to the other surfaces to form channels about four inches wide on each side. The flanges on the outer surface are spliced to the lower longerons on the aft skirt to form load and thrust distribution members.

The four larger skin panels that complete the tank barrel circumference are machined with eight "T" stringers on the inner surface of each panel. Panels are tapered with the thickest part at the aft of the tank.

Ten stretch-formed sheet aluminum ring "Z" frames are attached to the stringers and longeron panel standoffs at equal intervals with riveted brackets to further stabilize the walls. The conical bottom of the tank is formed by five welded sections. This conical dome provides for the Stage I fifth attachment point. The upper portion of the cone is formed from four quarter segments with chemically-milled areas on the outer surface. The center portion of the tank bottom is a press-formed cone with an inverted, truncated tip. Two 6-inch fuel outlet lines located 180 degrees apart at the inversion point provide fuel flow to the engine. Cross-baffle plates located over the line outlets prevent liquid swirl or vortex in the outlet flow.

The upper edge of the tank bottom is welded to the special extruded splice frame which attaches the bottom to the tank wall and aft skirt. This special frame is often referred to as a "K" (Figure 6-3). The frame is machined from three basic extrusions formed and welded into a circular frame with the back of the K as the outside surface. The tank bottom is welded to the lower angle of the K; the tank wall and

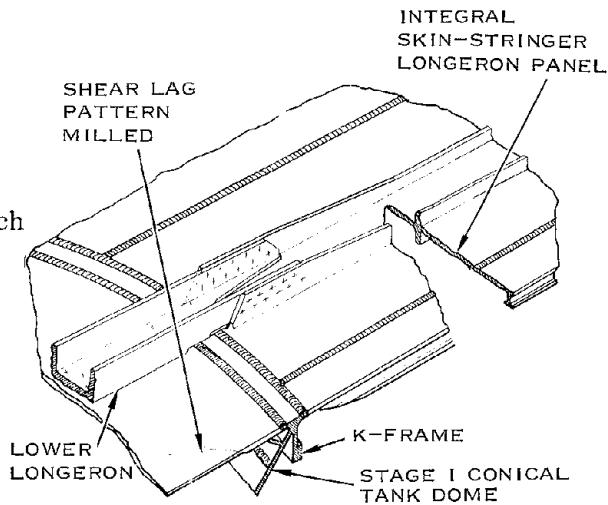


Figure 6-2. Fuel tank longeron panel.

aft skirts are welded to the back legs. The upper right angle of the K, extending at a right angle to the back, has a heavy edge and serves as a stiffener. An extension is formed on the lower side of the tank bottom leg to aid in positioning the aft skirt skins for welding to the K frame.

The aft skirt assembly is a stringer-frame-reinforced unit using the rivet-bolt construction principle. The assembly is 37.5 inches long exclusive of the longerons. The four longerons that extend from the aft frame into the tank skin longeron panels are 56.6 inches long. They are forged aluminum channels which interface with the top side of the aft frame at a point directly opposite the engine frame attachment and core vehicle support pads. Six holes drilled through both channel flanges of each longeron are used as attachment points for vehicle-handling items.

The four skin panels of the skirt are milled on the inner side. A series of shear lag patterns are milled into the skin between frames and stringers. The skins are spliced end-to-end under the longerons.

Two extruded ring frames are placed inside the skirt. The frame near the top edge is a "T" frame and the frame at the center of the skirt is of the "I" type. The frame at the aft edge of the skirt is a heavy forged aluminum member which is bolted to the aft end of the longerons and riveted to the skirt skins. The frame is approximately 16 inches wide, narrowing somewhat in the longeron area. Engine frame mating sockets (Figure 6-4) are machined into the frame at the longeron attachment points, and SRM attachment holes are drilled to mate with corresponding holes in the longerons.

The fuel tank top is an all-welded elliptical dome with manhole access and oxidizer conduit openings. The dome is fabricated from five press-formed skins. Four skins form the outer dome area, and a center cap completes the structure. The center cap has a cone-shaped conduit adapter welded in a center opening and extending into the dome.

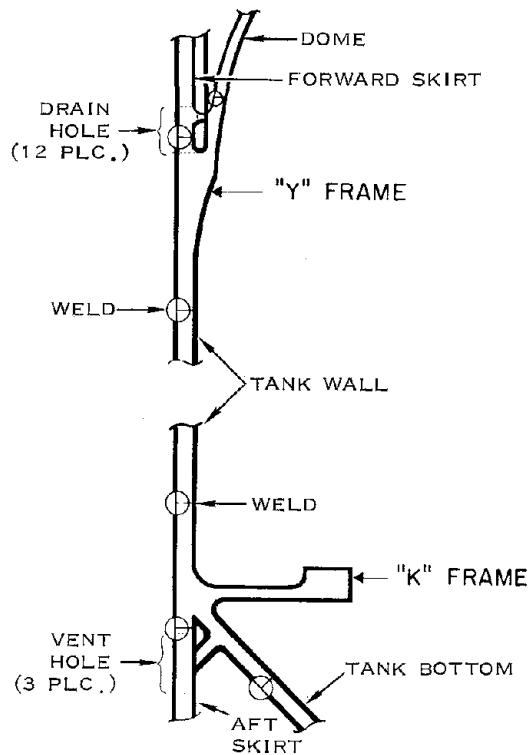


Figure 6-3. Tank K and Y frames.

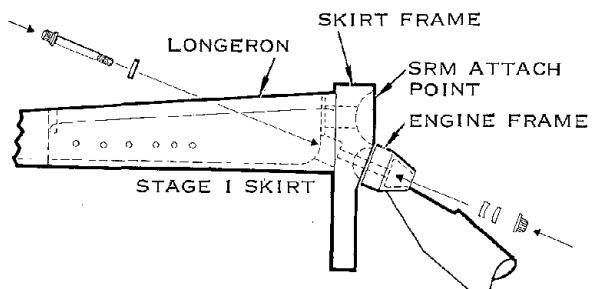


Figure 6-4. Engine attachment.

The dome assembly is welded to a special extruded Y frame (Figure 6-3). The frame is composed of three extrusions welded end-to-end and machined to final tolerances. The outer surface of the frame is vertical, making the bottom and one upper leg of the Y common members. The other upper leg of the Y provides the weld attachment point for the bottom edge of the dome. After fabrication of the dome is complete, it is welded to the tank barrel at the bottom leg of the Y frame.

The oxidizer feed line is not a fuel system component but is considered part of the fuel tank assembly. It is a 10-inch-diameter line welded to an inverted Y member at the lower end. The two outlet legs are seven inches in diameter. They have bolt-on flanges where prevalves are mounted to control flow to the vehicle engine oxidizer pumps. The Y line member has brackets attached to provide line support mounting on the engine truss unit at the bottom of the tank cone. The forward end of the 10-inch feed line attaches to the aft dome of the oxidizer tank.

The forward skirt assembly of the fuel tank is a riveted buildup structure with machined skins. The aft edge is welded to the fuel tank Y frame and the forward edge is bolted to the aft of the oxidizer tank assembly. Drain holes are provided at the base of the skirt assembly.

The skirt skin is formed by two bands. The lower band extends a little more than half the length of the skirt and is formed from three machined sheets. Short stringers are machined out of the skin stock at 177 places on the inner surface near the aft edge. These stringers support the skin in the area where the tank dome meets the Y frame.

An external conduit is attached to the tank assembly to bypass electrical wiring and tank pressurization lines around the tank from the aft fuel skirt to the forward fuel skirt.

Oxidizer Tank Assembly. The oxidizer tank assembly includes the oxidizer tank with forward and aft skirts. The tank structure is similar to the fuel tank in the wall and forward dome configuration. However, the tank bottom is an inverted dome rather than a cone.

The tank barrel section, 238.10 inches long, is assembled from four taper-machined panels similar to the larger fuel tank panels, welded edge-to-edge. The panels are machined with nine T stringers on each panel on the inner surface as integral parts of the units. Additional machining of the skins between stringers reduces the panel thickness. Twelve stretch-formed Z frames are attached to the top surface of the stringers, at equal intervals along the wall, to stabilize the tank walls.

The aft dome is similar to the fuel tank forward dome except for the oxidizer outlet center skin. The center skin is press-formed into a funnel shape to interface with the oxidizer feed line running through the fuel tank assembly.

The aft skirt, approximately 32 inches long, is a stringer-and-frame supported skin structure. The skin band that covers the skirt is assembled from three machined lengths spliced end-to-end by internal splice plates. The skin machining is similar to the machining on the fuel tank forward skirt skins. Thirty-six forged I stringers support the lower portion of the structure below the short machined stringers. A channel frame placed over the lower ends of the I stringers serves as a splice interface between the oxidizer tank assembly and fuel tank assembly. The frame is referred to as a tension splice frame. Seventy-two bolts (two per stringer) splice the stringers of the oxidizer tank aft skirt to the fuel tank forward skirt stringers by bolting through the tension splice interface frames. Three drain or vent holes are cut at the Y frame-to-skirt weld.

The oxidizer tank forward dome is similar in structure to the aft dome with the exception that the forward dome contains a manhole cover to provide entry into the tank if required. The dome is splice-welded to the tank barrel with a Y frame.

The forward skirt assembly of the oxidizer tank is similar in design to the aft skirt with the exception of oval-shaped blast ports at fourteen locations. The ports relieve the Stage II engine exhaust pressure in the skirt area. Larger openings in the interstage structure provide venting of the major portion of the gases. Thirty-six forged I stringers stiffen the skin. A T frame fastened to the skin and aft end of the I stringers at a point just above the machined stringers stabilizes the skin. A channel frame placed over the forward end of the I stringers completes the structure. This frame serves as an interface for the interstage structure and is referred to as the transportation splice frame. The interstage is attached to this interface by 72 bolts when the stages are assembled on the transporter. Four tapered shear pins are located on the frame to align the aft frame of the interstage structure with this frame to mate the bolt holes.

The inner surface of the skirt and the top of the oxidizer tank dome are coated with an ablative material called MMS-K421. The material, brick-red in color, is sprayed on to a thickness of 0.060 inch on all surfaces exposed to the Stage II engine exhaust. This delays heat rupturing of the dome or skirt during staging of the vehicle.

An external conduit bypasses electrical wiring and tank pressurization lines around the tank. This conduit is similar to the fuel tank conduit in design.

Stage I Engine Heat Shield. The engine heat shield (boattail) assembly protects the Stage I engine from the high temperatures radiated from the SRM nozzles.

The engine heat shield (Figure 6-5) is composed of an upper engine shroud, which attaches to the aft frame of Stage I and encloses the engine from the thrust chamber throats up; two lace-on jackets to cover the thrust chamber nozzles; and covers to close the open end of the chambers. The heat shield assembly is installed on the vehicle in the vertical integration building.

The upper engine heat shield assembly is an oval unit, constructed of aluminum material utilizing stringers, ring frames, and sheet cover. The base section of the shield is constructed of aluminum honeycomb approximately 1-1/2 inches thick. Openings in this section are provided for the engine nozzles to gimbal. The openings are sealed with a silicone rubber/glass cloth flexible boot. The entire structure is coated with ablative material and is attached to the Stage I aft frame structure.

The lace-on nozzle jackets are composed of Refrasil material with Inconel sheets for facing. Wire is used as lacing. A metal flange at the lower edge of the nozzle provides a latching interface for the nozzle closure unit and secures the lower edge of the nozzle Refrasil jackets.

The nozzle closures are constructed of aluminum honeycomb and are held in place by an aluminum band. In flight, the band is sheared by a primacord charge. Three aluminum snap-tabs then hold each closure in place until final release is accomplished by pressure buildup at Stage I engine start. Cork material covers the closures to protect them from the SRM radiant heat energy.

6.1.2 INTERSTAGE STRUCTURE. The interstage structure (Figure 6-1) is 117.5 inches long. The unit is a forward skirt extension of Stage I and serves as the launch interface for Stage II. Blast ports are provided in the lower area of the structure to relieve Stage II engine exhaust during staging.

Outrigger fittings are provided on the exterior surface at the forward end for attachment of the upper end of the solid rocket motors. These fittings are forged

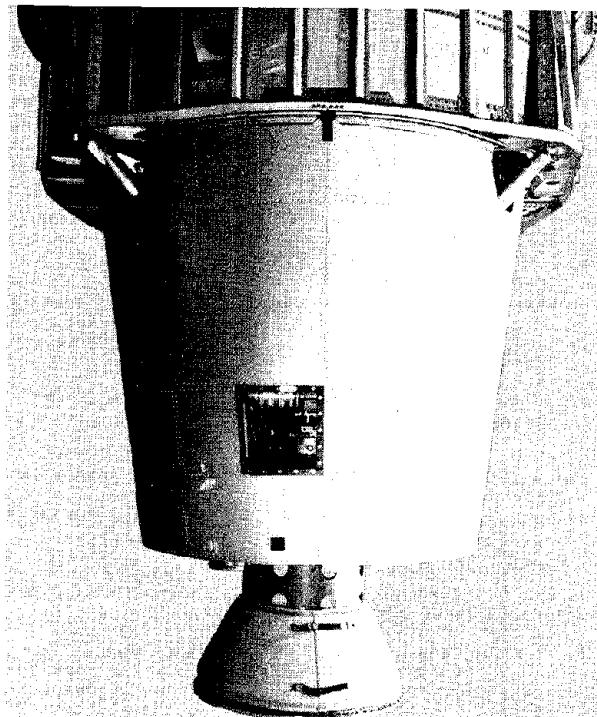


Figure 6-5. Stage I engine heat shield in manufacturing fit check.

aluminum units which provide a machined ball socket to receive the rounded end fitting of the four tubular SRM outrigger struts. The socket and outrigger strut ends are drilled at their center to receive a staging stud which holds the two in tension by gas-operated nuts. Access to the nuts is provided by access covers on the fittings and outrigger struts. Eight nuts (two per stud) are used. Shear tie fittings are also located at the forward end of the structure; they engage slots on the SRM to prevent horizontal shear motion between the SRM and the core vehicle.

The lower portion of the interstage structure contains blast ports, grouped into four areas. Each area encompasses seven stringers and reaches upward across two ring frames. The spacing of stringers and ring frames results in six larger windows at the bottom with four centered windows above. The stringers and frames in the blast ports are spray-coated with MMS-K421 ablative compound. The structure and internal surface of the skin between blast port areas are also coated with the material.

Gas-operated staging nut fittings are located on 12 stringers at the top of the structure. The fittings are forged members designed to contain the nut fragments and are provided with access covers located on the outer surface. Twenty-two taper shear pins extending from the top frame engage the aft frame of Stage II to absorb shear side loads and align the staging studs between Stages I and II.

6.1.3 STAGE II AIRFRAME. The Stage II airframe structure (Figure 6-1) consists of the fuel tank and its skirt sections, the oxidizer tank and its skirt sections, and the thermal barrier.

Fuel Tank Assembly. The fuel tank assembly consists of three major sections. These are the fuel tank, fuel tank aft skirt, and fuel tank forward skirt.

The fuel tank is an all-welded unit with a machined barrel section, a dome bottom with engine mount attachment, a dome top, and an oxidizer passage conduit.

The barrel section (Figure 6-6) has four skin panels which are machined from 0.660-inch stock. The inner surface of the panels is machined in a diagonal cross-grid pattern. The cross-grid machining forms 3-1/2-inch-square pockets.

The aft dome is similar in structure to the aft dome of the Stage I oxidizer tank. One major difference is of the off-center oxidizer outlet to mate with the off-center location of the Stage II engine fuel pump.

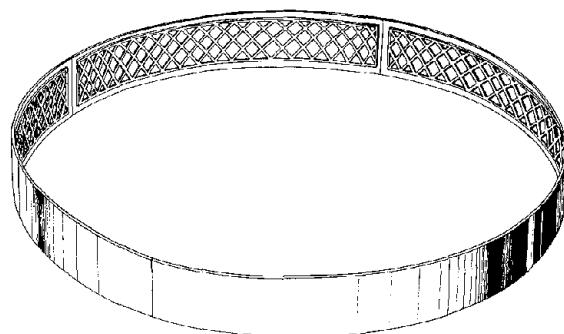


Figure 6-6. Stage II fuel tank barrel section.

An oxidizer feed line opening is located near the fuel outlet. The center dome skin is press-formed with the outlet neck welded on.

A stringer-reinforced truncated cone is welded to the aft dome to serve as an engine mount for the Stage II engine because Stage II has no external longerons for engine attachment. Openings for the fuel and oxidizer feed lines and level sensors are provided in the structure. A forged, machined ring frame is attached to the ends of the stringers at the apex of the cone. Twenty-four heli-coil-lined holes in the frame provide attachment for the engine frame cone.

The aft skirt assembly is 66.455 inches long, reinforced with 36 I stringers and four ring frames. Twelve stringers have gas-operated staging nut fittings at their lower ends, formed as part of the stringer to contain nut fragments during staging. The aft channel frame has 22 guide pin holes that mate with the 22 taper guide pins on the forward frame of the interstage structure. Twelve staging studs and 24 gas-operated nuts tie Stage II and the interstage structure together.

The Stage II engine roll nozzle is supported by a bracket attached to the aft edge of the skirt.

The fuel tank forward dome is similar in design to the Stage I forward fuel tank dome. The oxidizer feed line welded into the center of the dome is six inches in diameter. Tank pressurization, vent, and instrumentation ports are provided on the dome.

The fuel tank forward skirt is similar to the Stage I oxidizer tank aft skirt. The skirt is approximately 35 inches long and is reinforced by 36 stringers and two ring frames. A channel frame at the forward edge of the skirt interfaces with the oxidizer tank aft skirt and is referred to as the tension splice frame.

An external conduit located on the surface of the tank and skirts bypasses the wiring and tank pressurization lines from compartment-to-compartment around the tank area. The conduit is similar in construction to that on Stage I.

Oxidizer Tank Assembly. The oxidizer tank is similar in construction to the fuel tank assembly. The tank barrel is approximately 21 inches long and consists of four machined panels. All machining is on the inner surface and has the same diagonal cross-grid pattern as the fuel tank skins.

The aft dome assembly is similar to the aft dome of Stage I oxidizer tank. The center cap has a six-inch-diameter outlet.

The aft skirt assembly is similar in design to the aft skirts of the Stage II fuel tank. The skirt is approximately 55 inches long and is reinforced by 36 stringers and three ring frames. Attachment points are provided in the between-tanks structure for mounting of four retrorockets.

The forward dome of the oxidizer tank is similar in design to the forward dome of the Stage I oxidizer tank. The dome is welded to the tank wall using a Y frame.

The forward oxidizer tank skirt or forward Stage II skirt provides support for the Centaur interstage adapter. Flight control components, instrumentation, batteries, and other items are also housed in the skirt on a truss unit mounted on the inside of the compartment.

The skirt structure is 76-1/2 inches long from tank dome weld splice to forward frame. The skirt skeletal structure consists of 36 I stringers and three ring frames. Short ring segments are attached to the skin and stringers, parallel to the ring frames to stiffen the skin structure. These segments are located at five levels.

Umbilical disconnects for air-conditioning, electrical services and oxidizer tank vent are also located in this skin band. Rectangular, flush-mounted telemetry and command control antennas are located on the skin band.

The forward skirt channel frame provides four alignment pins located 90 degrees apart to mate with the aft edge of the Centaur interstage adapter. Two 3/8-inch holes through the frame and in the stringer ends of all 36 stringers provide for bolt-on attachment of the adapter.

The skin design incorporates a machine-tapered skin band at the lower edge of the skirt where the skins are welded to the oxidizer tank Y frame.

All access doors on Stages I and II used for component removal or checkout are backed up by removable stringer sections.

Thermal Barrier. A structural thermal barrier (Figure 6-7) is attached to the forward end of the Stage II airframe.

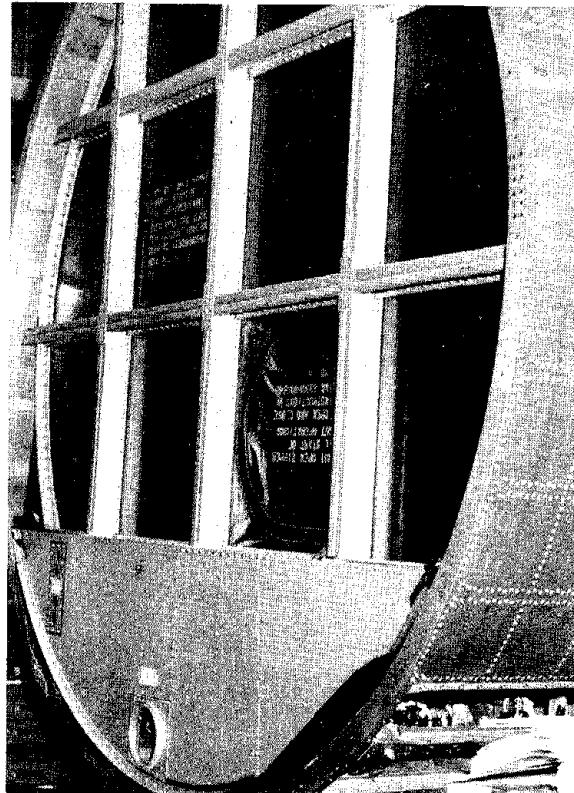


Figure 6-7. Thermal barrier.

The barrier is constructed of aluminum stringers and sheets and thin Dacron-polyester material. The purpose of the barrier is to prevent the environment of the Centaur thrust compartment from adversely affecting the Titan electronic components and the Stage II ullage gas and propellant and, conversely, the barrier prevents the Titan compartment environment from adversely affecting the Centaur thrust compartment during prelaunch and flight. The barrier is designed to withstand a one psi differential pressure between the Titan and the Centaur.

6.1.4 AIRFRAME ATTACHMENTS. Airframe attachments include the probes, umbilical connections, transducers, and pressure switches. Figures 6-8 and 6-9 illustrate the location of these various components on the Stage I and Stage II airframe structures.

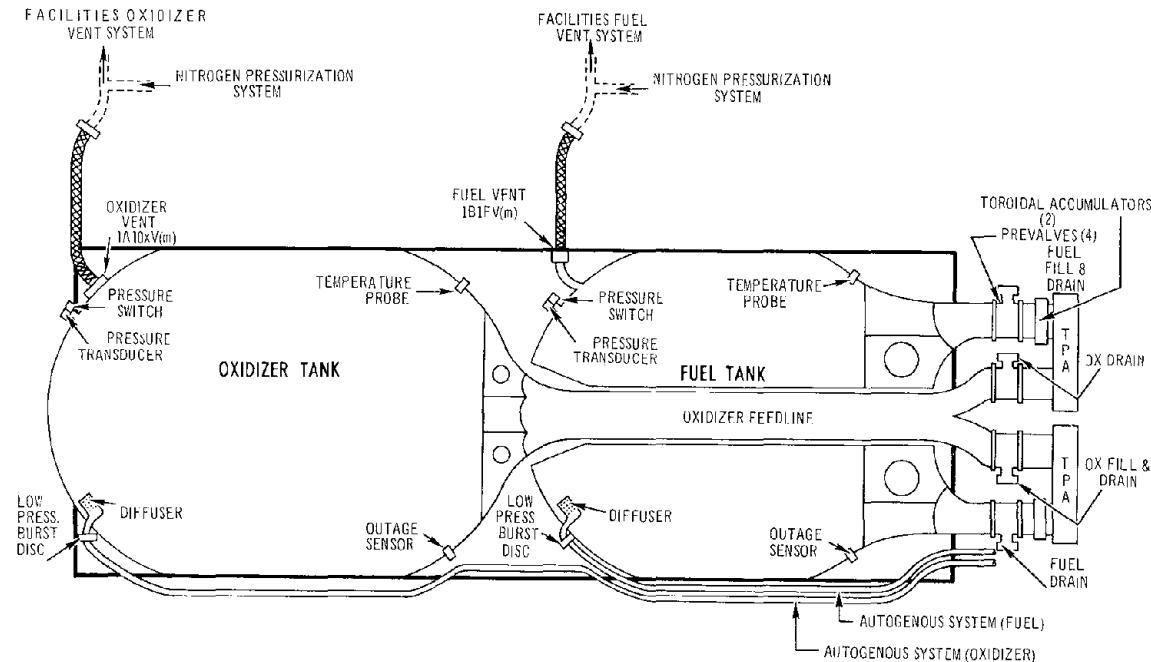


Figure 6-8. Stage I airframe attachments.

Pressure switches are located on the forward domes of all tanks (Figure 6-10). These switches provide an electrical output and resultant indication that tank pressures are above a predetermined high pressure setting. These settings are well below the tank burst pressure. Pressure switches provide indication during vehicle ground operations only. Pressure switch settings are given in Table 6-1.

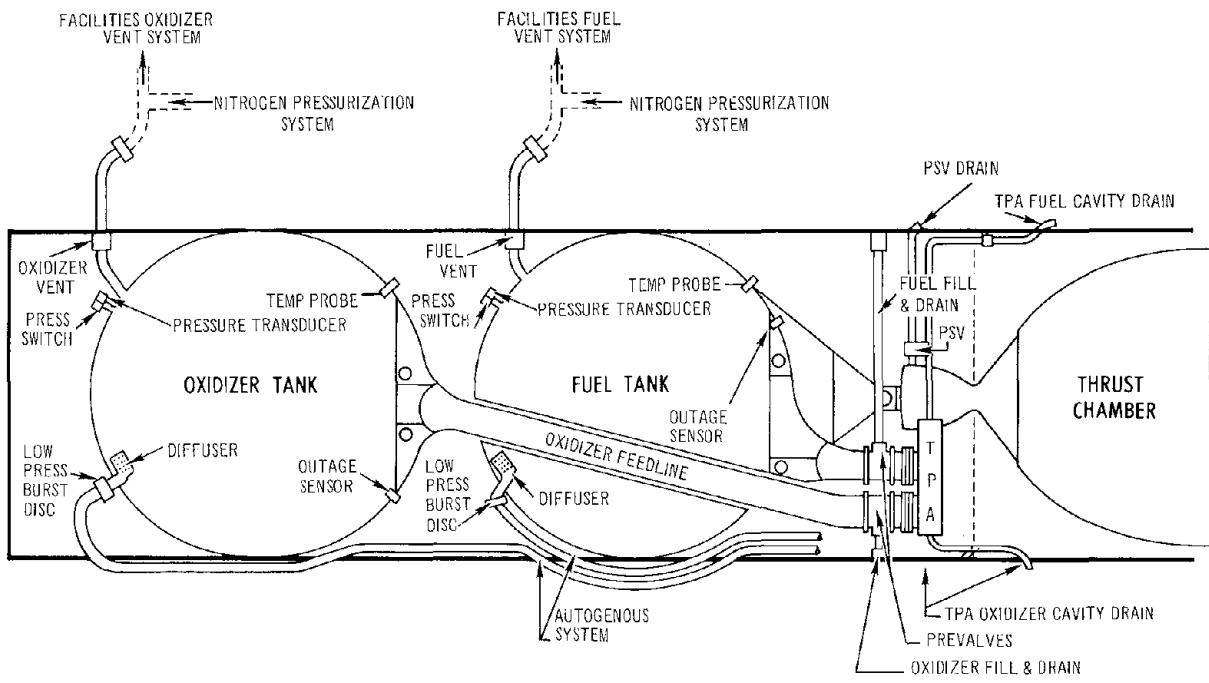


Figure 6-9. Stage II airframe attachments.

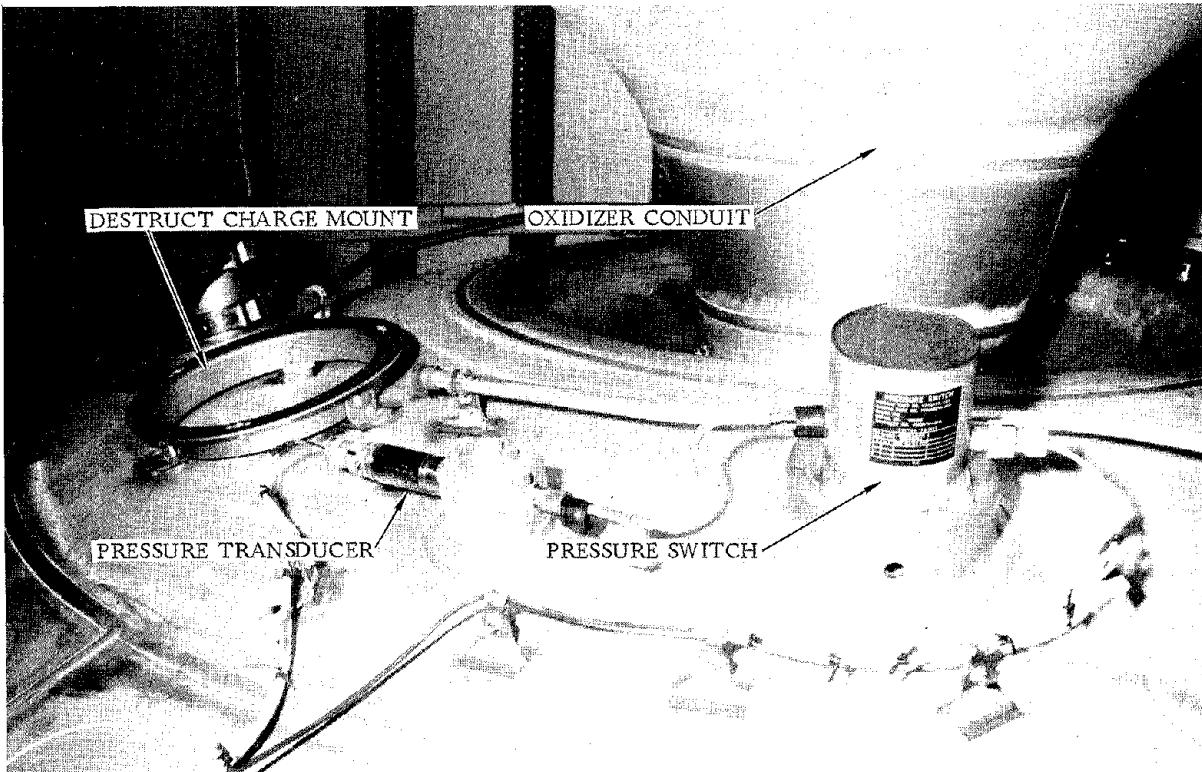


Figure 6-10. Stage I fuel tank dome.

Table 6-1. Pressure switch settings.

Tank	Pressure Switch Setting	Burst Pressure
Stage I		
	Oxidizer	56.4 psia
	Fuel	41.4 psia
Stage II		
	Oxidizer	56.4 psia
	Fuel	55.4 psia

Pressure transducers are used in conjunction with the pressure switches on all tank domes. These sensors provide an analog indication of varying tank pressures through the instrumentation system. They function during ground operation and inflight.

The manually operated vent lines permit venting of gases during propellant loading, and are used to apply tank flight pressure, using GN₂, to the vehicle tanks.

Temperature probes provide an analog readout of propellant temperatures during ground operations.

The low-pressure burst disc and diffuser are associated with the pressurant lines from the engines which provide inflight tank pressure.

The outage sensors provide an instrumentation readout during flight to determine inflight propellant mixture ratios in the engine systems.

Manually operated fluid umbilical connections are located on skirt sections to provide propellant and hydraulic service to the vehicle. All electrical umbilical connections provide vehicle power and checkout capabilities and are disconnected at liftoff. One air conditioning connection maintains the temperature requirements in the forward compartment of Stage II and is disconnected at liftoff.

Four solid propellant retrorockets are located between tanks on Stage II to provide the necessary force to retard the forward motion of Stage II at Centaur separation.

One truss assembly is located in the Stage II forward airframe compartment. Located on this truss are the major "black box" components of the various electrical/electronic systems of the vehicle.

6.2 PROPULSION SYSTEMS

ELEMENTS:

- | ELEMENTS: | Subsection |
|-------------------------------|------------|
| • Solid Rocket Motor | 6.2.1 |
| • Propellant Feed System | 6.2.2 |
| • Stage I Engine (LR87AJ-11) | 6.2.3 |
| • Stage II Engine (LR91AJ-11) | 6.2.4 |

FUNCTIONS:

- Provide Thrust for Propelling Vehicle
- Provide Vehicle Attitude Control

6.2.1 SOLID ROCKET MOTOR. Initial thrust for the Titan IIIE vehicle is provided by two government-furnished United Technology Center solid rocket motors. The solid rocket motor stage is sometimes referred to as Stage 0. This stage consists of two identical, segmented, solid propellant rocket motors. These 120-inch-diameter motors are mounted 180 degrees apart on the Titan liquid-propellant core vehicle. Each motor consists of a forward closure, an aft closure and five identical, interchangeable segments (Figure 6-11). Other components include a single six-degree canted nozzle, an igniter, fore and aft solid propellant staging rockets and a liquid injection thrust vector control (TVC) system. The TVC injectant, nitrogen tetroxide (N_2O_4), is carried in a tank mounted on the side of the motor and is pressure-fed into the nozzle exit section by nitrogen gas.

The motor case (segments and closures) is constructed of D6aC steel, heat-treated to an ultimate strength of 195,000 psi. Each joint is a pin-and-clevis type held together by 240 cylindrical pins. During assembly of the motor, the pins are inserted by hand (rather than force-fitted) and held in place thereafter by a retaining strap. A gas pressure seal between segments (and closures) is provided by an O-ring.

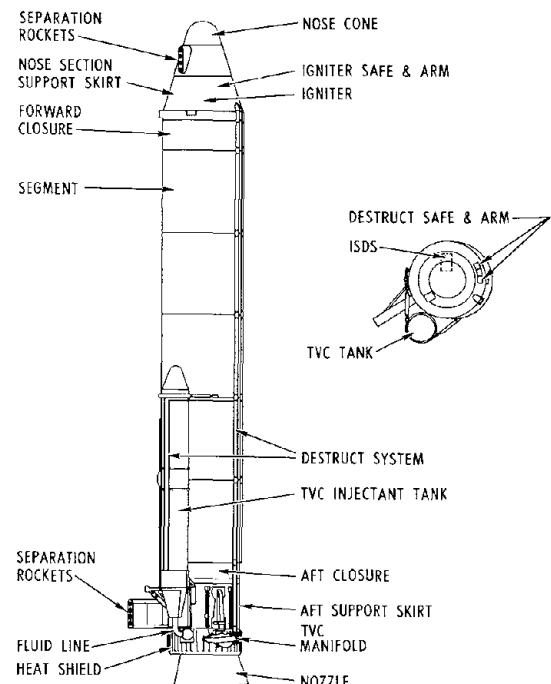


Figure 6-11. Solid rocket motor components.

Each center segment (Figure 6-12) contains approximately 72,400 pounds of propellant which contains powdered aluminum fuel and ammonium perchlorate oxidizer with polybutadiene acrylic acid acrylonitrile (PBAN) as a binder. The case-bonded propellant grain has a circular port which tapers 10 inches through the 10-foot length of the segment. The forward end has a smaller port. The purpose of this taper is to provide a 10-second controlled tail-off at the end of web-action time. The forward end of the segment is inhibited from burning by a rubber restrictor bonded to the propellant surface. Silica-filled, butadiene acrylonitrile rubber insulation protects the motor case from combustion gas during motor operation. The insulation is thickest in the segment joint areas where there is no unburned propellant to protect the case walls.

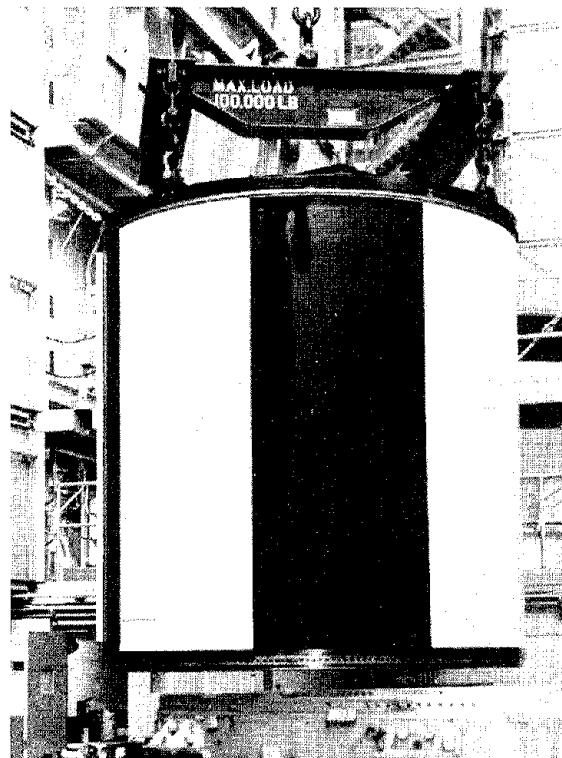


Figure 6-12. SRM center segment.

The closures contain the same type of propellant as the segments, and the forward closure has mounting provisions for the solid propellant igniter. Instead of the cylindrical grain shape of the segments, the forward closure (Figure 6-13) has an eight-point star internal burning grain configuration. The forward closure is 95 inches long and contains 37,000 pounds of propellant. The aft closure (Figure 6-14) contains approximately 20,300 pounds of propellant in a straight cylindrical bore configuration and projects 64 inches from the segment joint to a 57-inch diameter boss for nozzle attachment.

The propellant burns along the entire central port of the SRM and also on the aft of each segment between segments. The closures also burn on their ends. Three inches of clearance are left between the segments to permit this burning. The igniter burns for approximately one second to fill the grain bore with hot gas to ignite the motor.

The SRM has a regressive thrust-time curve (Figure 6-15) produced in part by the star configuration of the propellant grain in the forward closure of the motor. During the early phases of burning, this portion contributes much of the gas flow necessary to produce the high initial peak in the thrust-time curve.

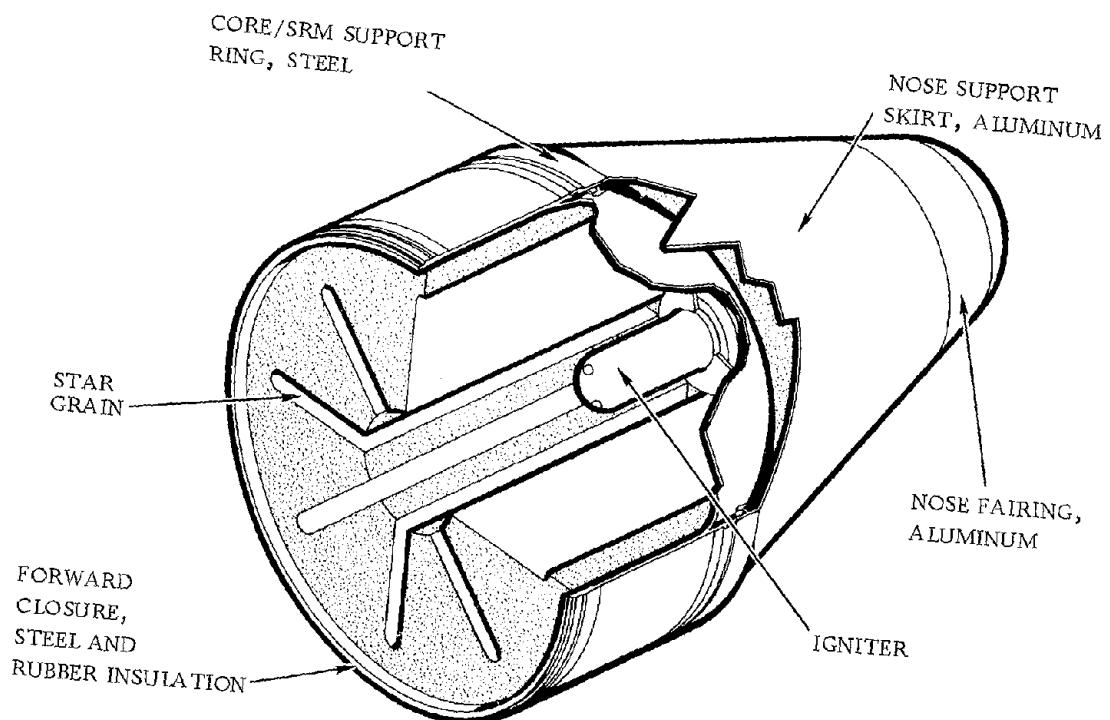


Figure 6-13. SRM forward closure.

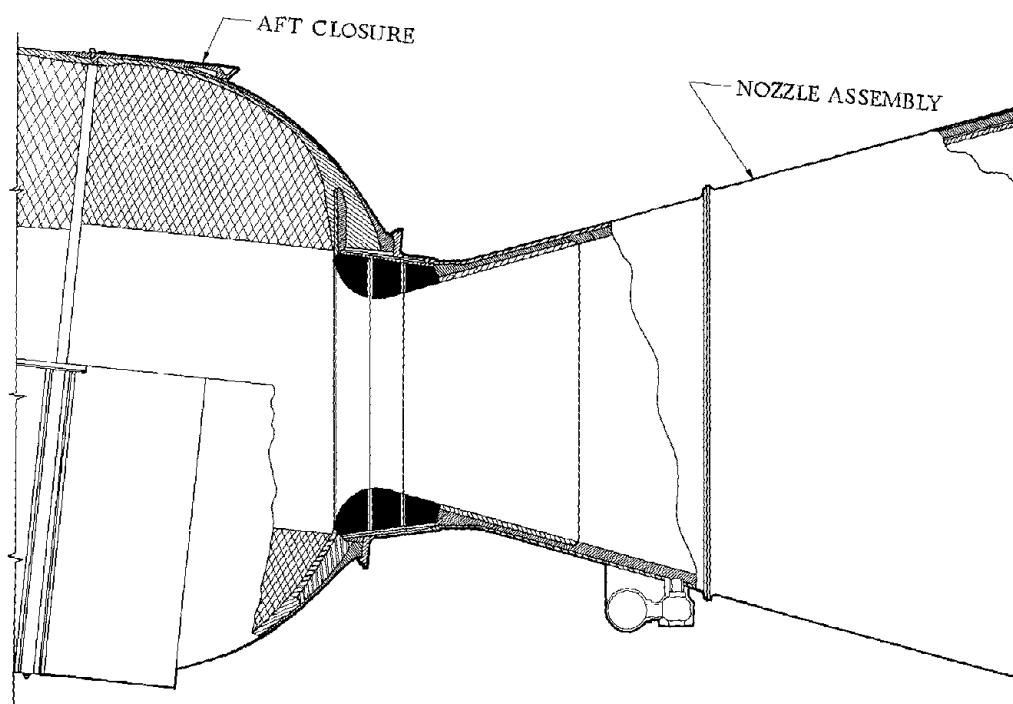


Figure 6-14. SRM aft closure assembly.

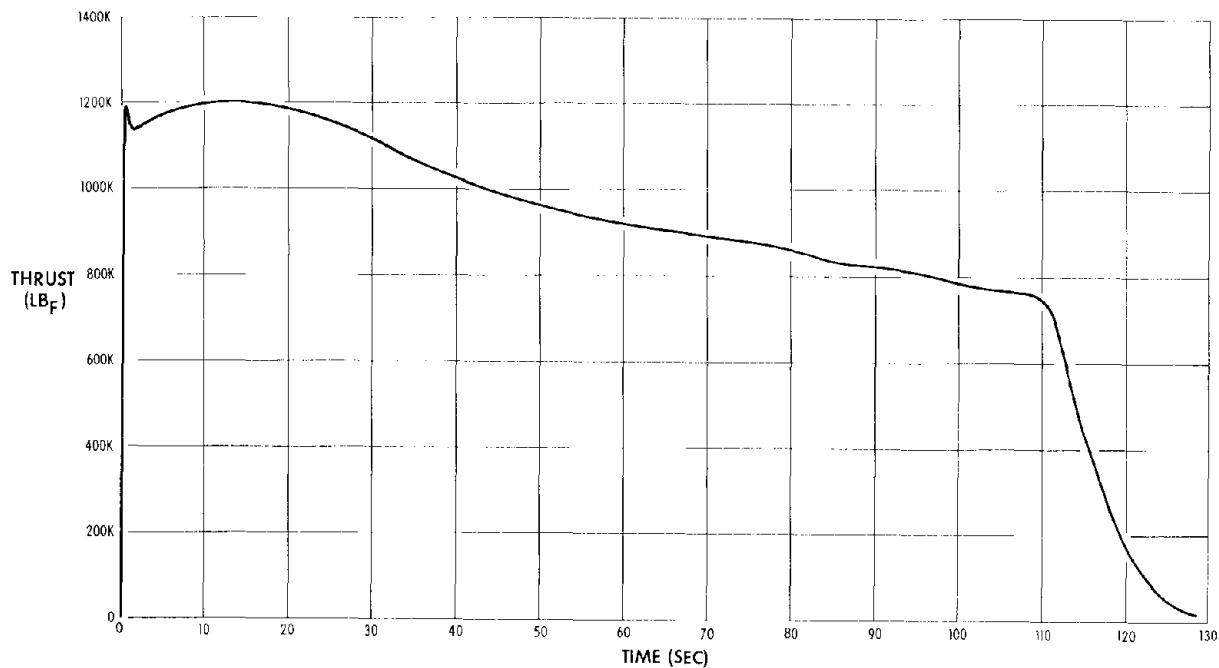


Figure 6-15. SRM thrust-time curve.

The SRM nozzle (Figure 6-16) consists of a throat section and a two-piece exit cone assembly. High-density graphite rings backed by a steel support shell and silica insulation are bonded in a steel housing to make up the nozzle throat section. The nozzle middle section consists of graphite and silica phenolic liners bonded to a steel outer shell. This section contains the thrust vector control injection ports. The exit section is an extension of the silica phenolic liner of the middle section except that its structural shell is aluminum honeycomb sandwiched between steel for lighter weight. The three sections are bolted together, forming an assembly approximately 11.5 feet long. Nozzle expansion ratio is 8:1, and the half-angle is 17 degrees.

The SRM liquid injection TVC system is capable of a vector angle of 5 degrees and a maximum side force of 110,000 pounds (each motor). The system has 24 valves located at an area ratio of 3.5:1. These valves are uniformly spaced on the periphery of the nozzle and operate in groups of 6 per quadrant. The valves modulate from zero position to full open and are controlled by electrical signals (0-10 volts) from the vehicle core. Mission capability may be retained with only 5 of the 6 valves in each quadrant operating. Nitrogen tetroxide is supplied to the valves through a toroidal manifold mounted above them. The injectant tank is approximately 22 feet long, 3.5 feet in diameter, and carries initially 8424 pounds of N_2O_4 and 636 pounds of gaseous nitrogen. Initial pressure is 1030 psia, and the system operates down to a minimum of 450 psia tank pressure.

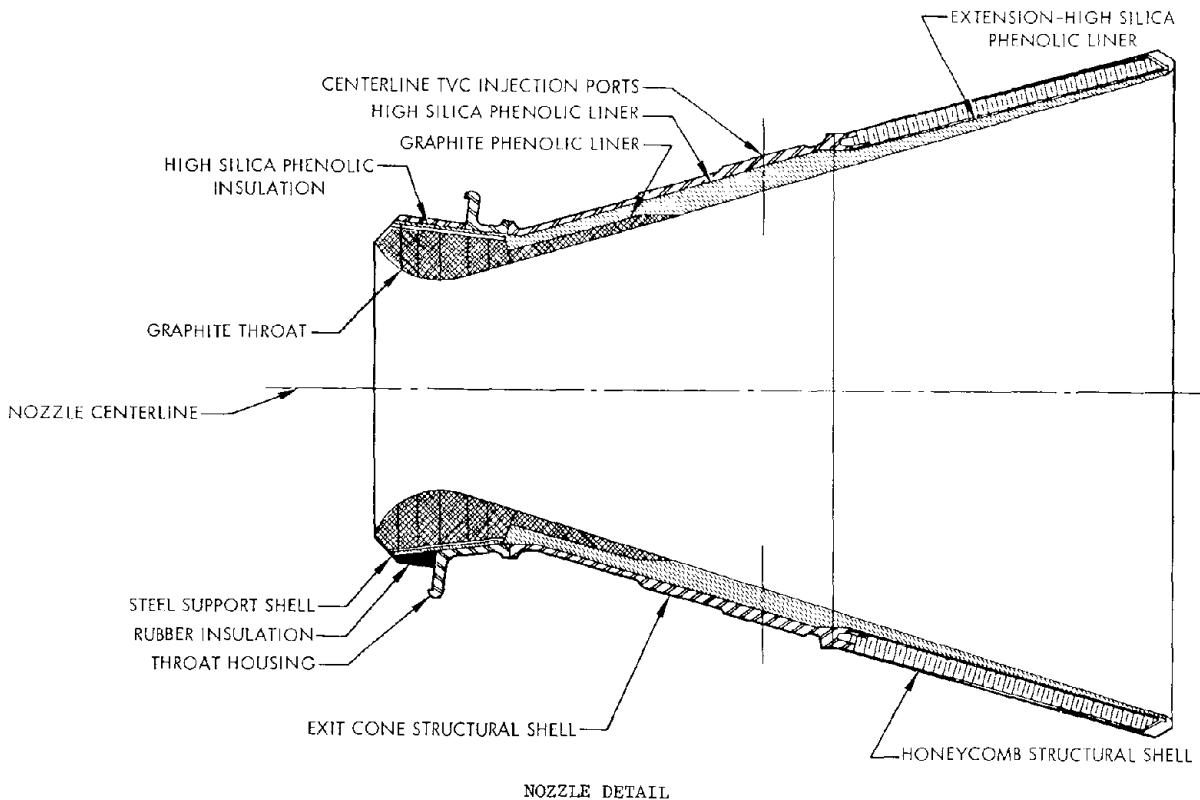


Figure 6-16. SRM nozzle detail.

Figure 6-17 illustrates a fully assembled solid rocket motor in the solid motor assembly building (SMAB).

6.2.2 PROPELLANT FEED SYSTEM. The propellant feed system for Stage I and Stage II is illustrated in Figures 6-8 and 6-9. Both fuel and oxidizer lines terminate at a set of electrically operated, reclosable prevalves (Figure 6-18). Propellants are loaded aboard the vehicle tanks through manually operated disconnects above the prevalves. The Stage I engine feed system also includes a set of toroidal accumulators in the fuel feed lines. These accumulators are designed to dampen pressure surges to the turbopump assembly and thus reduce the "pogo" effect between the engine and the airframe. Propellant tanks must be pressurized to maintain sufficient inlet pressure to the engine pumps for proper pump operation. The tanks are pressurized with gaseous nitrogen prior to engine start. Pressurizing gas is supplied to the tanks by the engine autogenous (self-generating) system at a controlled rate to make up for the removal of propellant from the tanks. The fuel tank is pressurized by gas from the turbine inlet, and the oxidizer tank is pressurized by oxidizer which has been heated to a gaseous state by a heat exchanger in the turbine exhaust stack.

Propellants. Stages I and II of the Titan IIIIE vehicle use the same storable, hypergolic liquid propellants. The fuel is Aerozine 50, a 50-50 mixture of hydrazine and unsymmetrical dimethyl hydrazine (UDMH), and the oxidizer is nitrogen tetroxide.

The use of propellants storable at ambient temperature and pressure, eliminates holds and delays inherent in handling cryogenic propellants. This feature gives Titan the demonstrated capability of meeting critical launch windows within two seconds of any preestablished time. Hypercolic action, spontaneous ignition upon contact, eliminates the need for an ignition system and related checkout and support equipment.

The physical properties of the propellants are:

	Aerozine 50	Nitrogen Tetroxide
Freezing Point, °F	19	12
Boiling Point, °F	158	70
Physical Description	Clear, colorless liquid	Red-brown liquid
Critical Temperature, °F	635	317
Critical Pressure, psia	1696	1469

6.2.3 STAGE I ENGINE (LR87AJ-11).

Thrust for vehicle flight during the period of Stage I powered flight is provided by one government-furnished Aerojet liquid rocket engine. Figure 6-19 shows the LR87AJ-11 rocket engine to be a pair of identical engines attached to a single frame and mounted on the vehicle. These individual engines, designated Subassembly 1, and Subassembly 2, are designed to operate simultaneously under a single control system.

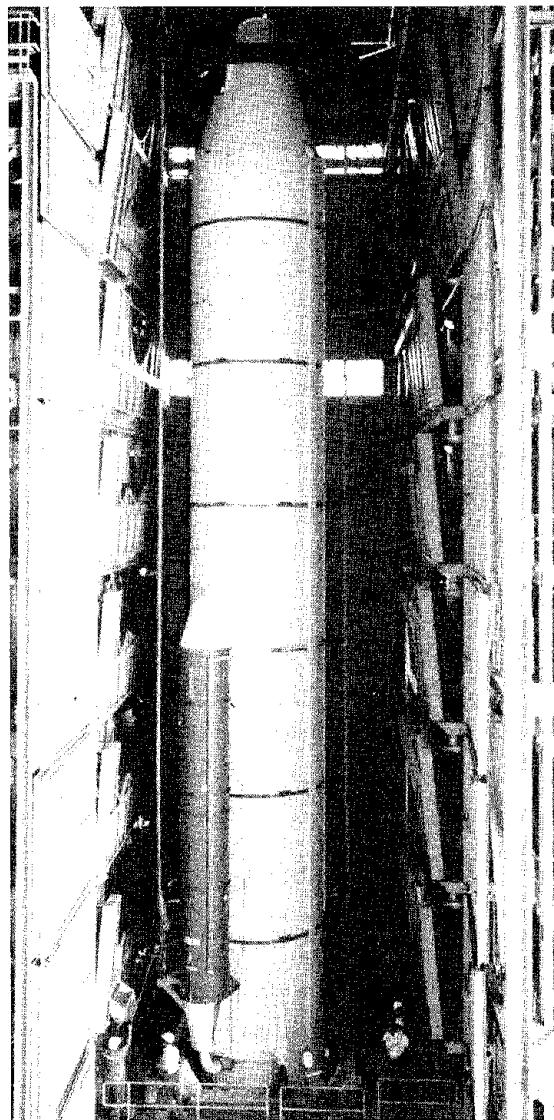


Figure 6-17. Assembled solid rocket motor.

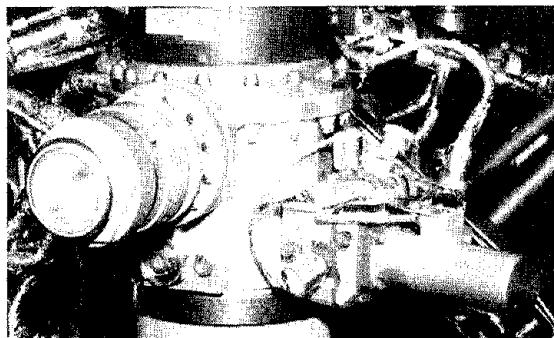


Figure 6-18. Stage I propellant prevalve.

Engine reliability is augmented through functional simplicity of design. The engine is hydraulically balanced and requires no thrust controls. It is preset to operate at a certain level (i.e., consume propellant at a fixed rate) by the use of orifices. Balance orifices in the propellant discharge lines and cavitating venturis in the gas generator bootstrap lines determine the steady-state level. The propellant flow rate established by the discharge line orifices is a function of both upstream and downstream pressures. The cavitating venturis establish a flow rate that is sensitive only to upstream pressure, maintaining a constant flow rate over a wide range of downstream pressures. The control of propellant flow rate to the gas generator results in a stabilized turbine speed.

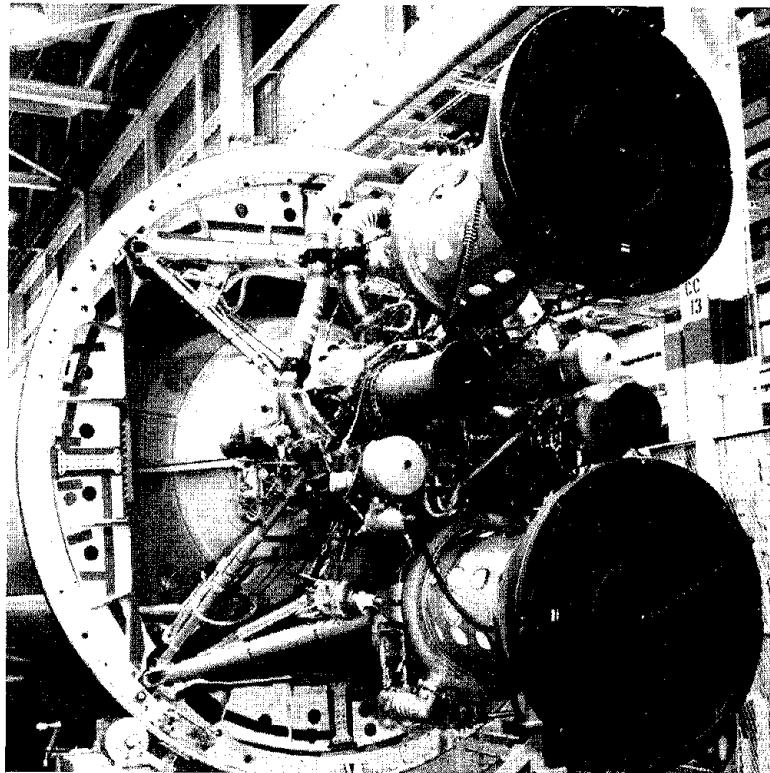


Figure 6-19. Stage I engine assembly (LR87AJ-11).

The Stage I engine consists of the following major components and subsystems: pump suction (inlet) lines, turbopump assemblies, pump discharge lines, thrust chamber valves, gas generator systems, thrust chambers, autogenous pressurization system, control and instrumentation harnesses, and engine frame.

The suction lines duct the fuel and oxidizer from the propellant tank lines to the turbopump assemblies, each of which is driven by a turbine rated at over 5000 horsepower. The fluid pressure is increased through the pumps by over 1000 psi to force the propellant through the discharge lines into the thrust chamber. Thrust chamber valves control engine start and shutdown. Gas generators are operated by propellant from the discharge lines to drive the turbines which maintain propellant flow. Combustion in the thrust chambers produces gas at pressures over 800 psia and temperatures over 5000°F. This gas is expanded through a convergent-divergent (DeLaval) nozzle and exhausted at supersonic velocity to produce thrust. Thrust vector control

(pitch, yaw, and roll) is achieved by pivoting the thrust chambers independently on gimbal bearing mounts. The gimbal action of the thrust chambers is provided by hydraulic actuators which operate in response to signals from the launch vehicle flight controls system.

An electrical control harness carries signals to start and shut down the engine, and an instrumentation harness carries information from the various transducers on the engine to the vehicle for transmission of engine performance data to ground stations.

A summary of engine performance data is given below:

Altitude Thrust, lb	520,000
Altitude Specific Impulse, sec	301.1
Total Flow Rate, lb/sec	1,727
Oxidizer Flow Rate, lb/sec	1,135
Fuel Flow Rate, lb/sec	592
Mixture Ratio	1.915
Operating Cycle, sec	165
Expansion Ratio	15:1

Operation. Prior to intended use of the LR87AJ-11 rocket engine, prevalves in the propellant tank lines immediately above the engine interface prevent propellant from entering the engine system. This allows loading of propellant tanks a long time before launch while protecting the engine systems from contamination by propellants. During the countdown, an arming signal is supplied to the launch vehicle. This signal accomplishes the following: (1) the prevalves are opened allowing fuel and oxidizer to fill the engine, and (2) the engine starting electrical circuits are readied for receipt of the firing signal.

Fill and Bleed. Opening of the prevalves, which are electrically operated, reclosable type valves, places the engine in the fill and bleed condition as shown by Figure 6-20. Both fuel and oxidizer fill the engine above the thrust chamber valves due to the static pressure of the propellants in the tanks above the engine. Air entrapped in the oxidizer lines travels through 3/8-inch flex lines, connected on each subassembly from the discharge line near the pump outlet flange (high point) to the

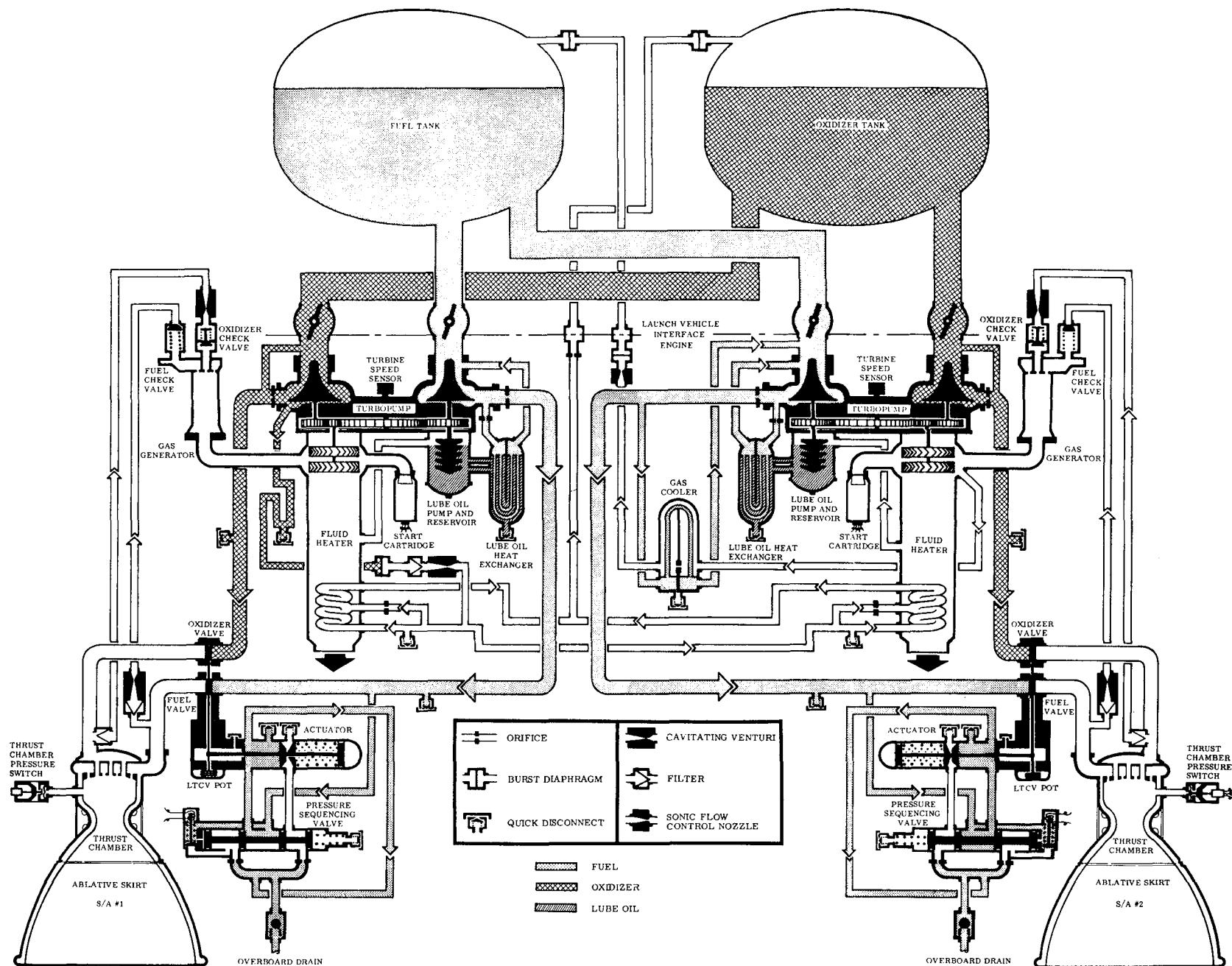


Figure 6-20. LR87AJ-11 propellant fill and bleed schematic.

suction line, up into the oxidizer tank. Air removal from the fuel lines must be complete, as fuel hydraulic pressure actuates the thrust chamber valves at engine start. Smooth operation of the valves depends upon a hydraulically hard system. Therefore, the fuel system is bled full rather than simply allowing entrapped air to bubble out under gravitational forces as in the case of the oxidizer lines.

The fuel-operated valve actuation system consists of a rod and piston mechanically linked to the thrust chamber valves (TCVs), held closed by springs and opened by fuel pressure. A pressure sequencing valve (PSV), also held closed by a spring and opened by fuel pressure, acts as a pilot valve to the TCV actuator. Clearly, all of the cavities and interconnecting lines of this system must be free of air during valve actuation. Fuel bleed is accomplished by allowing fuel to flow through a 1/2-inch flex line from the high point on the discharge line at the TCV and discharge line connecting flanges to the PSV inlet port. While in the bleed position, the PSV diverts the fuel into and through the closing side of the TCV actuator, through a 1/4-inch stainless steel vent line to an overboard manifold mounted on the PSV, and out an overboard drain line through a check valve which serves only to protect the PSV from contamination. A bleed orifice, located in the drain line and PSV manifold connection, controls the bleed rate to approximately 1200 cubic centimeters per minute per sub-assembly. As long as the engine remains in the fill and bleed condition, fuel is bled overboard in the manner and at the rate described. Minimum bleed duration prior to engine start is 30 seconds.

Start Sequence. After completing the bleed cycle, the engine is ready for operation. The start signal (Figure 6-21), fire switch 1 (FS-1), applies 28V d-c to the initiator charges of the solid propellant start cartridges mounted on the turbine inlet manifold of each subassembly and initiates separation of the exit closure from the ablative skirt. The start cartridge solid propellant ignites and supplies gas to the turbines causing them to accelerate. The turbine shaft of each subassembly is connected through a gear train to the fuel and oxidizer pump so that pump operation also begins. Since the thrust chamber valves are closed, no propellant flows, and pump acceleration produces only an increasing pressure in the discharge lines and valve actuation system.

When fuel discharge pressure reaches approximately 300 psia (295 to 325 psia nominal), the pressure on the opening end of the PSV spool produces a force which exceeds the spring force on the PSV spool closing end, causing the spool to shuttle from the bleed position to the operation position. This occurs approximately 0.25 second from FS-1. Relocation of the PSV spool to the operation position halts flow of fuel to the closing side of the TCV actuator, terminating fuel bleed, and allows fuel to enter the actuator opening side. The fuel pressure, increasing beyond the 300 psia PSV actuation pressure, is immediately sufficient to operate the TCV actuator, initiating opening of the thrust chamber fuel valves. The fuel and oxidizer

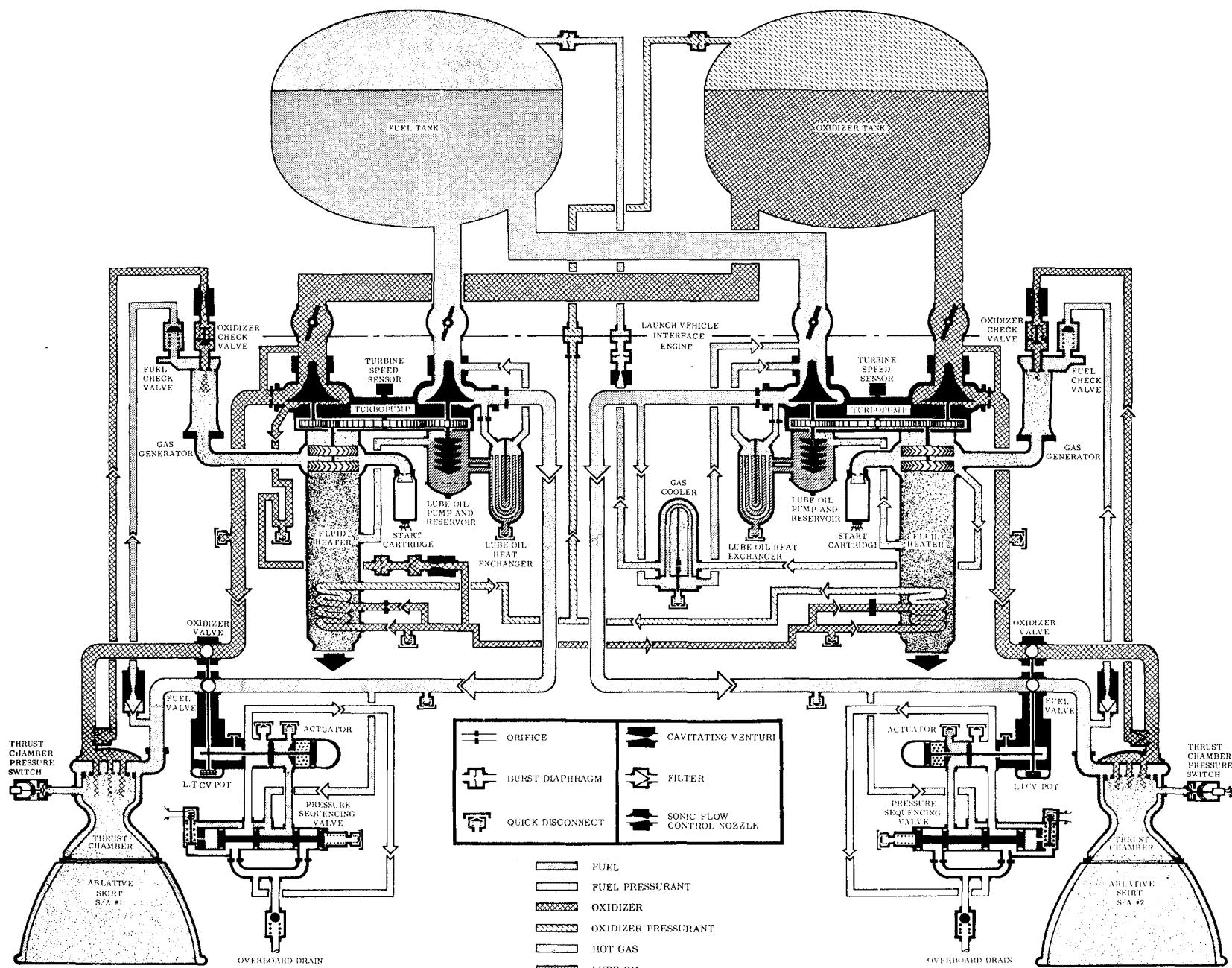


Figure 6-21. LR87AJ-11 engine start sequence.

valves are mechanically linked so that both move simultaneously. The rate of motion of the TCV actuator piston is controlled by an orifice located in the overboard drain manifold to PSV housing connection nearest the PSV opening end. This opening orifice controls the rate at which bleed fuel can be expelled from the TCV actuator closing side. Valve opening begins approximately 0.3 second after FS-1, and at approximately 1.1 seconds after FS-1, the valves are fully opened.

Propellants begin to flow to the thrust chamber at the time the valves begin to open. Oxidizer flows directly into the injector dome, filling the oxidizer injector manifold through orifices into the combustion chamber. Fuel is used to regeneratively cool the combustion chamber, and must fill a toroidal manifold and flow through the stainless steel cooling tubes that make up the chamber walls before reaching the injector fuel orifices and entering the combustion zone. The term "regeneratively cooled" is derived from the fact that heat removed from the system by the coolant is subsequently returned to the combustion process resulting in no net loss of energy. The larger volume that the fuel must fill before reaching the injector orifices results in an oxidizer "lead" into the combustion zone. This slight oxidizer lead (0.25 to 0.30 second) has been shown to provide a characteristically smoother start. Initial pressurization of the chamber during oxidizer lead ejects the skirt exit closures.

For the short period of time that the gas generator and start cartridge are operating concurrently, the engine will overshoot its nominal level, but this is expected and controlled.

Propellant Tank Pressurization. The Stage I fuel tank is pressurized by diverting a portion of the gas generator output of Subassembly 2 from the turbine inlet manifold to the tank. This gas must be cooled to be safely used, and cooling is accomplished by passing it through a heat exchanger with fuel. The fuel flows from the discharge line through the gas cooler back to the suction side of the pump as shown in Figure 6-21. Inside the gas cooler, a bypass orifice diverts a portion of the hot gas around rather than through the cooling tubes so that the bypass orifice diameter determines the fraction of total flow that traverses the entire heat exchanger. In this way, a tank pressurant temperature control is provided without affecting engine balance by altering the total amount of gas flow taken from the turbine. A sonic nozzle in the autogenous gas line near the engine-to-vehicle interface maintains a flow rate that is insensitive to tank pressure, and a 300 psid burst diaphragm prevents gas flow to the tank until gas generator operation begins.

The Stage I oxidizer tank is pressurized by heating oxidizer to the gaseous state and ducting it to the tank. Oxidizer is piped from the pump discharge flange to a heat exchanger (fluid heater) located in the turbine exhaust stacks. A cavitating venturi, located at the fluid-heater inlet, maintains a constant flow rate insensitive to downstream pressure, and a burst diaphragm prevents oxidizer autogenous flow until the

discharge pressure reaches approximately 300 psia. A loop in the inlet line traps air to ensure pneumatic operation of the burst diaphragm. An orificed bypass line on the fluid heater provides temperature control of the tank pressurant without affecting engine balance in the same manner as the bypass orifice in the fuel autogenous gas cooler. A back-pressure orifice, located at the engine-to-vehicle interface, provides sufficient residence time of the oxidizer in the fluid heater to achieve the proper gas temperature of the pressurant.

Engine Shutdown. Engine shutdown is initiated during a flight when either fuel or oxidizer is depleted. At this time combustion chamber pressure in one or both of the subassemblies decreases. Thrust chamber pressure switches (TCPS), located on each subassembly, sense the pressure decay and cause a 28V d-c signal, Fire Switch 2 (FS-2), to be sent to the solenoid of the PSV override valve (PSVOR). Moving the PSVOR to the shutdown position prevents fuel discharge pressure from entering the opening end of the PSV and vents this cavity. The PSV spring then shuttles the main spool to the fill and bleed position, which blocks fuel to the TCV actuator opening side and allows the actuator spring and any residual fuel discharge pressure to move the TCV actuator toward the closed position, venting the opening fuel pressure. The rate at which the fuel is vented is controlled by an orifice located in the PSV overboard-manifold-to-housing connection nearest the spring end of the PSV. Hence, this PSV closing orifice determines the closing time of the thrust chamber valves. The time from FS-2 to thrust termination is approximately 0.5 second.

6.2.4 STAGE II ENGINE (LR91AJ-11). Except for being somewhat smaller, the Stage II engine (Figure 6-22) is similar in construction and operation to a single subassembly of the Stage I engine. This engine is designed to produce approximately 101,000 pounds of thrust at altitude. Pitch and yaw thrust vector control is achieved by pivoting the thrust chamber on a gimbal bearing mount and roll control is provided by ducting turbine exhaust through a swiveled roll control nozzle.

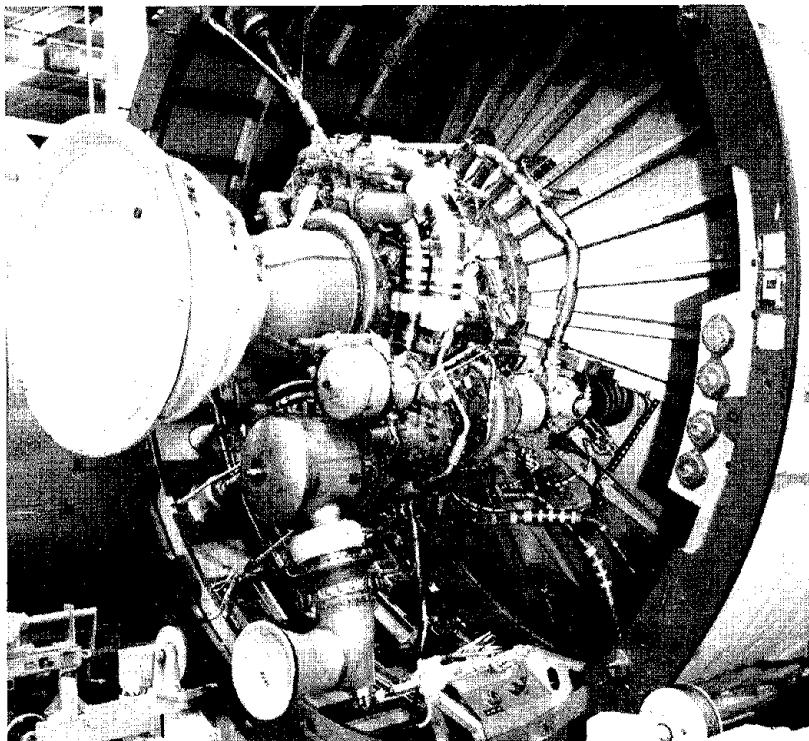


Figure 6-22. Stage II engine (LR91AJ-11).

A summary of engine performance data is given below:

Vacuum Thrust, lb (including Roll Nozzle)	101,000
Vacuum Specific Impulse, sec (including Roll Nozzle)	318.7
Total Flow Rate, lb/sec	317
Oxidizer Flow Rate, lb/sec	203
Fuel Flow Rate, lb/sec	114
Mixture Ratio	1.79
Operating Cycle, sec	225
Expansion Ratio	49.2:1

Operation. As indicated in Figures 6-23 and 6-24, the Stage II engine operates identically to one subassembly of the Stage I engine. Bleed-in of the Stage II engine occurs at the same time as Stage I, since all prevalves in the vehicle are opened simultaneously. Stage II engine shutdown occurs as a result of propellant depletion in the Stage II tanks. When an acceleration switch in the flight control system senses an acceleration decay to 1 g, it issues the Stage II shutdown signal to the PSVOR. This flight controls system signal is backed up by an identical signal from the Centaur guidance system.

6.3 TITAN IIIE THRUST VECTOR CONTROL SYSTEMS

ELEMENTS:	Subsection
• Solid Rocket Motor Thrust Vector Control System	6.3.1
• Stage I Hydraulic System	6.3.2
• Stage II Hydraulic System	6.3.3

FUNCTIONS:
• Provide Fluid Injection to Change the Solid Rocket Motor Direction of Thrust • Provide Force to Gimbal the Stage I and II Engines

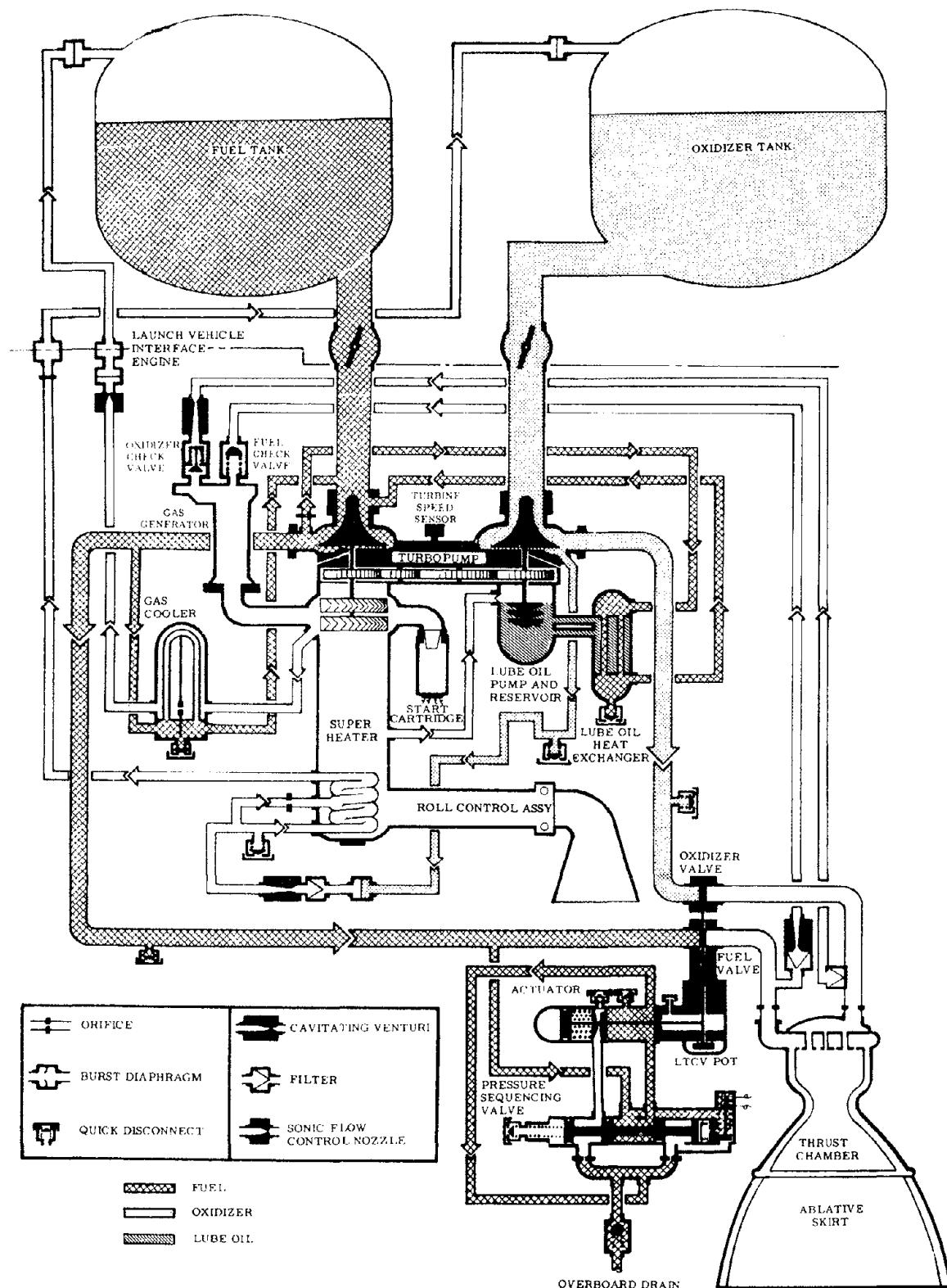


Figure 6-23. Stage II engine fill and bleed sequence.

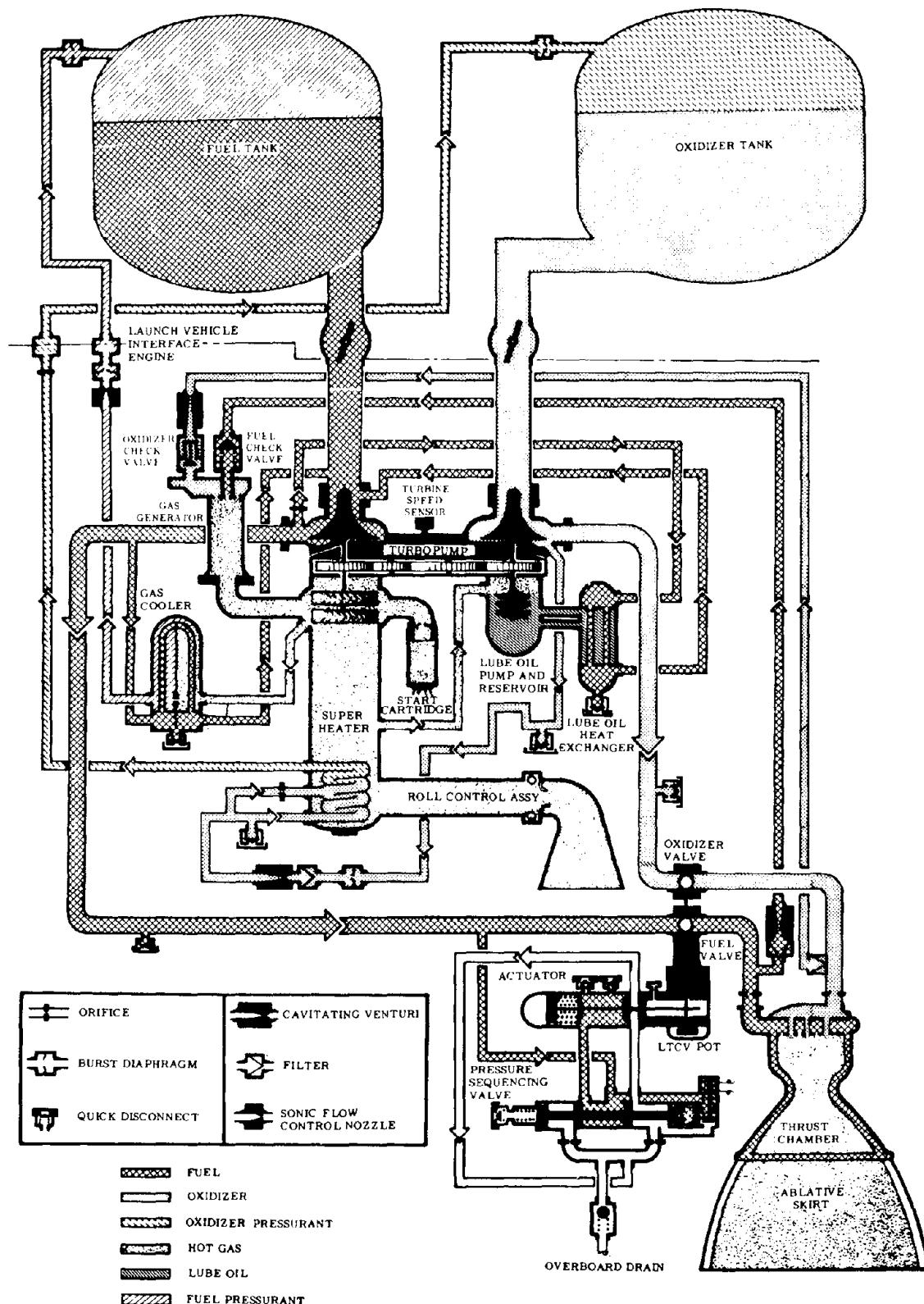


Figure 6-24. Stage II engine start sequence.

The three separate thrust vector control (TVC) systems provide the mechanical force to change the direction of thrust on the three propulsion systems of the Titan IIIE vehicle, resulting in changes of orientation of the body axes. Each system operates only during its respective portion of vehicle powered flight. Provisions are made to operate each system in a checkout mode prior to vehicle flight. Location of TVC actuation devices is shown in Figure 6-25.

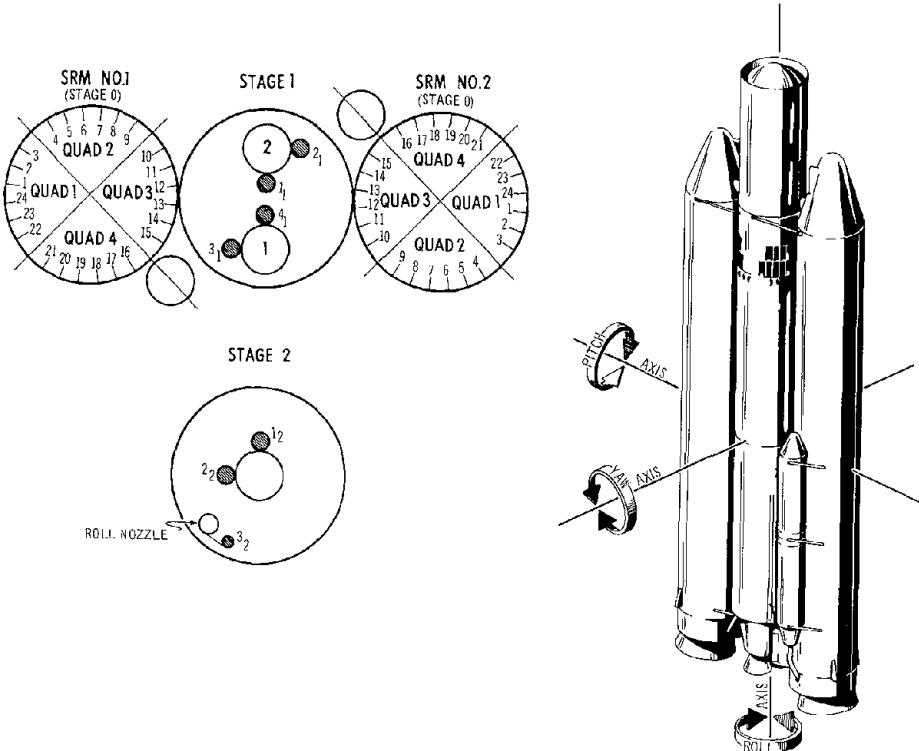


Figure 6-25. Thrust vector control actuation devices location.

6.3.1 SOLID ROCKET MOTOR THRUST

VECTOR CONTROL SYSTEM. The thrust vector control system used for the Titan IIIE 120-inch solid rocket motor provides side forces on each motor in response to command signals from the core vehicle flight controls computer. Nitrogen tetroxide is injected into the nozzle exit cone in each of four quadrants to provide side force in any direction normal to the nozzle centerline (Figure 6-26). The TVC system is designed to provide for a long launch pad hold period to make it compatible with the readiness criteria of the Titan/Centaur program. A subsystem is provided as part of the flight controls system which computes, based upon actual usage, the amount of excess injectant fluid, and necessary valve commands to dump this excess.

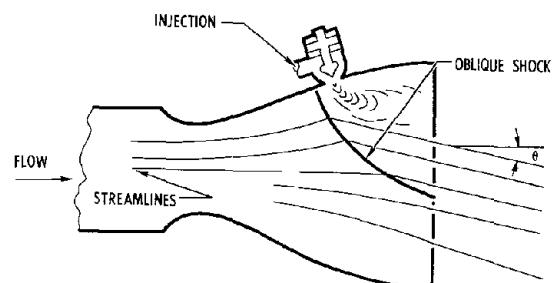


Figure 6-26. SRM thrust vector control injection.

Nitrogen Tetroxide Tank Unit. The unit is 42 inches in diameter and approximately 22 feet in length. As illustrated in Figure 6-27, this is a single tank structure using an ullage blowdown system. Total tank weight is 3817 pounds. Nominal load is 8424 pounds of N₂O₄ and initial pressure is 1030 psia. This pressure reduces to a minimum of 450 psia at SRM burnout.

A single feed line transfers the injectant from the tank to the nozzle distribution manifold.

Motor Nozzle Unit. The motor nozzle unit includes the motor nozzle and injector valve housing, injector valve assembly, injector manifold, pitch and yaw control, electrical power supply, electrical harness assemblies, battery assemblies, power transfer switch, and mounting supports. The package is capable of complete functional checkout without flowing propellant.

The injector valve housing is attached to the nozzle and provides a support for the distribution manifold and the 24 injector valve assemblies. In addition, the housing delivers the injectant from the manifold to the injector valves and provides a passage from the valve into the nozzle. Each of the valve housings is dowel-pinned and bolted to the nozzle expansion cone shell.

Valve actuation is provided by 24 proportional electromechanical valves (EMVs) which are complete servomechanisms in themselves. These valves operate from 28V d-c and consist of a servoamplifier-driven motor which positions a pintle attached to a lead screw. Pintle position is fed back to the servoamplifier to complete the position control loop. An electronic fail-safe detection logic is included to detect either a mechanical binding or electronic failure and drive the valve closed.

6.3.2 STAGE I HYDRAULIC SYSTEM. The Stage I hydraulic system provides the mechanical force necessary to gimbal the thrust chambers of the Stage I engine during all phases of checkout and flight. The single system provides a nominal 3000 psi pressure to change the thrust vector of both subassemblies of the Stage I engine system.

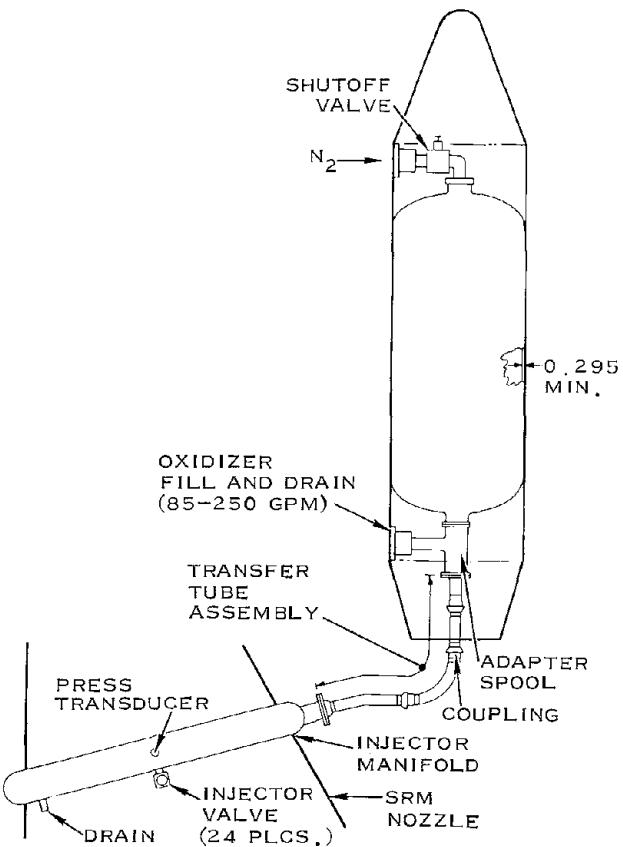


Figure 6-27. Thrust vector control tankage.

The hydraulic system (Figure 6-28) consists of a high-pressure turbine-driven pump assembly; an electric motor; a high-pressure, electric-motor-driven pump assembly, high-pressure lines and manifold; a filter; a regulating unit that includes an accumulator and reservoir; four linear actuators; instrumentation; and the airborne half of the manually operated ground disconnect.

Electric-Motor-Driven Pump. The electric-motor-driven pump (Figure 6-29) is used for checkout purposes only. Although it is an airborne component, it does not operate during flight.

Filter. The system filter is a replaceable cartridge fluid filter located in the discharge line from the motor-driven pump. The filter is equipped with a differential pressure indicator that indicates red when the pressure drop across the filter exceeds 70 ± 10 psi. The filter has a nominal 10 micron rating.

Regulating Unit. The regulating unit (Figure 6-30) is a combination accumulator-reservoir. Its functions are: (1) to maintain a constant system pressure, (2) to serve as a hydraulic system reservoir, and (3) to sense system pressure changes and hydraulic fluid level changes. The unit provides pressure to the linear actuators. It has a return line from the actuators to the reservoir, and suction lines to both hydraulic pumps.

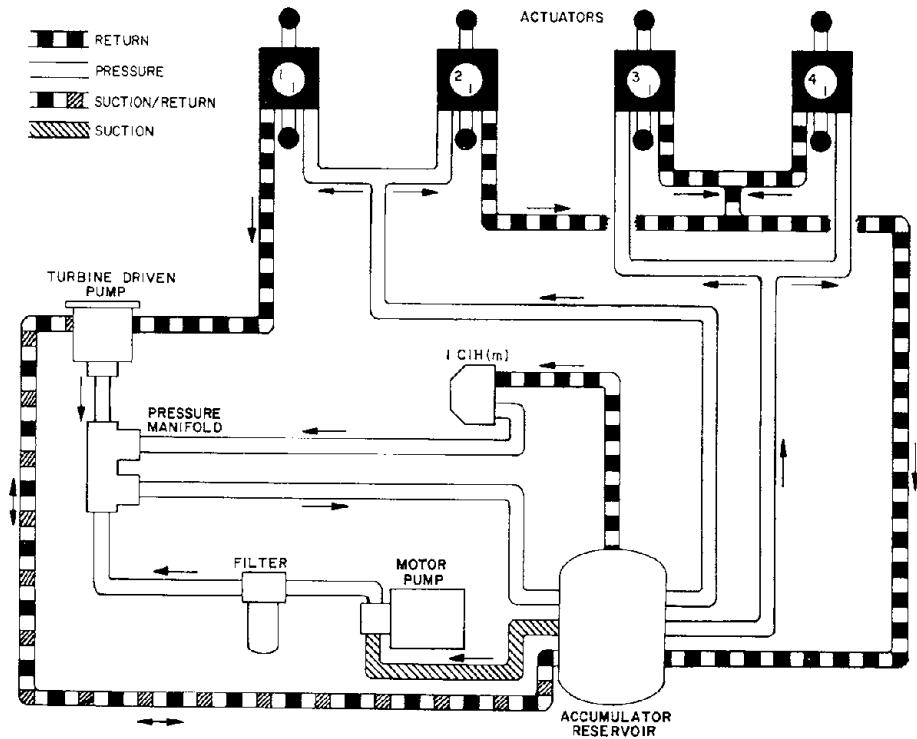


Figure 6-28. Stage I hydraulic system.

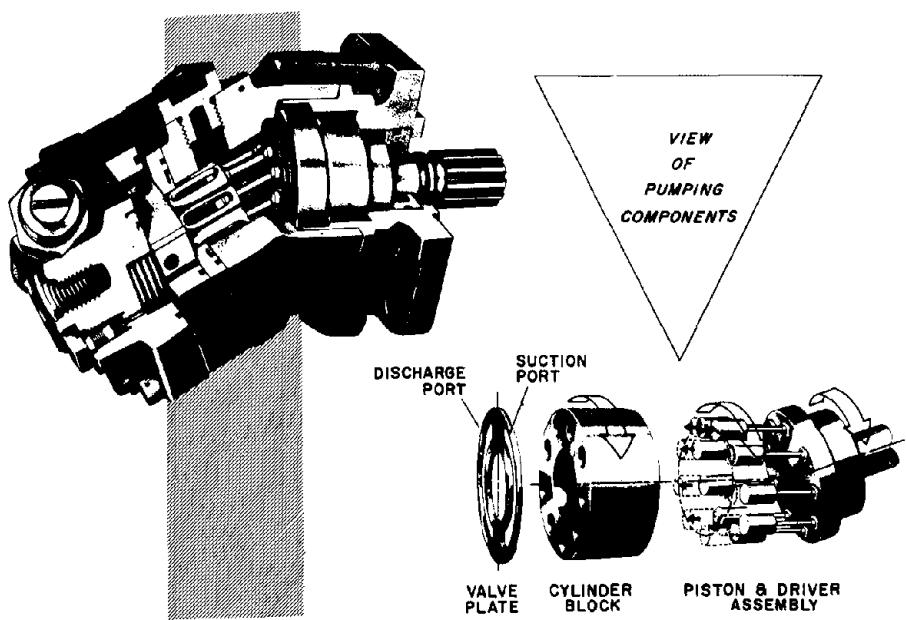


Figure 6-29. Electric-motor-driven hydraulic pump.

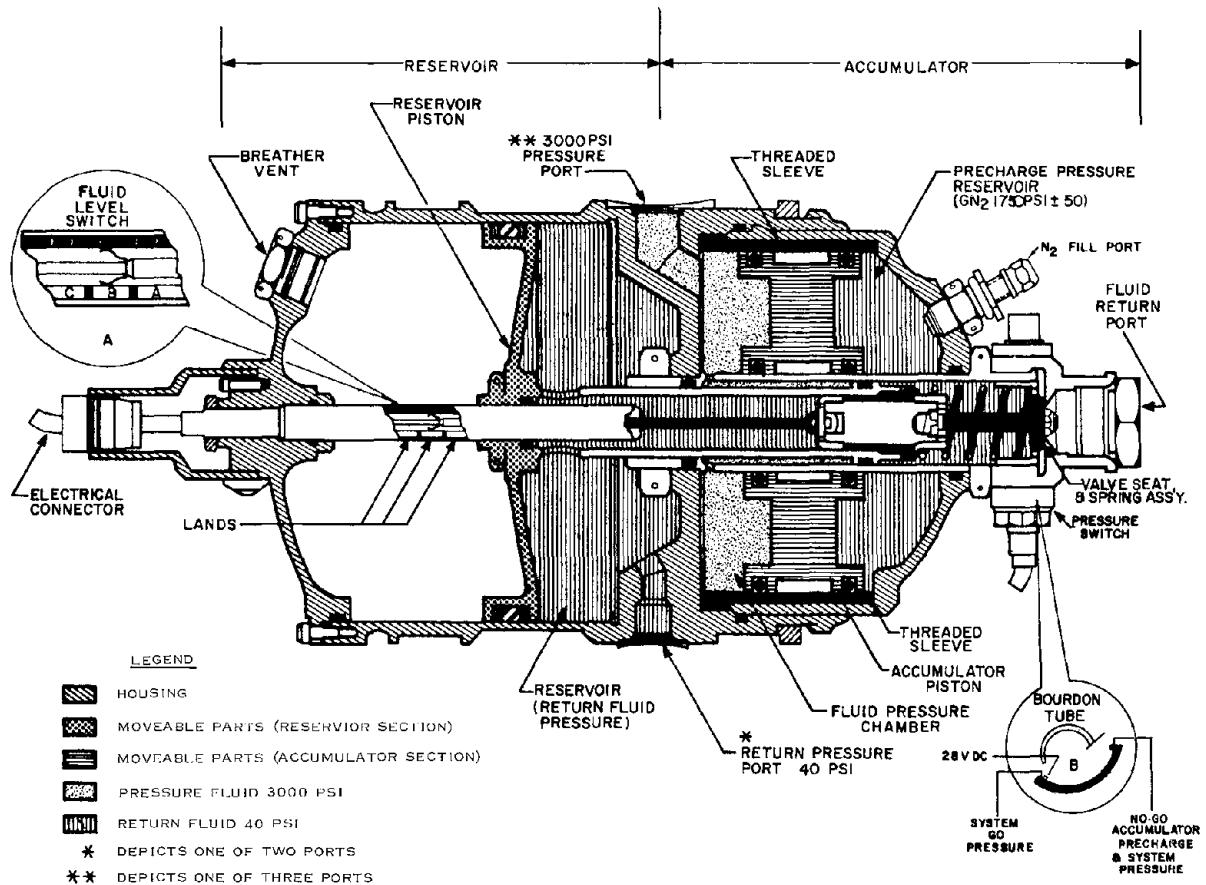


Figure 6-30. Hydraulic regulating unit cross section.

Turbine-Driven Pump. The Stage I turbine or mechanically driven pump (Figures 6-31 and 6-32) is a variable delivery, pressure-compensated hydraulic pump that supplies pressure and fluid flow during Stage I engine operation. The pump contains nine pistons which are driven by a wobble plate attached to the splined shaft of the engine turbopump assembly. Pressure compensation is provided by a sliding sleeve on each piston which allows the piston to provide an increase or decrease in system pressure dependent on downstream demands.

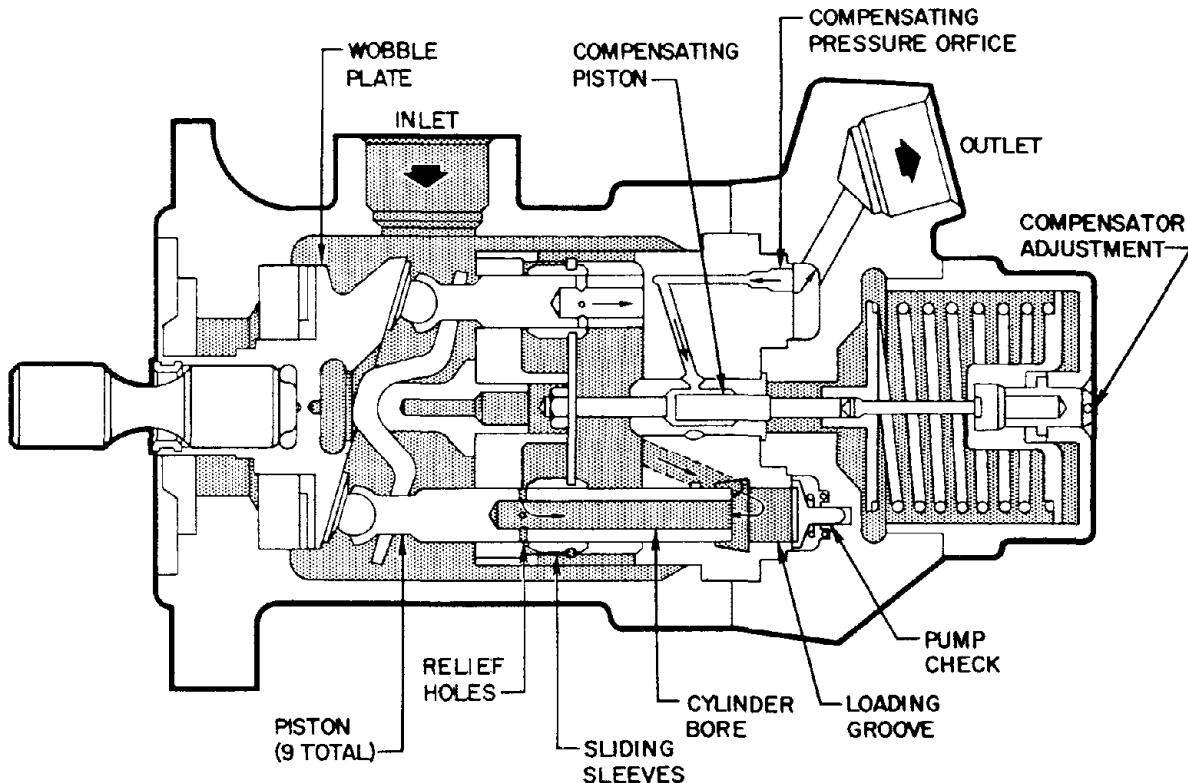


Figure 6-31. Turbine-driven hydraulic pump cross section.

Linear Actuators. Stage I has four linear actuators (Figures 6-33 and 6-34). They are hydraulically operated servo-mechanisms that gimbal the engine thrust chambers in response to electrical signals (± 40 ma) from the flight controls system. The servoactuator stroke of ± 1.095 inches limits the engine gimbal angle to ± 4.38 degrees pitch, and 4.12 degrees yaw from the neutral position. The output force of the actuator is 29,640 pounds at 3000 psid. The maximum actuator velocity is 13.74 degrees per second.

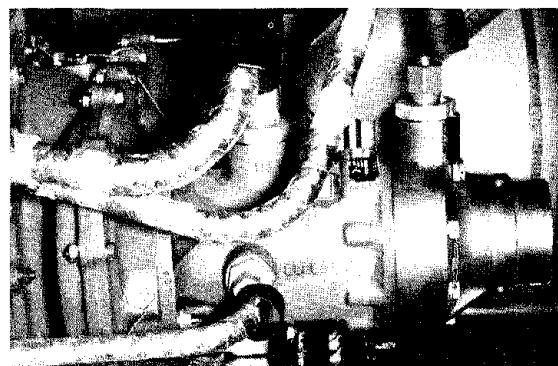


Figure 6-32. Turbine-driven hydraulic pump installed on Stage I.

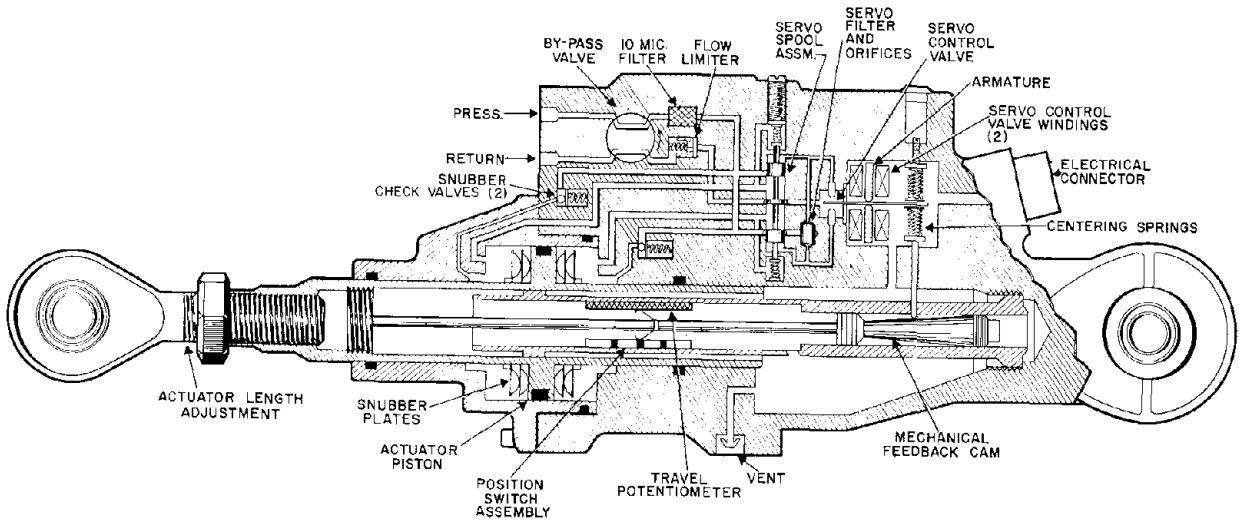


Figure 6-33. Linear hydraulic actuator cross section.

Operation. The total Stage I hydraulic system is proof-tested during factory tests to 4500 psi. During testing the system is also filled with hydraulic fluid (MIL-H-6083), flushed of all contaminants, and all air is bled from the system. These service functions are performed with the ground hydraulic servicing unit. After completion of these tests, the ground umbilical is disconnected and the system sealed. Pressure for all further ground operations is provided by the electric-motor-driven pump. During ground operations, the pump draws fluid from the reservoir section of the accumulator-reservoir unit. The hydraulic fluid is then sent under pressure through the 10-micron filter to the high pressure side of the accumulator-reservoir unit. The high pressure side is maintained at a nominal 3000 psi. The N₂ accumulator is designed to prevent pressure surges in the high pressure system, and to assure availability of 3000 psi at all four of the actuators. Pressure is returned to the reservoir at 40 psi. Actuator I pressure returns through the turbine-driven pump to provide for bleeding this pump during servicing.

During Stage I powered flight, the system receives hydraulic pressure from the turbine-driven pump. This pump is splined into the turbine on subassembly number 2

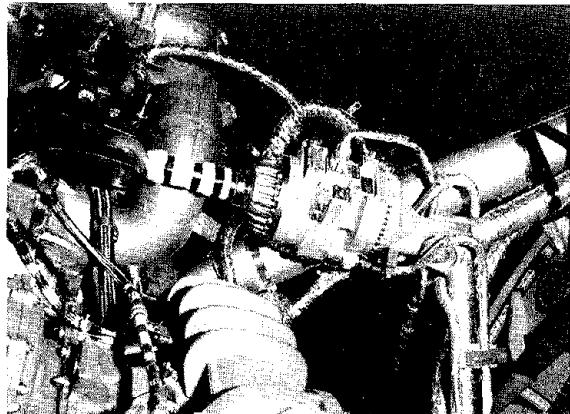


Figure 6-34. Linear hydraulic actuator installed on Stage I.

of the Stage I engine, and begins to operate when the engine is started. This pump then maintains the high pressure side of the accumulator reservoir at 3000 psi.

Instrumentation on the accumulator-reservoir unit provides readouts of system pressure and fluid level. A travel potentiometer is located on each linear actuator to provide ground readout of piston movement during test and flight.

Operating parameters of major Stage I hydraulic system components are:

Turbine-Driven Hydraulic Pump

Flow, gpm	15
Pressure, psi	3000 nominal
Speed, rpm	3760 minimum
Type	Variable volume, pressure compensated, piston

Electric-Motor-Driven Pump

Flow, gpm	3
Pressure, psi	3000 nominal
Speed, rpm	12,500
Type	Variable volume, pressure compensated, piston

Electric Motor

Voltage, V d-c	28
Speed, rpm	12,500 rpm
Current, amp	250 @ 12,500 rpm

Hydraulic Regulating Unit

Gas Volume, cu. in.	25
Precharge (N ₂), psig	1750
Capacity, cu. in.	95
High Pressure, psig	3000
Low Pressure, psig	40

System Filter

Micron Rating

10 nominal

Hydraulic Fluid

MIL-H-6083

6.3.3 STAGE II HYDRAULIC SYSTEM. The Stage II hydraulic system (Figure 6-35) is similar to the Stage I system. Differences include the use of a smaller turbine-driven pump and the use of a roll-control nozzle actuator.

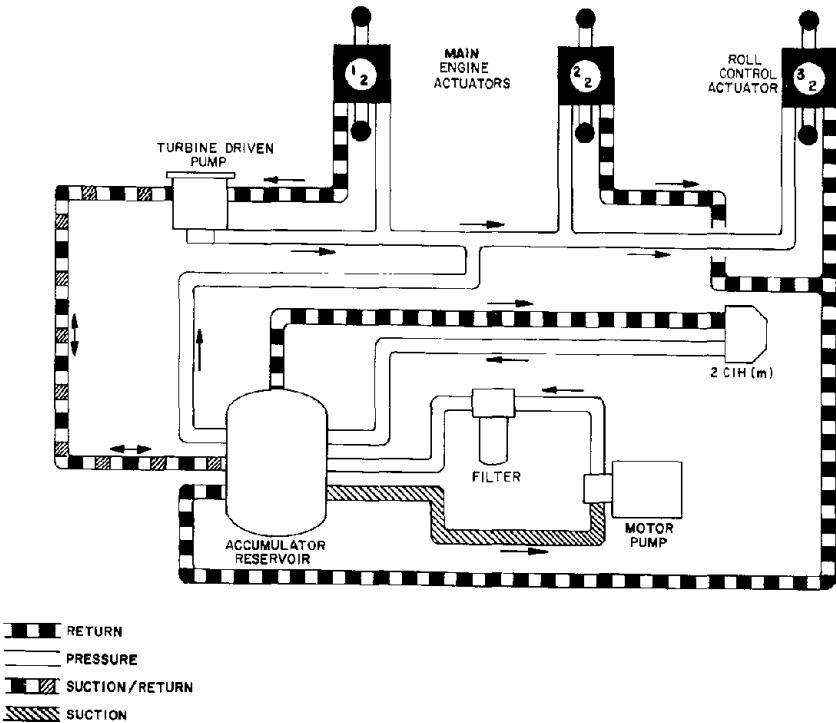


Figure 6-35. Stage II hydraulic system.

The pitch and yaw servoactuator stroke of ± 0.498 inch limits the engine gimbal angle to ± 2.038 degrees (including 0.26 degrees snubbing) from the neutral position. The output force of the actuators is 7650 pounds at 3000 psid. The maximum actuator velocity is 16.38 degrees per second. The roll servoactuator stroke of ± 1.4 inches limits the nozzle gimbal angle to ± 33.83 degrees (including 3.62 degrees snubbing) from the neutral position. The output force of this actuator is 1245 pounds at 3000 psid. The maximum actuator velocity is 8.75 degrees per second.

Operation. All components of the system operate identically to those on Stage I. Figure 6-36 shows the hydraulic regulating unit and the electric-motor-driven pump installed on Stage II.

Operating parameters of major Stage II hydraulic system components are identical to those of Stage I except for the turbine-driven pump:

Turbine-Driven Hydraulic Pump

Flow, gpm	5
Pressure, psi	3000 nominal
Speed, rpm	4025 minimum
Type	Variable volume, pressure com- pensated, piston

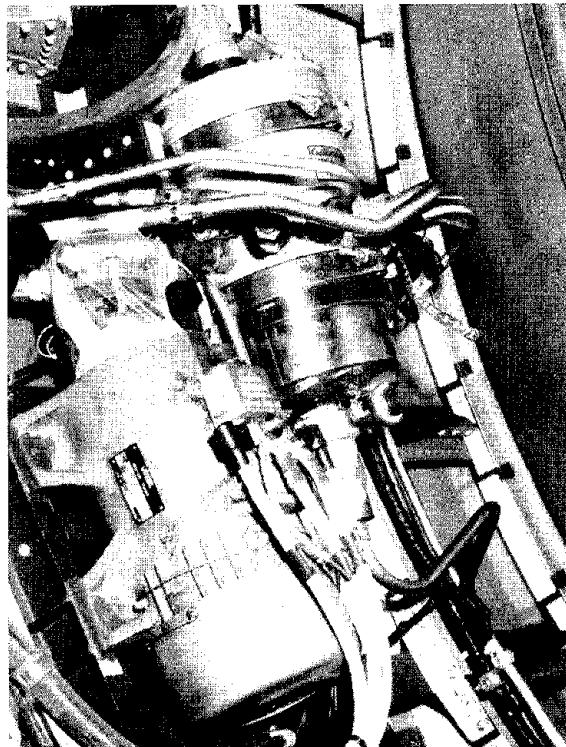


Figure 6-36. Electric-motor-driven pump and regulating unit installed on Stage II.

6.4 ORDNANCE ITEMS

ELEMENTS:	Subsection
• Solid Rocket Ordnance	6.4.1
• Stage I/II Propulsion System Ordnance	6.4.2
• Staging Release Ordnance	6.4.3
• Destruct Ordnance	6.4.4

FUNCTIONS:
• Perform Operational Sequence
• Provide Destruct Capability

The Titan IIIE booster ordnance items (Figure 6-37) perform two functions during launch and flight: operational sequence functions and booster destruct in the event of an aborted flight. Operational sequence functions are employed during a successful launch and flight and provide ignition of solid rocket motors, start of core engines, and staging separation. The destruct system is employed in flight during an abort or inadvertent separation. All ordnance items are either electro-explosive devices themselves or are ignited by an electro-explosive device.

6.4.1 SOLID ROCKET ORDNANCE ITEMS. Solid rocket items used on a IIIE vehicle consist of the 2 large solid rocket motors, 16 SRM staging rockets (8 on each SRM), and 4 retrorockets mounted between tanks on Stage II. Each of these motors is ignited by an igniter assembly, composed of an initiator unit and an igniter charge.

SRM Staging Rocket Motor. Each SRM staging rocket motor (Figure 6-38) produces 4700 pounds of thrust, is 60 inches long including its nozzle, is six inches in diameter, and weighs about 82 pounds. The motor has a mount pad on the motor case near the nozzle end and mounting holes bored and tapped around the igniter port on the motor head. The motor case is aluminum and the nozzle is asbestos phenolic with an aluminum burst seal disc. The 54.5-pound propellant uses a polybutadiene acrylic acid acrylonitrile binder with ammonium perchlorate oxidizer and aluminum fuel and is cast in a seven-point star hollow-core configuration.

Retrorocket Motors. Four small, solid retrorocket motors producing 511 pounds thrust each are mounted between tanks on Stage II. Their thrust nozzles are aimed forward. These rocket motors are fired to retard Stage II during Centaur separation from Stage II. The retrorocket motor (Figure 6-39) consists of a thin-walled,

nickel-chrome-plated steel chamber loaded with 4.6 pounds of Thiokol TPE 8035 case-bonded solid propellant. An aft bulkhead contains a graphite insert for the nozzle throat and supports a steel coupling which holds the asbestos phenolic exhaust cone to the bulkhead. A 0.014-inch aluminum nozzle closure (burst disc) provides a seal against entrance of moisture and foreign material into the chamber. A head cap closes the forward end of the chamber and mounts the igniter.

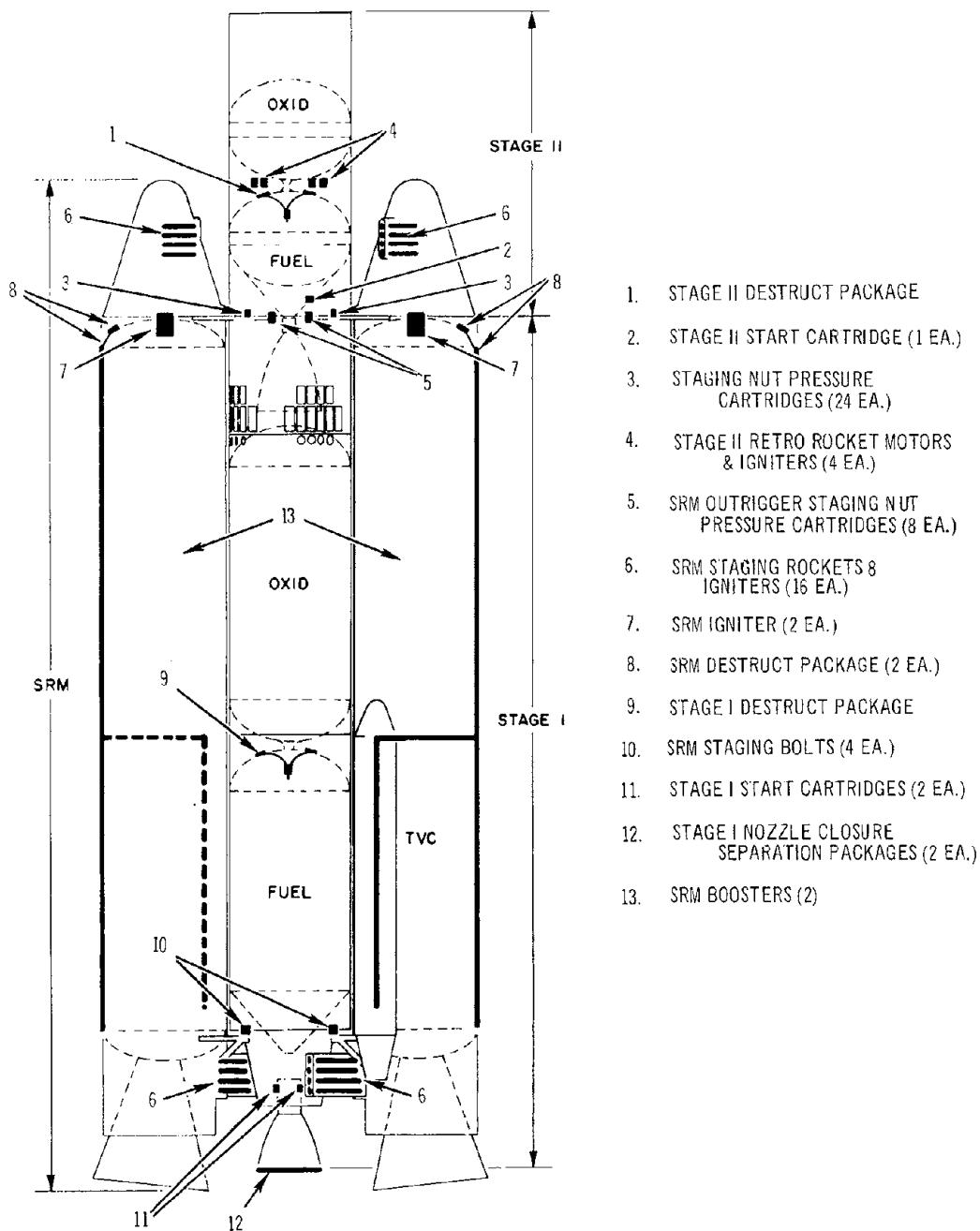


Figure 6-37. Location of ordnance items.

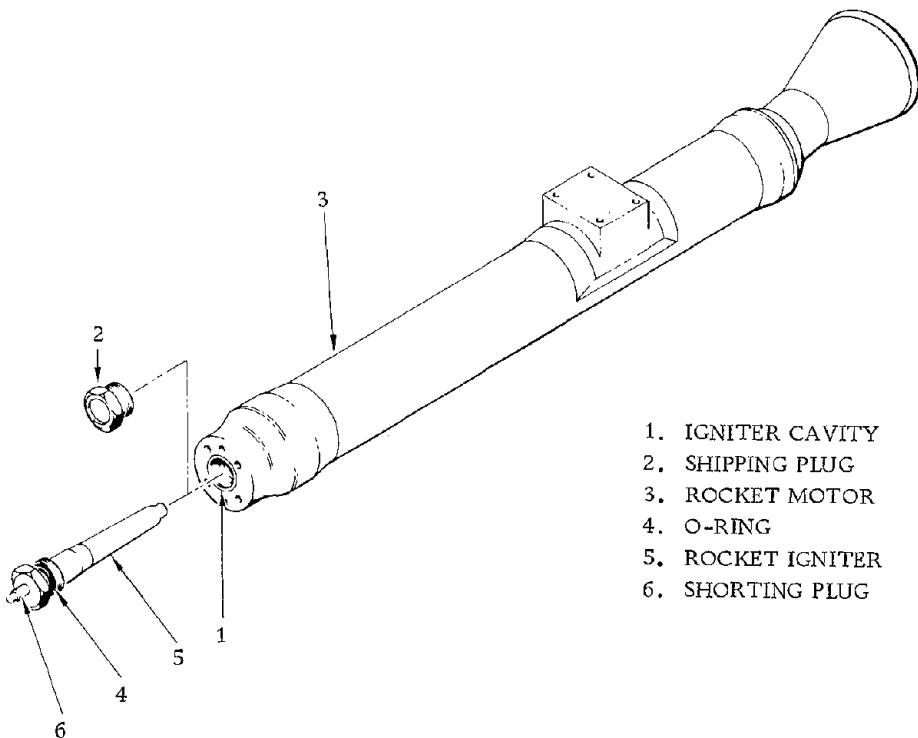


Figure 6-38. SRM staging rocket motor.

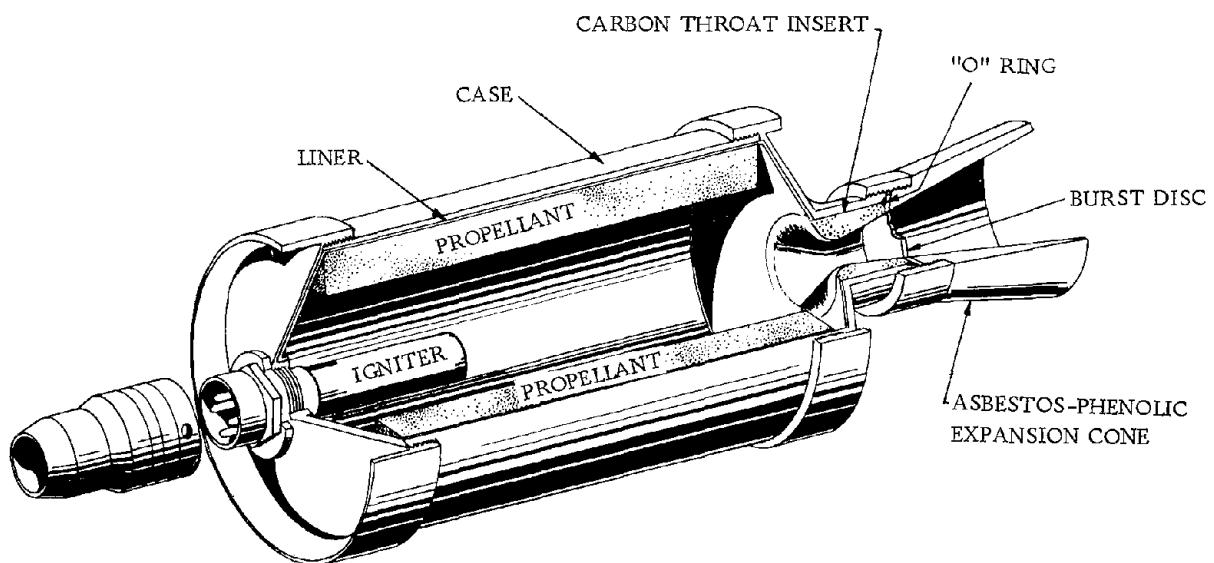


Figure 6-39. Stage II retrorocket motor.

6.4.2 STAGE I/II PROPULSION SYSTEM ORDNANCE ITEMS. Ordnance items employed in the Stage I and Stage II propulsion system are the engine start cartridges and the Stage I nozzle exit closure separation systems.

Engine Start Cartridges. The Stage I and II engine start cartridges (Figure 6-40) are designed to provide a flow of hot gas to accelerate the turbopump turbines to a

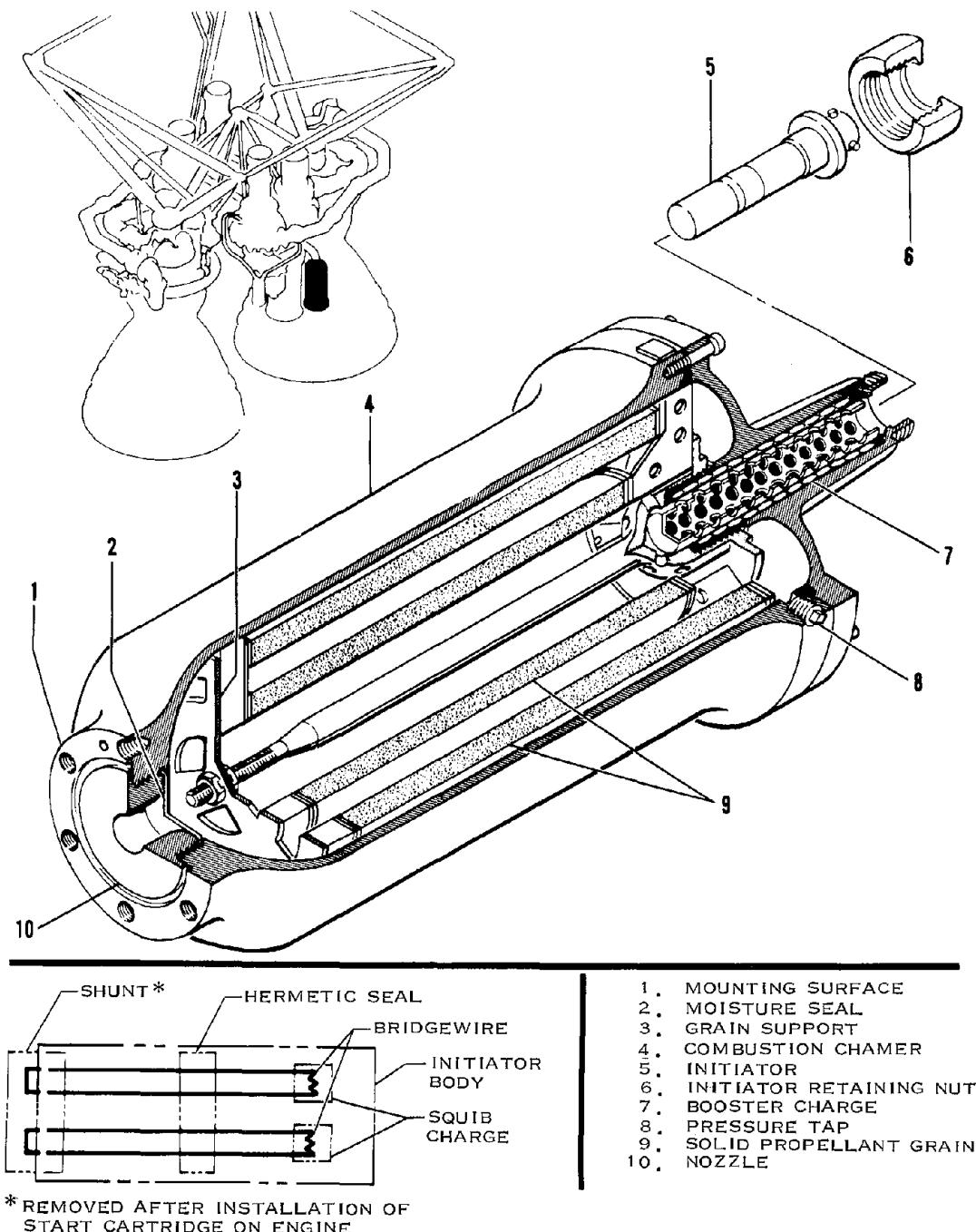


Figure 6-40. Titan IIIE engine start cartridge.

speed sufficient to allow the engine to self-sustain its operation. Each cartridge assembly consists of a combustion chamber with propellant load and an igniter assembly which includes two initiator squibs. The propellant grain is composed of rubber-based, ammonium nitrate propellant (AMR2506). The cartridges burn for approximately 1.1 seconds at 2200°F and produce a pressure of 2500 psia in the cartridge case.

Exit Closure Separation System. The Stage I exit closure separation system (Figure 6-41) consists of an upper and lower detonating manifold for each engine subassembly, an interconnecting confined detonating fuse (CDF) and a mild detonating fuse (MDF) that encircles the ring of each closure. The upper manifold contains a dual-bridgewire electric initiator which ignites two strands of CDF. Booster charges and a cross strand of MDF complete the explosive circuit inside the manifold. The lower manifold, located at the exit closure attaching ring, provides an ordnance junction for the CDF with the five-grain-per-foot MDF that cuts the attaching ring. A cross strand of seven-grain-per-foot flexible linear shaped charge connects the two CDF terminals within the lower manifold and ensures that both ends of the MDF ignite even if only one of the CDF strands operate.

6.4.3 STAGING RELEASE ORDNANCE

ITEMS. The ordnance items employed to cause release of the vehicle stages one from another at the proper sequence during flight are of two types: gas pressure cartridges and explosive bolts. The gas pressure cartridges are used to power gas-operated nuts which are a component of the staging set. The explosive bolt is a staging release unit within itself.

Gas Pressure Cartridges. The gas pressure cartridge (Figure 6-42) is an electro-explosive device employing independently operated dual-bridgewire circuits. The bridgewires fire a prime load, and through a booster and sustainer charge, fire the main load which will develop an operating gas pressure of approximately 2000 psig in a 10-cubic-centimeter chamber.

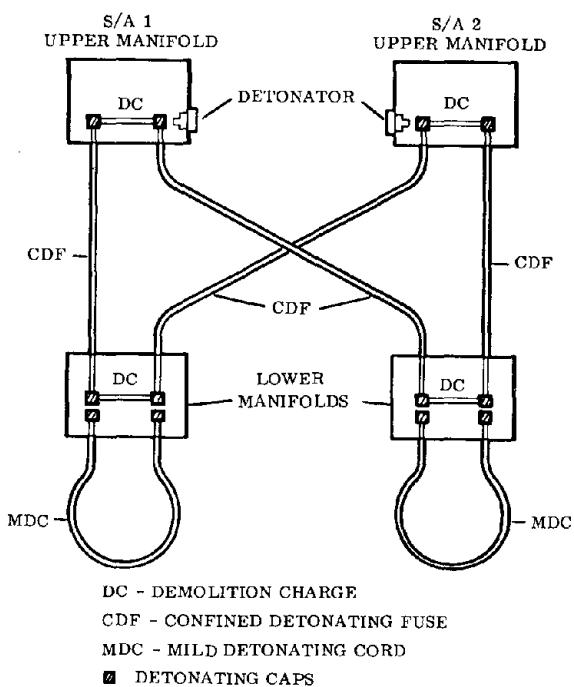


Figure 6-41. Engine exit closure separation system.

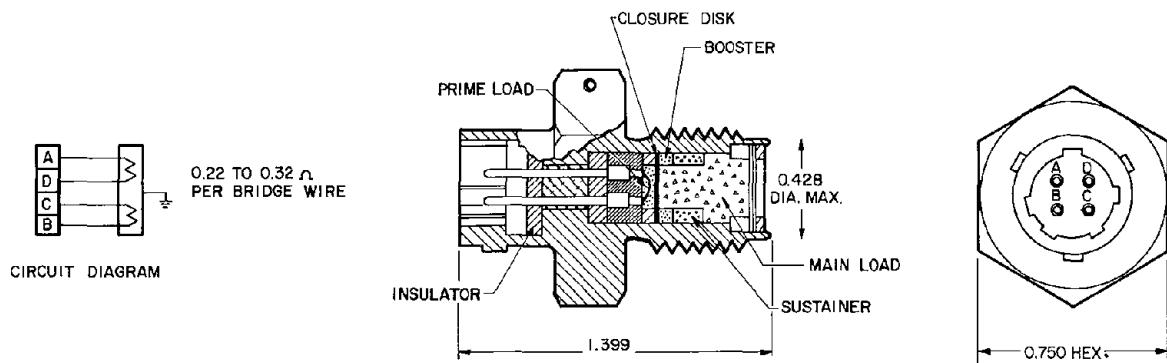


Figure 6-42. Gas pressure cartridge details.

The gas pressure cartridge is approximately 1.4 inches in length and 0.5 inch in diameter. The gas pressure cartridge has a 0.75-inch hexagon section with a lock type receptacle and four pins for the electrical connector. The metal body is threaded for insertion into a component and the end is sealed against moisture. The gas pressure cartridge requires 2.2 amperes at 28V d-c for an "all fire" characteristic and will withstand one ampere for five minutes without igniting.

Gas-Operated Nuts. The gas-operated nuts are designed to be threaded onto a standard fine thread bolt or stud. The nut provides the same holding force as a standard nut but incorporates a design allowing the nut to be disassembled in place by gas pressure (Figure 6-43). The pressure removal process slides the outer jacket off of four inner-threaded segments. The nut may also be removed with a wrench if desired.

Staging Release Studs. The staging release set studs (Figure 6-44) are 4.82 inches long and 0.874 inch in diameter at the center with 3/4-inch threaded ends. They have a minimum axial breaking tensile strength of 100,000 pounds. The stud threaded ends accommodate the staging nuts.

Aft Staging Set Explosive Bolts. The explosive bolts (Figure 6-45) are used to secure the four aft mating interface points between the Stage I airframe and the SRM, and provide a means of positively releasing these points

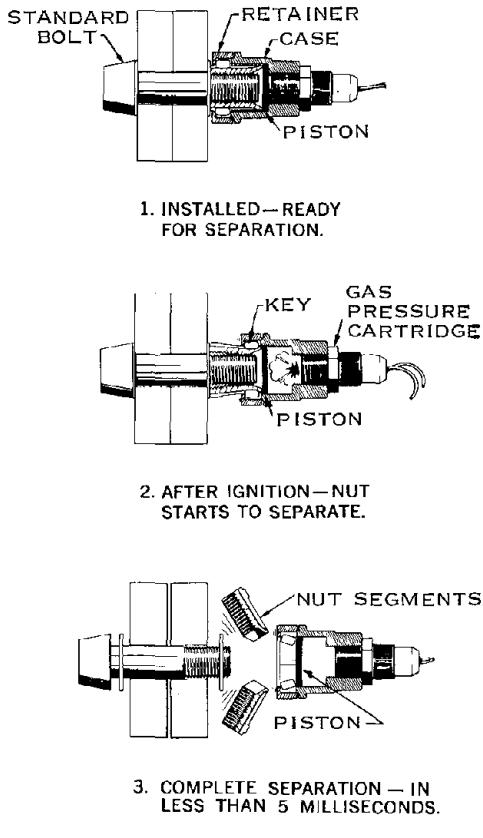


Figure 6-43. Gas-operated nut operation.

during the staging release sequence. The bolt is a stainless steel unit of standard bolt design and is approximately 10-3/4 inches long, 1.37 inches in diameter, and has a standard 1-3/8-inch threaded end. The bolt body contains two 18-grain RDX charges which are located internally, adjacent to two scored break-lines on the bolt shank. Detonator assemblies containing dual bridgewires, lead azide prime charges, and PETN booster charges fire the RDX charges. Electrical connectors at both ends provide electrical connection to core vehicle circuits.

In addition to the two redundant explosive charges encased in the bolt, additional reliability has been added by two cross-bored holes between the scored break lines, which act as a frangible break point. The bolt will fracture at this point as the SRM tips away from the core even if the explosive charges misfire or fail to separate the bolt.

6.4.4 DESTRUCT ORDNANCE ITEMS.

Destruct ordnance items are the SRM and TVC tank destruct package, the Stage I destruct package, and the Stage II destruct package.

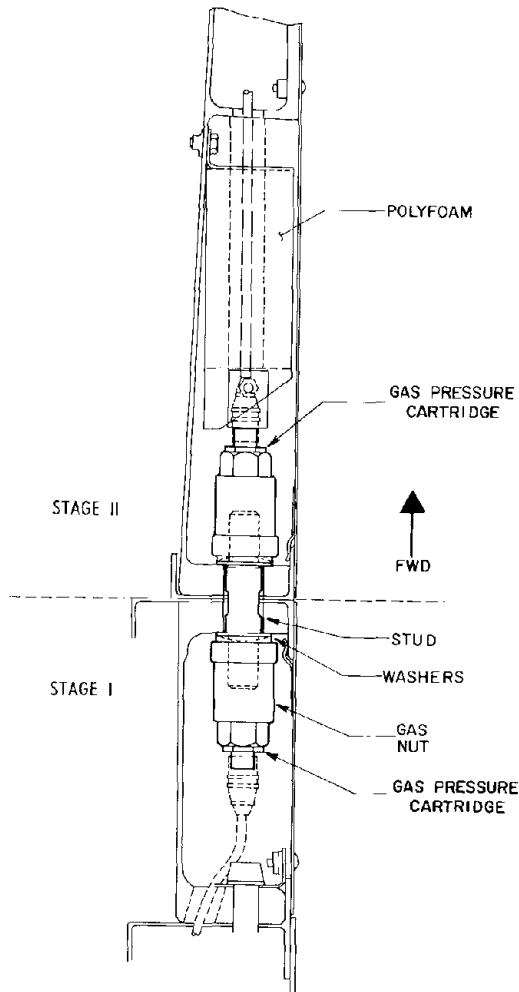


Figure 6-44. Stage I/II staging release set.

SRM and TVC Tank Destruct Package. The destruct system for each solid rocket motor includes two major assemblies: the safe and arm device and the explosive train assembly, which is designed to sever both the SRM motor case and the TVC tank.

The explosive train assembly incorporates an explosive transfer fuse, linear shaped charges, and jumper fuses between segments. Upon receipt of the proper firing signal, two detonators housed in the safe and arm assembly initiate a redundant transfer harness. The explosive stimulus is then transferred to two linear shaped charges running along each of the segments. An explosive transfer block initiates the transfer harness, leading to a similar explosive block to activate the linear-shaped charge on the TVC tank. Complete redundancy is incorporated from the detonators through the linear-shaped charges.

The linear shaped charges are sized to cut through 0.650-inch mild steel. In operation, the charge severs the SRM case and insulation and also severely cracks the propellant under the charges. These actions, in combination with the internal pressure of the SRM, provide a positive destruct mode. The charge load is 600 grains per foot of RDX; the transfer and jumper fuses are loaded with 70 grains per foot.

Stage I Destruct Package. This system consists of an initiator, two bi-directional "wafer" destruct charges, interconnecting primacord, and primacord boosters (Figure 6-46). Located between tanks, this system is fired simultaneously with the SRM destruct package and ruptures the domes of the fuel and oxidizer tanks. This allows propellant mixing, consequent ignition of the propellants, and resultant destruction of the Stage I core vehicle.

Destruct Initiator. The destruct initiator is approximately 4.6 inches long, 3 inches wide, 2.4 inches high, and weighs approximately 1.7 pounds. The explosive train in the destruct initiator consists of two primers, two lead charges and one booster charge.

The primers are electro-explosive devices employing a single bridgewire circuit. One primer receives a firing signal from command receiver number one, while the other receives a firing signal from command receiver number two.

The lead charge consisting of 166 mg of pure RDX is located in a cylinder which rotates 90 degrees. The lead charges can be ignited by the primers only after the two explosives are aligned.

The booster charge consists of 2500 mg of waxed RDX and is located on the opposite side of the cylinder from the primer. The booster charge is ignited by the lead charge in the cylinder.

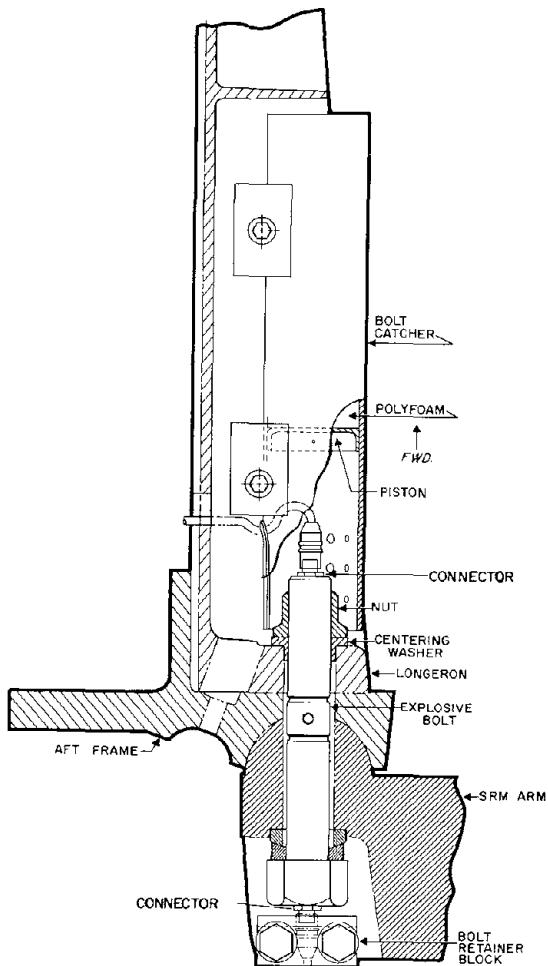


Figure 6-45. SRM/Stage I aft staging set.

The ordnance circuitry between initiator and destruct charges consists of 100-grain-per-foot primacord with auxiliary boosters at the terminal ends. Primacord also provides an explosive connection between the two destruct charges in a stage destruct package. The primacord burning rate is approximately 19,000 feet per second. Auxiliary 60E7-1 boosters are crimped to each end of the primacord. The primacord consists of PETN with a textile wrap covered with a polyethelene jacket.

Bi-directional Destruct Charges.

Two bi-directional destruct charges are mounted on the Stage I fuel tank dome and are held in place by a bracket assembly. Each charge is secured by three bolts and washers to the brackets.

The function of the bi-directional destruct charges is to demolish large areas of opposing fuel and oxidizer tank domes by producing a destruct pattern in two directions. The destruct charges consist of a 1.4-pound, 5/8-inch thick wafer charge of RDX composition C4 in a lead sheath. The destruct charge is 8-1/2 inches in diameter and 7/8 inch thick. Each destruct charge has two receptacles to provide attachment of the primacord boosters. These receptacles each contain a booster subassembly containing 2500 mg of RDX.

Stage II Destruct Package. This system is similar in design and function to the Stage I destruct package.

Ordnance Items Summary. The following summarizes the various types of ordnance items used on the Titan IIIE:

<u>Item</u>	<u>Force</u>	<u>Details</u>
SRM Igniter	1850 psi	Dual squib boron - Potassium nitrate booster
SRM Staging Motors	4700 pounds each	Same as above

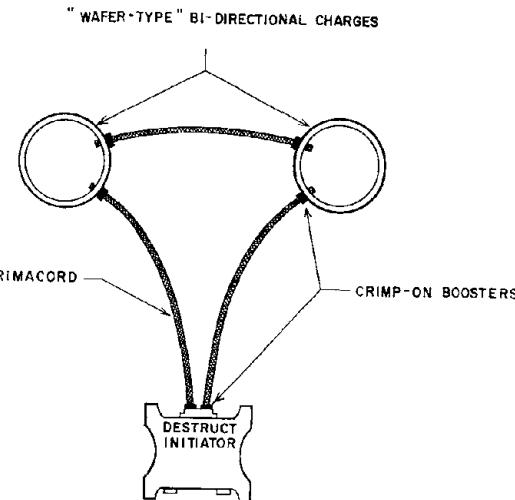


Figure 6-46. Stages I and II destruct system.

<u>Item</u>	<u>Force</u>	<u>Details</u>
Gas-Operated Nuts	2000 psig	Dual squib 346 mg boost charge
Explosive Bolts		Dual squib Dual RDX boost charge Frangible break point
Retrorockets	511 pounds each	Dual squib Boron booster Thiokol TPE 8035 Base charge
Stage I Engine Exit Closure System		Dual squib Redundant detonating fuse
Engine Start Cartridges	2500 psia	Dual bridgewire Alclo boost charge
Destruct Initiator		Dual primers 166 mg RDX
Primacord		100 grain-foot-PETN 19,000 feet/sec
Bi-directional Destruct Charge		1.4-pound RDX

7

TITAN IIIE FLIGHT CONTROLS SYSTEM

The major components of the Titan IIIE flight controls system (FCS) are physically mounted on the truss within the forward compartment of Stage II. Layout of these components is shown in Figure 7-1.

7.1 FLIGHT CONTROLS SYSTEM

ELEMENTS:	Subsection
• Rate Gyro System	7.1.1
• Three Axis Reference System (TARS)	7.1.2
• Flight Programmers (2)	7.1.3
• Flight Controls Computer (FCC)	7.1.4
• Staging Timer (ST)	7.1.5

FUNCTIONS:

- Provide Vehicle Attitude Stability
- Control Trajectory
- Regulate Solid Rocket Motor Fluid Usage
- Provide Discrete Commands

The flight controls system provides stable control of the vehicle during Stage 0, Stage I, and Stage II powered flight in response to the attitude rate command data issued by the Centaur guidance system (CGS) and by the FCS open loop programmed rates. Control is effected by angular displacement of the thrust vector during powered flight.

The basic block diagram of the flight controls system (including hydraulics) is illustrated in Figure 7-2. The component functions are described in the following paragraphs.

7.1.1 RATE GYRO SYSTEM. The rate gyro package is located between tanks on Stage I (Figure 7-3). The purpose of this component is to sense rate of change of vehicle attitude in the pitch and yaw planes during Stage 0 and Stage I flight. The outputs of the component are voltage analogs of the pitch and yaw angular rate, sent to the flight controls computer.

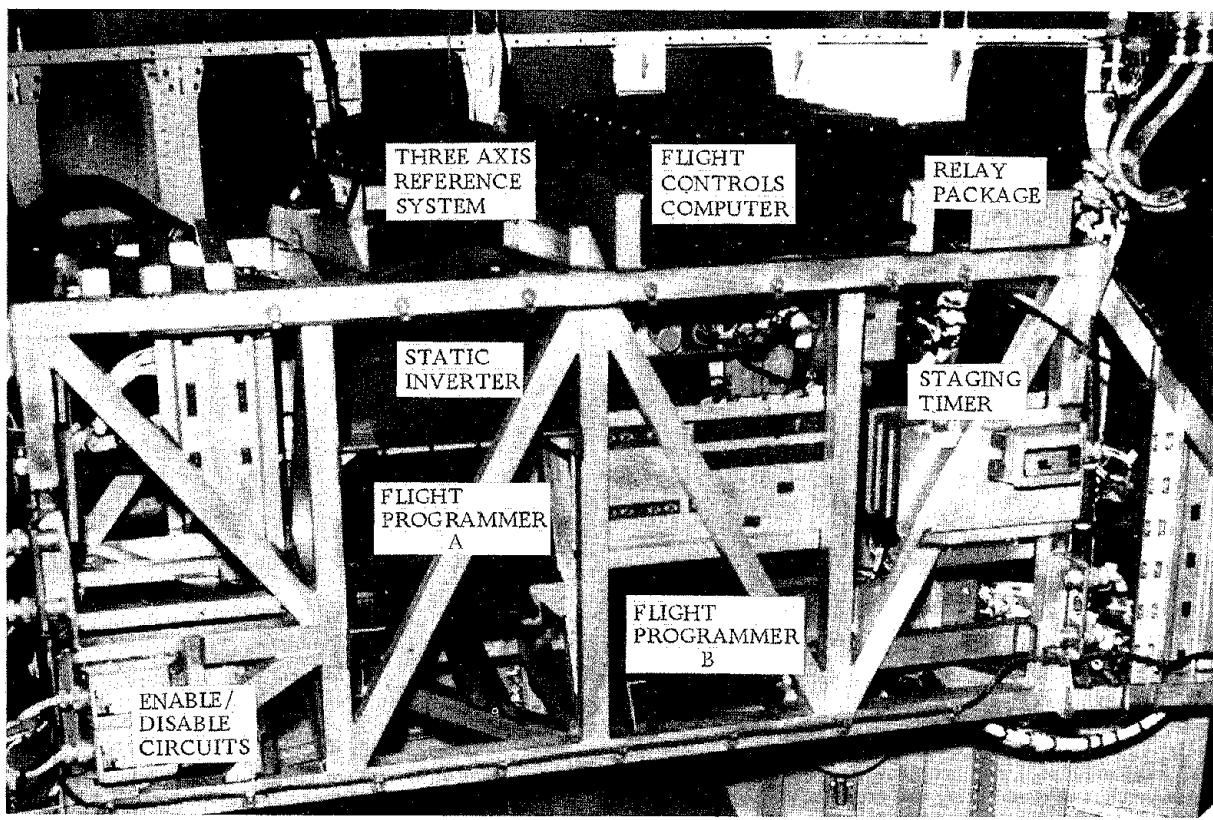


Figure 7-1. Flight controls component location.

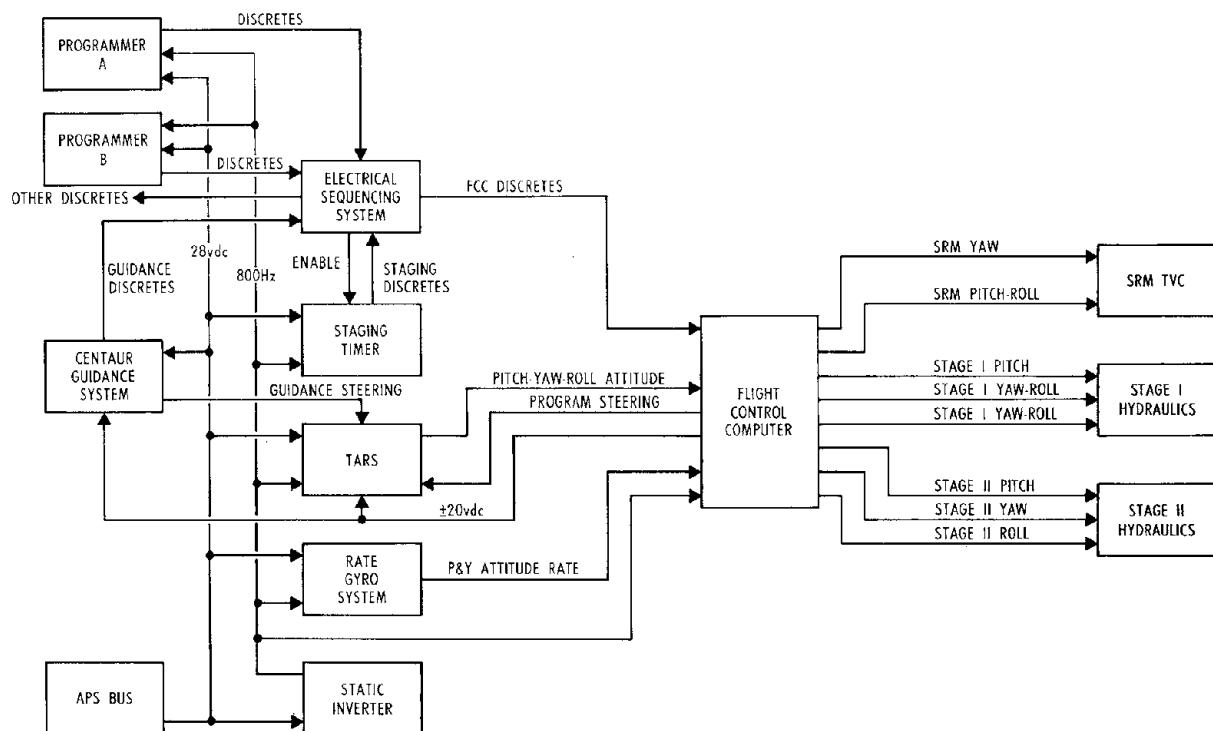


Figure 7-2. Flight controls system block diagram.

7.1.2 THREE-AXIS REFERENCE SYSTEM (TARS). The TARS package is located on the truss in the forward compartment of Stage II. This component provides signals proportional to the angular displacement of the vehicle from an established reference in each of the pitch, yaw, and roll axes. The output of the TARS to the flight controls computer is a voltage analog of pitch, yaw, and roll angular displacement errors. Torquer inputs to the package are provided by the Centaur guidance system (vehicle steering commands in the pitch, yaw, and roll axes) and the flight controls computer (the capability exists to program maneuvers in the pitch, yaw, and roll axes). During ground checkout, these commands are supplied by aerospace ground equipment in addition to guidance and the FCC.

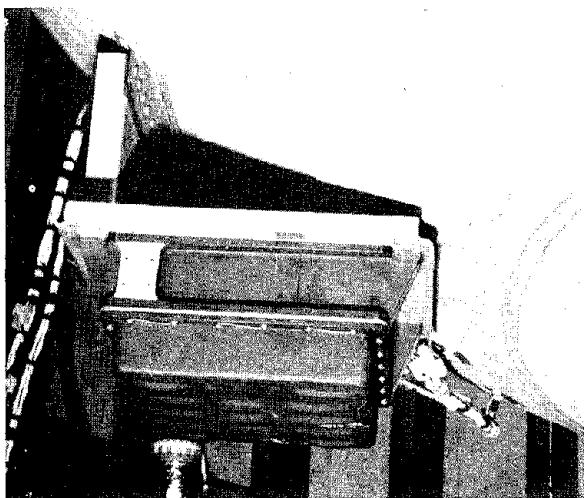


Figure 7-3. Rate gyro package.

7.1.3 FLIGHT PROGRAMMERS. The two flight programmers provide momentary and continuous discrete outputs as a prescribed function of time. The programmers are also capable of issuing a discrete signal upon sensing a reduction of acceleration. Each programmer has the capability of providing thirty time-based discretes and one acceleration discrete. System redundancy is provided by one programmer backup of the other during flight.

7.1.4 FLIGHT CONTROLS COMPUTER (FCC). The flight controls computer contains all the active circuitry for conditioning the vehicle control signals. It accepts attitude control signals from the TARS, rate gyro system, and the Centaur guidance system (via the TARS). It conditions, amplifies, mixes, and distributes these signals to the thrust vector control devices. The FCC accepts discrete inputs from the flight programmers, the staging timer, and the Centaur guidance system through the electrical sequencing system. The FCC uses these inputs to change channel gains and pitch rates, and to switch signal conditioning channels in and out at appropriate times. The FCC also controls the dump programming of the TVC injectant fluid by comparing the actual utilization against a nominal schedule and controlling the dumping of excess fluid accordingly.

7.1.5 STAGING TIMER (ST). The staging timer is capable of issuing discrete commands on both a timed and an acceleration base. It first issues a discrete to the electrical sequencing system for Stage I engine start (87FS-1) as a result of sensing vehicle deceleration. Next is a timed discrete for Stage 0/I separation. After issuing the Stage 0/I separation signal, the staging timer time base is automatically reset. The next discrete, timed from Stage I/II separation, provides a signal for Centaur shroud separation backup. The staging timer time base is again reset. The final timed discrete is presently used as a spare.

7.1.6 FLIGHT CONTROLS SYSTEM SEQUENCE OF EVENTS. The flight controls system sequences inflight events required to satisfy the open loop trajectory shaping requirements, maintain vehicle stability, and command certain flight critical functions, such as staging and engine start. In addition, the flight controls system accepts discrete signals from AGE and the Stage I thrust chamber pressure switch (TCPS) via the electrical sequencing system. AGE controls all ground checkout and countdown functions.

Prior to liftoff, the flight controls system receives an AGE discrete signal to start the flight programmers. The flight programmers then issue "run" verification discretes which are used as "hold" monitors in the countdown sequence. At SRM ignition, an AGE discrete uncages the TARS gyros which then function as displacement gyros to provide the flight vehicle attitude reference. Simultaneously the AGE issues a discrete to the flight controls computer to start the first of four SRM, TVC injectant fluid dump levels. TVC dump improves performance by augmenting solid thrust and decreasing vehicle weight. The remaining three dump levels are sequenced by Stage 0 gain change discretes.

At liftoff, the flight controls computer is in the initial Stage 0 gain and dynamic state. Additional gain and dynamic changes are then sequenced during flight as required to maintain vehicle stability margins.

The program maneuvers required to attain the open loop trajectory are sequenced by the flight programmers, and the torquing rates are determined by mission-dependent plugs on the flight controls computer. The Centaur guidance system provides the initial roll maneuver torquing rate through the TARS roll guidance torquer. In addition, the CGS provides closed-loop trajectory-shaping commands in the pitch and yaw axes during Stage I and II flight. Upon completion of the roll maneuver, the first of 10 possible pitch rates is commanded. The remaining nine rates may be programmed throughout Stage 0, Stage I, and Stage II flight.

During the mid portion of Stage 0 flight, the Centaur guidance system provides pitch and yaw open loop bias programming to provide structural load relief through the maximum dynamic load (max q) region of the atmosphere. Near the end of Stage 0 flight, the flight controls system enables the ordnance power bus to provide primary power for all subsequent ordnance functions. The next flight-critical function sequenced by the flight controls system is Stage I engine start. When the staging timer and flight programmer B sense reduction of Stage 0 acceleration to a predetermined level, they redundantly issue a discrete signal to start the Stage I engines. For a short period, the Stage I and Stage 0 engines are then thrusting in parallel. After a fixed time, the staging timer and flight programmer B redundantly issue discretes to command Stage 0/I separation.

The Stage I engine shutdown, Stage I/II separation, and Stage II engine start signals are issued simultaneously when either of the Stage I engine thrust chamber pressure switches senses a reduction in chamber pressure (propellant depletion).

The physical Stage I/II separation (staging connector disconnect) sends a discrete to the flight controls system to change the computer gains and pitch rate and to start the second time base in the staging timer. A fixed time after separation, the staging timer issues a backup shroud separation signal. The CGS issues primary and secondary shroud separation signals for redundant commanding of Centaur standard shroud jettison.

The Stage II engine burns to propellant exhaustion, at which time flight programmer A issues a Stage II shutdown discrete based upon vehicle reduction in acceleration. The CGS will also issue a (backup)

Stage II shutdown signal by sensing the reduction in acceleration.

This ensures Stage II shutdown and Centaur separation in the event of the loss of the CGS discrete. The primary Stage II/Centaur separation discrete is issued by the CGS based upon vehicle acceleration monitored by the inertial platform. A (backup) Stage II/Centaur separation discrete is issued by flight programmer A, a fixed time after it issues Stage II shutdown.

A sequence of events for a typical flight is given in Table 7-1.

A pictorial representation of the sequence of events is illustrated in Figure 7-4.

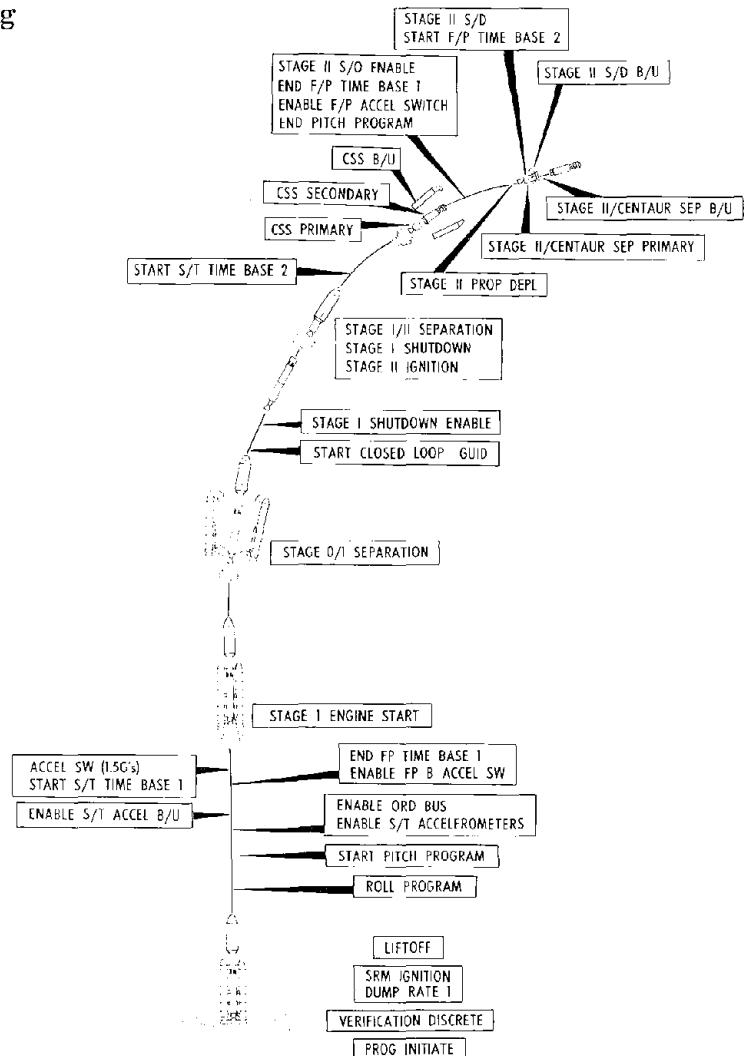


Figure 7-4. Titan III flight sequence.

Table 7-1. Typical sequence of events.

EVENT	APPROXIMATE TIME (SEC) FROM SRM IGNITION	SIGNAL SOURCE	REMARKS
1. Program initiate	-1.00	AGE	
2. Verification discrete	-0.70	F/P A&B	
3. SRM ignition; Start TVC Dump Rate 1.	0.0	AGE	
4. Liftoff; Start TVC Dump Rate 1 (B/U)	0.25	Umbilical Disconnect	
5. Roll program	6.0 - 10	CGS	Depending on amount of roll required.
6. Start Pitch Program	10	F/P A&B	Time constant over inclination range.
7. Enable ordnance bus; Enable S/T Accelerometers;	75	F/P A&B	
8. Enable S/T Acc. (B/U)	75	F/P B	
9. End F/P B Time Base 1; Enable F/P B Accel. Switch	96	F/P B	
10. Acceleration switch (1.5 G's); Start S/T Time Base 1; F/P B Time Base 2	112	S/T, F/P B	Time dependent on SRM performance
11. Stage I Start	112	S/T, F/P B	
12. Zero/I Separation;	124	S/T, F/P B	
13. Start closed loop guidance	132	CGS	
14. Stage I S/D Enable	245	F/P A&B	
15. Stage I S/D	258	TCPS	Stage I propellant depletion
Stage II start	258		
16. Start S/T Time Base 2	259	I/II Separation	
17. Centaur Shroud Separation Primary	268	CGS	
18. Centaur Shroud Secondary	268	CGS	500 ms ± 50 ms after primary CSS separation.
19. Centaur Shroud Separation (B/U)	288		
20. Stage II S/D Enable; End F/P A Time base 1; Enable F/P Accel. Switch; End pitch program	452	F/P A&B	
21. Stage II Propellant Depletion	468	--	
22. Stage II S/D	468	F/P A	Based on Accel. Decay; Activates at 1G.
Start F/P A Time Base 2			
23. Stage II S/D (B/U)	---	CGS	
24. Stage II/Centaur Separation Primary	475	CGS	
25. Stage II/Centaur Separation (B/U)	475.4	F/P A	7.4 sec after Stage II S/D

Abbreviations:

B/U	Backup	S/D	Shutdown
CGS	Centaur Guidance System	S/T	Staging Timer
CSS	Centaur Shroud Separation	TCPS	Thrust Chamber Pressure Switch
F/P A	Flight Programmer A		
F/P B	Flight Programmer B		

8

ELECTRICAL/ELECTRONIC SYSTEMS

This section presents a description of the Titan IIIE electrical/electronic systems.

SUBSECTION	SYSTEM	Page
8.1	Airborne Electrical Systems	8-1
8.2	Flight Termination System	8-6
8.3	Instrumentation and Telemetry System	8-10

The major components of the electrical/electronic systems are located on the Stage II forward compartment 2A truss (Figure 7-1).

8.1 AIRBORNE ELECTRICAL SYSTEMS

ELEMENTS:	Subsection
• Solid Rocket Motor Electrical System	8.1.1
• Core Vehicle Electrical System	8.1.2

FUNCTIONS:
<ul style="list-style-type: none">• Provide all vehicle d-c and 800-Hz a-c power• Transfer from ground to airborne power• Sequentially apply signals/commands as required

8.1.1 SOLID ROCKET MOTOR ELECTRICAL SYSTEM (Figure 8-1). Internal electrical power for each SRM is provided by three batteries. These batteries are 28V d-c silver-zinc, manually activated power supplies.

The SRM aft instrumentation power system (AIPS) battery has a two ampere-hour capacity and is located within the aft skirt section of each solid rocket motor. It provides all power requirements of the instrumentation components of the motor.

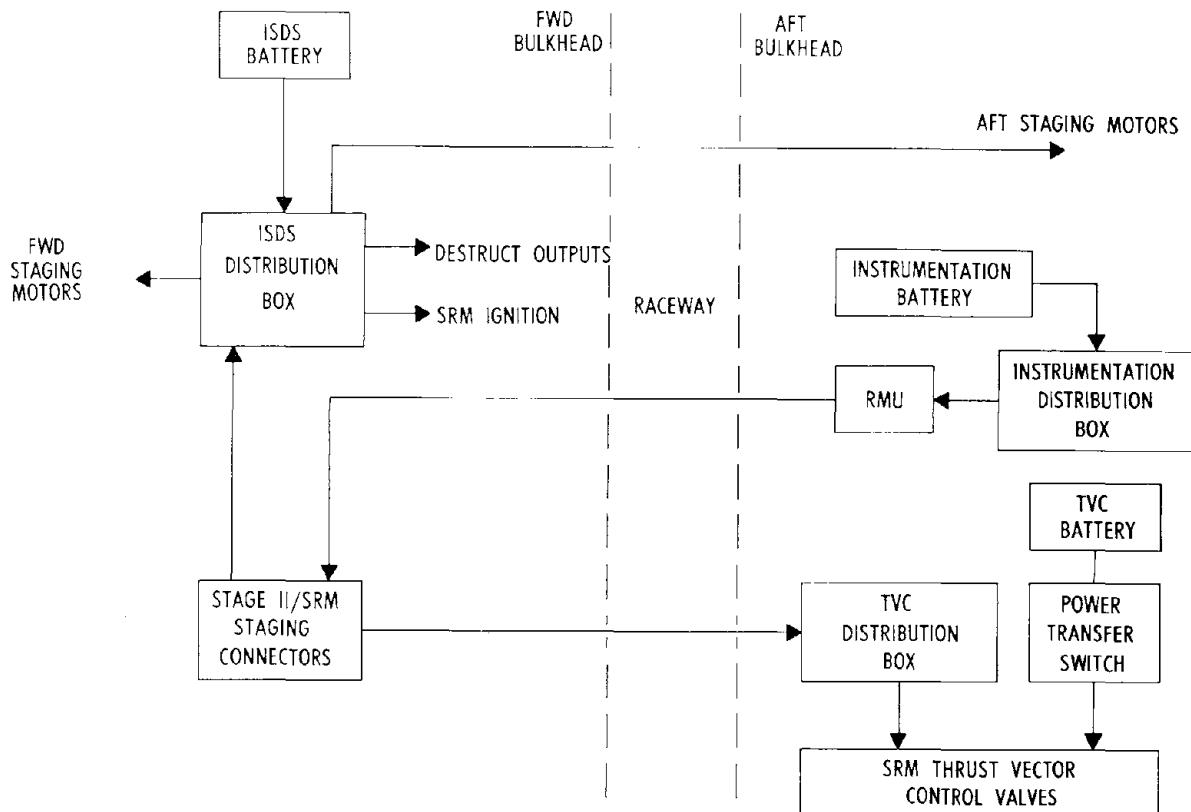


Figure 8-1. SRM electrical system block diagram.

The thrust vector control battery has a 27 ampere-hour capacity and is also located within the aft skirt section of the motor. It provides the power requirements to drive the electro-mechanical thrust vector control valves.

The inadvertent separation destruct system (ISDS) battery is rated at five ampere-hours and is located in the forward nose fairing of the SRM. This battery powers the SRM destruct system in the event of an SRM separation prior to the normal staging sequence.

System Operation. Initial operation of the system begins during the final phase of the countdown. AIPS power is transferred at T-31.7 seconds. The thrust vector control power transfer switch transfers TVC power from ground power to battery power at T-20 seconds. At T-0, the SRM ignition commands are transmitted through the Stage II electrical system and the ISDS distribution box to the igniter safe-arm device which ignites the SRM igniter. During flight, the Titan IIIE flight controls system will provide commands through the thrust vector control distribution box to control the opening and closing of the electro-mechanical valves.

In the event of a range-safety-generated destruct command, the destruct power is supplied through the Stage II/SRM staging connectors by the core vehicle command receivers and their associated power systems.

Power for the solid rocket motor staging functions is provided by the core vehicle transient power supply system. This power fires the forward and aft SRM separation ordnance and also provides power through the SRM ISDS distribution box for igniting the forward and aft staging motors.

The inadvertent separation destruct system battery provides power for destruct through the ISDS distribution box in case of separation of an SRM prior to normal sequenced separation, i.e., vehicle breakup.

The instrumentation battery provides power through the instrumentation distribution box to power all components of the SRM instrumentation system. Power is supplied to transducers, signal conditioners, and other instrumentation system components.

8.1.2 CORE VEHICLE ELECTRICAL SYSTEM (Figure 8-2). The function of the electrical system is to furnish and distribute power to the various airborne systems. Power is furnished to these systems from ground power supplies during checkout and readiness procedures. The electrical system includes power switches for transferring the systems to the airborne power supplies prior to launch. The airborne electrical system also includes a flight sequencing system that converts command signals from the flight controls programmers and/or guidance into events signals to perform flight operations.

The core vehicle airborne electrical system has been designed for maximum circuit simplicity with the following maintenance requirements in mind. Components are replaceable without requiring adjustment, calibration, or alteration. Components are standard and are all clearly marked, as are the mounting areas. Those components that may require servicing during readiness are located close to access doors and can be replaced without removing other equipment. In general, all mounting equipment and cable connections are designed to be as simple as possible. All critical items are tested or checked during system and/or subsystem tests.

The airborne electrical system consists of the following components:

- Batteries
- Motor driven switches
- Static inverter
- Relay packages
- Enable-disable circuits
- Squib firing circuits
- Diode and resistor assemblies
- Staging connectors
- Cabling, wiring, and connectors

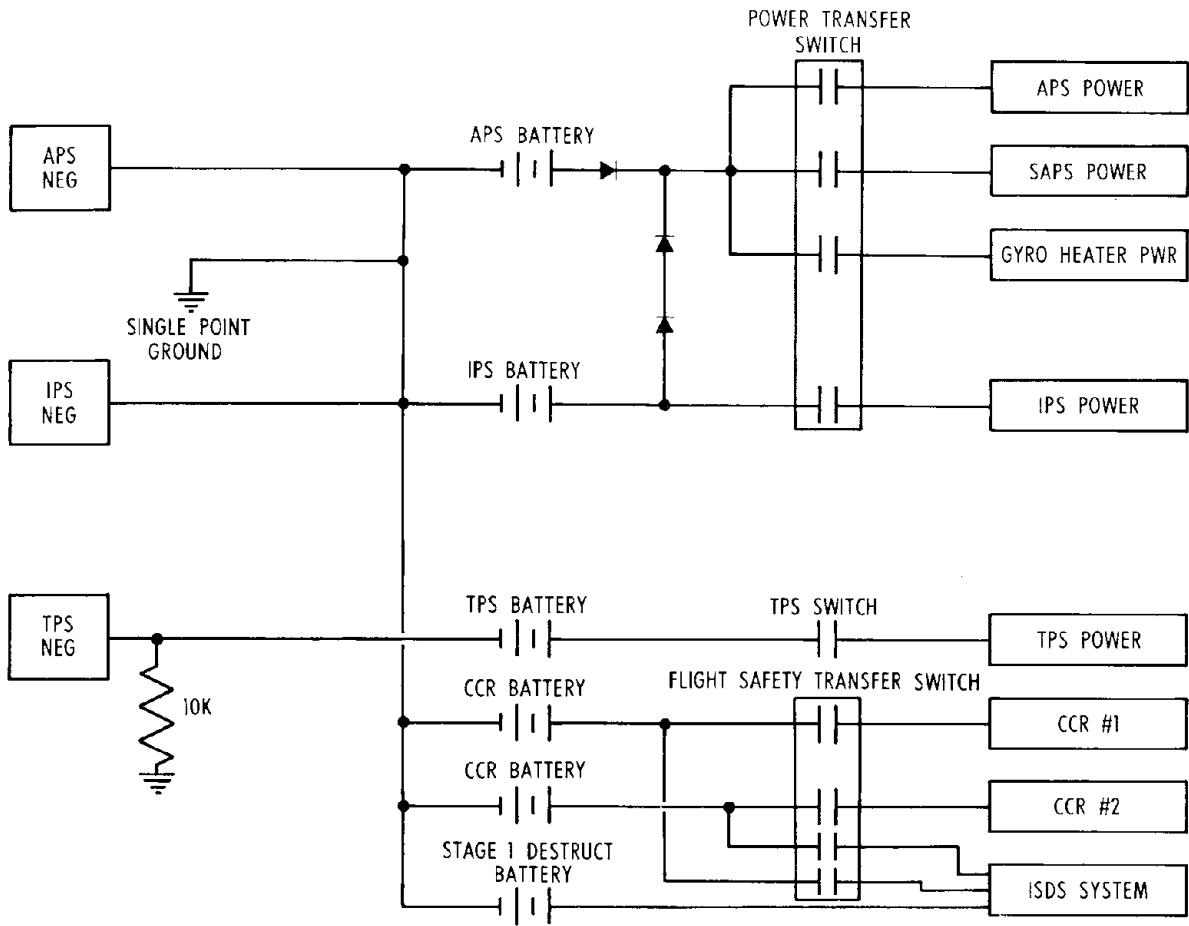


Figure 8-2. Core vehicle electrical system block diagram.

The core vehicle has five airborne power systems complete with distribution systems. They are the:

- Accessory power system (APS)
- Instrumentation power system (IPS)
- Command control receiver power system (CCRPS)
- Transient Power System (TPS)
- Stage I destruct power system

The primary batteries used in the airborne electrical systems are 28V d-c silver-zinc manually-activated batteries. Each battery cell is activated with electrolyte provided in a separate container. After activation, the battery must stand for a minimum of two hours before supplying its full rated capacity. The batteries are capable of remaining fully activated for 30 days when not connected to a load, and then are capable of providing full power when required. There are 5 four-ampere-hour batteries: A PS, IPS, CCRPS (2), and Stage I Destruct. The TPS battery is rated at 25 ampere-hours.

Accessory Power System. The accessory power system consists of one 4-ampere-hour battery to provide power to equipment essential to flight. The battery and associated power transfer switch are located on the Stage II forward compartment truss. Contacts of the Stage II power transfer switch are used to supply the output from this battery to the gyro heaters on both Stage I and Stage II, the flight controls static inverter, and to the sequencing system. The system also provides the power and signal requirements to the flight controls system, the TVC interface, the liquid rocket engine interface, and the Centaur vehicle interface.

Instrumentation Power System. The instrumentation power system is also located on the forward compartment truss and provides power through the same power transfer switch as the APS battery. After power transfer, this battery provides power to satisfy two requirements. Its primary output is to power all core instrumentation/telemetry components. Loss of APS power would result in a loss of mission; therefore, the IPS battery will back up the APS battery if this loss should occur, thus assuring mission success. This redundancy is indicated in Figure 8-2. Analysis of vehicle power requirements indicate that the IPS battery is capable of powering both systems during flight of the vehicle.

Transient Power System. The transient power system (TPS battery and the associated TPS power transfer switch) is also located on the forward compartment truss. This 25-ampere-hour battery is used to provide power to fire all operationally sequenced airborne ordnance systems. Power to the ordnance bus is supplied through the TPS power transfer switch. Power transfer on this bus is accomplished in flight by discrete signals from flight programmers A & B approximately 75 seconds after liftoff. The TPS system then provides power for all the sequenced ordnance functions during the remainder of the flight.

Command Control Receiver Power System. The command control receiver power system consists of two 4-ampere-hour batteries to power the two command receivers. It also provides power to the hot wire portion of the inadvertent separation destruct system (Subsection 8.2.2). Transfer of power to command receivers 1 and 2 is accomplished by the flight safety power transfer switch.

Stage I Destruct Power System. The Stage I destruct battery is located between tanks on Stage I and provides power for Stage I destruct in the event that Stage I separates from Stage II prior to the normal staging sequence. The battery supplies power to the inadvertent separation destruct system.

Power Transfer Switches. The three motor-driven switches are positive-action switches for transfer of power supply voltages. Two handle currents up to 20 amperes. The third, the TPS power transfer switch, will handle 200 amperes. Contact closure is accomplished by means of a motor-driven gear train that positions a contact mounting block to open or close the contacts. The switches are hermetically sealed and filled with dry inert gas. The contacts are designed in such a way that a wiping action occurs

as the contacts are opened or closed. This wiping action ensures a positive contact. Contact action is break-before-make. In addition to the power contacts, each switch contains indicator contacts which are used to indicate whether the switch is open or closed. One switch, Stage II power transfer switch, transfers the APS and IPS busses. Another switch, TPS power transfer switch, is used to transfer the TPS bus. The third switch, flight safety power transfer switch, is used to control power to the command receivers and to the inadvertent separation destruct system.

Power transfer on all busses except the TPS occurs at approximately T-32 seconds in the countdown. As indicated above, the transient power transfer switch is operated in flight.

Static Inverter. The static inverter (Figure 7-1) converts 28V d-c from the APS bus to 800 Hz for the flight controls computer, the staging timer, the TARS and rate gyros, and the acceleration sensors. The inverter function is shown in the flight controls system block diagram, Figure 7-2.

8.2 FLIGHT TERMINATION SYSTEM. The flight termination system comprises two safety systems: the command control system and the inadvertent separation destruct system. These systems are compatible with the range ground systems and are almost entirely redundant.

ELEMENTS:	Subsection
• Command Control System	8.2.1
• Inadvertent Separation Destruct System	8.2.2

FUNCTIONS:
• Shutdown Stage I and II engines • Destroy Entire Vehicle Upon Command • Automatically Destroy Stage 0 and Stage I in Case of Vehicle Break-up

8.2.1 COMMAND CONTROL SYSTEM. The command control system (Figure 8-3) is used during vehicle flight by range safety personnel. The Titan IIIE vehicle is tracked by the Cape Kennedy Air Force Station (CKAFS) ground tracking stations, utilizing the Centaur C-band tracking system. This system provides continuous vehicle position to the Air Force Range Safety Officer. The command control system provides capability for initiating engine shutdown and vehicle destruction. There are two identical command receivers (Figure 8-4), individually powered from separate batteries to assure reliability. Both receivers and batteries are located on the Stage II truss.

The command receiver contains solid-state receiver and decoder sections constituting a compatible link for reception and decoding of tone-coded FM radio signals from a command transmitter at a ground station.

Audio tones detected by the receiver section are supplied to three channel filters in the decoder section. Channel 1 and 5 outputs must be present for an engine shutdown command. Destruct logic output requires inputs from channels 1 and 2.

The receiver supplies sufficient power to operate the vehicle engine-shutdown circuits when this command is detected. A destruct command causes a receiver output of sufficient power to operate all destruct initiators.

The antenna system for the command control receivers includes two antennas and a four-port junction. Each antenna consists of a radome, incorporating printed circuitry, mounted on an aluminum cavity. The antennas provide coverage for reception of the CW and FM-CW signals transmitted by command control ground stations. They have left-hand circular polarization. Their efficiency is at least 30 percent for reception from a linear source and 70 percent for circular reception, to provide sufficient signal for receiver operation when within 1,000 miles of a ground destruct transmitter.

The four-port junction is a hybrid circuit network. It electrically connects the two antennas with the two receivers while maintaining at least 16 db isolation between each antenna and each receiver. The unit provides two antenna connectors and two receiver connectors.

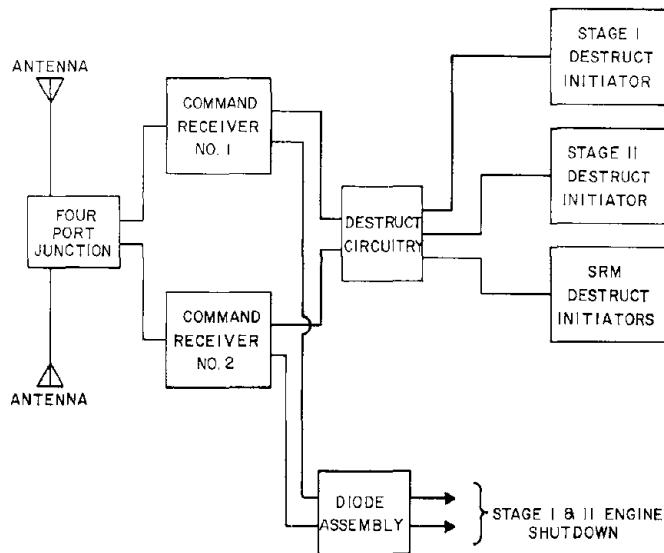


Figure 8-3. Command control system block diagram.

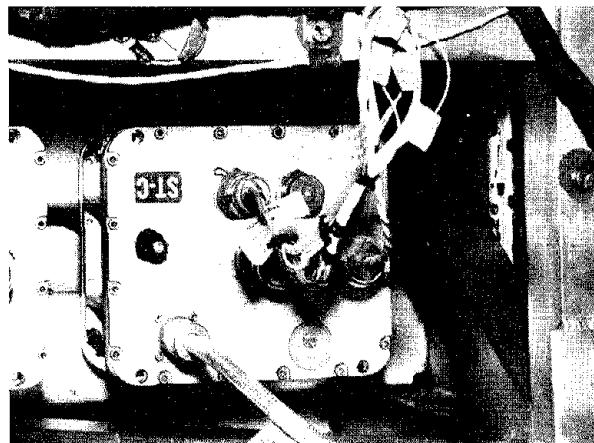


Figure 8-4. Command control receivers.

The destruct system provides vehicle destruction by intermixing of propellants in Stage I and II. Destruction of the solid rocket motors (SRM) is accomplished by explosively ripping open the SRM cases. The destruct system can be initiated by signals from the command control receivers or from the inadvertent separation destruct system. The ordnance destruct system consists of initiators, primacord, and destruct charges. The primacord and destruct charges are discussed in Section 6.

The initiators are equipped with a safety-pin that locks the units positively in the safe position during handling, transport, storage, and installation. There are two types of initiators. Type I is the live initiator employed for igniting the destruct charges. Type I initiators are installed prior to vehicle launch and are identifiable by a red tag attached to the safety pin. Type II is the initiator simulator, utilized for destruct system tests conducted by the ground checkout equipment. The initiator simulator in conjunction with standard ordnance circuit verification units (SOCVUs) electrically simulates the operation of the live unit with the exception of explosive detonation.

8.2.2 INADVERTENT SEPARATION

DESTRUCT SYSTEM. Titan IIIE contains three inadvertent separation destruct system (ISDS) circuits. An ISDS circuit is contained in SRM1, SRM2, and Stage I (Figure 8-5). The system automatically generates destruct signals in the event of vehicle breakup or premature stage separation. The system will destroy the prematurely-separated stage, or stages, which cannot be destroyed with the command control receivers. The system is connected in such a way as to make maximum utilization of components provided for command destruct. The components for the SRM ISDS are supplied by United Technology Center.

The ISDS is a hot-wire system. The loss of the 28V d-c signal (breaking the hot wire) in conjunction with loss of ground connections to the ISDS in any stage results in destruction of that stage only.

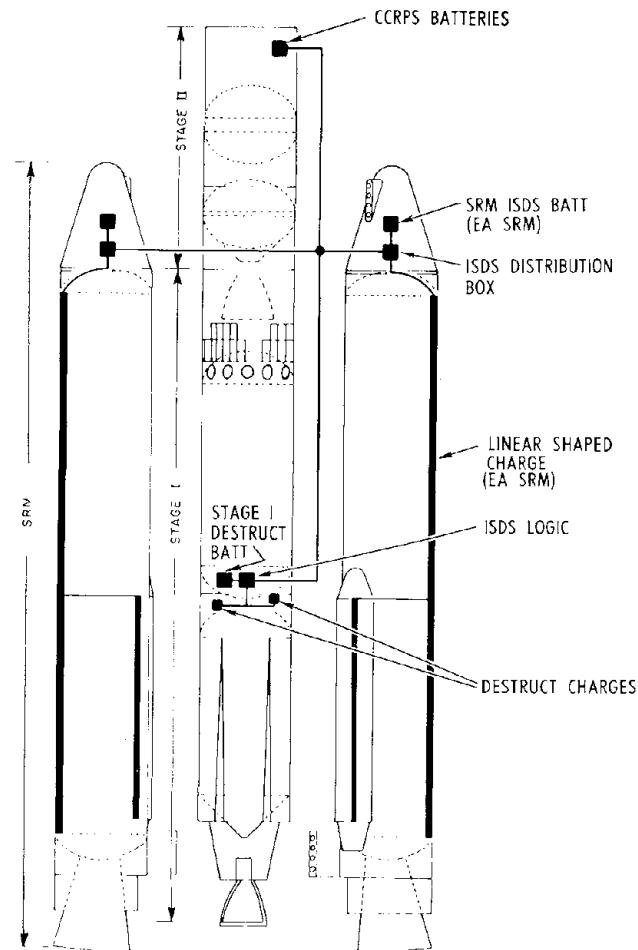


Figure 8-5. ISDS component locations.

The ISDS in each stage contains a destruct battery, a destruct monitor circuit, an arm/safe switch, and two squib firing circuits.

Batteries supplying power for the ISDS circuits are: SRM1 ISDS battery, SRM2 ISDS battery, and Stage I destruct battery. These batteries furnish necessary power for destruct as previously discussed.

The ISDS destruct monitor circuits (Figure 8-6) are "not" gates, which supply an output signal with no input present. They monitor the ISDS hot wire. Both hot wires must break for an output from the destruct monitor circuit.

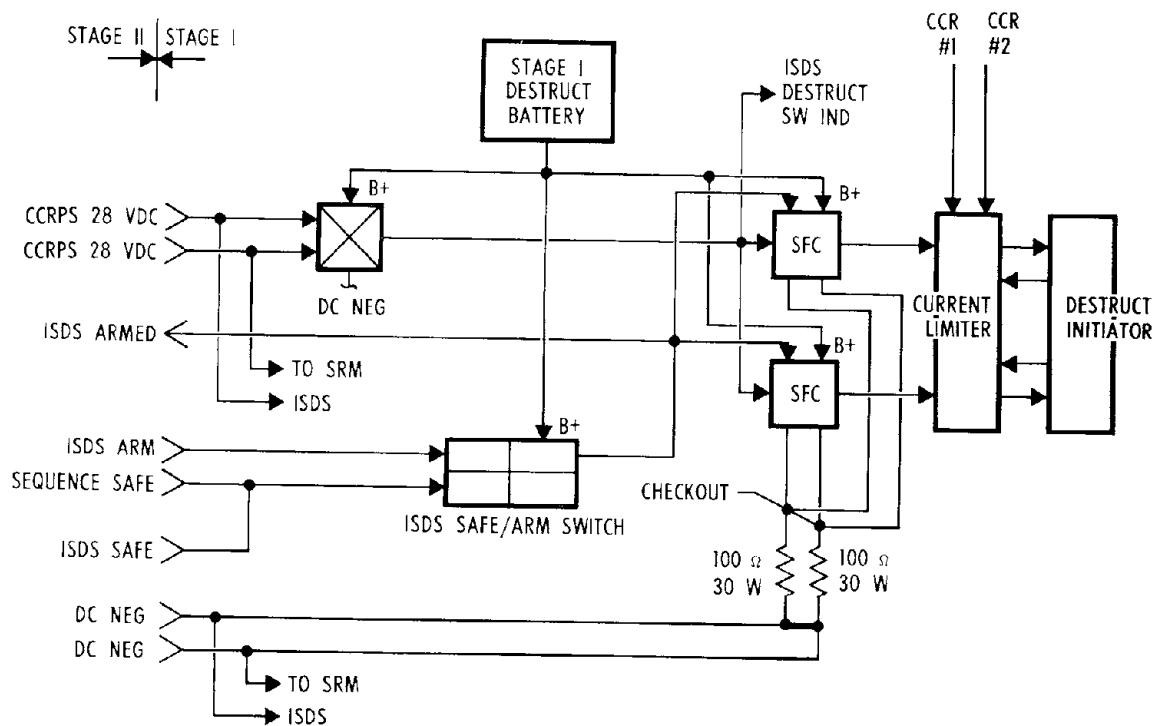


Figure 8-6. ISDS destruct monitor block diagram.

ISDS arm/safe switches are enable-disable circuits. Their outputs supply power to the squib firing circuits (SFC). They provide for simultaneous arming or safing of all ISDS circuits. They also permit disarming individual ISDS circuits prior to normal staging. When an arm/safe switch is safe, the SFC is disabled and no destruct signal can be generated when normal stage separation occurs. All ISDS arm/safe switches are armed during launch countdown and their outputs must be present for a "GO" condition. They are armed and monitored by the AGE. In addition to safing the Stage I arm/safe switch, the safing of Stage I destruct initiator is accomplished at Stage I separation.

The squib firing circuits of the ISDS are used to provide high firing current to the initiator primers. Two SFCs are used in each stage controlled by the ISDS to provide redundancy in firing both destruct initiator primers. The SFC produces a fire signal upon receiving an input from the arm/safe switch, a fire signal from the ISDS monitor not gates, and a loss of both ISDS grounds.

8.3 INSTRUMENTATION AND TELEMETRY SYSTEM

ELEMENTS:

- Measurement Devices
- Signal Conditioners
- Remote Multiplexer Units (RMUs)
- Converter Unit (CU)
- Telemetry Transmitter and Antenna

FUNCTIONS:

- Collects and Transmits Vehicle Data
- Provides Data for Post-Flight Analysis

The airborne instrumentation and telemetry system consists of transducers, signal conditioners, the remote multiplexer instrumentation system (RMIS), the transmitter, and the antenna. The RMIS consists of the remote multiplexer units (RMUs) and a converter unit (CU) which are illustrated in Figure 8-7. A simplified block diagram of all components is shown in Figure 8-8. The system collects, multiplexes, encodes, and transmits analog and 28V d-c bilevel measurements made on the airborne systems during checkout, launch, and flight of the Titan IIIE vehicle.

The pulse code modulation (PCM) sequence of transmission of vehicle measurements is programmable within the RMIS. This allows flexibility in the PCM format and permits any reasonable choice of sample rate.

8.3.1 SYSTEM OPERATION. Test data inputs to the instrumentation system originate from sensors monitoring physical parameters and electrical signals from the various vehicle subsystems. The sensors

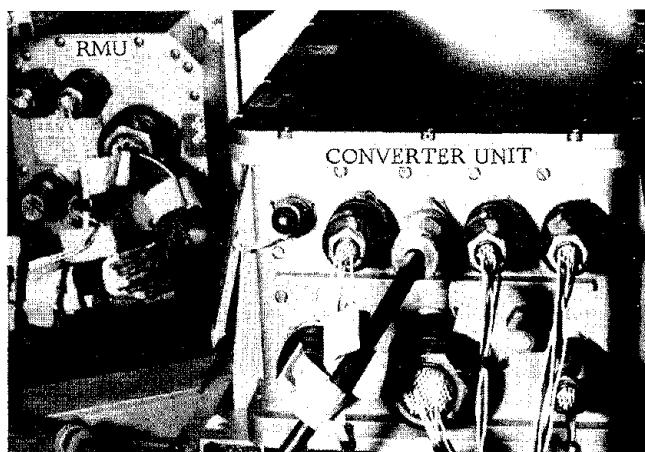


Figure 8-7. Airborne instrumentation and telemetry system components.

convert mechanical conditions, liquid levels, gas and liquid pressure, temperature, and acceleration to proportional electrical signals.

All data signals are either analog or bilevel dc. Signals not compatible with the required inputs to the encoding equipment are routed to signal conditioners.

The data signals are sampled and encoded by the RMIS (Figure 8-9), which comprises a group of up to 16 remote multiplexer units and a single converter unit. The CU is capable of accepting up to 80 bilevel inputs in addition to the analog RMU outputs. Up to 32 analog signals having a range of 0 to 40 mvdc can be accepted by an RMU. The RMU samples, amplifies, and holds the assigned low-level input signals to provide a serial pulse amplitude modulated (PAM) output train to the converter unit. The output of each RMU is connected to a pair of redundant data lines that feed the multiplexed PAM data to the converter. Logic control signals are addressed and transmitted from the converter programmer to each of the RMUs on a pair of redundant address lines. Each RMU has a unique address. The converter performs analog-to-digital conversion of the PAM input signal and provides a serial PCM data train output to an S-band PCM/FM transmitter.

The transmitter accepts the RMIS serial pulse train from the converter and generates a frequency modulated RF signal for transmission to ground receiving stations. The transmitter output power is 24 watts nominal. The RF output of the transmitter is routed by coaxial cable to a single broad-beam antenna. The

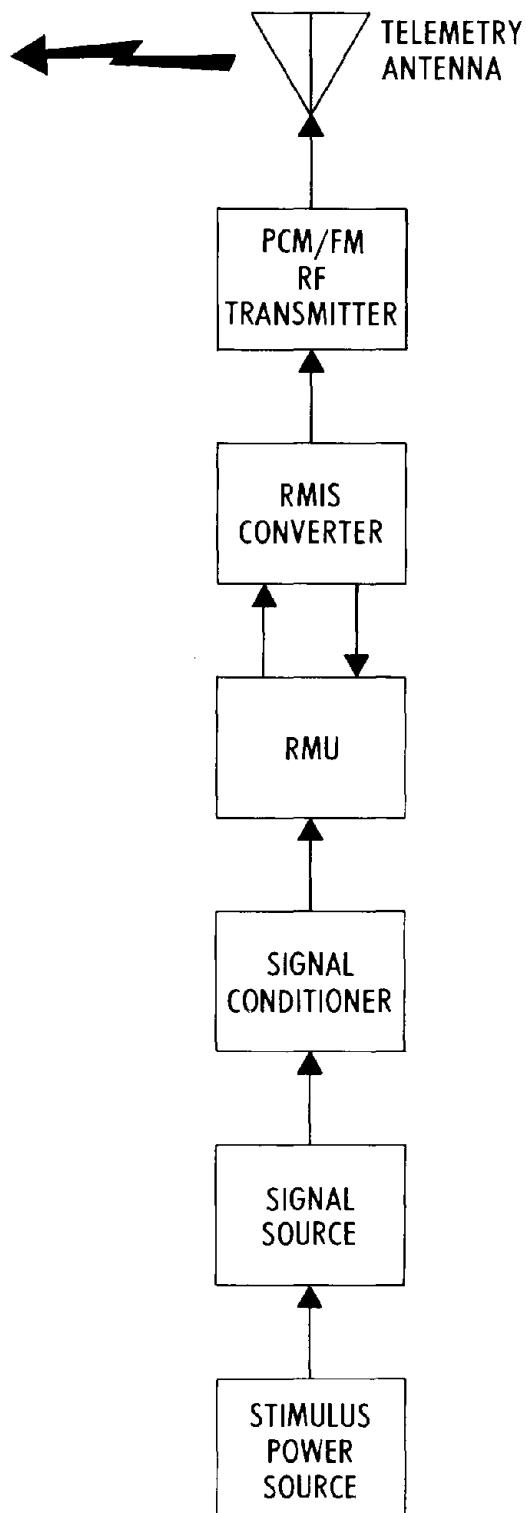


Figure 8-8. Airborne instrumentation and telemetry system simplified block diagram.

radiation pattern of the antenna is such that it provides continuous adequate coverage to the telemetry ground stations during checkout and flight.

A block diagram of the RMIS presently used on the Titan IIIE vehicle is illustrated in Figure 8-9.

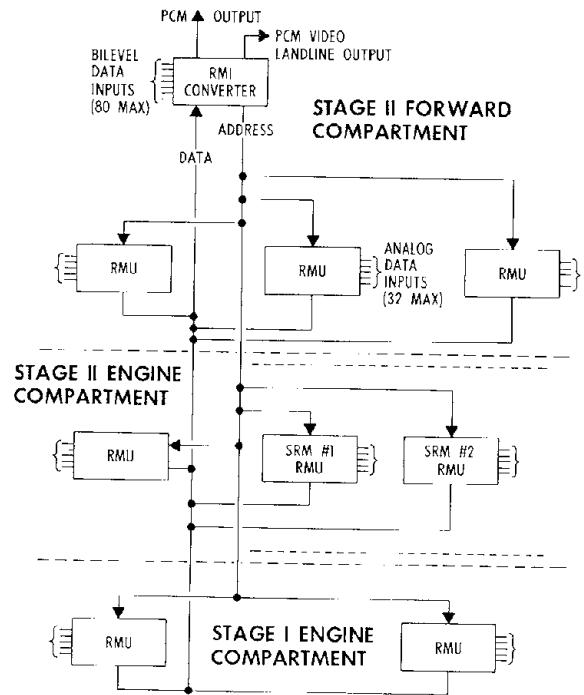


Figure 8-9. Titan IIIE RMIS block diagram.

9

TITAN IIIE AEROSPACE GROUND EQUIPMENT

This section describes the Titan IIIE aerospace ground equipment.

SUBSECTION	SYSTEM	Page
9.1	Checkout and Launch Control Equipment	9-1
9.2	Ground Instrumentation Equipment	9-5

9.1 CHECKOUT AND LAUNCH CONTROL EQUIPMENT

ELEMENTS:	Subsection
• Equipment General Description	9.1.1
• Detailed Component Functions	9.1.2

FUNCTIONS:
<ul style="list-style-type: none">• Perform Detailed Checkout of Vehicle Systems• Control all Phases of the Countdown• Monitor Status of Vehicle Systems• Record Data for Post-Test Analysis

Checkout, control, and monitoring of the various Titan airborne systems is a function of the aerospace ground equipment (AGE). This includes: system calibration and functional testing of the flight controls system, flight safety system, electrical system, and the instrumentation system. The AGE is capable of performing tests on individual components, and total systems and subsystems tests, including end-to-end vehicle tests. Major components of the AGE are located in the launch control van (Figure 9-1) with the exception of the launch control console which is located in the vertical integration building (VIB) control center. The following is a list of AGE components and their functions.

<u>Item</u>	<u>Function</u>
Vehicle Checkout Set (VECOS)	Checkout Flight Controls System
Flight Safety Checkout Equipment	Checkout Flight Safety System
Data Recording Set (DRS)	Record all Data
Launch Control Console (LCC)	Control Countdown
Control Monitor Group (CMG)	Sequence Functions
Van Power Distribution Control (VPDC)	Provide Power Requirements
Data Transmission Set (DTS)	Transmit Data

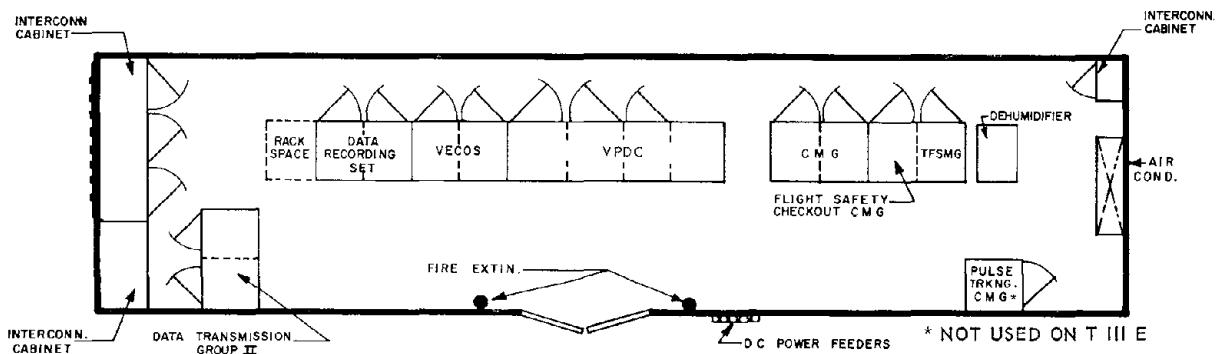


Figure 9-1. Launch control van.

9.1.1 EQUIPMENT GENERAL DESCRIPTION (Figure 9-2). Prior to launch pad evacuation, the vehicle checkout equipment (VECOS, FSGM, AND DRS) provide the capability to check out the major airborne systems of the Titan IIIE vehicle. These system tests are performed on individual components, and culminate in complete end-to-end tests of the individual systems. During these tests, the DRS monitors each system and provides post-test data for comparison with a predetermined criteria. After pad evacuation, these same tests may be performed by utilizing remote units in the VIB.

The launch countdown sequence is initiated and monitored from the launch control console in the launch control center (Figure 9-3). The Titan IIIE countdown is divided into three phases; the R count, the T count, and the automatic sequence. The R count begins at T-5 days and includes those checkout and service functions necessary to bring the vehicle to an operational readiness condition. The T count is picked up at approximately T-620 minutes and continues to T-32 seconds where the final phase, known as the automatic sequence, is initiated. All functions performed during this period of time are sequenced automatically by the control monitor group.

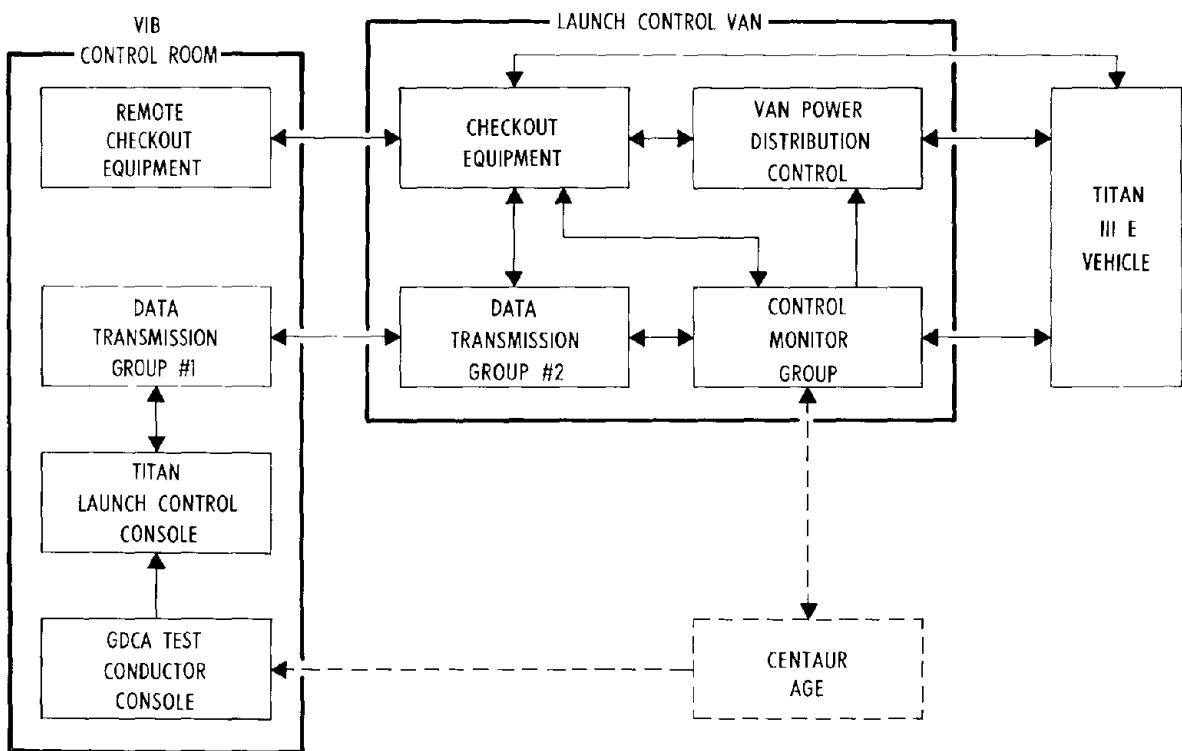


Figure 9-2. Checkout and launch control equipment block diagram.



Figure 9-3. Launch control center.

9.1.2 DETAILED COMPONENT FUNCTIONS. Functional descriptions of each major AGE component are given in the following paragraphs.

Vehicle Checkout Set. The vehicle checkout set (VECOS) provides the stimuli and monitoring for checking the flight controls system, hydraulic systems, and level sensors. Vehicle systems are verified in an end-to-end manner by applying the proper stimuli (analog and discrete) input command to the appropriate subsystem within the vehicle. The output of this subsystem is then monitored by the vehicle checkout set for correct performance. Malfunctions are identified to a black-box level, and are displayed on vehicle checkout set front panel indicators. A punched-tape programmer automatically controls tests.

Flight Safety Checkout Equipment. The flight safety monitor group (FSMG) and the flight safety checkout CMG (FSCMG) are used to control and monitor the vehicle flight safety subsystem. They are dual rack units that monitor functions of the airborne command receivers, airborne initiators, and inadvertent stage separation safe/arm switching devices. They also monitor ground power to the airborne flight safety equipment.

Data Recording Set. During all tests and the launch countdown, the data recording set (DRS) records on tape initial conditions and all subsequent changes on a time basis. It has a capability of receiving 750 inputs from various systems. This data is used to verify proper operation of AGE and airborne systems, and to support malfunction isolation if a problem occurs.

Launch Control Console. The launch control console (LCC) is a one-man unit housing controls, monitors, and communications for launch functions. Pushbuttons on the launch control and monitor panel control the following functions: commence countdown, reset countdown, manual hold, launch, shutdown, and reset shutdown. Applying and monitoring of ground power to the vehicle is also accomplished from this panel. System holds, water spray control, and core prevalve controls are all located on this panel. The LCC provides the capability of initiating test and launch activities, monitoring launch and test activities, and controlling corrective action on the pad.

Control Monitor Group. The control monitor group (CMG): (1) controls a time-based countdown for the complete vehicle, (2) provides automatic control and monitoring of Titan/Centaur vehicle launch and prelaunch functions, (3) provides hold and shutdown capability during the launch sequence from T-90 minutes to T-0.5 seconds, (4) has reset capability, (5) provides capability for patching input and output signals, (6) provides input simulation of signals and control of simulator-testers during the combined systems test, and (7) drives countdown readout indicators.

Van Power Distribution Control. The van power distribution control (VPDC) is located in the launch control van and provides or controls all ground and airborne power required during checkout and countdown. The unit operates under the control of the various items of launch control and checkout equipment. As an example, the control monitor group may issue the discrete command, "open core prevalves" to the van power distribution control. The power required for the functions is then supplied directly from the VPDC to the applicable airborne system, and the function is performed.

The VPDC also provides or controls all continuous power required to the vehicle until vehicle power transfer, which occurs during the final phase of the launch countdown. Subsequent to power transfer, all vehicle power requirements are provided by the airborne batteries.

Data Transmission Set. The data transmission set (DTS) consists of two groups: data transmission group I and data transmission group II.

Data Transmission group I is a two-rack unit housed in the data transmission room in the instrumentation control center area. Group I conditions and multiplexes command signals from the launch control center and transmits to data transmission Group II. It also receives and demultiplexes monitoring signals from group II for readout in the control center.

Data transmission group II is a two-rack unit located in the launch control van. It receives and demultiplexes signals from data transmission group I and transmits these signals to the launch control and checkout equipment in the van. Monitoring signals are transmitted by data transmission group II to data transmission group I in the instrumentation control center for readout.

9.2 GROUND INSTRUMENTATION EQUIPMENT

The Titan III ground instrumentation equipment (GIE) is located in two separate areas: the instrumentation van (Figure 9-4) and the vertical integration building instrumentation control center (Figure 9-5). These areas are interconnected by a landline system (Figure 9-6). The GIE is designed to process instrumentation measurements from the Titan III and Centaur vehicles.

9.2.1 INSTRUMENTATION VAN. During ground operation, SRM and Centaur measurements are supplied directly from measurement devices as analog voltages and bilevel signals to a landline PCM encoder in the instrumentation van. The PCM encoder output is converted into an FM signal and routed by coax lines to the ground instrumentation equipment in the VIB. The PCM output of the RMIS converter unit may also be routed by vehicle umbilical connection to the van where it is modulated onto an FM carrier and sent through coax cable to the VIB.

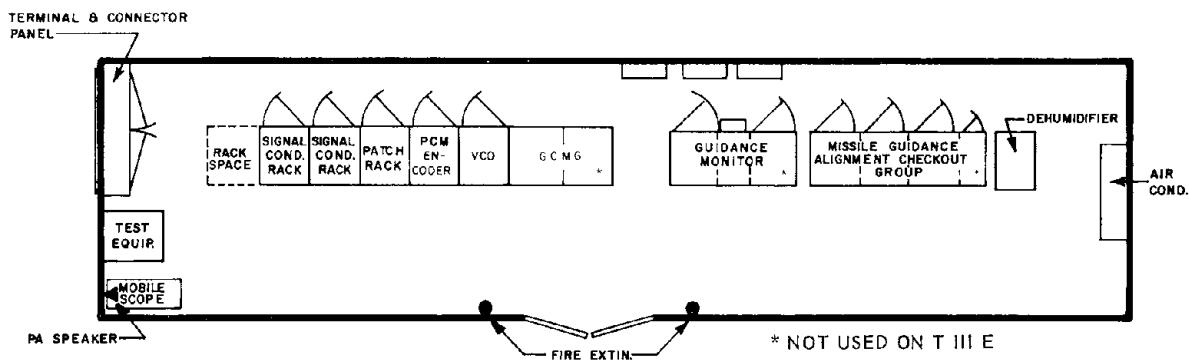


Figure 9-4. Instrumentation van.



Figure 9-5. Instrumentation control center (VIB).

Prior to launch, the Centaur PCM data is supplied to the Centaur mobile transfer room (MTR) by coax cable where it is converted to an FM signal and routed to the VIB.

9.2.2 VIB INSTRUMENTATION CONTROL CENTER. The Titan and Centaur PCM landline signals are routed to patch racks in the instrumentation control center. From the patch racks, the Titan PCM data is processed through a programmable decoder which decommutes and decodes the signals to reproduce the original analog voltages and bilevel measurements. These outputs are supplied to graphic recorders and meters for data analysis and "quick look" purposes.

The Centaur measurements are also routed to patch racks for distribution to the computer controlled launch set (CCLS) located at Launch Complex 36. These Centaur measurements are also routed to the Titan programmable decoder in the VIB. This group of measurements is supplied to graphic recorders, meters, and lamps on Centaur equipment consoles in the VIB.

When the airborne telemetry transmitters are operating prior to launch, the Titan and Centaur PCM/FM signals are received at the VIB through an antenna on the roof. During flight, this data is received by the central instrumentation facility (CIF) and routed by coax cable to the VIB instrumentation control center for recording and future playback.

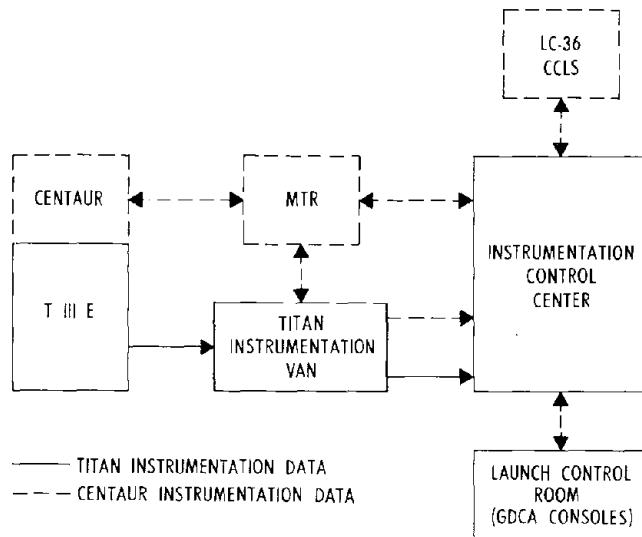


Figure 9-6. Landline system data flow.