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REUSABLE CENTAUR STUDY

VOLUME II ✦ FINAL REPORT

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VOLUME II • FINAL REPORT

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REUSABLE CENTAUR STUDY

Volume II – Final Report

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FOREWORD

This final report on the Reusable Centaur Study was prepared by the Convair Aerospace Division of General Dynamics Corporation for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-30290. The NASA Study Manager was James B. Brewer.

The study results were developed during the period from June to December 1973. A five-volume "data dump" was furnished MSFC on 25 September. Final presentations were made at NASA/MSFC on 17 January 1974 and at SAMSO/Aerospace Corporation on 25 January 1974. This report consists of two volumes:

Volume I Executive Summary

Volume II Final Report

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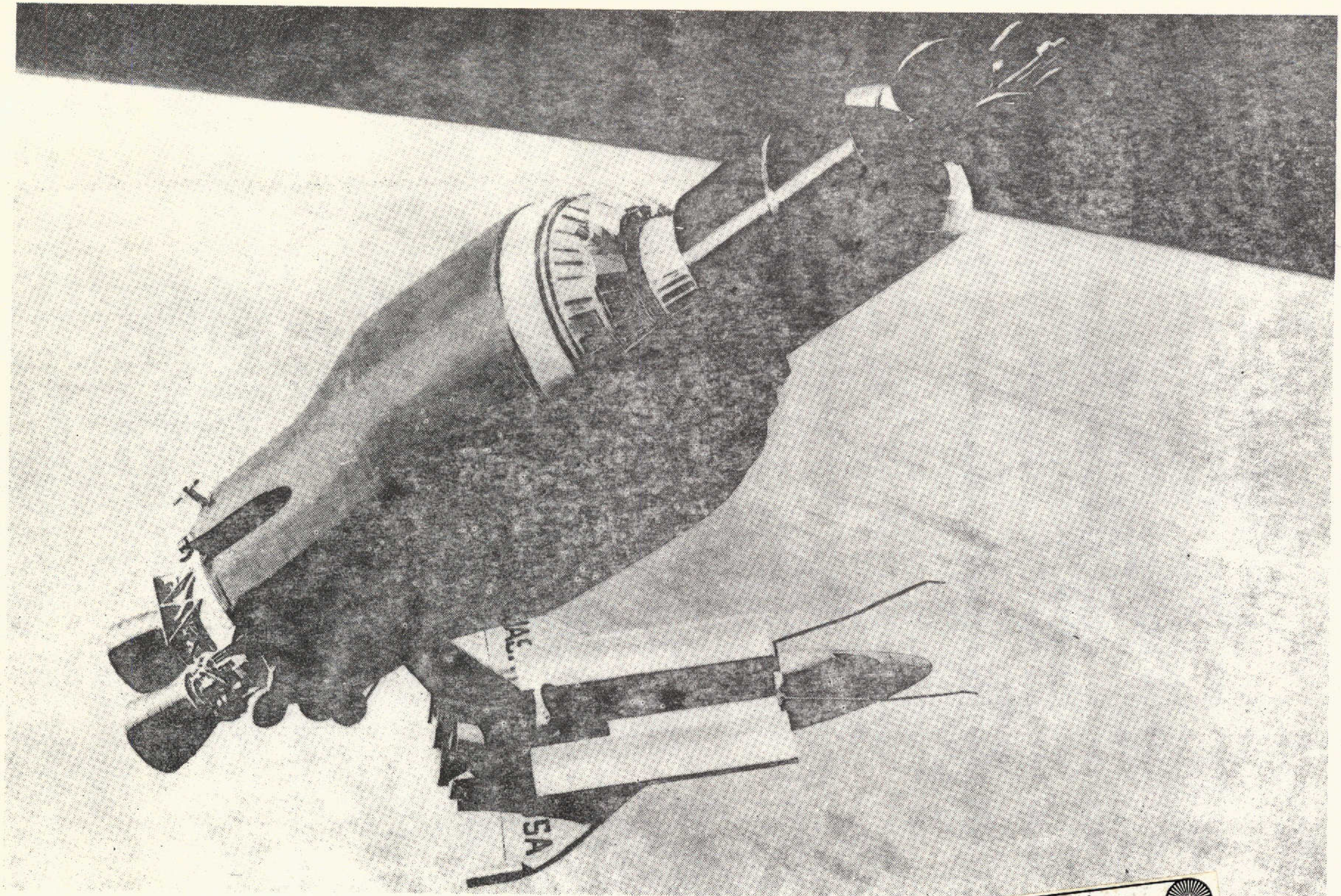


Reusable Centaur in Flight

SUMMARY

The Space Shuttle will deliver large payloads to low earth orbit. About half of the planned missions require an upper stage to reach higher orbits, such as synchronous, or to escape for planetary probes. Minimum development funding will be available for a new upper stage (Tug) until after 1978 when Shuttle expenditures begin to decline. Therefore existing upper stages, modified for Shuttle compatibility and reuse at minimum cost, are being considered as an initial Orbit-to-Orbit Stage (OOS).

This study developed data for OOS versions based on the current Centaur high energy upper stage. This vehicle has flown 23 operational missions, and has a future mission backlog through 1979. Centaur uses 30,000 pounds of liquid oxygen and liquid hydrogen propellants contained in a pressure-stabilized, stainless steel tank. The liquid oxygen tank is aft, separated from the liquid hydrogen by a double-wall, evacuated intermediate bulkhead. Twin Pratt & Whitney RL10A-3-3 engines provide the main impulse. A hydrogen peroxide auxiliary system provides attitude control during coast as well as turbine drive for the tank-mounted boost pumps. Aluminized Mylar radiation shielding provides thermal protection during mission coast periods, significantly reducing space heating rates. A fully integrated astrionics system uses a central digital computer for software control of the vehicle during flight operations.

The D-1T Centaur is the key element in NASA's near future space program. It is scheduled to fly Viking, Helios, and Mariner-Jupiter-Saturn missions.

During the Reusable Centaur and related studies, many configurations were considered, each with advantages, each satisfying different priorities. Configurations studied range from 22 to 35 feet long with 30,000 to 53,000 pounds of propellants. The Reusable Centaur version used in the Government assessment during November and December 1973 is 28 feet long with 47,000 pounds of propellant.

Reusable Centaur is a low risk development since it needs no new technology. Sixty-three percent of the components are flying today on Centaur, 25% are modified existing hardware, and only 12% are new. The Reusable Centaur's main engines are uniquely suitable for OOS application in that they possess a combination of features not found in other engines: (1) multi-flight reuse without any change to the existing RL10A-3-3 qualified configuration, (2) clean, non-toxic propellants, (3) high performance, and (4) minimum turnaround maintenance between flights.

Reusable Centaur has the high performance inherent in a liquid oxygen/liquid hydrogen cryogenic stage: it will place 4600 pounds of payload in synchronous equatorial orbit and return the stage to the Shuttle for reentry and reuse without an expensive, expendable, solid upper stage. This high performance allows multiple payloads to be flown and gives confidence in actually achieving OOS performance and reliability requirements. Centaur has the potential to achieve payload retrieval capability should that eventually be desired.

Centaur is inherently reusable. The Shuttle ascent environment is very similar to Titan's including acoustics and vibration. A review of current Centaur component specifications and qualification test data indicates that most D-1T components have received sufficient life testing to confirm their reusability for 10 to 20 Reusable Centaur missions.

Reusable Centaur is a safe configuration compatible with manned operations in the Shuttle. All Centaur subsystems are at least fail safe, many are first fail operational. The Centaur vehicle can be safed in any flight or ground abort mode. No single failure of any Centaur component will preclude Orbiter abort capability.

Initial development cost, which varies with vehicle capability, is estimated to be $77.2M for the Reusable Centaur. Thirty-five percent of this is for a dedicated flight test program and ground test program. The investment includes six new vehicles, which will be flown a maximum of 16 times each. The cost of a four-year program to put 112 payloads in orbit is $212M.

Centaur's high performance and inherent reusability means investment in a small fleet and few expendable kick stages. This means that routine reuse of the Centaur will cost $800,000 a launch, which is truly low recurring cost. Its low life cycle cost makes Centaur a very cost-effective OOS candidate.

SECTION 1

INTRODUCTION AND BACKGROUND

Under NASA Marshall Space Flight Center Contract NAS8-30290, Convair Aerospace Division of General Dynamics performed an eight-month study of the Reusable Centaur for early use as an initial upper stage with the Space Shuttle. The primary study objective was to provide realistic capability and cost data to be included in the Government's overall Space Tug program assessment.

Due to funding constraints, it may be necessary to delay major new Tug development until the Space Shuttle itself is operational. Therefore, it is logical to consider the initial use of an existing stage to provide the necessary additional capability, at minimum development cost. The currently operational Centaur stage, with modification for reusability and for improved performance, represents a cost-effective development solution in the face of present and projected funding constraints. Several versions suitable for a Shuttle initial upper stage are possible depending on mission performance needs and the available development dollars.

Convair Aerospace identified two 35 foot (10.7m) long Reusable Centaur configurations, one with enlarged propellant tanks, and one current size with a velocity package. A third Reusable Centaur configuration only 28 feet (8.55m) long, which is described in Section 10, was created to satisfy DoD requirements for special long payloads.

These three Centaur versions, typical of the many configurations possible for various program requirements, all share common strong points:

a. Very high performance inherent in the use of cryogenic propellants.

b. Low program risk resulting from the use of a majority of existing Centaur hardware.

c. Low recurring and low user costs due to the inherent reusable nature of the Centaur vehicle.

Therefore, the Reusable Centaur is a very attractive candidate for the initial upper stage or orbit-to-orbit stage.

1.1 BACKGROUND

The Space Transportation System (STS) includes a propulsive stage, sometimes called a Space Tug, that is carried into low earth orbit in the Space Shuttle Orbiter payload bay. The primary function of this upper stage is to extend the STS operating regime beyond the Shuttle, including plane changes, higher orbits, geosynchronous orbits, and beyond. It is desirable that an upper stage be available at or about Shuttle initial operating capability (IOC) to provide the maximum operational, performance, and cost benefits.

Current resource constraints preclude the coincident development of both the Space Shuttle and the ultimate Space Tug. NASA and DoD are therefore evaluating alternatives for providing an interim, lower cost vehicle. These alternatives, illustrated in Figure 1-1, include the use of:

a. Existing expendable stages modified for use with the Shuttle, followed by a Space Tug at a later date. This alternative has the advantage of minimum early-year development funding and the disadvantage of high cost per launch for the expendable stages.

b. Modified existing stages with a capability for reuse and an increased propellant capacity giving greater performance and stage recovery capability, followed by a greater capability Space Tug at a later date. The Reusable Centaur is in this group.

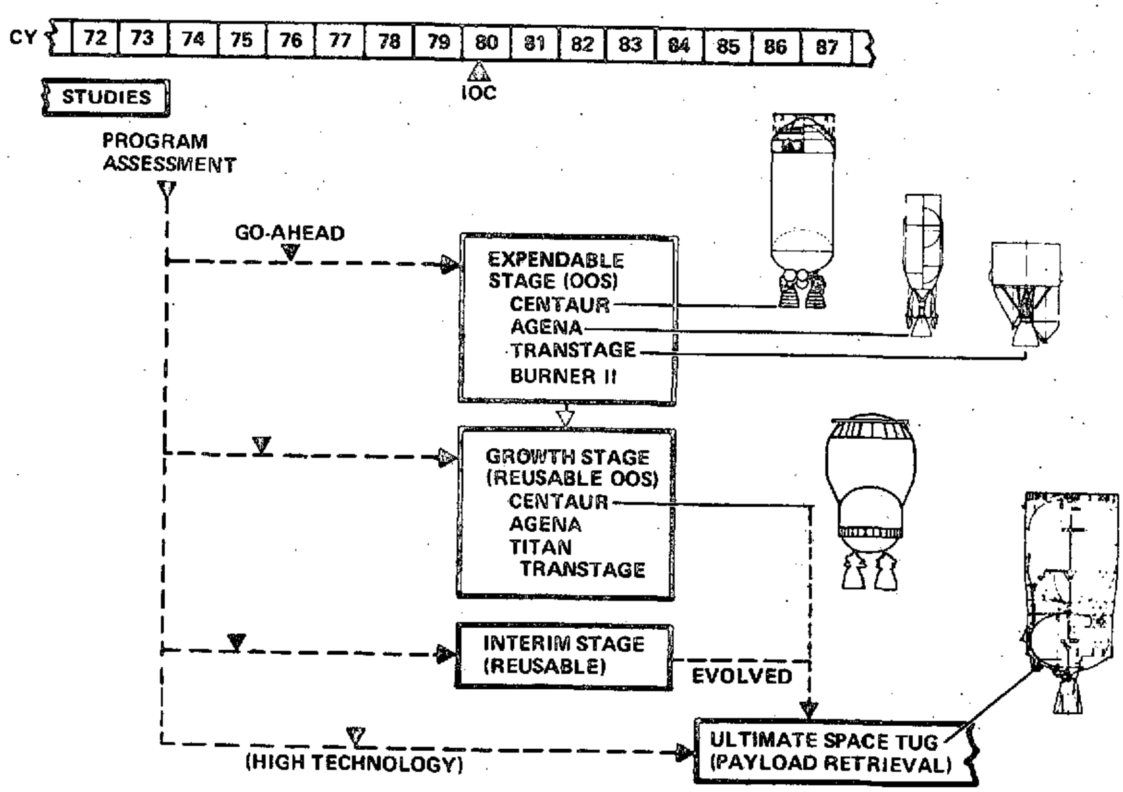


Figure 1-1. OOS/Tug Program Options

c. A low development cost, reusable, interim Space Tug available at or about Shuttle IOC that could be evolved to greater system capabilities at a later date (phased development paths). This approach was part of the Space Tug Systems Study. It would provide Tug options initially having limited operational and performance capabilities that could be increased to match anticipated future mission requirements and capitalize on future technology advances as additional development resources became available.

d. Greater capability Tugs to satisfy specific mission requirements at specified dates after Shuttle IOC (direct development paths). This last is not an interim solution. It would be planned and developed as a high-performance Tug without regard to an evolutionary approach. This Tug, however, would not be available until 1982 or later.

These program options can be complementary or competitive. Each has advantages and disadvantages. The Government will evaluate and select a program concept using extensive data such as that presented here on a Reusable Centaur.

1.2 STUDY PLAN

This study is based on actual data from the current operational D-1T Centaur program, which is summarized in Section 1.3. A preliminary feasibility study of a Centaur used as an expendable Tug was conducted by Convair Aerospace for NASA/LeRC under contract NAS3-14389. The second phase of that work, Centaur/Shuttle Integration Study, was completed in December 1973 under contract NAS3-16786 (Reference 2). Much data from that study is directly applicable to a Reusable Centaur.

Data and requirements definition from the Space Tug Systems Study (Cryogenic), NAS8-29676, were integrated into this study, where applicable, particularly in the missions and payloads areas. In addition, refurbishment, maintainability, and ground operations data from the KSC-sponsored Space Tug Launch Site Services Study, NAS10-8031, conducted by Convair Aerospace, were used (Reference 3).

The basic Reusable Centaur study objective was to develop realistic technical programmatic data for a complete "data dump" by mid September 1973 to be used in the NASA/DoD program assessment. This was accomplished by Reference 1. Key drivers considered in this study were:

a. Low development costs (especially before 1978 while the Orbiter is being developed.)

b. Safety, reliability, and manned compatability.

c. Low program risks by maximum use of proven hardware.

d. Low operating costs due to reusability and high performance.

e. Possible phased development using the building block approach.

f. Extremely long DoD payloads.

Principal study guidelines were:

a. Base configuration on operational D-1T Centaur.

b. Required mission is payload deployment and Centaur return to Orbiter (Payload retrieval is not required). Placing a 3500-pound satellite in synchronous equatorial orbit is the baseline.

c. Satisfy Tug program missions in first 4*, 6, or 11 years of operation.

d. Meet interim Tug requirements, including Orbiter installation, from STSS.

e. Probability of mission success of 0.97 is a goal.

f. Design must be fail-safe (not jeopardize Orbiter or its crew).

g. Loads on payloads should be less than current designs. Specifically payload acceleration should be <3.6 g.

h. Payloads are allowed to "walk" to final position.

i. Ground tracking for position and velocity update may be considered.

j. Installed length of the Reusable Centaur (and kick stage, if required) shall not exceed 35 feet. (Some versions 28 feet and 25 feet long are considered for DoD missions.)

k. Other applicable requirements from the MSFC Space Tug Systems Data Package dated April 1973.

The study schedule is shown in Figure 1-2. Major milestones are noted below (including some related DoD discussions of additional company funded studies).

2 Jul 73	Contractual go-ahead
7 Aug 73	Concept selection presentation at MSFC
26 Sep 73	Program definition/midterm presentation and "data pump" at MSFC
24 Oct 73	"Short" Centaur for long DoD payloads presentation at SAMSO
2 Nov 73	"Short" Centaur presentation at MSFC
21 Dec 73	Draft of final report
17 Jan 74	Final presentations at MSFC
15 Mar 74	Issue final report; contract completion

*For the 4-year program, it did not seem logical to activate WTR, so only ETR launches were covered, including WTR missions flown out of ETR using a dog-leg.

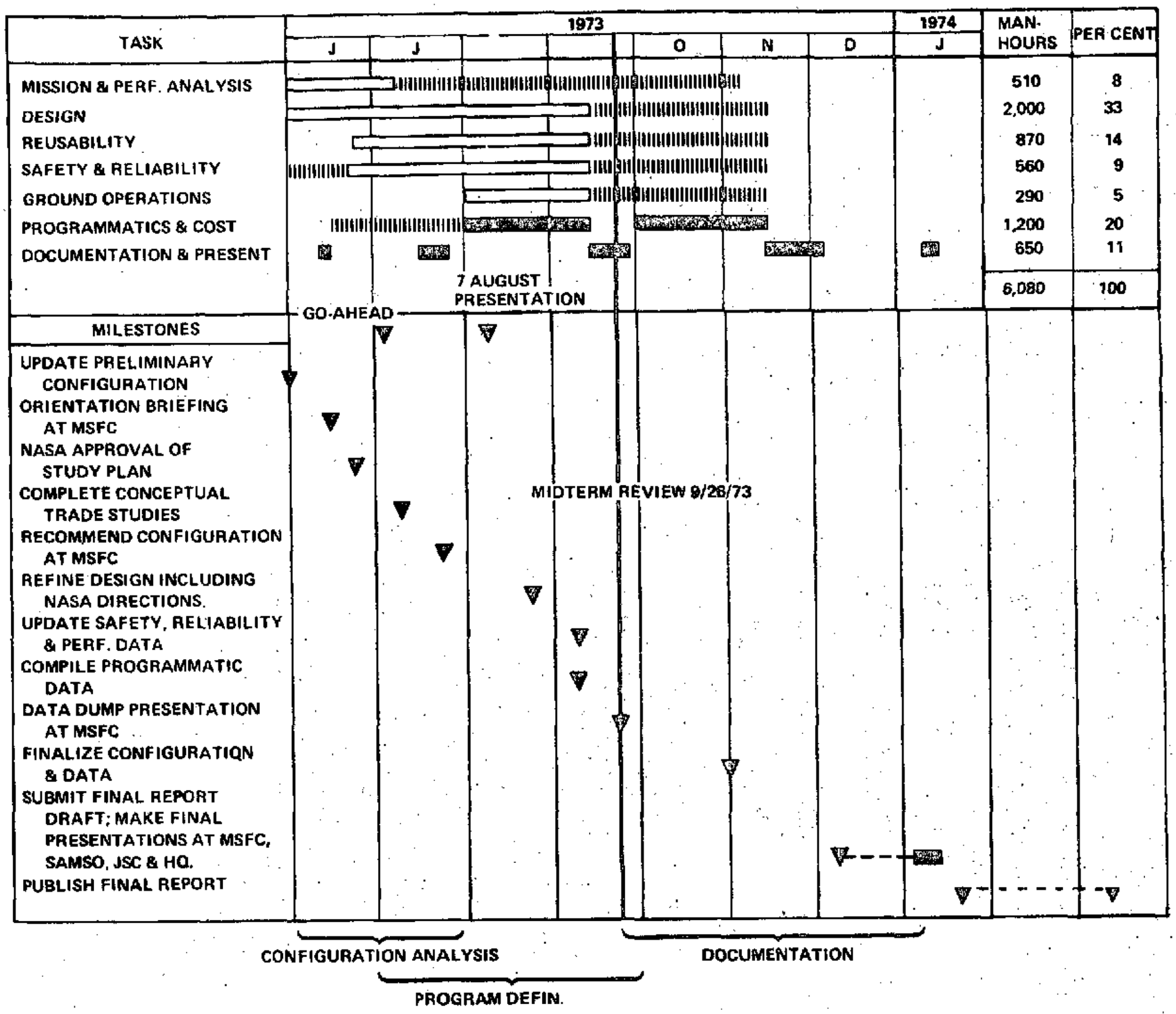


Figure 1-2. Study Schedule

The study initially focused on a reusable large tank Centaur (RLTC) with propellant capacity increased about 50% above the current vehicle. Then the building block approach was considered, where at least initial versions would be like the current size vehicle called D-1S(R) (size of the current D-1, integrated with Shuttle, and reusable) using velocity packages for added performance when required. Each approach has advantages. The optimum solution is dependent on development funding limits, the relative importance of low user costs, as well as the final mission model, including primary and backup allocation of payloads to the Shuttle and/or expendable launch vehicles. Since the configuration cannot yet be optimized, a range of features was considered in each area, from which other versions could be conceived.

Reusable Centaur configurations can be divided into two groups: one with current 10 foot (3.05 m) diameter tanks and the other enlarged to 14-1/2 foot (4.43 m) diameter. Velocity packages or kick stages are usually required with the smaller versions but usually not with the RLTC. Therefore it was logical to generally group the low development items together on the D-1S(R) such as hydrogen peroxide ACPS and batteries, while higher performance systems were grouped together on the RLTC, like hydrazine ACPS and fuel cell. In almost every case there is a tradeoff among performance, development requirements, and user costs.

1.3 EXISTING D-1T CENTAUR

The Centaur D-1T configuration was used as the basis for evolving Centaur as a Shuttle third stage. The Centaur high-energy upper stage has flown 24 operational missions, and currently has a future mission backlog through 1979. A complete vehicle description is contained in Reference 4.

The improved Centaur vehicle currently used as the high-energy upper stage of the Atlas is designated the D-1A - "A" for Atlas. On 5 April 1973 it successfully launched Pioneer 11 with a spin-stabilized kick stage. There also were successful D-1A launches in August and November for COMSAT and Mercury-Venus respectively. This Centaur incorporates a fully integrated astrionics system using a central digital computer for software control of vehicle and flight operations.

D-1T Centaur is configured for launch on the Titan. The proof flight of this vehicle is scheduled for February 1974. In addition to the integrated astrionics system, D-1T incorporates a space radiation shield insulation system, and subsystem modifications to improve vehicle reliability. The proof flight of the D-1T will accomplish several mission objectives of particular interest relative to eventual Centaur/Shuttle use, including a four-burn mission and a coast duration of 5.25 hours.

Figure 1-3 shows the general arrangement of the existing D-1T, which is 31.5 feet (9.6 m) long and 10 feet (3 m) in diameter, with a mass fraction of 0.88. The Centaur stage carries 30,000 pounds (12,000 kg) of liquid hydrogen and oxygen in pressure-stabilized stainless steel tanks. Oxygen is aft, separated from the hydrogen by a double-wall, evacuated intermediate bulkhead. Two Pratt & Whitney RL10A-3-3A engines produce 30,000 pounds thrust at a nominal Isp of 444 seconds. The current operational stage uses peroxide monopropellant for propellant settling and attitude control motors. Tank weights are minimized by low operation pressures made possible by boost pumps in the propellant feed system. Prestart tank pressurization is provided by a short "burp" from ambient helium bottles; boiling propellants make the system self pressurizing during outflow. Propellant heating in space is reduced by radiation shields on the LO_2 tank aft bulkhead, a vacuum intermediate bulkhead, and multi-layered insulation blankets on the forward LH_2 dome. The D-1T has three radiation shields on the LH_2 sidewall. Thrust vector control is powered by hydraulics with engine-driven power packs. An advanced feature of the Centaur D-1 is the use of an onboard digital computer unit (DCU) for navigation, guidance, control, sequencing, propellant utilization, tank pressurization and venting, instrumentation, and telemetry. This permits many functions to be done by software, eliminating some hardware components. Pulse code modulated (PCM) telemetry data is downlinked by S-Band. The navigation function is

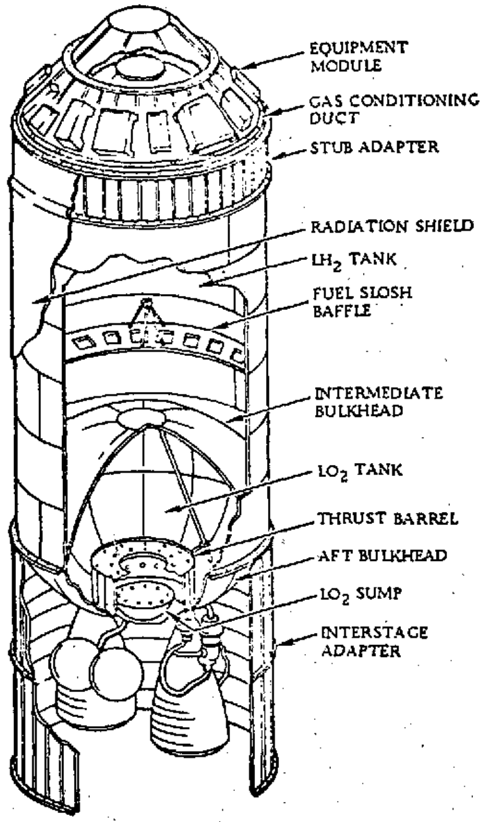


Figure 1-3. Centaur D-1T

done onboard by an inertial platform. The power supply is a 150 ampere-hour battery with a servo inverter unit to provide 400 Hz to the PU, gyros, and engine position indicators. Testing of the Centaur D-1 astrionics is controlled and monitored by a ground computer-controlled launch set (CCLS) that includes an XDS930 computer.

1.4 SPACE SHUTTLE INTERFACE

This study is based on Space Shuttle Program definitions from JSC 07700, Volume XIV, Payload Accommodations (Reference 5). The Shuttle configuration, capabilities, and interfaces continue to evolve. During the time of this study, NASA selected Rockwell International to develop the Orbiter. General Dynamics/Convair Aerospace received the subcontract for the Orbiter mid-fuselage. Valuable data was exchanged at a meeting at Rockwell International on 15 November 1973 at an Interface Presentation (Reference 6). Figure 1-4 shows the general arrangement of the Shuttle. Some of the basic interfaces are noted below:

a. Reusable Centaur with its payload and peculiar Orbiter interface equipment cannot exceed 65,000 pounds (29,500 kg) for a due east launch from ETR to 160 n.mi. (296 km). Total Shuttle payload must be below 36,000 pounds (16,350 kg) for polar launches from WTR.

b. The payload bay is 60 feet (18.3 m) long and 15 feet (4.57 m) in diameter.

c. Structural support is a four point, statically determinate mounting arrangement. Tentative attachment locations are shown in Reference 5).

d. Fluid and electrical service interface panels are located in the lower portion of the aft bulkhead of the payload bay.

e. Centaur will be tanked during the last 2 hours of countdown, while vertical, with the payload bay doors closed.

f. Centaur will provide the Orbiter (mission specialist console), and the ground, those caution and warning functions critical to Shuttle safety, while operating in or near the Orbiter. For this study, it was assumed to be preferable for Centaur to provide in the Orbiter its own monitor and control equipment including dedicated computer capacity.

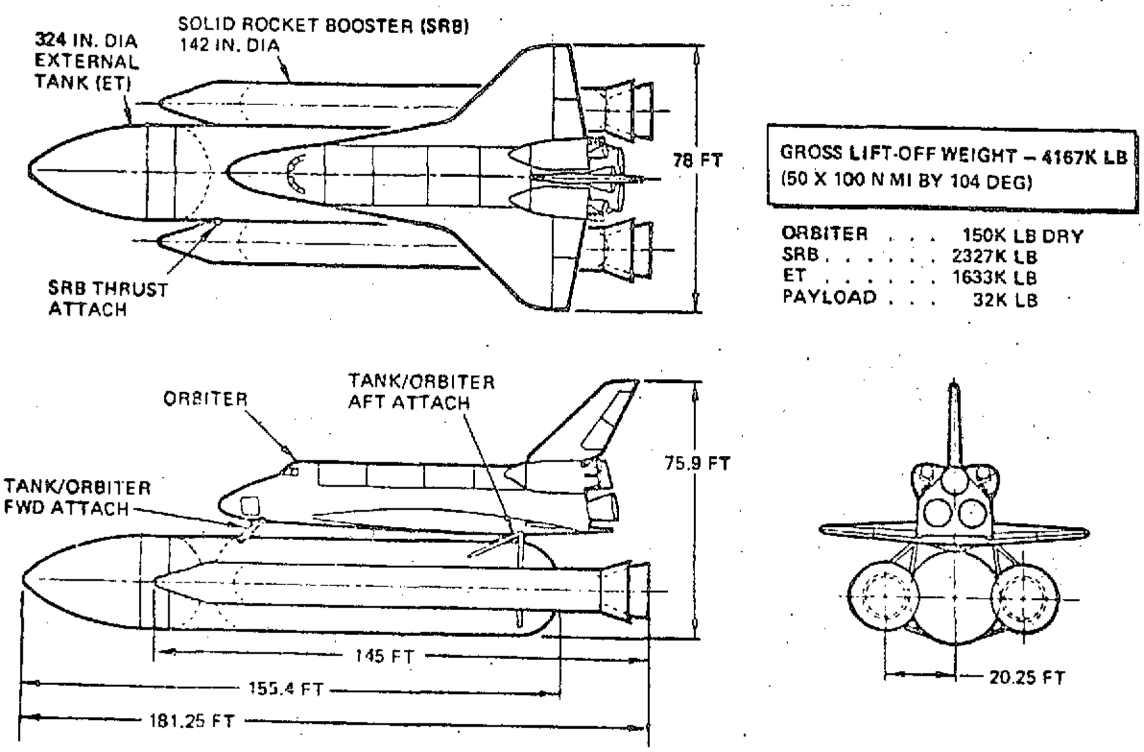


Figure 1-4. Shuttle Vehicle

g. Inflight Shuttle abort modes occur after solid rocket booster (SRB) burnout, allowing return to launch site, orbit, or once around. In this study, it was assumed preferable to drain in-flight Centaur main propellants prior to return to the landing site.

h. Environmental interfaces are being clarified, including vibration, deflections, and temperature ranges.

SECTION 2

MECHANICAL SYSTEMS CONFIGURATION

Two versions of Reusable Centaur are discussed in depth. Either one, or a hybrid could be the optimum arrangement. Basically the D-1S(R) is closest to the existing D-1T vehicle, while higher performance options are grouped together on the larger RLTC.

D-1S(R): Figure 2-1 shows the general configuration of D-1S(R). Emphasis has been to use existing Centaur D-1T hardware and designs. The main engine is the P&W RL10A-3-3 operating at a 5.8:1 mixture ratio. The structure is a pressure-stabilized 301 CRES tank with aluminum and titanium skin stringer adapters. The ACPS is hydrogen peroxide monopropellant with helium pressurized bladder tanks. Sixteen peroxide ACPS engines in clusters of four provide all axis maneuver capability. The H_2O_2 also provides power for the LH_2 and LO_2 engine boost pumps. The propellant tank pressurization system raises tank pressures prior to each engine start with ambient temperature helium. The tank vent system uses both ground vent valves and zero-g vent mixers. The fill and drain system also provides in-flight propellant dump

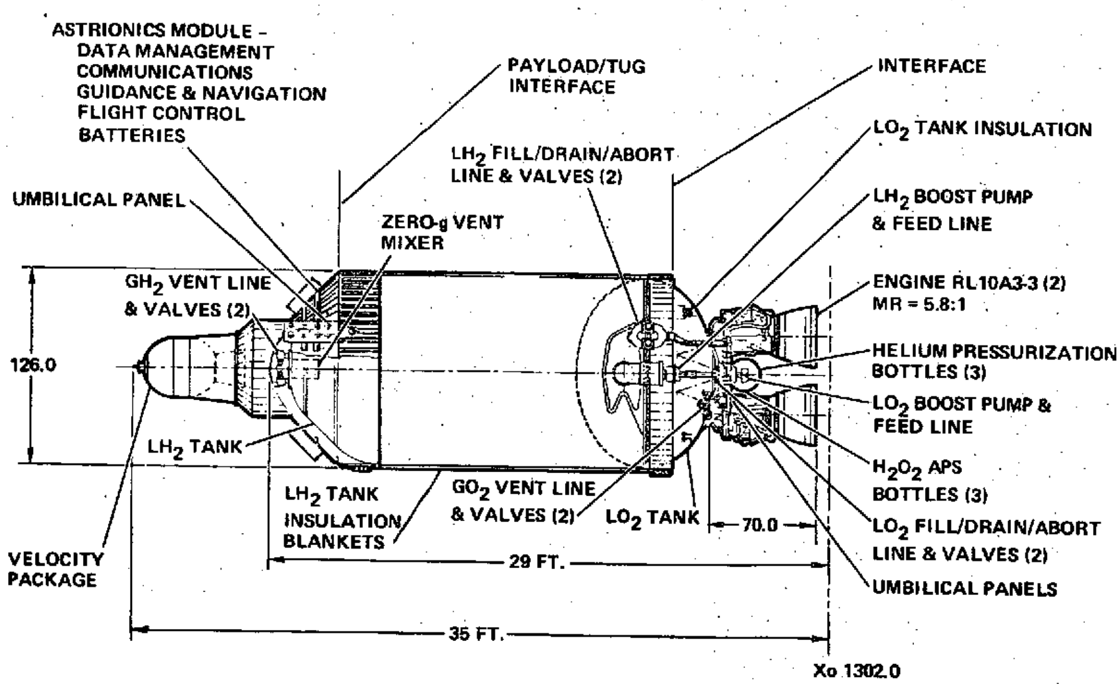


Figure 2-1. D-1S(R) Configuration

capability in the event of an abort. The tank insulation system is multilayer goldized Kapton blankets for ground hold, and three-layer radiation shielding for deep space. The D-1S(R) flight support equipment includes a flight pallet for handling and deploying the vehicle.

RLTC. Figure 2-2 shows the general configuration of RLTC. The RLTC provides an increased propellant capacity to make full use of the Space Shuttle boost capability of 65,000 pounds (29,400 kg). Performance has been maximized while at the same time maintaining a large percentage of the existing operational D-1T Centaur subsystems. The main engine is the P&W RL10A-3-3 operating at a 5:1 mixture ratio. The structure is a pressure-stabilized 301 CRES tank with composite sandwich construction adapters. The ACPS is hydrazine monopropellant with helium pressurized bladder tanks. Sixteen ACPS engines in clusters of four provide all axis maneuver capability. The hydrazine also provides power for the LH_2 and LO_2 engine boost pumps. Helium for tank pressurization is stored cryogenically within the LH_2 tank and warmed prior to use by a heat exchanger. The tank vent system utilizes both ground vent valves and zero-g vent mixers. The fill and drain system also provides in-flight propellant dump capability in the event of an abort. The tank insulation system is multilayer goldized Kapton blankets for ground hold, and three-layer radiation shielding for deep space. The RLTC flight support equipment includes a deployment adapter.

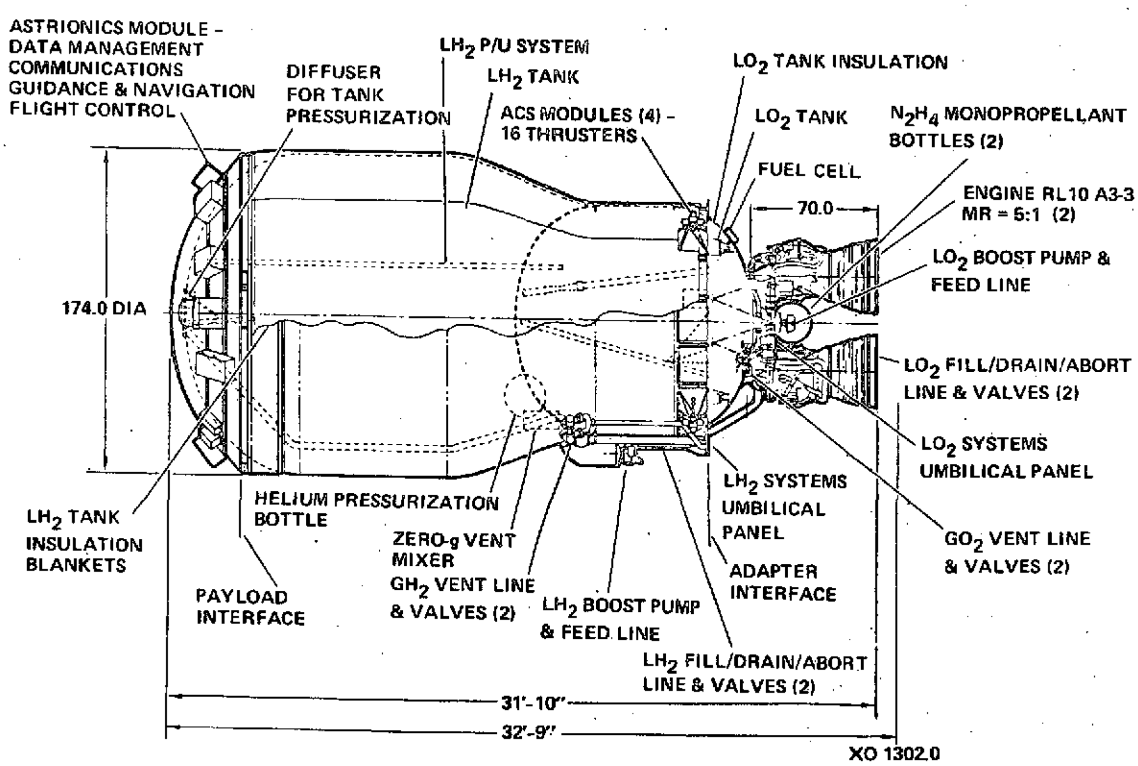


Figure 2-2. RLTC Configuration

2.1 SHUTTLE REQUIREMENTS/ENVIRONMENTS AND MANNED COMPATIBILITY CONSIDERATIONS

The evaluation of the Centaur for use as a reusable space Tug has shown that no major integration problems will occur if attention is paid to the requirements and constraints of both the Shuttle and the cryogenic Tug during the early design phase. The use of the high-energy Centaur upper stage as a Shuttle payload requires servicing and control functions which must be planned in the Shuttle development. Proper upper stage design of support systems will minimize the system peculiar effects on the Shuttle, provide smooth phasing of the cryogenic stage in both ground and flight operations, and result in a safe system for all anticipated Space Shuttle operations.

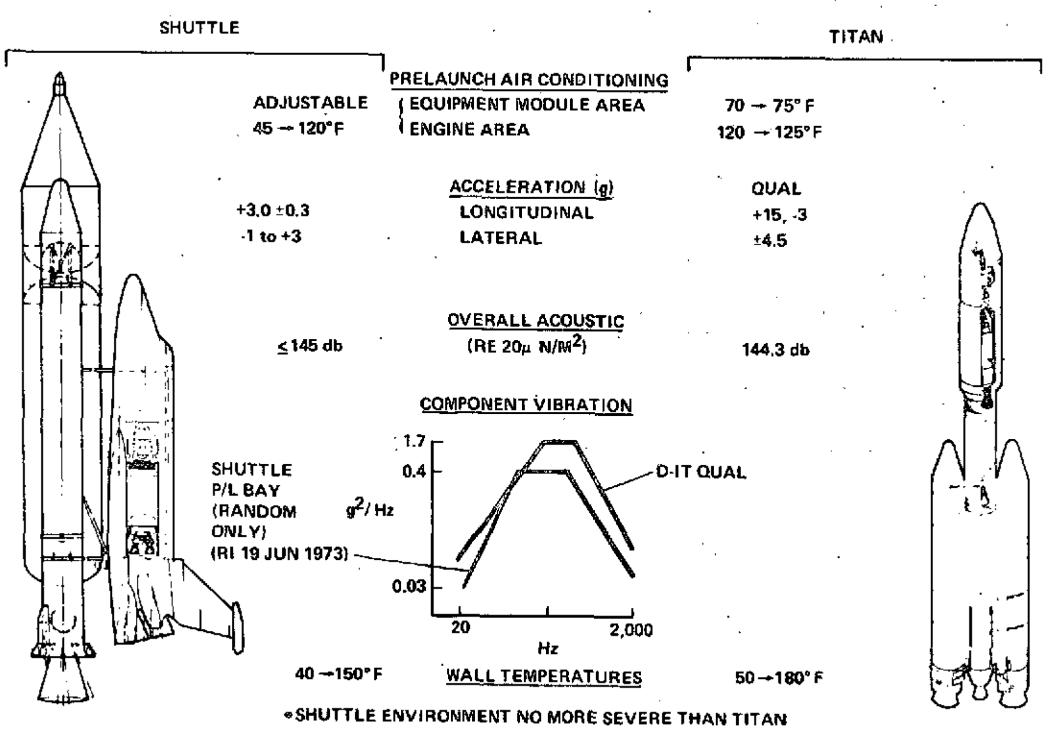
The design concept evolved during the study used two major reusable support items: the Centaur support adapter, and the Centaur monitor and control system. The Centaur support adapter provides the structure for evenly distributing the Shuttle-induced concentrated four-point flight loads imposed by the Shuttle payload retention system. It also provides necessary inflight functions such as umbilical disconnects, docking equipment, abort helium supply, and interface junctions. Most of the electromechanical functions occurring during flight, deployment, or retrieval take place within the support adapter and impose no requirements on any Shuttle system.

The Centaur monitor and control system equipment is located in the Orbiter crew compartment and contains the necessary hardware and software for safety monitoring, system status verification, warning and action signals, and deployment, retrieval and abort sequencing. The Shuttle provides some of this control and monitoring equipment for use by all payloads, but a portion of this flight monitor and control system is also reusable payload-peculiar equipment which is supplied for each flight and recycled for use on subsequent flights.

Based on the use of this ancillary equipment, the only Shuttle requirements for Centaur interfacing are the structural support for the support adapter and Centaur within the payload bay, sufficient space for payload or Obiter-supplied support equipment to satisfy the Centaur monitor and control function in the crew compartment, the provision for service and ground umbilical connections in the Orbiter payload bay and skin line, electrical power supply for Centaur and its ancillary equipment during flight operations in the Orbiter, and the provision for obtaining a Centaur guidance alignment update from the Orbiter. These interfaces are all Orbiter-provided for general payload use.

While the physical Shuttle/Centaur interface requirements (structural, mechanical, fluid and electrical) are satisfied with ancillary equipment, the environmental and manned interface compatibility must be inherent in the Centaur vehicle design. Fortunately the launch environment of the Centaur D-1T, the vehicle from which the Reusable Centaur is derived, is similar to the predicted Shuttle environment. Because of this similarity, a great majority of Centaur's subsystems will have been previously qualified and flight tested to Shuttle compatible accelerations, vibration and acoustic spectra, and temperature ranges. Additionally, the Centaur in Shuttle payload bay prelaunch (tanked) thermal conditioning environment closely parallels that for D-1T in the Centaur standard shroud. Use of the Centaur proven ground hold insulation system will result in a benevolent thermal environment for Centaur's payload and the surrounding Shuttle structure. A comparison of the Shuttle and Titan launch environments imposed on Centaur are presented in Figure 2-3.

The Centaur, as an unmanned vehicle which is launched by a manned vehicle and operates at times in relative proximity to man, must achieve manned compatibility to the extent that it does not violate any Shuttle man-rating requirements. The structural safety factor revision needed to achieve manned compatibility is accomplished by reducing Centaur propellant tank and storage bottle pressures to levels consistent with the required Shuttle factors and the reduced flight loads. The Reusable Centaur design factors for operations in or near the Shuttle are shown in Table 2-1. The ability of Centaur to meet manned compatibility requirements from a system safety standpoint was also evaluated. The safety analysis systematically analyzed the safety of the Centaur within the framework of manned vehicle operations and identified potentially hazardous situations arising from those operations. Once identified, these hazards were eliminated, or the likelihood of occurrence reduced, by subsystem redesign and/or hazard controls. The Reusable Centaur incorporates propellant isolation valves, redundant propellant dump and vent valves, sealed propellant tank purge bags,



(REFERENCE JSC 07700, VOL XIV, REV. B.)

Figure 2-3. Shuttle Environment

Table 2-1. Design Factors - Centaur/Shuttle Operations

Component	Yield	Ultimate	Proof	Applied On
Main Propellant Tanks	1.10	1.40	1.00	Max relief valve pressure only (ALL SHUTTLE OPERATIONS)
	1.10	1.40	-	Loads (+ limit pressure) (ALL SHUTTLE OPERATIONS)
	1.10	-	-	Proof pressure (ALL SHUTTLE OPERATIONS)
Airframe Structure	1.10	1.40	-	Loads from all Shuttle operations
Pressure Vessels	1.50	2.00	1.50	Max operating pressure - all Shuttle operations
Pressurized Lines + Fittings (Hydraulic & Pneumatic)	2.00	3.00	2.00	Max operating pressure - all Shuttle operations
Pressurized Lines & Ducts (Main Propellant Supply & Vent System)	1.50	1.87	1.50	Max operating pressure - all Shuttle operations

Note: Main propellant tank and airframe structure ultimate design factor = 1.0 for crash landing loads.

backup retrieval command systems, etc., to ensure the safety of the Space Shuttle vehicle and its crew.

The Reusable Centaur vehicle is compatible with all Shuttle integration requirements including those associated with physical, functional, environmental and manned considerations. Compatibility has been achieved with ancillary equipment, and by incorporating selected vehicle changes resulting from three years of integration work, on an existing upper stage which has been designed and flight tested to Shuttle-similar environments.

2.2 STRUCTURE

The D-1S(R) and RLTC are both modifications of the existing D-1T Centaur with revised propellant capacity and changes to enable use in the Space Shuttle Orbiter. The basic modifications involve the relocation of fluid disconnects from the locations of the present expendable launch vehicles to locations suitable for reuse in the Orbiter. The structural adapters are revised to enable introduction of the four point loads for mounting in the Orbiter payload bay for both vertical launch and horizontal landing. The main structural elements of D-1S(R) and RLTC are shown in Figures 2-4 and 2-5.

2.2.1 PROPELLANT TANKS. The D-1S(R) Centaur propellant tanks contain the vehicle liquid oxygen and liquid hydrogen for main engine operation and also provide the major portion of the vehicle structure. The tanks are pressure-stabilized 301 CRES monocoque structure the same as on the existing D-1 series Centaurs except a cylindrical section of the LH_2 tank has been removed to make the stage compatible with the 35-foot (10.67 meter) length requirements. (See Figure 2-6).

The LO_2 tank aft bulkhead is made of gore sections, stretch formed to a 1.38:1 ellipsoidal contour with a 24-inch (0.61m) flange for tank access and for mounting the LO_2 outlet sump and boost pump. A 120-inch (3.05 m) flange on the bulkhead provides attachment for the aft adapter structure. Machined doublers and rings provide engine thrust structure mounting.

The intermediate bulkhead separating the LO_2 and LH_2 propellant tanks is the same as used on D-1T, and consists of two bulkheads that enclose an evacuated insulation cavity. There are no common surfaces or welds between the two propellant tanks as shown in detail in Figure 2-7. The forward bulkhead is a 1.38:1 ellipsoidal contour and attaches to the LH_2 cylinder. The bulkhead pressure loads are carried by tension in the forward

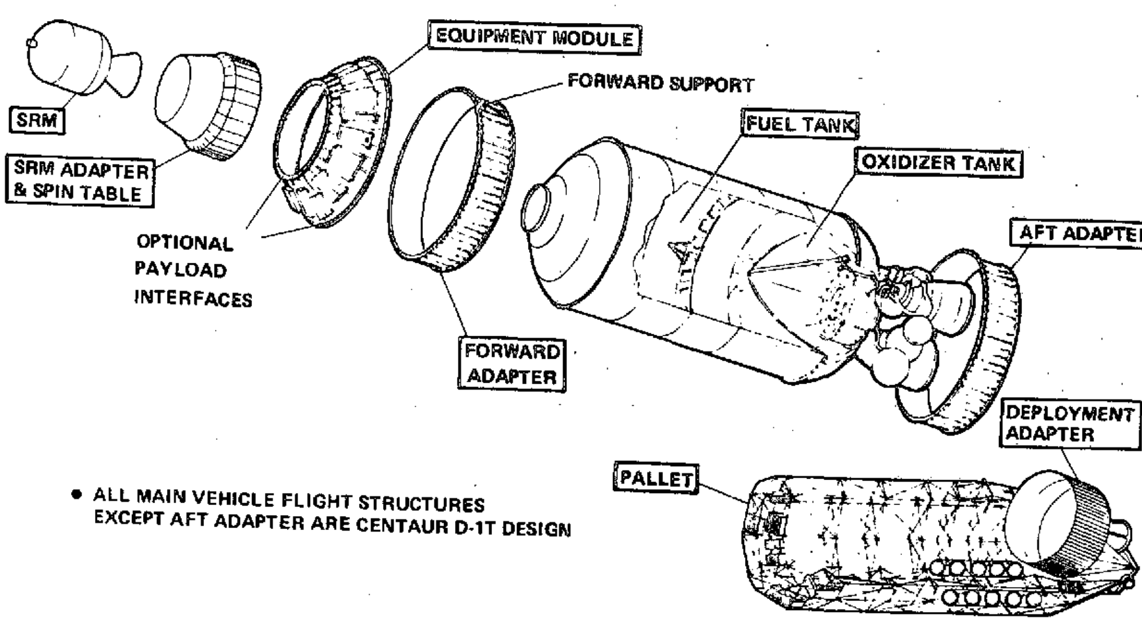


Figure 2-4. D-1S(R) Main Structures

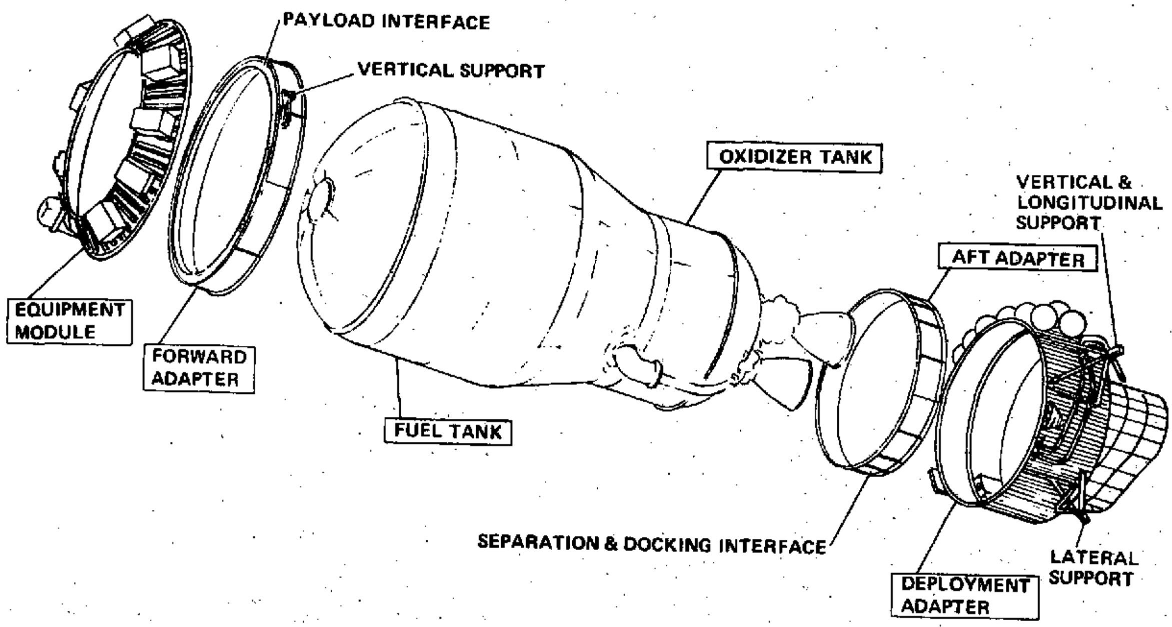
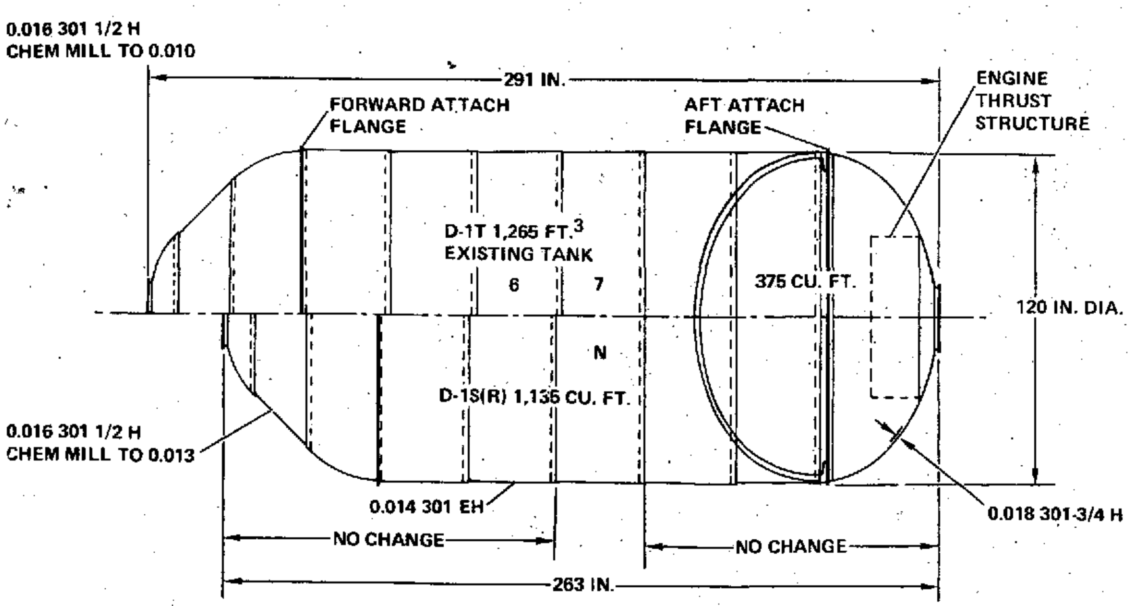


Figure 2-5. RLTC Main Structures



NOTE: EXISTING SKIN "6 & 7" REPLACED BY SKIN "N" TO SHORTEN TANK

Figure 2-6. D-1S(R) Propellant Tank

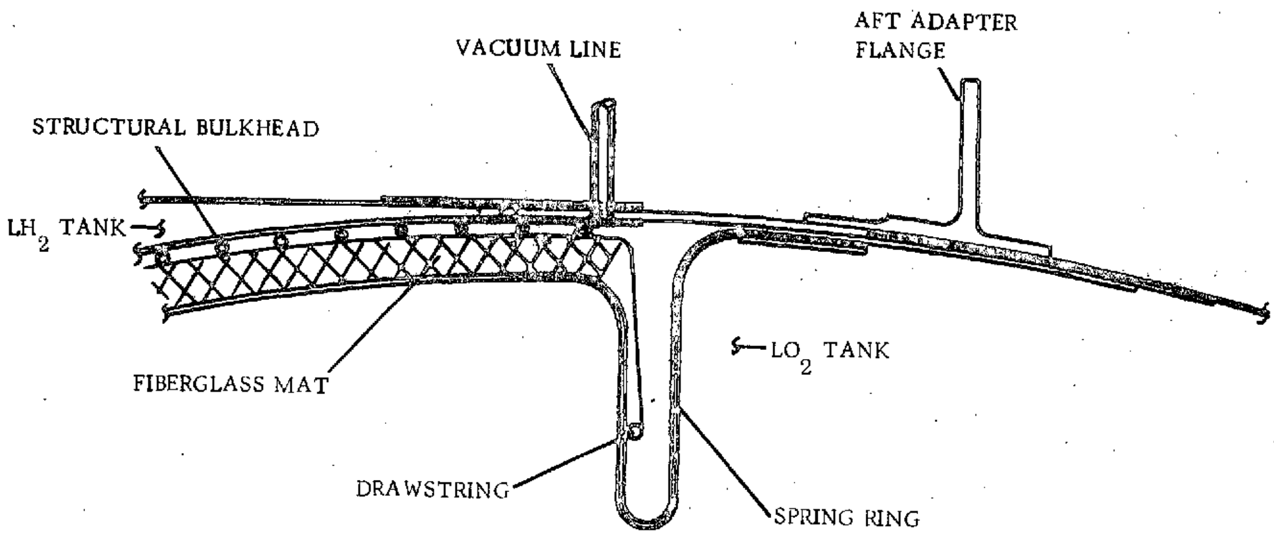


Figure 2-7. Intermediate Bulkhead

skin. The intermediate bulkhead insulation system consists of a fiberglass mat enclosed in a mylar bag which has a plastic screen on one side. The insulation blanket is positioned on the LO_2 tank side spring ring bulkhead skin which is welded in place in the LO_2 tank. The spring ring bulkhead has a single convolute which allows it to unload by expanding until it compresses the insulation against the intermediate bulkhead. It is mandatory that the LO_2 tank pressure equals or exceeds the hydrogen tank pressure at all times in order to prevent structural failure of both bulkheads by reversing them.

Prior to shipment from the factory, the Centaur vehicle intermediate bulkhead cavity (and insulation) is evacuated and backfilled with GN_2 at atmospheric pressure and sealed. During tanking, the cavity cryopumps down to a pressure below 10 microns. The intermediate bulkhead cavity is vented through a checkvalve to provide an overboard vent path in the remote event of an intermediate bulkhead leak.

Leak testing the bulkhead is accomplished by evacuating the bulkhead cavity first with the LO_2 tank only pressurized with helium and then with both tanks pressurized with helium. By using a vacuum pump and a helium mass spectrometer on the vacuum line, verification of bulkhead integrity is assured.

The LH_2 tank forward bulkhead has two ellipsoidal sections and a conic section. A 24-inch (0.61 m) flange and cover provide access to the tank. The forward adapter attaches to a 120-inch (3.05 m) diameter flange on the forward bulkhead.

The RLTC propellant tanks (Figure 2-8) are similar to the D-1S(R) tanks except a cylindrical section is added to increase the LO_2 tank volume and the LH_2 tank diameter

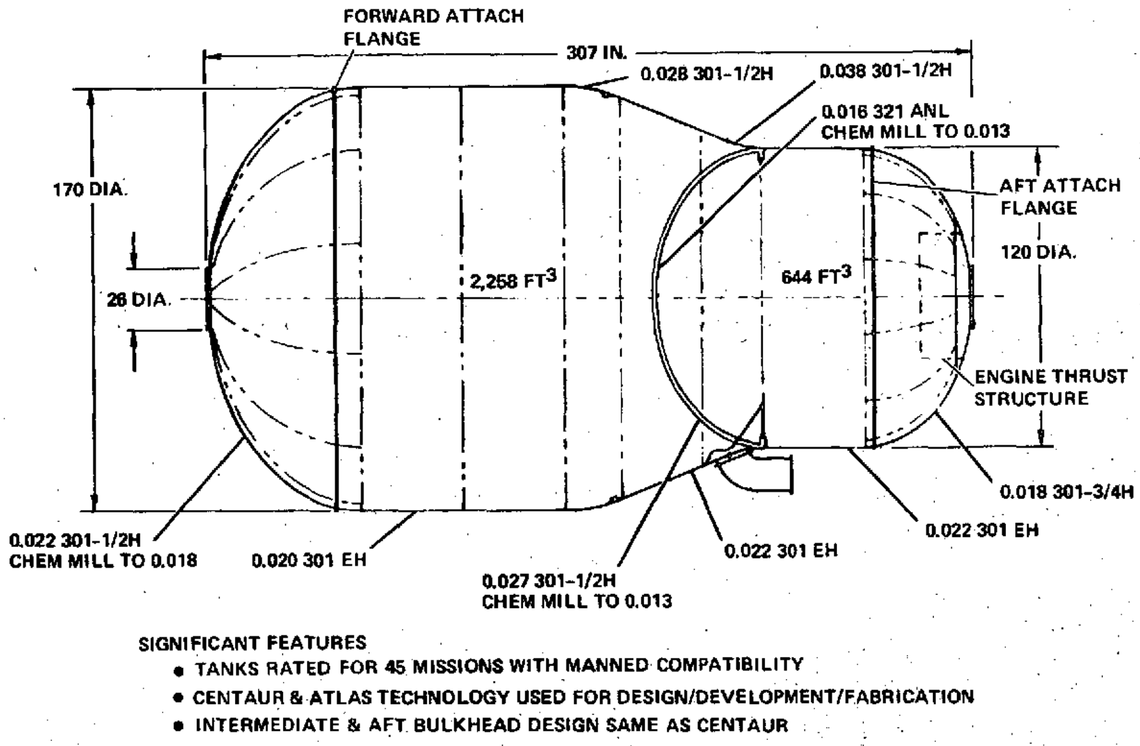


Figure 2-8. RLTC Propellant Tank

is increased to 170 inches (4.32 m) through a transition developed from sections of two tori and a cone. The forward bulkhead is 1.38:1 ellipsoidal contour. A 168-inch (4.27 m) diameter flange on the forward bulkhead attaches the forward adapter and transmits the payload and Orbiter support loads into the tank. A similar enlarged tank was previously analyzed and proposed to NASA as a "GT" version (Reference 8).

Propellant Tank Life. The predicted life of the Centaur propellant tanks is based on the fatigue strength of 301 CRES extra hard resistance welded joints at -423°F (24°K). The fatigue strength is actually the minimum number of cycles at a statistical 99% "A" value which causes leakage in a resistance weld (and not a parent metal failure). Since weld leakage is the tank failure mode and multiple leakages can occur before any degradation of structural strength, the tank structure falls well within a leak-before-fail criteria. The leak containment and overboard venting precludes fire/explosive hazards. Leak testing between flights or by hydrogen detectors in flight is an excellent method of verifying the tank structural integrity for the next flight. The tank life is therefore not a predetermined value based on an assumed life cycle but is the integrated effects of the stresses and cycling that actually occur on a specific tank.

For predesign, the tank life is calculated based upon the material properties and assumptions as given in Reference 1, Data Dump Volume 5.

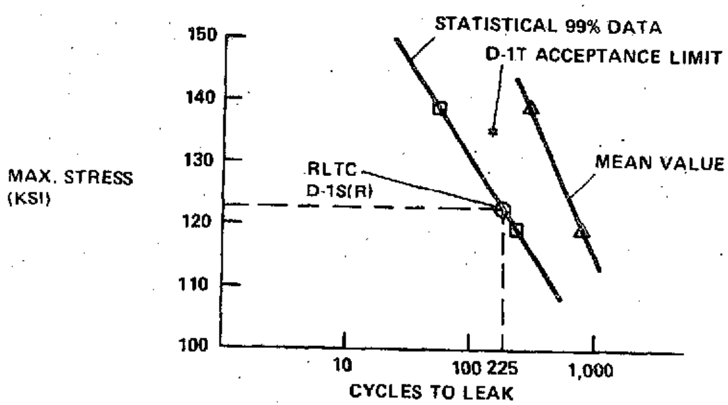
The predicted life of the Centaur as used in the Space Shuttle is a minimum of 45 missions. This life is based on the material fatigue properties of CRES 301 EH at LH_2 temperatures of 225 cycles to leak and an applied factor of 5 to account for mission variables, i.e., 225 cycles/5 = 45 mission life. This life is conservative since it is based on one pressure cycle per mission to an applied stress of 123 ksi which occurs only during an abort dump. (A nominal flight maximum pressure causes a stress of 110 ksi or 550 cycles to leak.) This data is shown in Figure 2-9.

The factor of 5 is used to account for the following effects not present in the coupon fatigue testing:

a. Stress buildup at "hard points" on the tank; i.e., bosses, doublers, etc.

b. Thermal cycling during reuse which will exceed the thermal cycling effects experienced by the test specimen.

WELDED SPECIMENS OF EACH CENTAUR LH_2 TANK ARE FATIGUE TESTED > 200 CYCLES FROM 0 TO 135 KSI WITHOUT FAILURE



REUSABLE CENTAUR STRESS REDUCED TO 123 KSI FOR MANNED COMPATIBILITY

ALLOW FACTOR OF 5 FOR BIAXIAL STRESS, SECONDARY FLUCTUATIONS, ETC.

CENTAUR TANK LIFE EXPECTANCY ≈ 45 MISSIONS

Figure 2-9. Reusability of Centaur Tanks

2.2.2 ADAPTERS

Engine Thrust Structure. The Centaur D-1S(R) and RLTC mount the two engines directly to fittings on the LO_2 tank aft bulkhead. A 50-inch (1.27 m) diameter by 16 inch (41 cm) long cylindrical thrust structure is attached to the two engine mount fittings and to a ring on the inside of the LO_2 tank to distribute the engine thrust loads. The thrust structure is a skin stringer assembly stiffened on the forward end with a perforated baffle which serves as anti-vortex flow control. The LO_2 tank vent pipe and propellant utilization (PU) probes are supported from the thrust structure. The thrust structure for D-1S(R) and RLTC is the same as used on D-1T.

Forward Adapter and Equipment Module. The forward adapter attaches to the flange of the LH_2 tank and distributes the loads to the tanks. The astrionics equipment module, the Centaur forward support point, the ground handling support points, and scientific payload mounting are provided on the forward adapter. The adapter has both a flange and 12 hard points which can be used discriminately to mount payloads.

The equipment module is a conic ring which provides a mounting platform for the astrionics equipment. The astrionics packages are mounted on the exterior surface of the cone providing excellent access for installation, checkout, and replacement.

D-1S(R) Forward Adapter and Equipment Module. The forward adapter and equipment module for D-1S(R) is the same as the present D-1T design with the addition of fittings for support in the Orbiter. The adapter is a conventional skin stringer frame cylinder 120 inches (3.05 m) in diameter and 25 inches (64 cm) long. Titanium skin and stringers are used for low thermal conductivity to reduce heat transfer into the LH_2 tank. An aluminum internal frame and longerons at the forward end provide the mounting hard points for Orbiter and payload.

The equipment module is a 90-degree ($\pi/2$ radians) included angle truncated cone 30 inches (76 cm) high with 120 inches (3.05 m) base diameter and attaches directly to the forward adapter. The module is aluminum skin stringer construction with two fiberglass hat section rings for mounting the astrionics equipment. The fiberglass provides thermal isolation of the astrionics units. The center portion of the equipment module is closed with an aluminum cap providing an enclosed cavity for the insulation purge helium.

RLTC Forward Adapter and Equipment Module. The forward adapter and equipment module for RLTC is sandwich graphite/epoxy construction with two laminated graphite/epoxy hat section equipment mounting rings.

A boron/epoxy reinforced aluminum extrusion forms the attachment ring which joins the equipment module to the forward adapter. It also serves to distribute the forward Orbiter support reaction loads into the forward adapter. Boron-reinforced aluminum extrusions are developed and marketed by Avco Systems Division. The Avco reinforced extrusions are specifically designed to utilize the very high stiffness, tension, and compression strength properties of boron/epoxy. Boron reinforced extrusions are fabricated by inserting boron/epoxy into the formed aluminum extrusions and step curing it in place.

The forward adapter supports the equipment module and the payload adapter, and distributes their load into the LH_2 tank. It provides the forward attachments for the sidewall insulation blanket and radiation shield.

Aft Adapter. The Centaur D-1T is supported on the expendable booster stage by a skin stringer cylindrical adapter which is separated with a linear shape charge approximately one inch (2.5 cm) from the LO_2 tank flange. For both D-1S(R) & RLTC, the aft adapter is a 15-inch (28 cm) long cylinder which distributes into the tank structure the 16 deployment latch loads from the Orbiter deployment adapter. The D-1S(R) adapter is titanium skin stringer with aluminum ring construction to provide stiffness and low thermal conduction.

The RLTC adapter is a skirt type, graphite/epoxy sandwich shell structure. It is bolted directly to the LO_2 tank and contains the APS thrusters and the hydrogen disconnect panel, and provides the aft mount for the sidewall insulation and radiation shield. Titanium bathtub fittings introduce the concentrated docking mechanism latch loads into the shell.

Fluid and electrical disconnect panels are mounted on the Aft adapter plus docking features. The adapter mounts probes which feed into drogues in the Deployment adapter for alignment during docking. The Deployment adapter, which performs the other half of the release and docking functions, is discussed in Section 2.7; flight operations deployment and retrieval are described in Section 5.4.

2.2.3 PAYLOAD/CENTAUR INTERFACE. The D-1S(R) can mechanically mount payloads on either the 60-inch (1.52-m) diameter of the equipment module or the 120-inch (3.05-m) diameter of the forward adapter. The forward adapter includes 12 hardpoints enabling either a distributed flange type payload mount or a truss type mount. Typical payload installations are shown in Figure 2-10. Electrical and fluid connections are provided on the equipment module for payload servicing.

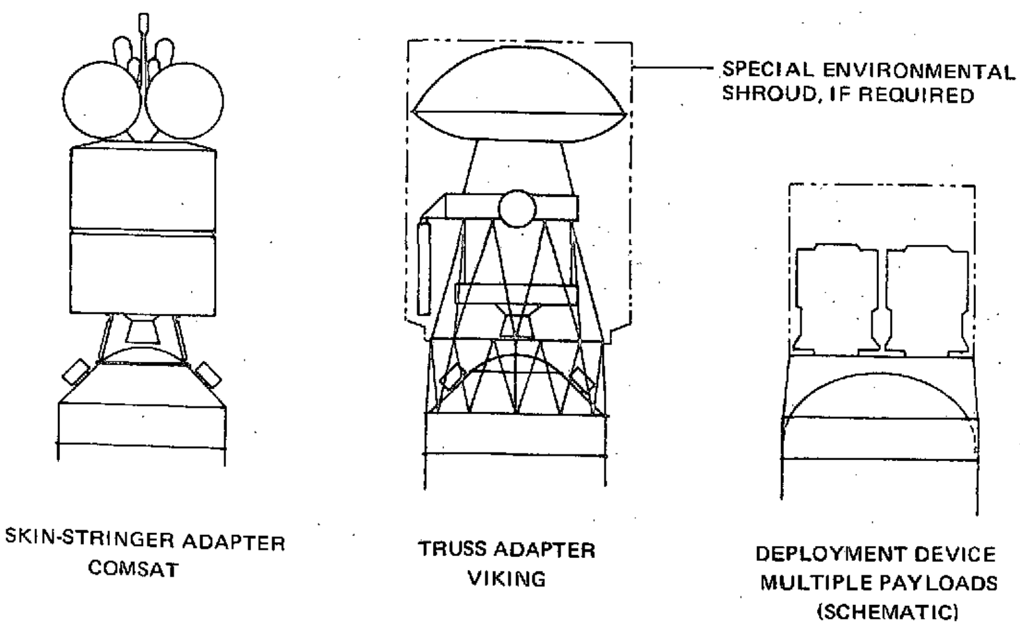


Figure 2-10. Typical Payload Installations

The RLTC mounts the payload from either a flange or 12 hardpoints on the 168-inch (4.26-m) diameter forward adapter. Smaller payloads utilize truss type adapters for support.

The hard support points can also be used to mount a standardized docking adapter.

2.3 MAIN PROPULSION SYSTEM

The main propulsion system includes the main engines and all the equipment involved in the flow, conditioning, and pressurization of propellants from the main tanks to those engines.

2.3.1 MAIN ENGINES. Thirty-thousand (30,000) pounds (133,400 newtons) of primary vehicle thrust is provided by the existing Centaur main engines, with minor modifications for reusability. The dual engines are basically the GFE, P&W RL10A-3-3 engines developed in conjunction with the Centaur. These engines have evolved to their present state of operational high energy, high reliability through extensive engine and system testing.

The engines use liquid hydrogen and oxygen as propellants, and are a constant thrust, turbopump-fed, regeneratively cooled design capable of multiple restarts after long space coast periods. After a long coast, the engines are thermally conditioned to accept cryogenic propellant before the combustion process is started. The RL10A-3-3 engine is shown in Figure 2-11.

Reusability. The existing expendable RL10A-3-3 specification guarantees 4000 seconds of life, while P&W F.R. 5523, 1/31/73, Application of RL10 Engine for Space Tug Propulsion predicts 5 hours (18,000 seconds); P&W letter to Walt Mitton, Convair, 8/5/69, documented testing of single engine builds in excess of 4 hours. Converting these burn times to the number of maximum propellant load flights for both D-1S(R) and RLTC indicates what actual test results reflect in the RL10A-3-3 usable life. For a detailed discussion, see Reference 1, Volume 5, pages 103-107.

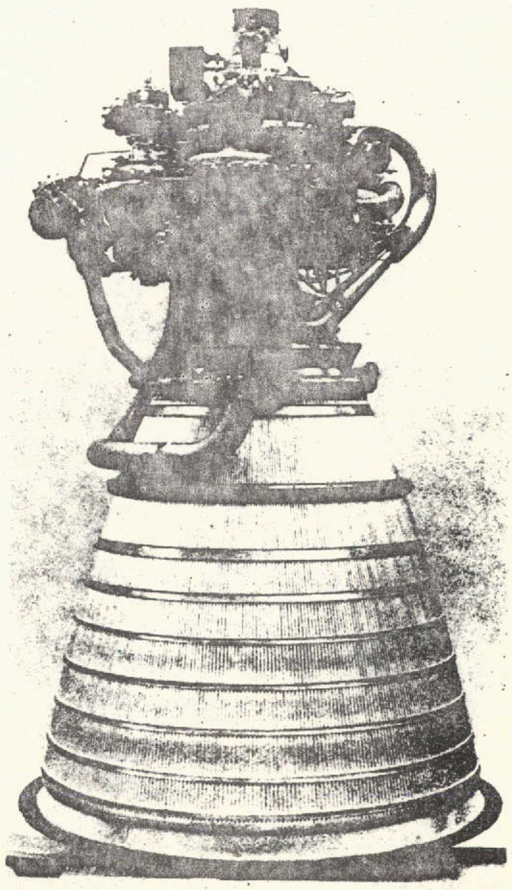


Figure 2-11. RL10A-3-3 Engine

Engine Life By		D-1S(R) (425 sec max/mission)	RLTC (786 sec max/mission)
Specification	= 4,000 sec	10 Missions (average)	6 Missions
P&W Recommendation	= 8,000 sec	18 Missions	10 Missions
Test Experience (High Time for One Build)	= 14,400 sec	34 Missions	18 Missions

Residual hydrogen and oxygen propellants vaporize within the engine chambers, and are purged with dry helium before and after the mission for safety reasons. The engine is essentially clean and ready for inspection and reuse after each mission.

5.8:1 Mixture Ratio (D-1S(R) Only). Operating the engines at 5.8:1 mixture ratio allows the hydrogen tank to be shortened, achieving a total vehicle length, including the SRM, of 35 feet (10.67 m). The engine mixture ratio is changed by resetting the thrust control, and retaining the current PU excursion of ±0.5. The engine nominal I_{sp} decreases from 444 seconds to 439.6 seconds.

The RL10A-3-3 has demonstrated the capability of operating at nominal mixture ratios up to 6:1 with no appreciable effect on thrust chamber life. Operation at 6:1 rather than 5:1 reduces the number of chamber thermal cycles from 200 to 190, which is still far in excess of the vehicle life requirements.

2.3.2 SLOW COOLDOWN. The existing Centaur cooldown techniques must be revised for the Shuttle. The currently used ground chilldown with liquid helium and high rate overboard dump for the suborbital start are not effective in the Shuttle sequence where the first start is orbital, about an hour after launch. Furthermore, the longer coast duration on many missions would result in warmer feed system temperatures and therefore revised cooldown. Therefore a revised, "slow cooldown" technique is proposed on Reusable Centaur. This basically involves reduced flowrates from the engine cooldown valves, active engine temperature measurements, and dual speed boost pumps.

Reduced Cooldown Valves Flow. The revised engine cooldown valves, in conjunction with the revised thermal conditioning sequence, reduce propellant chilldown losses on multistart missions, i.e., D-1S(R) (five starts) from 694 pounds (312 kg) to 125 pounds (56 kg) and the RLTC (six starts) with longer coast times, from 1200 pounds (544 kg) to 150 pounds (68 kg).

The chilldown valve areas were selected to minimize the chilldown losses, within the current valve design and start transient operational limits. The pacing valve is the "fuel interstage cooldown valve", being limited to 0.1 sq in (0.65 sq cm). A smaller area, and more efficient use of propellants for thermal conditioning, would require valve modifications plus significant engine start transient requalification. Optimizing the thermal conditioning about this limitation, the "hydrogen discharge valve" area of

0.05 sq in (0.32 sq cm) and "oxidizer control valve" area of 0.04 sq in (0.26 sq cm) produce acceptable reductions over the expected coast temperature ranges without upsetting the proven RL10A-3-3 start characteristics.

Active Engine Temperature Measurements. Incorporating the same critical temperature measurements used in engine qualification testing eliminates the wide possible space environment variations and the resulting allocation for propellant childown losses; i.e., D-1S(R) 120 pounds (54 kg) and RLTC, 350 pounds (245 kg) (Reference 1, Volume 5, pages 135 and 156). The actual engine temperatures are read by the onboard computer, which then presets the thermal sequence times.

2.3.3 FEED SYSTEM. The propellant feed system of the existing Centaur is used as is, with minor modifications for reusability and manned compatibility. The system transfers propellants from the main tanks for engine use. Involved in the propellant transfer is the thermal conditioning of the ducts and engines before engine ignition and the pressurization (i.e., pumping) of those propellants to the engine turbopumps. The pumping is accomplished by the boost pumps both to prevent main engine turbine stall during engine start and to supply pressurized liquids to the main engine during operation in sufficient quantities to prevent main engine turbopump cavitation. Figure 2-12 illustrates the main propulsion system.

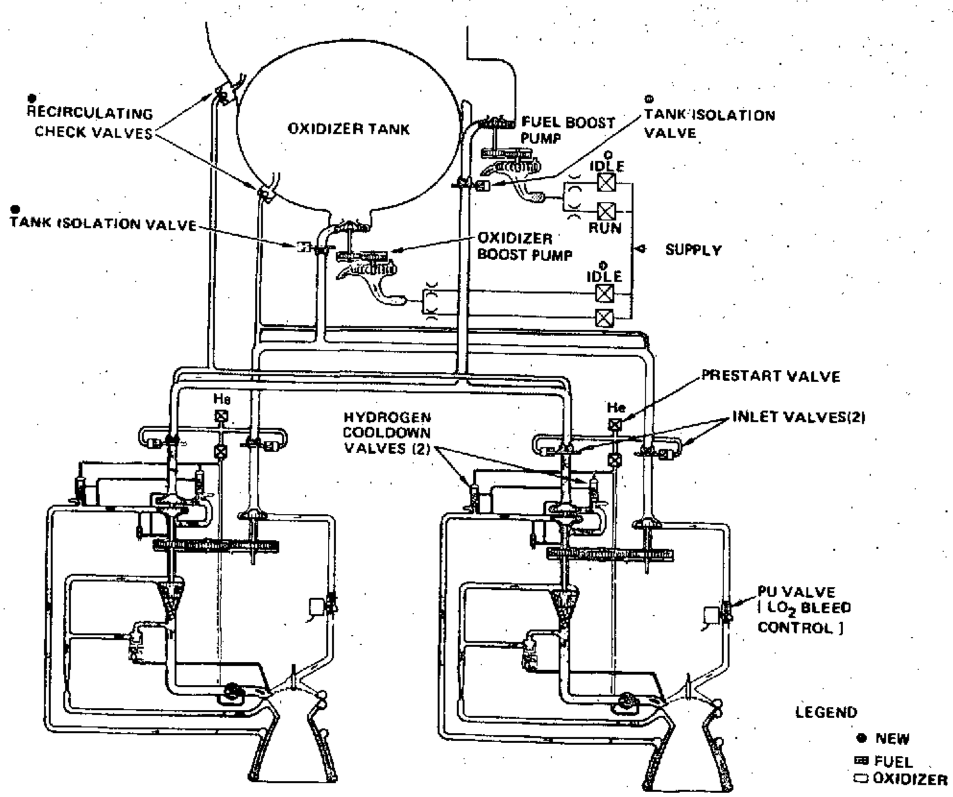


Figure 2-12. Main Propulsion System

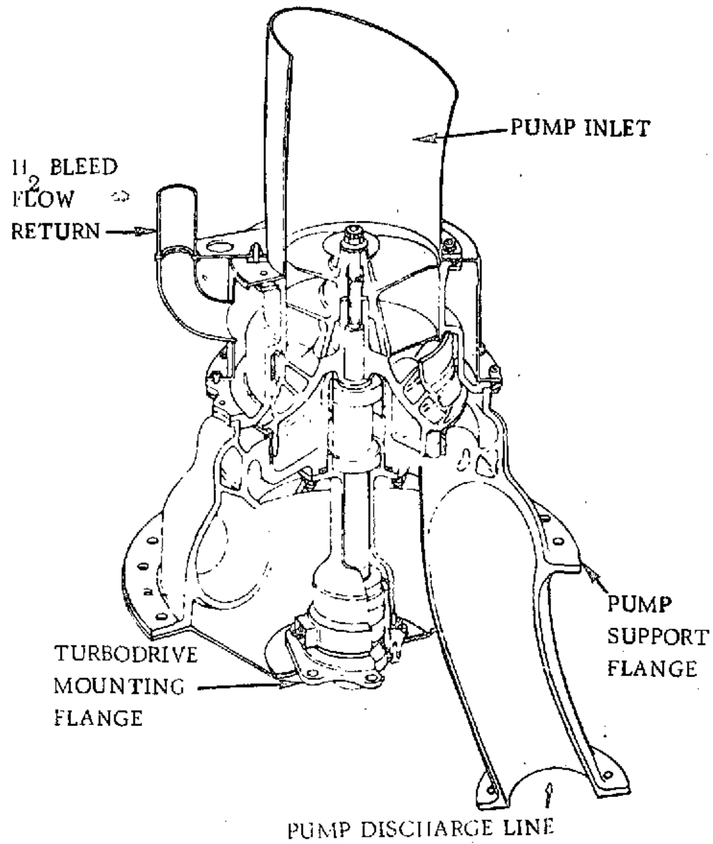


Figure 2-13. Fuel Boost Pump Cutaway

Boost Pumps. The existing Centaur turbine-driven boost pumps are retained for the Reusable Centaur. The Fuel Boost pump is shown in Figure 2-13. Secondary idle speed controls are added for thermally conditioning the propulsion system at low flow, low pressure recirculating propellants (Figure 2-12). Pressurized monopropellants are fed through control orifices and into catalytic beds, where the monopropellants are decomposed into hot gases for turbine power. The high speed turbines drive the boost pumps via gear reduction units, exhausting the hot gases overboard. The turbine monopropellants are common with the vehicle APS supply; D-1S(R) uses H_2O_2 and the RLTC uses hydrazine. The D-1S(R) H_2O_2 APS supply lines and catalyst beds are kept warm by individual thermostatically controlled 28 Vdc electric heaters. The RLTC keeps the system and catalyst beds warm by circulating the hydrazine fluid through the fuel cell heat exchanger. The hydrazine absorbs the fuel cell waste heat to make up for the thermal energy the APS system loses through space radiation.

Isolation and Check Valves. The Reusable Centaur has additional propellant isolation and check valves to keep both propellants contained within the main tank/purge bags whenever the vehicle is either in or near the Orbiter. The ducting is purged with helium, which is exhausted through the engines, collected within the deployment adapter and vented overboard through the Orbiter disconnects. Upon return and reconnection to the Orbiter, the isolation valves are reshut and the propulsion system purged with helium as a part of the safing operation. During reentry, the helium purge pressure is maintained at a slight positive pressure over the Orbiter payload bay pressure. The new isolation and check valves are shown in Figure 2-12.

Propellant Feed Lines. The existing Centaur D-1T propulsion feed lines are used intact, except as modified for the longer main LO_2 tank section (RLTC only) and additional isolation/check valves interconnection. With the lines dry and purged while the vehicle is in the Orbiter, no ground hold insulation is required. Radiation shielding similar to D-1T will be provided to reduce space heating.

Thermal Conditioning. The Reusable Centaur thermal conditioning has been slowed and resequenced, but terminates in the existing and proven Centaur start conditions. The Reusable Centaur thermal conditioning reduces the total propellant losses to less than 15% of the existing rapid start technique (Reference 1, Volume 5, pages 120 to 162). Current Centaurs use a liquid helium ground chilldown, which would not be effective on a Shuttle launch because of the much longer time from launch to first burn.

The expendable Centaur (D-1T) accomplishes a rapid start by concurrent flushing propellants through the total propulsion system and venting those vaporized and mixed phase propellants out of the engine.

The Centaur dual engines produce their full thrust in less than 3 seconds after the start command. To ensure that the engines will start and continue to run, the entire propulsion system must be primed with pressurized cryogenic propellants. System priming is done by thermal conditioning, which precools the hardware and pressurizes the propellants to the engine turbopump inlets. Thermal conditioning achieves a specific range of temperature and propellant pressure conditions before each engine start.

The Reusable Centaur thermal conditioning separates the functions into a series of individual steps, with all but the final engine conditioning using recirculating propellants. The engine conditioning (prestart) is done at slower, more efficient heat transfer rates. This reduces the propellants vented overboard as chilldown losses. Figure 2-14 compares the "rapid" and "slow" chilldown.

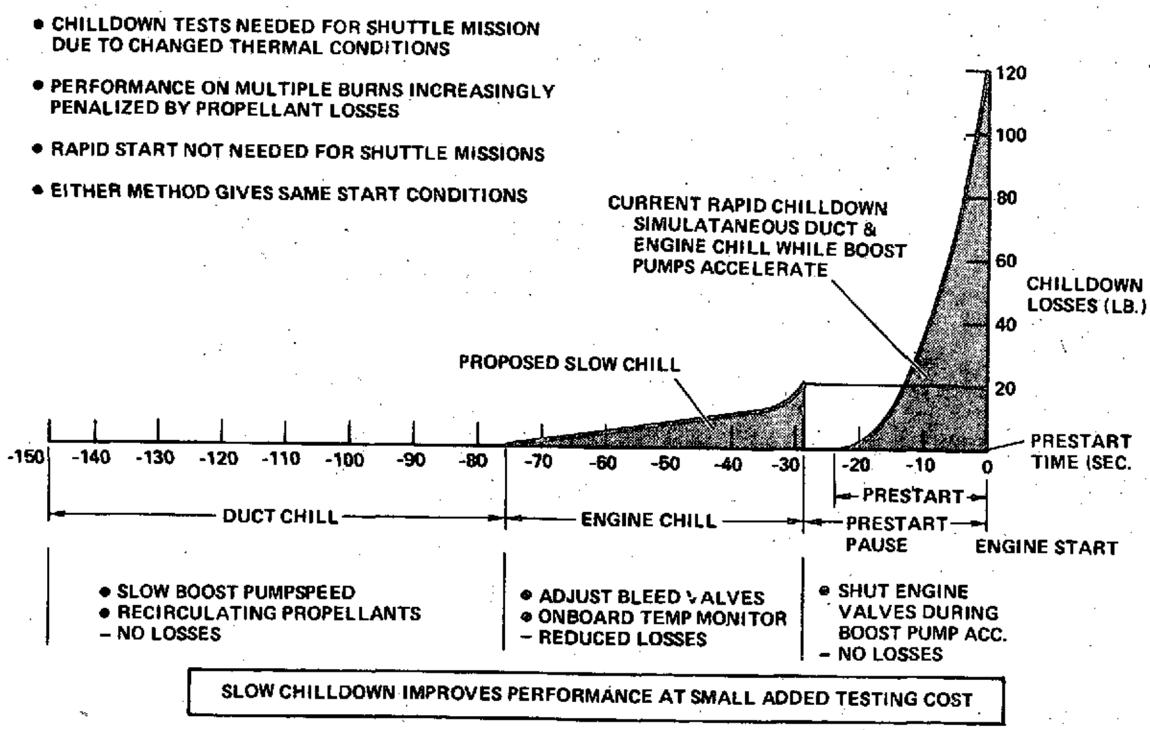


Figure 2-14. Slow Chilldown

During the Reusable Centaur coast period between each engine burn, the propulsion system is warmed by the sun, creating a unique thermal conditioning requirement for each coast. Slow thermal conditioning also incorporates actual engine turbopump temperature measurements. These measurements are directly used to autonomously set the real time thermal conditioning sequence for each coast, and eliminate dumping overboard extra propellants after the engines are chilled.

The Centaur rapid start was developed to reduce gravitational losses of the first suborbital burn. To achieve this rapid chilldown, all the propulsion associated equipment is chilled at the same time. All the chilldown propellants are passed through the engines and dumped overboard. For slow chilldown, functions have been separated and thermal conditioning occurs in the following sequence:

Duct Chill — Recirculate propellants, from the tanks to the engine and return vaporized fluid to tanks for recondensation.

Prestart (Engine Chill) — Slow the flow and extend the time, enabling the fluids to extract more heat from engine mass before being vented overboard.

Boost Pump Acceleration — After the entire system has been chilled, reshut the engine inlet valves during boost pump acceleration which varies between 14 to 28 seconds. This prevents the extra propellants from being dumped overboard.

The slow Reusable Centaur's chilldown arrives at the same engine conditioning requirements of temperatures and pressures for engines start as the existing Centaur's rapid start chilldown.

The maximum potential performance loss associated with thermal conditioning has been used in the thermal conditioning analysis; maximum number of burns per mission in combination with the worst case of full sun exposure. Fewer burns per mission, shorter coast/mission times, oblique sun impingement and shadowing effects would normally occur, reducing the actual operational thermal conditioning losses shown in Table 2-2.

Propellant losses through the main engines include starting and shutdown transient losses, and leakage through the engine inlet valves during coast phases. Leakage during main engine burn periods is accounted for in the engine I_{sp} value used. Starting and shutdown losses, propellant from which no useful impulse is obtained, were estimated by Pratt & Whitney at 22 pounds (10 kg) per firing based on total propellant flowing through the engine and measured start and shutdown impulse.

Leakage during coast is based on maximum values (0.63 lb/hr) measured by Convair and P&W for all delivered RL10's. This value has been increased by a factor of three to allow for increased leakage over the life of the engine. Total leakage is based on time from opening of the feed line isolation valves, after separation from the Orbiter, until completion of the final main engine burn. Table 2-3 presents propellant loss data.

Table 2-2. Main Propulsion System Thermal Conditioning

Burn No.	Boost Pump Idle: Duct Chill (sec)	Boost Pump Idle: Prestart sec/lb_b	Boost Pump Idle: (lb_a)	Prestart Pause (lb_a)	Start Losses (lb_b)	Total Non ΔV Losses $lb_a + lb_b = lb_{Total}$
D-1S(R)						
1	84	69/38.6	1.1	2.4	10	3.5 + 48.6 = 52.1
2	71	47/19.8	1.0	2.4	10	3.4 + 29.8 = 33.2
3	30	22/23.6	0.5	2.4	10	2.9 + 33.6 = 36.5
4	70	46/18.4	1.0	2.4	10	3.4 + 28.4 = 31.8
5	30	22/23.6	0.5	2.4	10	2.9 + 33.6 = 36.5
		125_c				$16.1_a + 174.0_b \sim 190$

a = Monopropellant non ΔV losses. b = Main propellant non ΔV losses. c = Prestart - "Childown" losses.

Table 2-3. Dual Main Engine Propellant Losses

	D-1SR	RLTC
Number of firings	5	6
Start, lb	50	60
Shutdown, lb	60	72
Leakage time, hr	13.2	39.8
Leakage loss, lb	25	75

2.4 MAIN ENGINE SUPPORT SYSTEMS

The main engine support systems for D-1S(R) and RLTC include all subsystems that contribute to functioning of the main engines and propellant flow from the tanks to the engines. The main engine boost pumps and propellant feed lines are main engine support systems, but they are included in Section 2.3 for clarification of the description of the entire propulsion system.

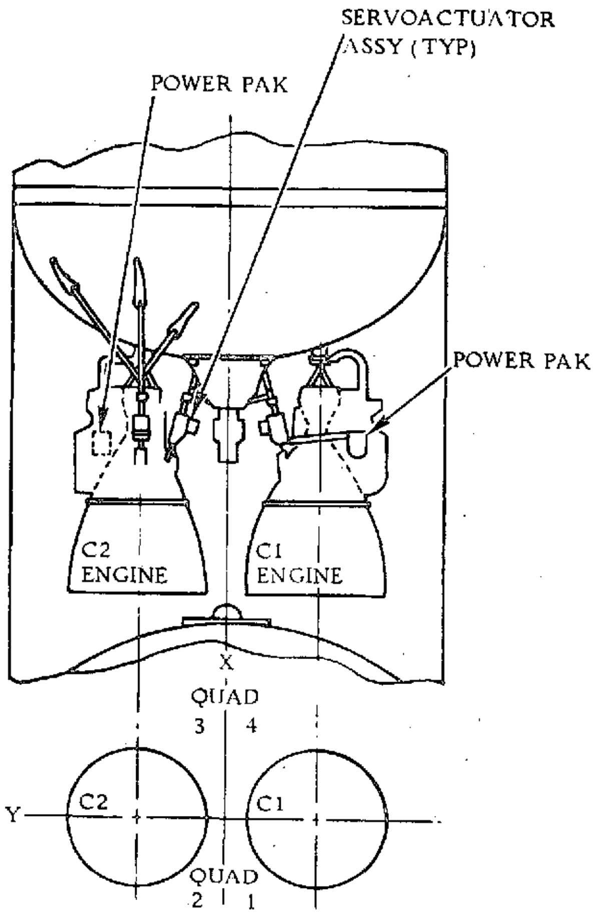


Figure 2-15. Centaur Hydraulic System Orientation

2.4.1 THRUST VECTOR CONTROL (Reference 1, Volume 5, Section 5.2). The thrust vector control is accomplished by use of the existing Centaur hydraulic systems, which provide the mechanical force required to gimbal the Centaur main engines during ground checkout and flight. Each system consists of an integrated hydraulic power package, two servo-controlled engine gimbaling actuator assemblies, a manifold, and miscellaneous fittings and connecting tubing. The systems operate at nominal pressures of 100 psig (69 N/sq cm) during non engine firing and 1000 psig (690 N/sq cm) during engine firing. Hydraulic fluid temperatures range from -30° F to +275° F (-34.5° C to +135° C). Figure 2-15 illustrates the hydraulic system orientation.

The existing Centaur hydraulic systems are used intact, with additional thermostatically controlled electrical heaters. A single heater is installed in the body of each hydraulic power package reservoir. The heater was sized (41 watts) based on the maximum environmental conditions of heat losses that could occur during long coast (Reference 1, Volume 5, page 280). The existing circulating system thermostats control the heater, in addition to the circulating motors. The existing circulating motors will be retested/replaced for multiple missions long coast life.

Existing test data substantiates that both the actuators and the power pack have adequate life in terms of cycles and running time for both RLTC and D-1S(R) missions. Some additional testing may be required, however, to demonstrate adequate fatigue life for these components to enable them to be used for the entire life of the vehicle without replacement.

2.4.2 PRESSURIZATION SYSTEM (Reference 1, Volume 5, Section 5.2)

D-1S(R) Pressurization System. Figure 2-16 shows the pressurization system for D-1S(R). This system provides the helium necessary to pressurize the LH_2 and LO_2 tanks to the pressure levels required during engine prestart, and provides regulated helium pressure for the engine controls systems and the auxiliary propulsion system.

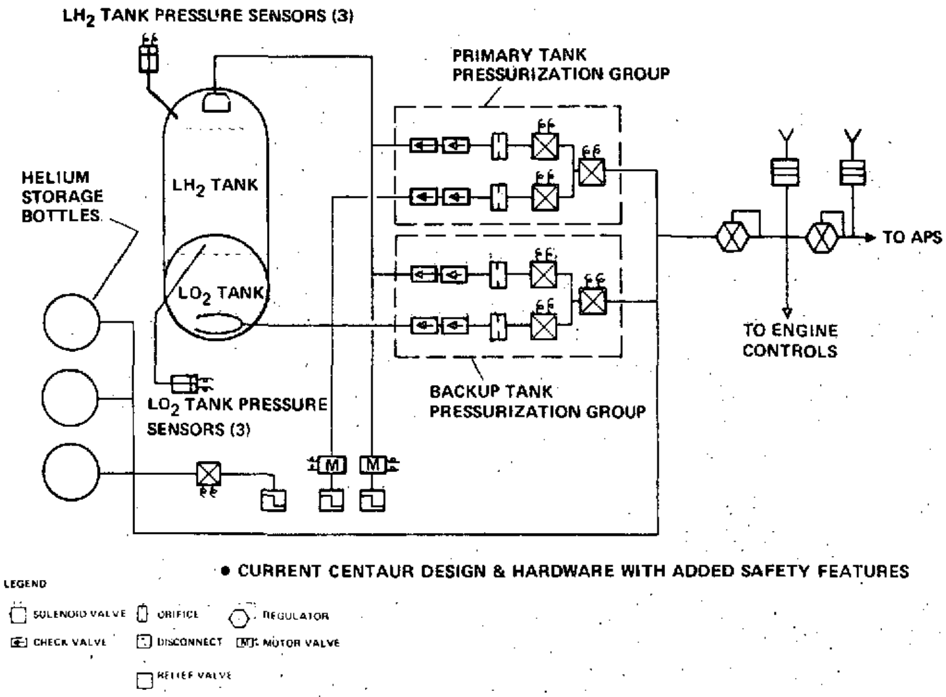


Figure 2-16. D-1S(R) Pressurization System

The system hardware and its mounting is the same as Centaur D-1T, except the tank pressurization solenoid valves are rearranged to meet Orbiter safety considerations, and pyrotechnic shutoff valves in the tank pressurization lines have been changed to motor valves for reusability.

Prior to each engine start, the ullage pressure in each main propellant tank is increased 3 psid (2 N/cm^2) to provide sufficient margin over the required net positive suction head (0.1 psi (0.069 N/cm^2) LH_2 and 1.5 psi (1.03 N/cm^2) LO_2), for the main engine boost pumps. The technique for providing this prestart pressurization is the same as for Centaur D-1T. The digital computer unit reads triple redundant transducers in each tank, and opens and closes tank pressurization solenoid valves as required. No helium pressurization is required during engine burns. Pressurization helium is supplied from the ambient temperature helium storage bottles to a regulator (with downstream relief valve protection) which provides 450 psig (310 N/cm^2) for operation of the engine control valves. A second regulator further reduces this pressure to 290 psig (200 N/cm^2) for pressurization of the auxiliary propulsion system bottle to expel H_2O_2. From these two regulated supplies are also furnished constant bleed purges to the LO_2 and LH_2 tank pressurization lines, the LO_2 tank sensing line, and the LO_2 tank standpipe to keep them free of liquid propellants during periods of zero-g coasting. The hardware for pressure regulation and purges is the same as used on D-1T Centaur.

The LH_2 tank pressurization line terminates inside the top of the LH_2 tank in a helium energy diffuser (D-1T design) that reduces the velocity of the incoming helium to minimize the amount of helium required for tank pressurization. The LO_2 tank pressurization line terminates in a perforated tubing ring mounted inside the LO_2 tank at a level below the liquid surface. During a pressurization sequence, helium from the ring bubbles up through the LO_2 to provide an efficient method of tank pressurization by minimizing helium usage. Flow control orifices are installed downstream of the tank pressurization valves, and series redundant check valves are installed to ensure separation of LO_2 tank and LH_2 tank ullage pressures.

During prelaunch operations, the helium bottles are charged to flight pressure (approximately 2800 psig) (1931 N/cm^2) through a sealing disconnect and solenoid valve (Centaur D-1T) components. During ground operations, tank pressures are maintained by the pallet pressurization system through sealing disconnects (Centaur D-1T components) until the start of propellant tanking. Propellant boiloff then maintains tank pressures (by vent valve control) throughout the tanking operation and during Shuttle ascent.

D-1S(R) pressurization system parameters are given in Table 2-4, and helium requirements are listed in Table 2-5.

Table 2-4. D-1S(R) Pressurization System Parameters

Helium Storage Bottle Fill Pressure (max)	2800 psia (1930 N/cm^2)
Helium Storage Bottle Final Pressure (min)	500 psia (345 N/cm^2)
Helium Storage Bottle Charge Temp. (nom)	530°R (294K)
Helium Storage Bottle Volume	12.8 ft^3 (0.36 m^3)
Usable Helium Available	17.9 lb (8.1 kg)
Engine Controls Supply Pressure (nom)	450 psia (310 N/cm^2)
APS Bottle Supply Pressure (nom)	290 psia (200 N/cm^2)

Table 2-5. D-1S(R) Helium Requirements

LH_2 Tank Pressurization	7.13 lb	(3.23 kg)
LO_2 Tank Pressurization	1.80 lb	(0.82 kg)
APS Propellant Pressurization	1.50 lb	(0.68 kg)
Engine Start Usage	0.44 lb	(0.20 kg)
Engine Helium Leakage	0.18 lb	(0.08 kg)
LO_2 Bubbler & Zero-g purges	1.70 lb	(0.77 kg)
Total Required	12.75 lg	(5.78 kg)
Available Contingency (40%)	5.15 lb	(2.34 kg)
Total Available	17.90 lb	(8.12 kg)

The D-1T pneumatic system components to be used on D-1S(R) have generally demonstrated adequate functional life to meet reusability requirements for the life of the vehicle without routine changeout. In general, however, existing vibration testing may be augmented with more vibration testing to demonstrate adequate fatigue life.

RLTC Pressurization System. The RLTC pressurization system schematic is shown in Figure 2-17. The system provides the helium necessary to pressurize the LH_2 and LO_2 tanks to the pressure levels required during engine prestart, and provides regulated helium pressure for the engine controls system and the auxiliary propulsion system. Most of the system operating techniques and the hardware components required for the pressurization system are the same for RLTC as are currently used on Centaur D-1T.

Prior to each engine start, the ullage pressure in each main propellant tank is increased 3 psid (2 N/cm^2) to provide sufficient margin over the required net positive suction head (0.1 psi (0.069 N/cm^2) LH_2 and 1.5 psi (1.03 N/cm^2) LO_2) for the main engine boost pumps. The technique for providing this prestart pressurization is the same as for Centaur D-1T. No helium pressurization is required during engine burns. Pressurization helium is supplied to a regulator (with downstream relief valve protection) which provides 450 psig (310 N/cm^2) for operation of the engine control valves and for pressurization of the auxiliary propulsion system bottle to expel N_2H_4. From this regulated helium supply are also furnished constant bleed purges for the LO_2 and LH_2 tank pressurization lines, the LO_2 tank sensing line, and the LO_2 tank standpipe to keep them free of liquid propellants during periods of zero-g coasting. The hardware for pressure regulation and purges is the same as used on D-1T Centaur.

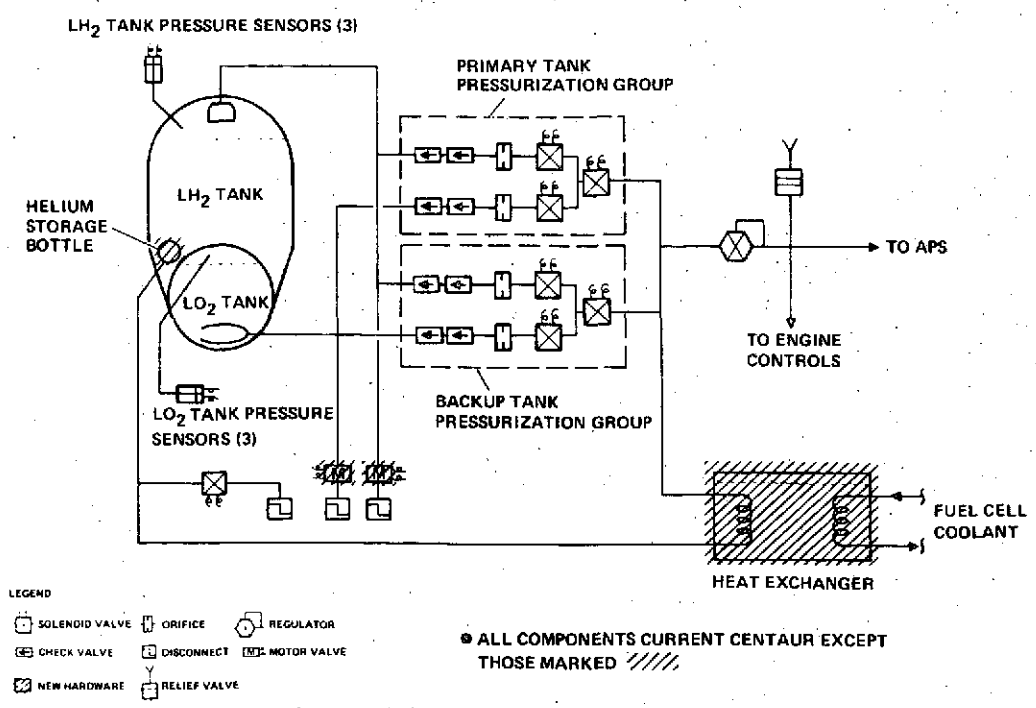


Figure 2-17. RLTC Pressurization System

Like D-1S(R) and D-1T, the RLTC LH_2 tank pressurization line terminates in a diffuser and the LO_2 tank pressurization line terminates in a bubbler ring.

During prelaunch operations, the helium bottle (mounted inside the LH_2 tank) is charged to flight pressure (approximately 2600 psig) (1793 N/cm^2) through a sealing disconnect and solenoid valve (Centaur D-1T components). During ground operations, tank pressures are maintained by the support adapter pressurization system through sealing disconnects (Centaur D-1T components) until the start of propellant tanking. Propellant boiloff then maintains tank pressures (by vent valve control) throughout the tanking operation and during Shuttle ascent.

A trade study was performed to compare storage of helium at ambient temperature (530°R) (294K) with storage at LH_2 temperature (38°R) (21K). The results indicated that cryogenic storage improved payload capability about 740 pounds (336 kg).

RLTC pressurization system parameters are given in Table 2-6, and helium requirements are listed in Table 2-7.

Table 2-6. RLTC Pressurization System Parameters

Helium Storage Bottle Fill Pressure (nom)	2600 psia (793 N/cm^2)
Helium Storage Bottle Final Pressure (min)	500 psia (345 N/cm^2)
Helium Storage Bottle Charge & Use Temp. (nom)	38°R (21K)
Helium Storage Bottle Volume	4.28 ft^3 (.12 m^3)
Usable Helium Available	28 lb (12.7 kg)
Engine Controls & APS Supply Pressure (nom)	450 psig (310 N/m^2)

Table 2-7. RLTC Helium Requirements

LH_2 Tank Pressurization	14.01 lb (6.35 kg)
LO_2 Tank Pressurization	2.90 lb (1.32 kg)
Engine Start Usage	0.53 lb (0.24 kg)
Engine Leakage	2.25 lb (1.02 kg)
APS Propellant Pressurization	3.30 lb (1.50 kg)
LH_2 Tank Helium Dissipator Purge	2.42 lb (1.10 kg)
LO_2 Sense Line & Bubbler Purge	0.86 lb (0.39 kg)
Misc. Unplanned System Leakages	1.73 lb (0.78 kg)
Total	28.00 lb (12.70 kg)

The D-1T pneumatic system hardware to be used on RLTC generally has already demonstrated by test its functional reusability for the mission life of RLTC. Fatigue life of the components for the RLTC missions may have to be demonstrated by additional vibration testing.

2.4.3 VENT SYSTEM. (Reference 1, Volume 5, Section 5.2) The main propellant tank vent system for D-1S(R) is shown in Figure 2-18. Figure 2-19 shows the RLTC system. The function of the vent system is to maintain proper propellant tank ullage pressures during all phases of the mission, from the start of ground tanking through Orbiter landing, except during engine prestart pressurization sequences and engine firing periods.

During ground tanking, Shuttle ascent, and Orbiter return, parallel redundant vent valves control tank pressures. These solenoid-operated valves are the same as used on D-1T Centaur, and operate in either a relief or shut-off mode. When in the relief mode, the vent valves maintain preset, absolute tank pressures, and act as upstream pressure regulation devices. When in the shut-off mode, the valves remain closed regardless of tank pressure. Selection of the proper operating mode is done by the Digital Computer Unit (DCU).

Two LH_2 and LO_2 tank vent valves are mounted in parallel to meet fail-closed safety requirements. The primary LH_2 and LO_2 tank vent valves will maintain tank pressures during ground tanking and Shuttle ascent between 19.0 and 21.5 psia (13.1-14.8 N/cm^2) and 29.0 to 32.0 psia (20 to 22 N/cm^2), respectively. The secondary LH_2 and

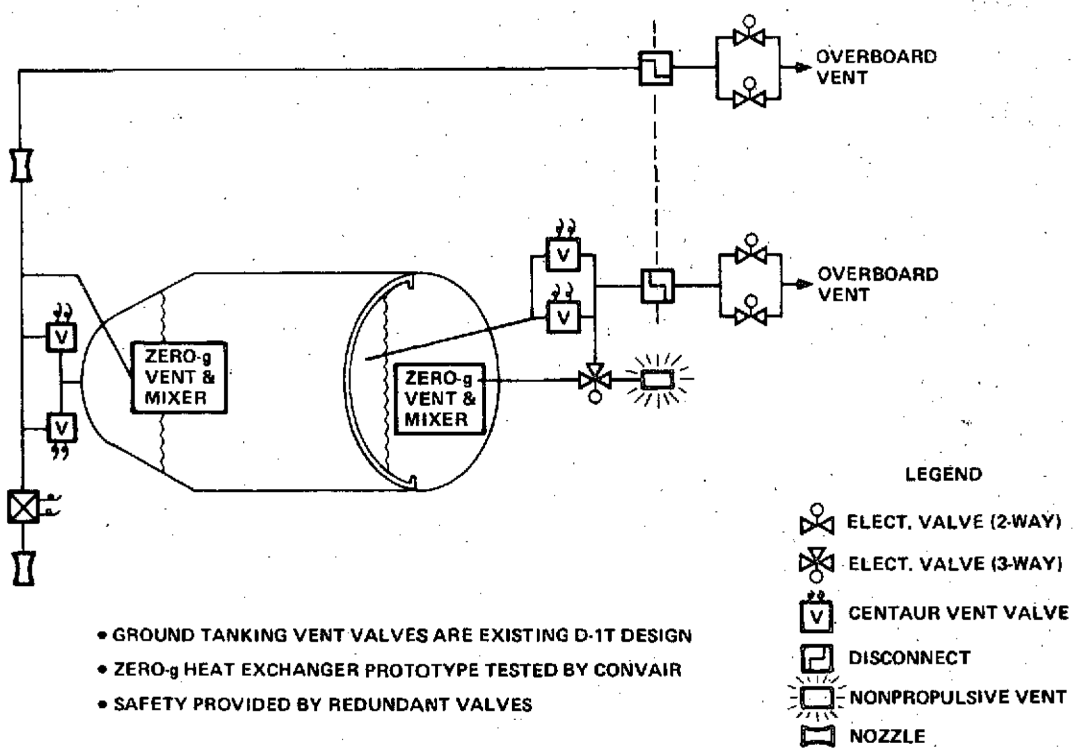


Figure 2-18. D-1S(R) Propellant Tank Vent System

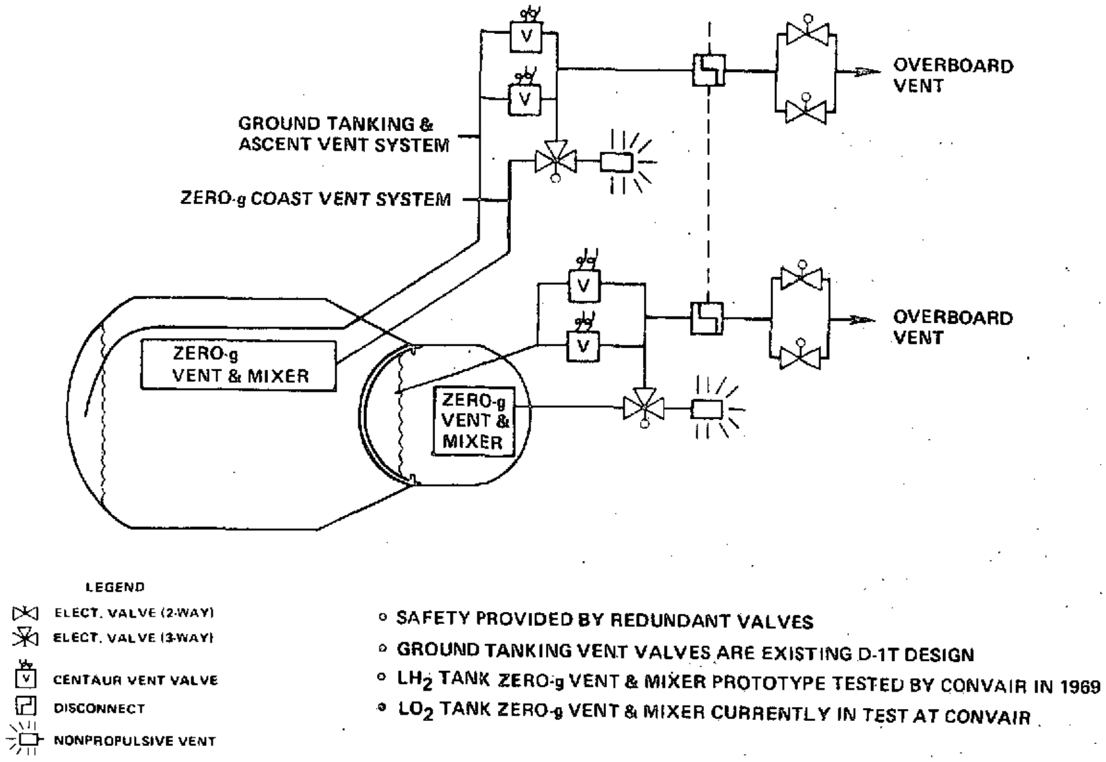


Figure 2-19. RLTC Vent System

LO_2 tank vent valves will provide safety backup to the primary valves, and in addition, during Orbiter reentry upon abort with the payload installed, will maintain the LH_2 tank at 25.0 to 26.5 psia (17.2 to 18.3 N/cm^2) and the LO_2 tank at about 34.0 to 37.0 psia (23.4 to 25.5 N/cm^2). In the overboard vent line for both LH_2 and LO_2 vent systems, on the deployment adapter, are two parallel mounted backup shut-off valves. These are normally open, but would be closed (and modulated open as required) only if a Centaur vent valve failed open. Control of the backup valves is from the Centaur monitor and control systems (CMACS). During Orbiter reentry, the overboard LH_2 vent duct will be purged with helium to insure that a combustion hazard does not build up due to entry of air into the line between venting periods.

During periods of D-1S(R) or RLTC zero-g coasting, with the vehicle either in the Orbiter bay or in space, a zero-g venting system is required in each tank. The system consists of an electrically driven pump/mixer to circulate the cryogenic propellant, a regulator to expand the propellant to be vented, a coil heat exchanger to chill the bulk fluid being circulated by the pump and to heat the expanded vent gas, and a solenoid shut off valve to actuate the vent system. The mixers are designed to run continuously during zero-g coast periods, with the system venting intermittently on demand. Figure 2-20 is a schematic and a photograph of an LH_2 tank prototype unit designed and tested at Convair in 1969. The tests proved that the vent device works equally well with either liquid or gas at the heat exchanger inlet, and that propellant settling prior to venting (as is currently done on D-1T) is not required. A prototype LO_2 tank vent

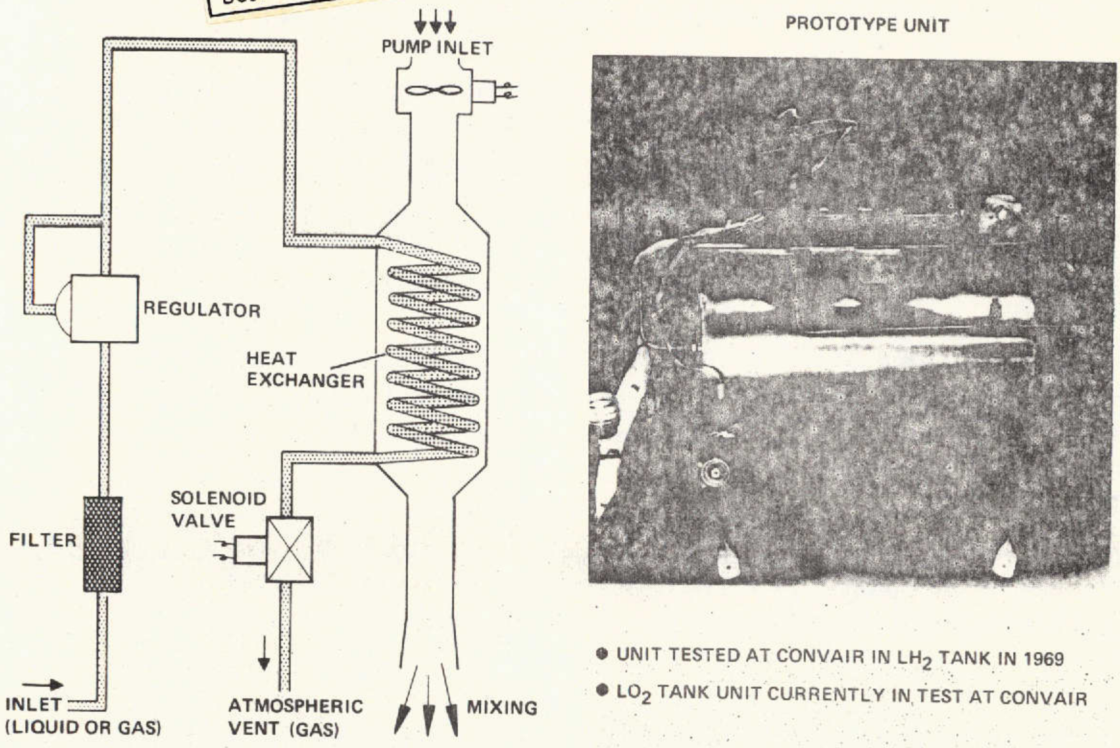


Figure 2-20. Zero-G Vent Device

device has been designed and built, and is currently undergoing testing at Convair. Two of these zero-g vent devices are mounted inside the Centaur, one in the LH_2 tank, and the other in the LO_2 sump and boost pump housing.

For both D-1S(R) and RLTC, two vent paths are included for the LO_2 zero-g vent system. One vent line ties into the ground vent ducting, and the other leg is routed to a non-thrust vent device. This second leg of the vent system is used during zero-g coast when the Centaur is out of the payload bay. An electrically actuated three-way valve connects the non-thrust vent to the vent line that is routed to the aft umbilical panel. During Orbiter deployment of the Centaur, this valve will be activated to close the vent line to the aft umbilical panel and open the line to the non-propulsive vent.

The current Centaur D-1T hydrogen vent system employs a balanced thrust vent ducting mounted on top of the LH_2 tank into which both LH_2 vent valves exhaust. For D-1S(R), the hydrogen ground vent system is similar except for the addition of a shutoff valve in one leg of the balanced vent ducting, and revision to the cant angle of the two exit nozzles. When the D-1S(R) is in the Orbiter bay, H_2 must vent through the single leg of the vent system that is ducted overboard through the flight pallet. The additional valve prevents venting from the other leg until the D-1S(R) is deployed and the valve opened. The nozzle cant angle is changed to conform to the pallet installation, umbilical retraction, and deployment techniques to be used with the Shuttle.

For D-1S(R), the hydrogen overboard zero-g vent tube passes through the LH_2 tank forward door and is connected into the ground vent line downstream of the tank-mounted vent valves. For RLTC, the hydrogen vent has two paths. While the RLTC is within the payload bay, LH_2 tank venting exhausts through the ground ducting. After the RLTC has been deployed from the Orbiter, the LH_2 tank vent gases are exhausted through a nonpropulsive vent device like the LO_2 tank vent system.

The deep space vent rates listed in Table 2-8 represent the steady state vent rates that would eventually occur during coast in the absence of engine firings. Actual venting will not occur during coast until the tank pressure reaches the vent pressure of the zero-g vent device. As shown in Figure 2-21, the engine firing sequence for D-1S(R) reduces LO_2 tank pressure so that the LO_2 tank does not actually vent any propellants overboard after the first Centaur engine firing, and the LH_2 tank vents overboard only during a portion of the coast time between burns. Figure 2-21 shows LO_2 and LH_2 tank ullage pressure history for D-1S(R) as controlled by the pressurization and vent systems.

Testing to date on the D-1T vent valves has demonstrated that the repeated lockup capability of the valves far exceeds the reusability requirements of both D-1S(R) and RLTC. Additional operating life cycling and vibration (fatigue) testing may be required to demonstrate these reusability requirements, however.

Table 2-8. Zero-g Vent System Characteristics

Nominal System Characteristics	D-1S(R)		RLTC	
	LO_2	LH_2	LO_2	LH_2
Tank Heat Rate (Deep Space), Btu/hr (watts)	573 (168)	2178 (638)	592 (173)	2634 (771)
Zero-g Vent Design Flow Rate, lb/hr (kg/sec)	89.5 (0.0113)	33.1 (0.0042)	90.5 (0.0114)	39.8 (0.0050)
No Vent Pressure Rise Rate (Payload Bay Doors Open, Tanks Full), psi/hr (Newton/cm^2 sec)	0.5 (9.6×10^{-6})	1.0 (19.2×10^{-6})	0.3 (5.7×10^{-6})	0.6 (11.5×10^{-6})
Vent Rate (Deep Space), lb/hr (kg/sec)	9.4 (11.8×10^{-4})	10.8 (13.6×10^{-4})	9.7 (12.2×10^{-4})	13.0 (16.4×10^{-4})

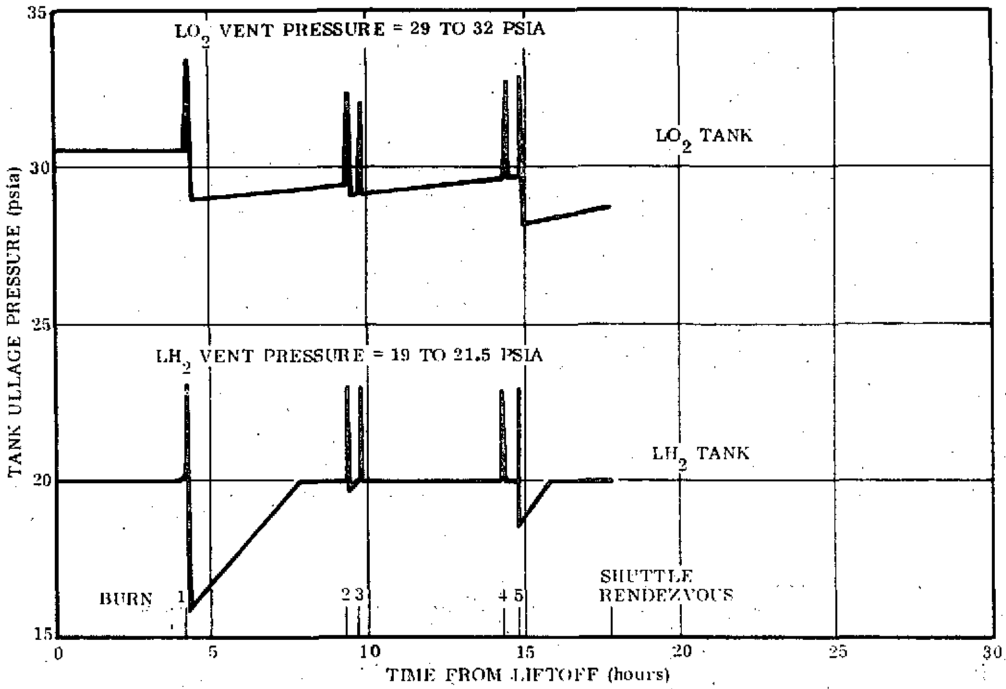


Figure 2-21. D-1S(R) Propellant Tank Ullage Pressure vs Time From Liftoff

2.4.4 PROPELLANT UTILIZATION. (Reference 1, Volume 5, Section 5.2) Propellant utilization is accomplished by the existing Centaur PU system, which provides engine oxidizer mixture control to simultaneously deplete both propellants. The PU system maximizes total impulse while minimizing unequal burn-out residuals. The system consists of tank probes, electronic circuit modules, motor-driven servopositioners, and feedback transducers to modulate the engine O_2 flow control valves (PU valves). All control functions are accomplished by the astrionics digital computer unit (DCU) and sequence control unit (SCU). Figure 2-22 illustrates the propellant utilization system orientation.

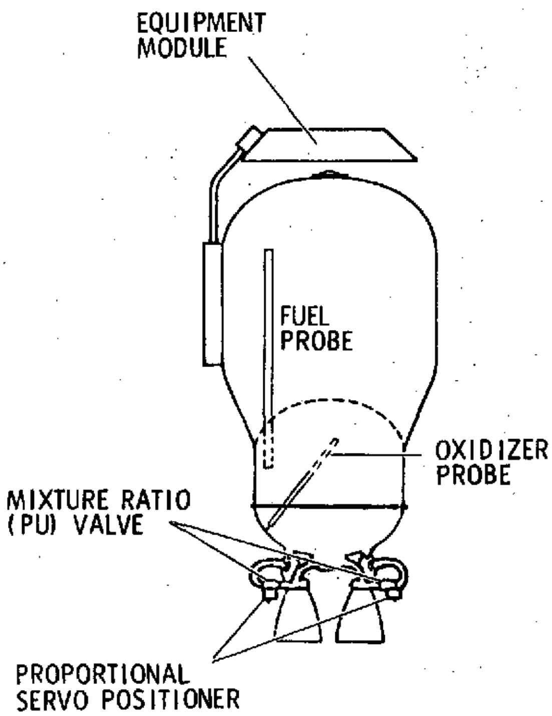


Figure 2-22. PU System Orientation

The D-1T type probes will be resized and tested to accommodate the revised tank shapes for D-1S(R) and RLTC. This testing will include vibration (fatigue) tests to demonstrate adequate reusability for the vehicle life.

2.4.5 INTERMEDIATE BULKHEAD VACUUM SYSTEM. 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)(<6.7 N/m^2).

The system for both D-1S(R) and RLTC is the same as used on Centaur D-1T (Figure 2-23). It consists of a check valve, pressure transducers, and tubing connecting the intermediate bulkhead cavity and the aft umbilical panel.

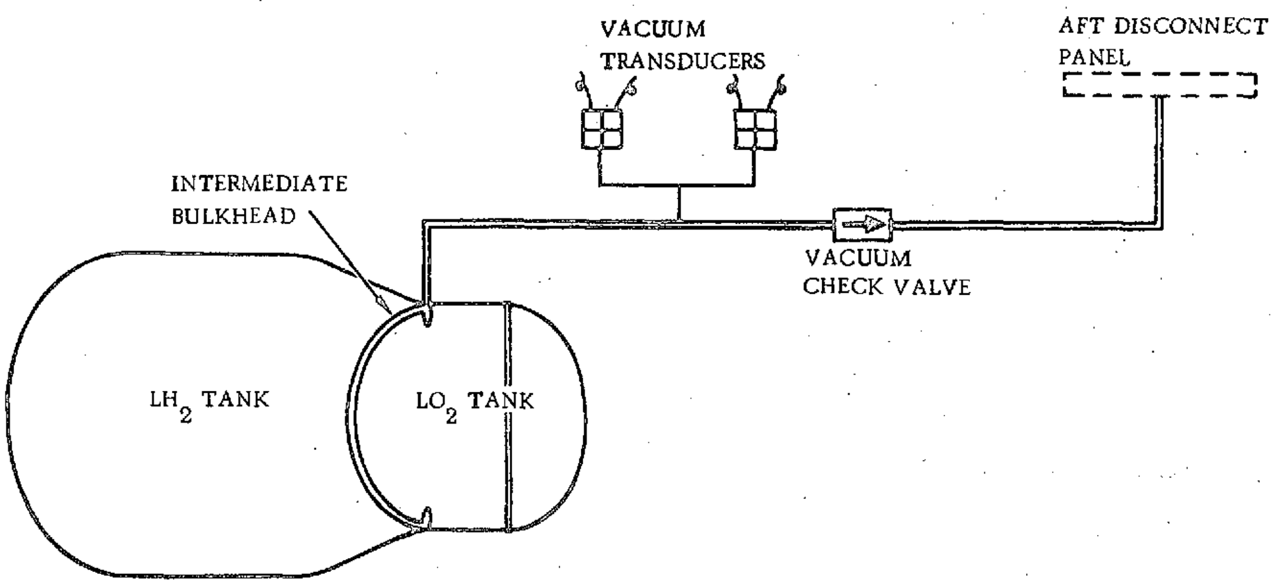


Figure 2-23. Intermediate Bulkhead Vacuum System

2.4.6 PROPELLANT TANK FILL AND DRAIN SYSTEM. The main propellant tank fill and drain systems for both D-1S(R) and RLTC are utilized to both fill the tanks prior to launch, and to quickly dump tanked propellants during flight if required by an abort condition. Both propellants will be dumped at altitudes greater than 220,000 feet (67 km) by supplying helium pressurant from storage bottles located on the deployment adapter and from vehicle stored helium. Parallel redundant dump valves are installed in each propellant line on the vehicle to meet fail-closed safety requirements. These dump valves are backed-up by redundant dump valves on the adapter which guard against failures in the failed-open mode.

Vehicle	LO_2 Line Dia.	LH_2 Line Dia.
D-1S(R)	4.0 in. (10.2 cm)	4.75 in. (12.1 cm)
RLTC	5.2 in. (13.2 cm)	6.8 in. (17.3 cm)

The lines were sized for 250-second duration dump of the full propellant tanks until the start of two-phase flow in the dump lines at tank pressures of 35 psia (24.1 N/cm^2) and 24.5 psia (16.9 N/cm^2) for the LO_2 and LH_2 tanks, respectively. The orientation of the propellants during dumping (caused by the Orbiter turnaround maneuver) results in a maximum of 180 pounds (82 kg) of LH_2 and 60 pounds (27 kg) of LO_2 left in the tanks at dump termination.

Figure 2-24 shows the LH_2 fill and drain system, with an alternative valve and axial dump line shown penetrating the Orbiter engine compartment aft bulkhead heat shield. All propellant lines aft of the retractable umbilical panel are vacuum jacketed. The LO_2 system is similar.

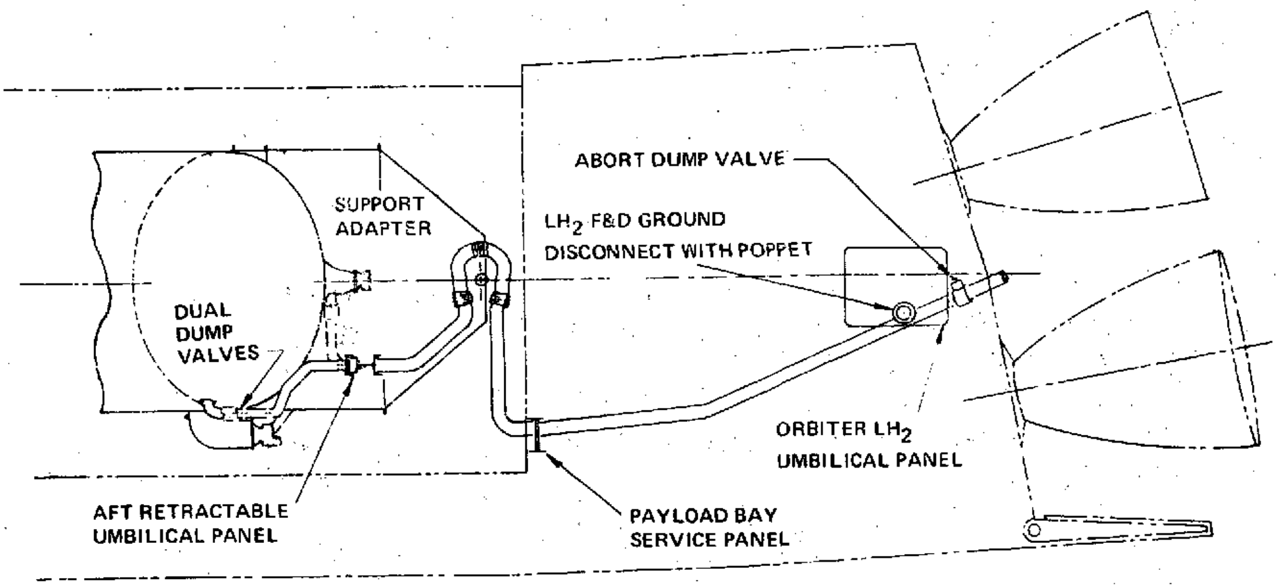


Figure 2-24. LH_2 Fill and Drain System

2.5 AUXILIARY PROPULSION SYSTEM

2.5.1 D-1S(R) AUXILIARY PROPULSION. The system consists of the D-1T H_2O_2 monopropellant supply to power the boost pump turbine and the reaction motors for attitude control, small velocity corrections, retromaneuvers and main propellant settling. An integral part of the system is the electric heater subsystem with thermostatic controls to maintain monopropellant temperatures within operational limits.

The Centaur D-1T operating hardware is retained (motors, valves, and supply bottles), with additions for manned Orbiter safety, redundancy controls, and remote tanking. These additions generally inhibit the APS system while the Centaur is in or near the vicinity of the Orbiter. Isolation and remote fill and drain provisions are added to the D-1T system. Remote monopropellant loading delays charging the system until just prior to main propellant tanking; this enables the system to be drained and safed while

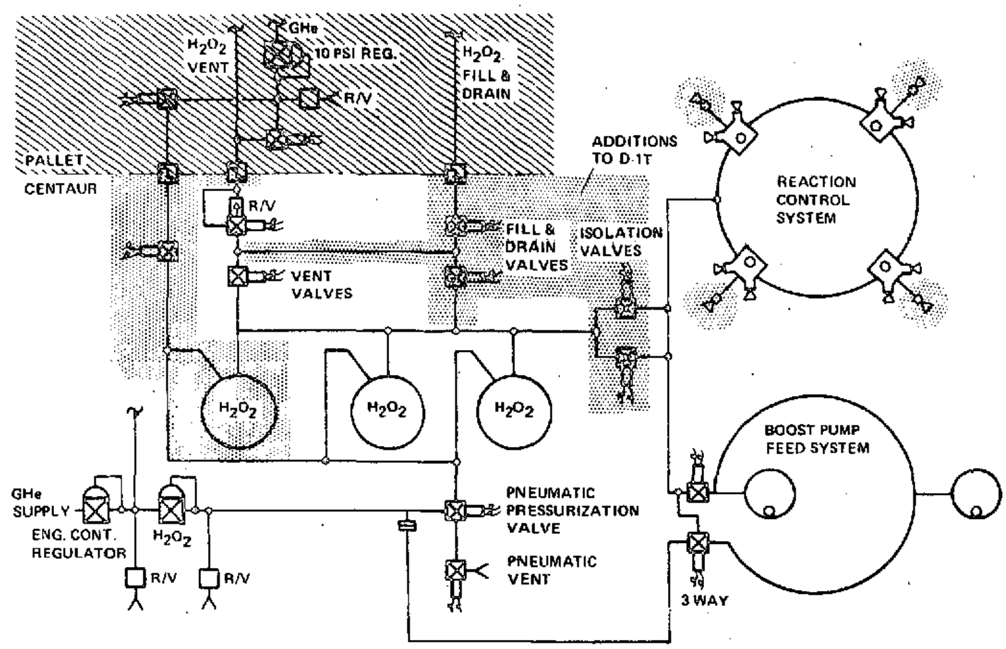


Figure 2-25. Auxiliary Propulsion System for D-1S(R)

the Centaur is still in the Orbiter and enables any reaction generated gases to be expelled safely through the inclosed Orbiter disconnect piping. The system is shown schematically in Figure 2-25.

Four retro motors (identical to the other motors, but forward facing) are added to provide retromaneuvers for Orbiter safety and payload placement. This necessitates the clusters to be relocated from the aft bulkhead outboard on tripod mounts on the latch skirt. This position reduces impingement and increases effective moment arm, for more effective propellant usage.

The APS supply has been sized to the maximum impulse anticipated (maximum payload delivered to synchronous orbit). Propellants can be offloaded for lighter payloads or less demanding missions if required. The motors currently being used on the Centaur are rated at 6 pound thrust, with 155 seconds I_{sp} under steady state operation and a nominal pulse mode attitude control I_{sp} of 120 seconds (Reference 1, Volume 5, Section 5.2).

Tests to date on the D-1T thrusters and H_2O_2 storage bottle indicate functional life reusability of at least eight to ten missions for D-1S(R) vehicles. Additional vibration testing to demonstrate fatigue life may be required.

Table 2-9 is a summary of the APS requirements for the D-1S(R).

Table 2-9. D-1S(R) APS Requirements Summary

Event	Impulse (lb-sec)		H_2O_2 (lb)	(kg)
ΔV	22,980		148	67
Settling	16,100		104	47
Attitude Control	34,000		284	129
Boost Pumps	7,600		49	22
	80,000		585	265
5% Contingency			29	(13)
Usable with contingency			614	(278)
2% Residual			12	(5)
(Capacity is 242 lb/bottle)		Total Tanked	626	(283)

2.5.2 RLTC AUXILIARY PROPULSION SYSTEM. The system consists of the hydrazine monopropellant supply to power the boost pump turbines and the reaction motors for attitude control, small velocity corrections, retromaneuvers, and main propellant settling. An integral part of the system is circulating hydrazine through a heat exchanger to use fuel cell waste heat to maintain monopropellant temperatures within operational limits. System design changes from D-1T are:

a. Change monopropellant to hydrazine.

Reason: Reduce operational complexity (by using only one propellant) and increase performance.

b. Increase reaction motor thrust to 10 pounds thrust.

Reason: Use modified rocket Research MR-50A. Life exceeds 20 missions.

c. Add isolation valves and overboard vent through Orbiter disconnects.

Reason: Orbiter safing.

d. Add cluster isolation valves.

Reason: Orbiter safety, enable shutoff of "failed on" motor.

e. Add hydrazine circulating system.

Reason: Provide system thermal control and reduce electrical power requirements.

The system is new to Centaur, but consists of components proven on other programs. The system operates at ambient temperature using 450 psi helium pressure to force hydrazine from the two positive expulsion bottles. The bottles feed the two boost pumps and the four clusters of four thruster modules as their individual valves open. Each cluster contains four 10-pound force thrusters and a cluster isolation valve for fail operational/fail safe operations. The Centaur is capable of continuing to function satisfactorily with one cluster out. The system is shown schematically in Figure 2-26.

The APS supply has been sized to the maximum impulse anticipated to deliver a maximum payload to synchronous orbit. Propellants can be offloaded for lighter payloads or less demanding missions. The very high impulse required for vehicle velocity changes dominate the system requirements (Reference 1, Volume 5, page 225).

Table 2-10 is a summary of the APS requirements for the D-1S(R).

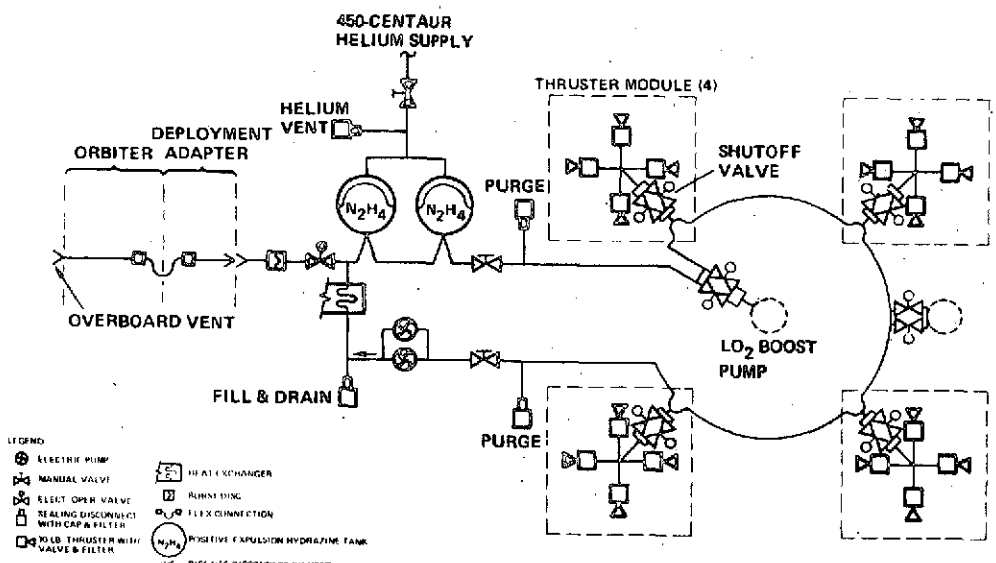


Figure 2-26. Stored Propellants Vented External to Orbiter

2.6 THERMAL CONTROL

Thermal control systems are provided for both D-1S(R) and RLTC to maintain the various vehicle systems and components within acceptable temperature limits during all phases of the mission (Reference 1, Volume 5, Section 5.2). The thermal control systems are both active and passive. Electric line heaters and blankets provide active control for the APS system and for the boost pump catalyst beds. An electric

Table 2-10. RLTC APS Requirements Summary

Event	Velocity (fps)	Impulse (lb-sec)	N_2H_4 Propellant (lb)	(kg)	
ΔV					
Separation From Orbiter	10	19,750	86	(39)	
Midcourse Correction	50	59,400	258	(117)	
Payload Separation	10	7,100	31	(14)	
Midcourse Correction	50	15,700	68	(31)	
Terminal Rendezvous	15	3,170	14	(6)	
Docking With Orbiter	10	2,120	9	(4)	
	145	107,240	466	(211)	72%
Propellant Settling		14,800	65	(30)	10%
Attitude Control		15,740	75	(34)	12%
Boost Pumps		8,740	38	(17)	6%
		154,150	644	(292)	
5% Contingency			33	(15)	
Usable with contingency			677	(307)	
2% Residual			13	(6)	
(Capacity is 419 lb/bottle)		Total Tanked	690	(313)	

heater in the hydraulic system reservoir provides thermal control for the TVC. Passive thermal control is provided by insulation blankets on the main propellant tanks, radiation shielding on TVC components and the main propellant tanks, thermal paints and finishes, and the vacuum environment provided between the propellant tanks by the intermediate bulkhead. For RLTC only, the fuel cell waste heat exchanger provides thermal control of the APS N_2H_4 and the pressurization system helium.

2.6.1 TANK INSULATION SYSTEM. The main propellant tank insulation system is based on designs developed for Centaur D-1T. The forward hydrogen tank bulkhead and the aft oxygen tank bulkheads use the Centaur D-1T insulation blanket concept, but the material has been changed to meet fire-resistant safety requirements, and the metallized surface has been changed to improve reusability. On the tank sidewalls, the D-1T type radiation shield concept is augmented by addition of insulation blankets to provide thermal control compatible with Shuttle requirements. The insulation is purged with helium during ground operations and orbiter reentry to prevent condensation of air or nitrogen from degrading the thermal properties of the insulation. The insulation blanket thickness is designed to insure that the payload bay environment is not chilled excessively during pre-launch operations.

The basic elements of the insulation system are:

a. Forward bulkhead insulation.

b. Tank sidewall radiation shield and insulation.

c. Intermediate bulkhead insulation.

d. Aft bulkhead radiation shield and membrane.

The functions of the insulation system are:

a. Insulate the propellant tanks during ground and Shuttle ascent operations.

b. Minimize heat transfer between the propellant tanks.

c. Provide radiation protection during space operations.

d. Protect equipment module components from excessive cooling.

e. Provide propellant tank meteoroid protection.

f. Provide separate containment of the propellant tanks to permit capture and venting of any propellant leakage.

Table 2-11 gives D-1S(R) and RLTC heating rates for the main propellant tanks, and Table 2-8 gives deep space vent rates for both vehicles. Table 2-12 gives insulation blanket performance data for various flight conditions.

Table 2-11. Propellant Tank Heating Summary (Nominal Values)

Flight Condition	D-1S(R) Btu/hr (watts)		RLTC Btu/hr (watts)	
	LH_2	LO_2	LH_2	LO_2
Closed Payload Bay, Prelaunch	115,645 (33,872)	43,346 (12,696)	157,000 (45,985)	62,300 (18,248)
Closed Payload Bay, 0-480 sec	40,744 (11,934)	14,801 (4,335)	59,488 (17,434)	21,190 (6,207)
Open Payload Bay, 480 sec to low altitude orbit	4,258 (1,247)	3,607 (1,056)	4,877 (1,428)	3,689 (1,081)
Low Altitude Orbit	2,326 (681)	692 (203)	2,876 (816)	712 (209)
Deep Space, Continuous	2,178 (638)	573 (168)	2,634 (771)	592 (173)

Table 2-12. Tank Sidewall Insulation Blanket Performance

Flight Condition	LH_2 Tank Heat Flux Btu/hr ft^2 (watts/m^2)	LO_2 Tank Heat Flux Btu/hr ft^2 (watts/m^2)
Prelaunch	148 (466)	127 (400)
Launch + 10 Hours	0.132 (0.416)	0.134 (0.422)
Deep Space (Equilibrium)	0.143 (0.450)	0.121 (0.381)

2.6.2 FORWARD BULKHEAD INSULATION. Insulation covers the forward tank bulkhead surface from just forward of the tank forward ring to the access door mounting flange. This assembly consists of two separate blankets each approximately 3/4-inch (1.9 cm) thick laid one on top of the other to produce a total assembly thickness of 1-1/2 inches (3.8 cm) and covered with a gas containment membrane made of plain Kapton sheets bonded to both sides of Beta glass scrim. Each blanket consists of 11 non-metallized, dimpled sheets of Kapton, 1/2 mil (0.0013 cm) thick and 10 goldized, flat sheets of Kapton, 1/4 mil (0.00064 cm) thick. The outer layers of the blanket are made of one flat sheet of Kapton bonded to Beta glass scrim. The dimpled and flat intermediate layers alternate, and the dimpling produces a separating gap of 0.06 inch (0.152 cm). The forward bulkhead insulation (blanket assembly and containment membrane) is installed on the forward adapter before the adapter is installed on the tank. Figure 2-27 shows the insulation blanket as installed on the forward bulkhead of a D-1T vehicle.

2.6.3 TANK SIDEWALL INSULATION AND RADIATION SHIELD. Two 3/4-inch (1.9-cm) thick insulation blankets and a three-layer radiation shield cover the entire sidewall of the LH_2 tank, and for RLTC, the cylindrical section of the LO_2 tank. Figure 2-28 shows the RLTC insulation. D-1S(R) tank insulation is similar.

The external radiation shield for both D-1S(R) and RLTC is similar to the D-1T Centaur radiation shield. The shield is comprised of three layers of a goldized Kapton Beta glass scrim. The shield outer layer is of similar construction except that one of the Kapton sheets is plain, not goldized. The three layers are laid one on top of the other to form a total shield. The plain Kapton surface of the outer layer is the outside of the shield. The outer and middle layers have 1/4-inch (0.64 cm) 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 and forms a containment membrane.

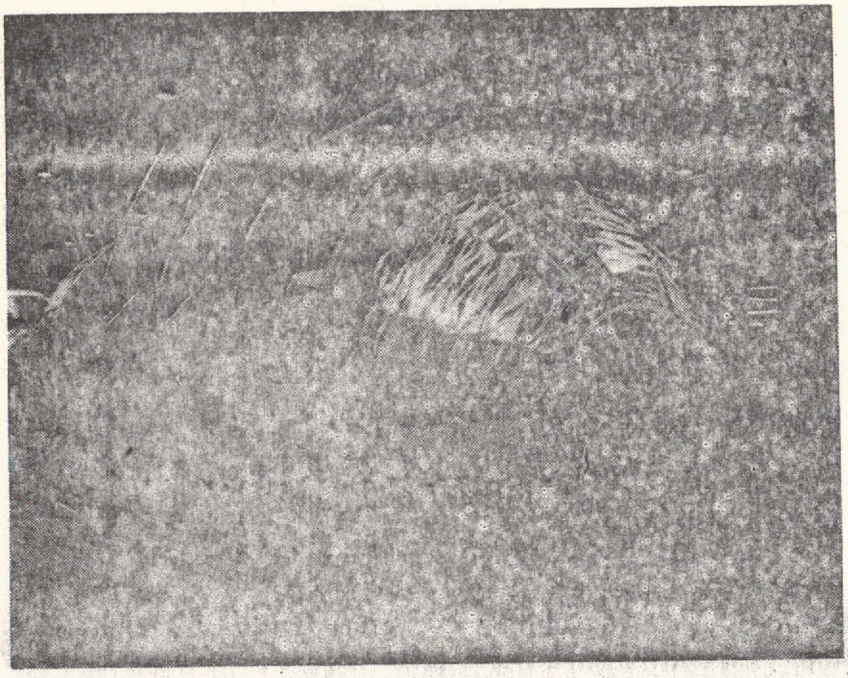
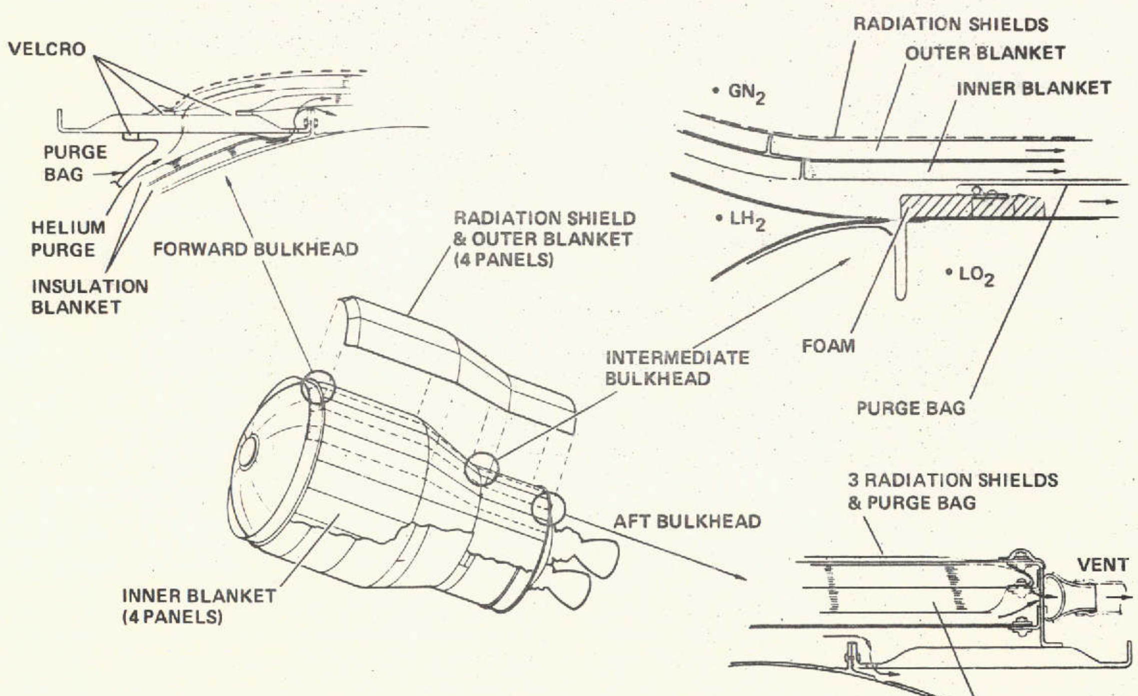


Figure 2-27. D-1T Insulation Blanket



• INSULATION BLANKET & RADIATION SHIELD SYSTEMS DERIVED FROM CENTAUR D-1T TECHNOLOGY

• PURGE BAGS INDIVIDUALLY ISOLATE PROPELLANT TANKS

Figure 2-28. RLTC Tank Insulation

To complete the separate containment of LH_2 and LO_2 gases, an additional LO_2 tank membrane seal is placed over the entire cylindrical section (for RLTC) of the LO_2 tank and portion of aft adapter. An annular foam pad, with a Kapton tape seal, covers the tank and provides a separation seal of LO_2 and LH_2 gas leakage that may occur as a result of a leak developing in either, or both, main propellant tanks. This pad also provides a forward mounting surface for the LO_2 tank membrane seal and helium purge tube.

2.6.4 INTERMEDIATE BULKHEAD INSULATION. This insulation is an integral part of the intermediate bulkhead construction. This bulkhead is a double-walled, ellipsoidal structure separated by plastic mesh and fiberglass mat insulation, as shown in Figure 2-7. The volume is 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.

2.6.5 AFT BULKHEAD RADIATION SHIELD AND MEMBRANE. The radiation shield on the LO_2 tank aft bulkhead, similar to that on D-1T, is a rigid assembly made of laminated nylon fabric with goldized Kapton on its inner surface and white polyvinyl fluoride on its outer surface. 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 1 inch (2.4 cm) from the tank bulkhead surface. A membrane of goldized Kapton (Beta glass laminate identical to the sidewall radiation shield material) is used 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. This "purge bag," mounted inside the rigid shield, is sealed with tape around the penetrations, like the existing Mylar membrane.

2.6.6 INSULATION PURGE. Both D-1S(R) and RLTC groundhold insulation blankets on the LH_2 tank forward bulkheads and sidewall are purged with helium during prelaunch tanking, and repressurized with helium during Orbiter reentry. This prevents condensation of air or nitrogen within the blanket from degrading the thermal properties of the insulation. Purge helium is supplied from a ring of perforated distribution tubing beneath the containment membrane on the forward bulkhead.

During ground tanking, a collection manifold at the base of the aft fairing retains the blanket purge and ducts it overboard. At liftoff, the purge is stopped and the manifold is permitted to vent into the payload bay during Shuttle ascent, to prevent the buildup of bursting pressure beneath the blanket. The bay contains nitrogen which in turn vents overboard.

Upon Orbiter reentry, the purge control valve in the vent duct is placed in the ground hold position, and the purge reestablished from the pallet (deployment adapter) supply. The purge is reestablished above 110,000 feet altitude (34 km) to prevent formation of a flammable mixture of potential hydrogen leakage and the warm air used to repressurize the Orbiter payload bay. The insulation blanket helium purge is continued until

30 minutes after landing when GSE assumes the responsibility for payload bay conditioning. During vehicle turnaround, the tank sidewall blankets and radiation shields may be removed and stored to protect the insulation from moisture or handling damage and provide accessibility to the tank skin for leak tests and inspection for meteoroid damage.

2.6.7 GOLDIZED KAPTON. Kapton is used in place of mylar because Kapton is more fire resistant. Goldized Kapton is used rather than aluminized Kapton because of the degrading effect moisture has on the aluminized surface over a period of time. Various NASA and NASA contractor studies of multilayer insulation systems have verified this degradation. General Dynamics/Convair has firsthand experience with this problem on its D-1 Centaur vehicles. The use of goldized material for only the flat sheets in the insulation buildup makes the system even more cost competitive with the aluminized system. The nonmetallized dimpled spacer material causes no appreciable increase in total vehicle heat transfer.

2.6.8 USE OF INSULATION BLANKETS AS A METEROID BARRIER. A Convair study, and extrapolation of Boeing data from NASA Reports CR121103 and CR121104, indicate that the insulation blankets will provide adequate meteoroid protection for D-1S(R) and RLTC tanks within the NASA minimum requirement of providing a probability greater than 0.995 of no meteoroid penetration.

2.7 ORBITER/CENTAUR INTERFACE

The Centaur with adapters is designed to use the Orbiter four-point support technique. The D-1S(R) has a flight pallet configuration as an intermediary structure to adapt from the Orbiter to the Centaur (Figure 2-29). The RLTC deployment adapter contains three support points with the fourth or forward Z support located on the forward adapter (Figure 2-30).

The D-1S(R) Centaur flight pallet (CFP) supports Centaur in the payload bay by tying into the four Shuttle payload bay support points and transmitting the loads through a reusable aft deployment adapter (which stays with the pallet subsequent to Centaur deployment) and the Centaur forward adapter ring. The CFP provides a variety of functions for Centaur, besides structurally supporting it in the payload bay. The CFP includes Centaur in-orbit support equipment, deployment and separation mechanisms, and separation disconnects, allowing complete testing of these functions on the ground before installation in the Orbiter. Hard connections are used between the Orbiter service panels and the CFP, obviating the need for disengagement testing with the Orbiter.

The support adapter includes structure for Centaur engine support (primarily for ground handling and landing) and the 16 separation latches. It also includes the aft retractable umbilical panel and the flex lines and propellant gimbal joints connecting the support adapter with the pallet truss.

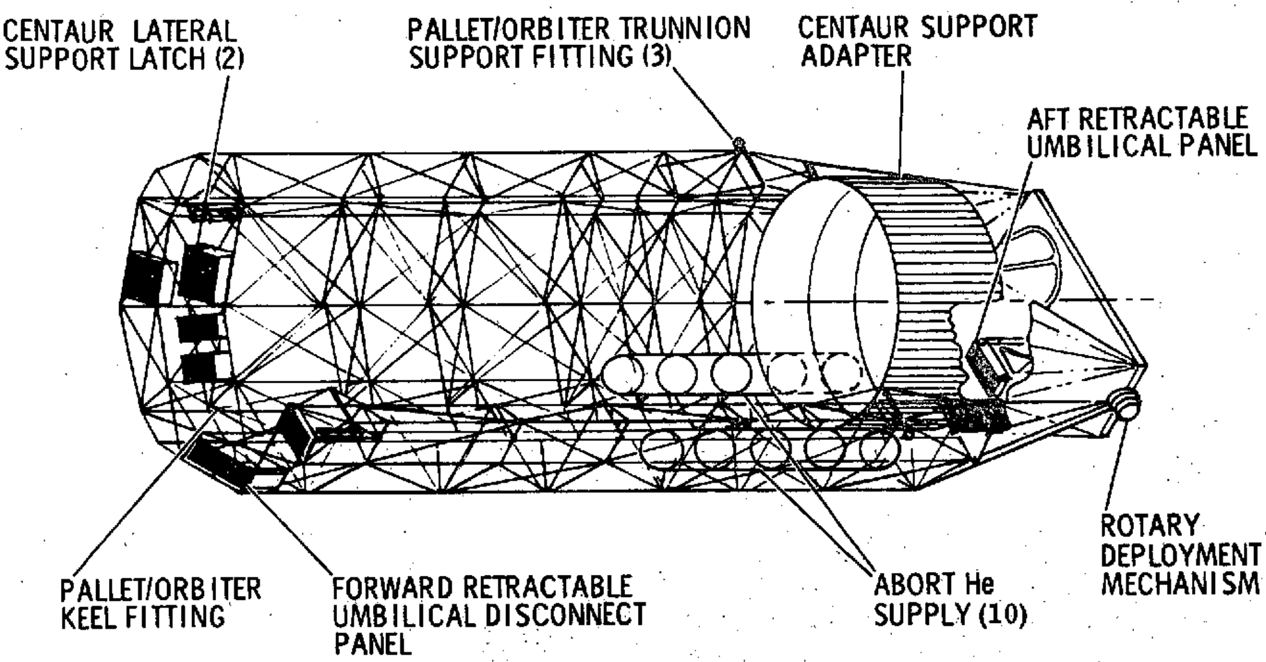


Figure 2-29. Centaur Flight Pallet Configuration

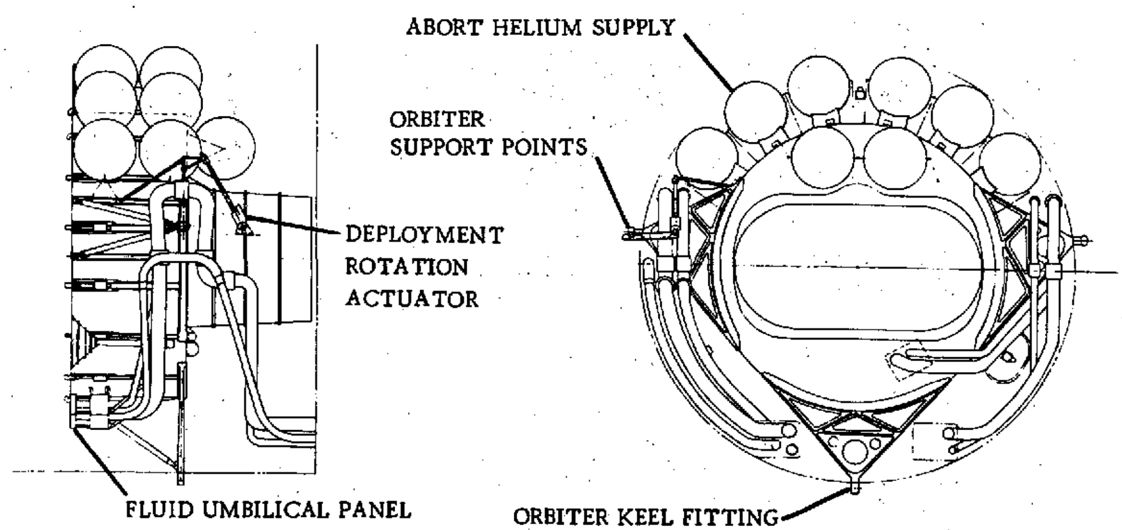


Figure 2-30. RLTC Deployment Adapter

The pallet truss provides supporting structure for the Centaur forward retractable umbilical panel and three pallet-to-Orbiter/ground "hardline" panels plus the line routing between them. The pallet also supports the emergency helium supply and pneumatic control panel for propellant dumping during Orbiter abort. Included in the truss are two forward lateral support latches, two aft lateral support latches, two rotary deployment actuators, Centaur/deployment adapter air conditioning fans, and Centaur stretch actuators. Also mounted on the truss is the pallet electrical control and power equipment, electrical Centaur/Shuttle interface equipment. Figure 2-31 is a schematic of the CFP fluid systems.

The RLTC interface consists of a deployment adapter and the astrionics interface equipment. The deployment adapter has the two aft X and Z direction supports which also serve as deployment rotation points and the single Y direction support point.

The forward Z load attaches directly to the RLTC forward adapter at Shuttle Station 947.5. All fluid and electrical services for the RLTC are routed aft on the vehicle and interface with the deployment adapter eliminating any forward umbilical disconnects.

The fluid interface between the Centaur and the Orbiter are at the aft bulkhead of the payload bay. Flight disconnect of the Centaur lines occurs in the support adapter, independent of Orbiter deflections.

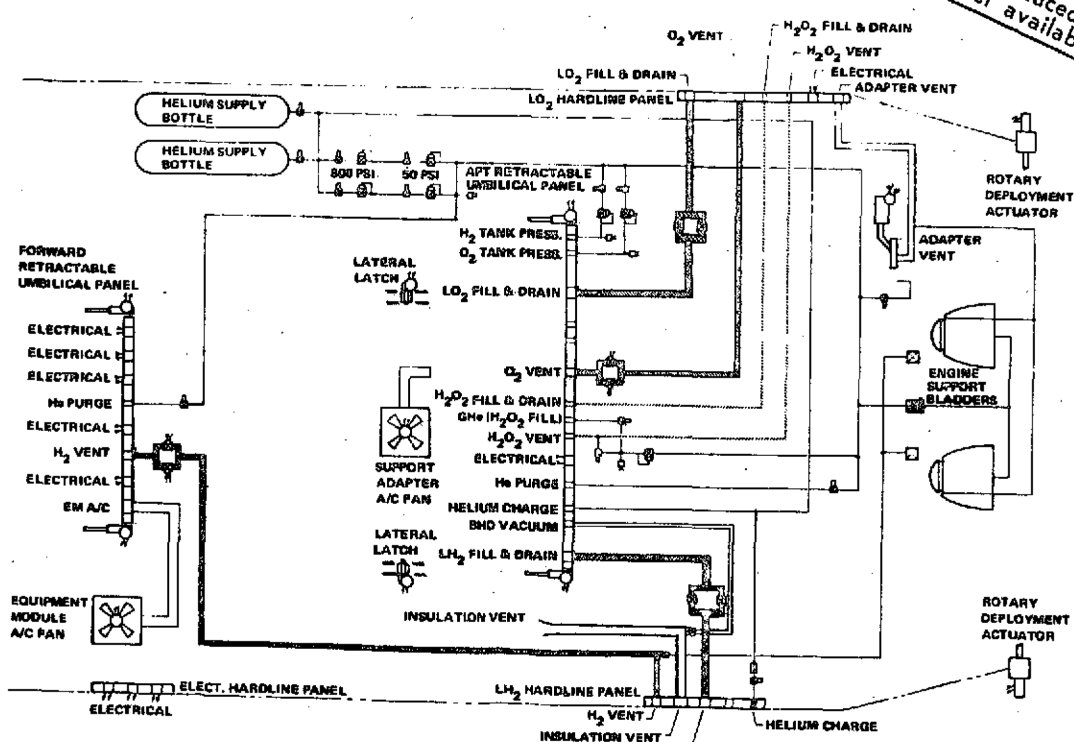


Figure 2-31. Pallet Schematic

SECTION 3

ASTRIONIC SYSTEMS

Evolution of the astrionic subsystems for the Reusable Centaur was based on the current operational D-1 Centaur. A recent redesign of the D-1A/D-1T astrionics has a large capacity central digital computer, reprogrammable, which provides an excellent starting point for increased capability upper stages. Verification of the improved astrionic and vehicle design has been accomplished with three operational flights during 1973.

Where possible, consistent with the mission and reusability objectives, the existing Centaur designs for electronic equipment and the same type of electrical interface are used for the Reusable Centaur to minimize risk and to avoid unnecessary development costs. Space Tug missions require more time than the usual D-1 flights (up to 7 hours); this increases the importance of the reliability of the design so it can meet the mission success goal of 0.97. For the longer RLTC 47-hour nominal mission, dual redundancy (and sometimes triple) is employed throughout the astrionic control and management function.

3.1 REQUIREMENTS

The prime driving requirements for the Reusable Centaur astrionic designs are shown in Figure 3-1. They are defined as follows:

Reusability — Changes to the stage design necessary to make the Centaur reusable. This focuses on new parts and redesign; existing Centaur parts have been shown to be inherently reusable.

Mission Reliability — Mission success goal for the total vehicle is 0.97. Existing single string designs have been examined, and for the longer missions redundancy will be needed.

Mission Duration — This parameter is sensitive in the establishing of electrical power, guidance accuracies, reliability, and software requirements.

Shuttle Integration — Man-rated safety is provided through monitoring, communications, control and arm-safe functions on board the Orbiter. The Centaur monitor and control system (CMACS) with pallet/adapter establishes the man-machine interface. Centaur redundancy is also provided in safety critical areas.

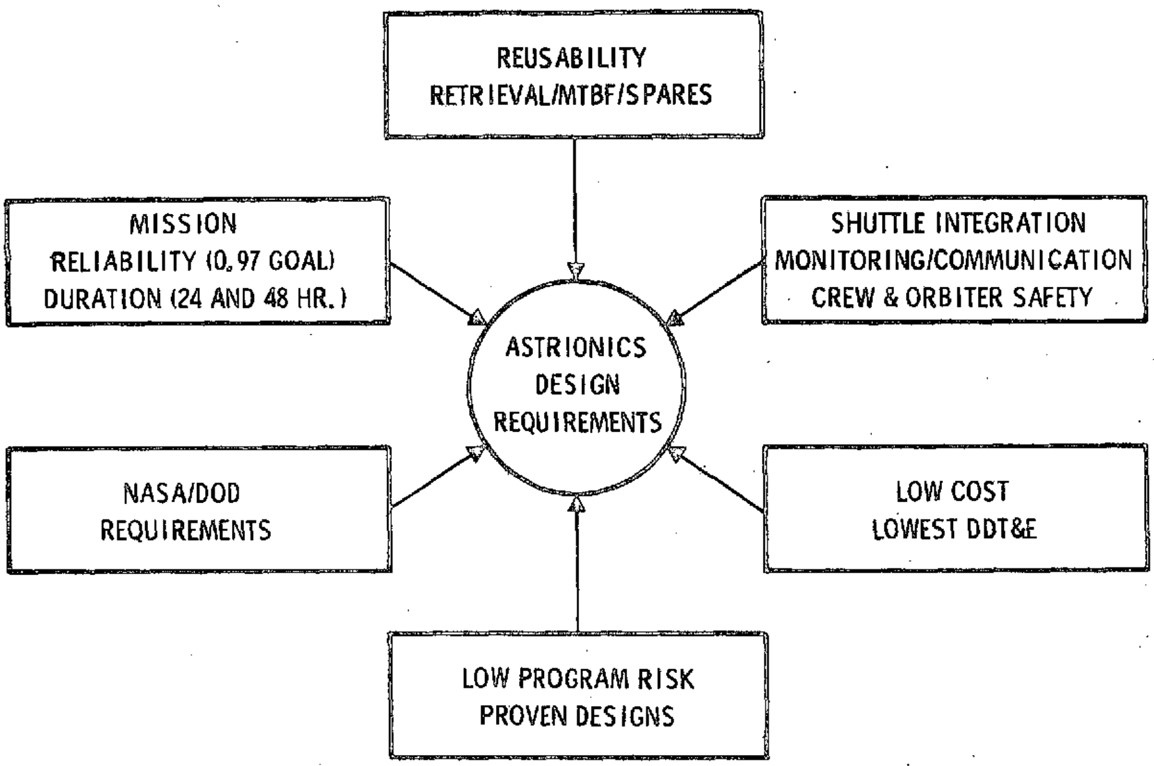


Figure 3-1. Reusable Centaur Requirements

Low Cost/Low Program Risk — This has been approached by the selection of proven and current Centaur electronic designs as applicable.

NASA/DoD Requirements — Most of these are incorporated in the overall study requirements. Communications vary between NASA and DoD because of different ground support networks, allocated frequencies, and cryptographic provisions.

Table 3-1 presents the specific astrionics design requirements for Reusable Centaur and their implementation for D-1S(R) and RLTC.

3.2 D-1S(R) ASTRIONICS

Existing tank configuration Centaurs were used in the recent detailed study of an expendable cryogenic upper stage (D-1S) performed for the NASA Lewis Research Center, contract NAS 3-16786, during 1972 and 1973. Returning the Centaur stage to the Shuttle Orbiter requires changes to the astrionics of this expendable D-1S (which is

Table 3-1. Astrionics Design Requirements

New Design Requirements	Implementation	
	D-1S(R)	RLTC
Increased Guidance & Navigation	Guidance update	Guidance update
Increased Electrical Power	Lightweight batteries	Fuel cell & backup battery
Increased Data Management	5k-word added software, existing computer	33k-word added software, triple computers
Revised Communications	USB/SGLS crypto	USB/SGLS crypto
Autonomy	Level III	Level I
Maximum Use of Centaur & Existing Designs	D-1T plus additions	Revised D-1T & additions
Interface Provisions	CMACS and pallet	CMACS & deployment adpter
Reliability	24-hr. mission (0.981) single string	47-hr. mission (0.995) dual redundancy
Safety	Arm-safe & dual communications attitude control backup	Redundancy & arm-safe
*Total Vehicle (0.97)	(0.968)	(0.975)

a minimum revision to the D-1T Centaur sufficient for integration with the Orbiter). The following astrionic areas are impacted:

a. Flight software

b. Increased computer main memory

c. Revised communications-

    1. Dual unified S-band (USB) communications (NASA)

    2. Dual SGLS communications (DoD), interchangeable

d. New additions:

1. Sun and horizon sensors for attitude update

2. Tape recorder

3. Docking attitude control for safety backup

Mission times under consideration are shorter than for RLTC (nominal 24 hours versus 47 hours); by carefully selecting components this permits the realistic examination of a simplex set of astrionics. During the period when the upper stage is within the Orbiter bay, redundancy and arm-safe switching are required to prevent potential safety hazards. Statusing of the Reusable Centaur must be done before any attempt to return the stage into the Orbiter payload bay.

Figure 3-2 is the astrionics system block diagram for the D-1S(R). The additions and deletions to the Centaur D-1T are noted. Subsystem grouping is the same as for the RLTC and for the Space Tug System Study (Cryogenic), NAS8-29676. The astrionic subsystems are:

a. Data management system (DMS)

b. Guidance, navigation and control (GN&C)

1. Flight control

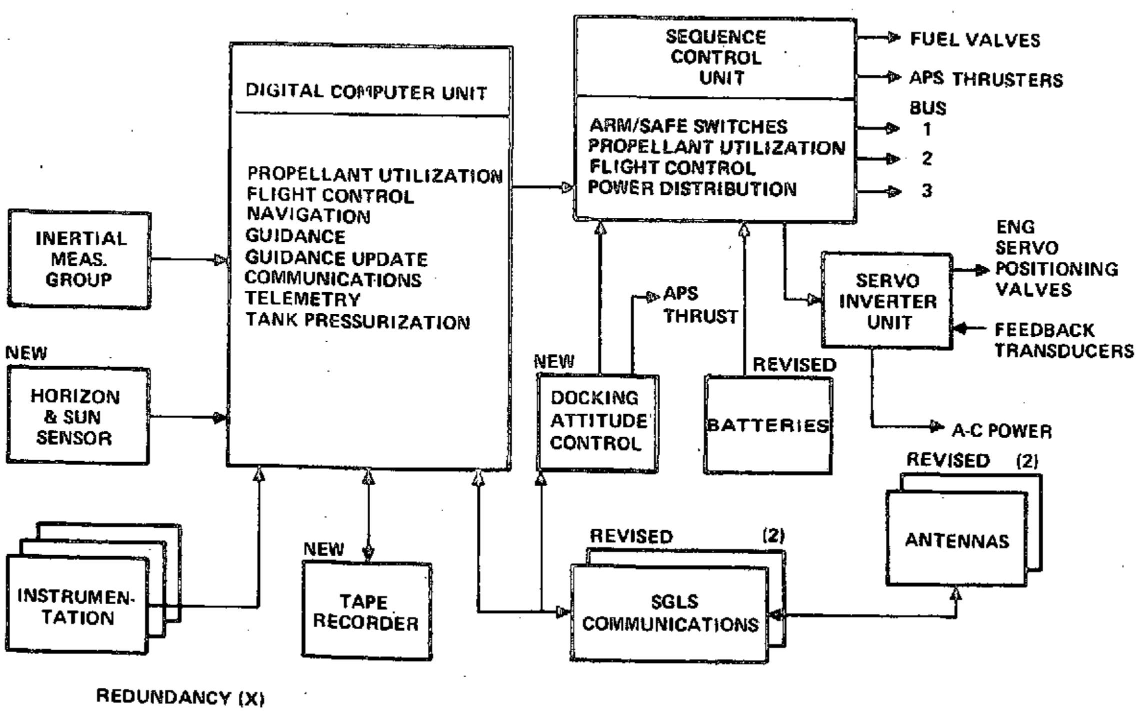


Figure 3-2. Astrionics System - D-1S(R)

2. Guidance and navigation
3. Guidance update
4. (Rendezvous and docking are deleted)

c. Communications

d. Instrumentation

e. Electrical power

f. Power conversion and distribution

Existing D-1T interfaces between subsystems have been retained along with the same input/output structure for the digital computer. Shuttle Orbiter interfaces do not require significant change of the present Centaur telemetry and GSE data and electrical interface. Provision has been made for additional essential supporting equipment in the Orbiter in the form of a pallet/deployment adapter and the CMACS.

Significant changes to the GN&C subsystem are the inclusion of an attitude update capability to support the ground tracking position and velocity data, and the provision of a safety backup system for docking attitude control during the retrieval process.

The normal Centaur arm-safe switching has been modified to ensure man-rated safety. This requires changes in the sequence control unit and in the flight control sequence.

Communications have been completely modified. For NASA missions, dual USB equipment will be used which is compatible with the Space Tracking and Data Network (STDN), the Orbiter, and the planned Tracking and Data Relay Satellite (TDRS). DoD missions will require the interchanging of space-ground link system (SGLS) modules. These are identical with the communications defined for the RLTC and for the STSS Study (Program 1). Instrumentation is similar to the expendable D-1S Centaur as described in the NAS3-16786 study results. Electrical power is the same as the D-1T Centaur except that lightweight batteries are included to satisfy operation over the extended mission times.

The adaptation of Centaur with minimum change and minimum new development continues the Level III type of autonomy with dependence on ground support of the flights. The RF uplink capability to provide update data and/or commands is a new function added to D-1S(R).

3.3 RLTC ASTRIONICS

Figure 3-3 is the functional system diagram for the RLTC astrionics.

Central computer data management using a digital computer has been adopted from the D-1T Centaur, and this has been made triply redundant for improved reliability

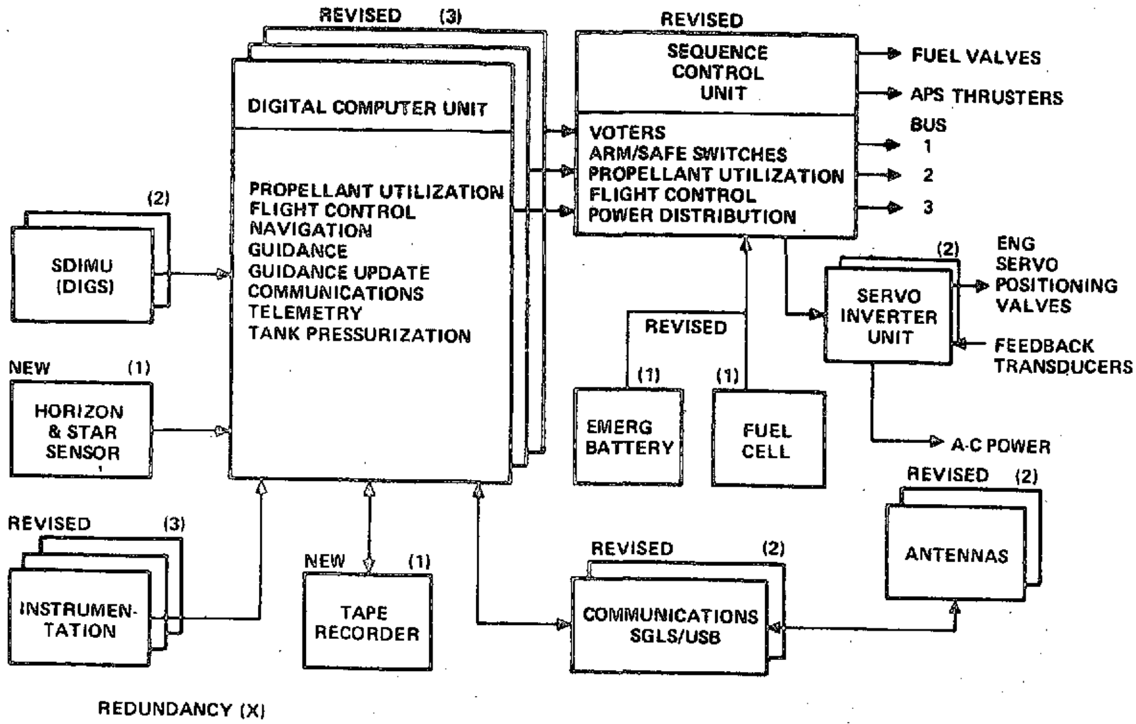


Figure 3-3. Astrionics System - RLTC

to satisfy 47-hour mission success goals. Voting of the three computers avoids the complexity of the software program logic that would be required with only dual computers. The "masking" and coverage limitations (typically 0.9) plus many of the involved problems of redundancy management within the computer are thereby avoided. A tape recorder was added to the data management system to keep flight history data of subsystems as required, particularly the engine burn history, plus enable delayed uplink commands and down telemetry where ground station contact availability may be limited at the time of event occurrence.

Other astrionic subsystems are dual redundant with minor exceptions in areas where the reliability is adequate and critical safety is not a factor.

In the GN&C system the inertial reference unit is a redundant design currently under NASA review as a D-1 Centaur improvement; it consists of two delta inertial guidance system (DIGS) strapdown gyro platforms with skewed axes, a total of six gyros. This design provides the equivalent of triple redundant platforms and can sustain two gyro failures and still fail operational.

Communications is a dual system providing two-way S-band links to ensure uninterrupted data and commands flow to and from the Tug. Interchangeable modules are available for either NASA or DoD mission requirements. Encryption and decryption are employed as required for DoD or special missions.

Instrumentation is based upon the existing D-1T Centaur transducers and signal conditioners, modified as needed to add redundant functions, to accommodate the revised missions, and to provide man-rated safety.

For the electrical power subsystem, a lightweight fuel cell design was adopted which uses the gaseous boiloff from the main cryogenic fuel tanks as the source of hydrogen and oxygen.

A guidance update function has been provided to limit the guidance dispersions for the longer mission times and to furnish independence from ground tracking support requirements.

Level I autonomy has been selected on the basis of reduced overall program costs, which are primarily in the operations support area. Additional software is required for onboard checkout and sequencing, for redundancy management, and for the increased guidance and navigation functions for the extended Tug missions (relative to D-1T Centaur), and for the onboard guidance update.

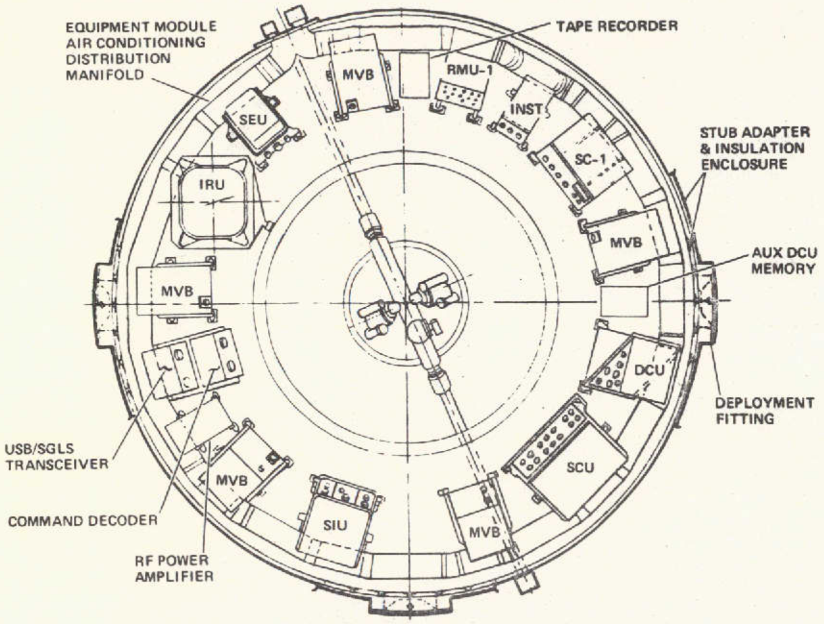
3.4 D-1S(R) ASTRIONIC SUBSYSTEMS

Astrionic subsystems for the D-1S(R) as a group contain identical equipment to the D-1T Centaur plus the necessary additions for reusability. The electronic components for these subsystems are for the most part mounted to the equipment module ring located on the forward end of the Centaur vehicle as shown in Figure 3-4.

3.4.1 DATA MANAGEMENT SYSTEM. Equipment identified for the data management system (DMS) are listed in Table 3-2. These units provide the required data

Table 3-2. DMS Equipment

	Usage	Wt (lb)	Power (W)
Digital Computer Unit (DCU)	D-1T modified	70 (31.75 kg)	190
Remote Multiplexers (2)	D-1T as is	40 (18.14 kg)	25 avg
Tape Recorder	Agena	12 (5.44 kg)	10 avg

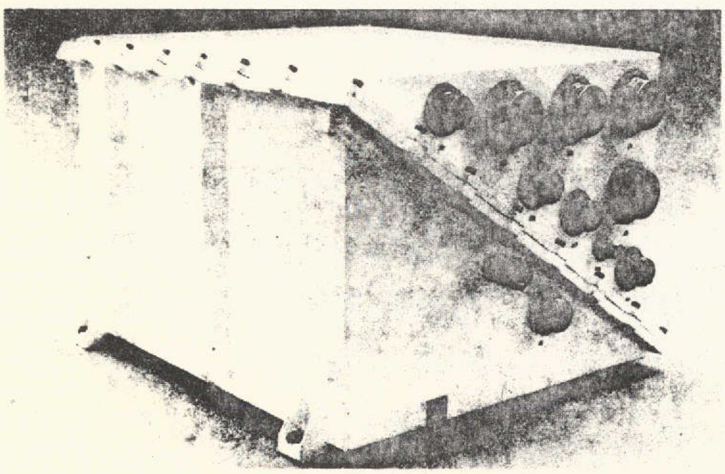


DCU	Digital Computer Unit
SCU	Sequence Control Unit
MVB	Main Vehicle Battery
SIU	Servo Inverter Unit
IRU	Inertial Reference Unit
SEU	Systems Electronic Unit
RMU-1-2	Remote Multiplexer Units
INST	Instrumentation Box

Figure 3-4. Astrionics Equipment Module - D-1S(R)

processing, reporting, and control functions for the D-1S(R) vehicle and are designed to function in a system having man-rated safety.

3.4.1.1 Digital Computer Unit. The DCU (Figure 3-5) is a stored program, random access core machine. Memory is composed of 24,576 words of 24 bits each. The



Weight: 70 lb (31.8 kg)

Size: 16 × 14 × 11 in.
(40.5 × 35.4 × 27.9 cm)

Power: 190 watts (28 Vdc)

Figure 3-5. Digital Computer Unit

standard Teledyne TDY-300 internal memory 16,000 words has been modified by 8,000 added core words to comfortably accommodate the estimated 17,200-word D-1S(R) software requirement. Memory cycle time is nominally three microseconds.

Instruction Set. The DCU has a set of 25 hardware instructions in addition to input/output instructions.

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

Interrupts. The DCU has five interrupt channels. The priority of processing them 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 3-3.

3.4.1.2 Flight Software. A software analysis and estimate was made at the subsystem level based upon the D-1T Centaur experience at General Dynamics in developing, designing, and verifying the software for NASA, plus the various analyses in related Space Tug studies as performed by General Dynamics and Rockwell International. Table 3-4 presents the output of this estimate.

Table 3-3. DCU Input/Output Capabilities

Inputs	Comments
Serial - 24 Bits	GSE uplink
Parallel - 12 Bits	28 Vdc (spare capability)
Discretes - 8 Channels	28 Vdc testable (SKD's)
Incremental - 4 Channels	ΔV's and real time
Analog - 12 Channels	11 bits plus sign, $\pm$12 Vdc
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, $\pm$5 Vdc
AC Analog - 6 Channels	11 bits plus sign, 3.5 Vrms
Telemetry - 2 Channels	1 to transmitter, 1 to downlink

Table 3-4. D-1S(R) Flight Software

Module	D-1T Ref. (words)	D-1S(R) (words)
Executive	2,388	2,500
Attitude Control	1,439	1,500
Communications	1,398	5,000
Guidance	2,120	2,200
Guidance Update	-	-
Navigation	369	400
On-board Checkout	-	500
Sequencing	2,600	2,800
Thrust Vector Control	1,205	1,200
Titan Booster Steering	464	-
	11,983	17,200

3.4.1.3 Tape Recorder. A flight tape recorder unit is provided to temporarily store vehicle status data to be transmitted to the ground. It is also available to supplement the computer main memory to store data needed only at preplanned times during a mission. The initial selection is the Lockheed Electronics MTR 2500, which is being developed for NASA-MSFC use on the High Energy Astronomical Observatory program. This recorder is capable of recording and reproduction at a rate of 51,250 bits per second. Total data storage is 4.15×10^8 bits for a maximum tape time of 270 minutes.

3.4.1.4 Remote Multiplexer Units. Inputs to the DCU from the instrumentation transducers are passed through signal conditioners as required and are then tailored by the fore and aft remote multiplexers to the current digital format. The signal conditioners convert the signal and measurement voltages to ranges compatible with the remote multiplexer unit (RMU) acceptance limits.

3.4.2 GUIDANCE, NAVIGATION, AND CONTROL. Included in the GN&C subsystem are the functions of:

Guidance and navigation

Guidance update

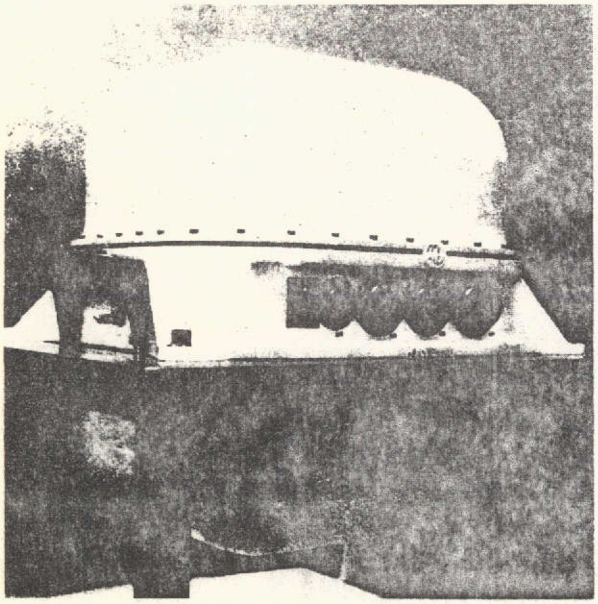
Flight control

Table 3-5 lists the GN&C equipment for the D-1S(R).

Table 3-5. GN&C Equipment

	Usage	Wt (lb)	Power (W)
Inertial Reference Unit	D-1T as is	64 (29.03 kg)	-
System Electronic Unit	D-1T as is	25 (11.34 kg)	185
Horizon Sensor (Barnes 13-161)	X-15	10 (4.54 kg)	5 avg
Sun Sensor (Adcole 1402)	NASA	4 (1.81 kg)	2 avg
Sequence Control Unit	D-1T modify	75 (34.02 kg)	17 avg
Servo Inverter Unit	D-1T as is	45 (20.41 kg)	40 avg
Docking Attitude Control	New	40 (18.14 kg)	5 avg

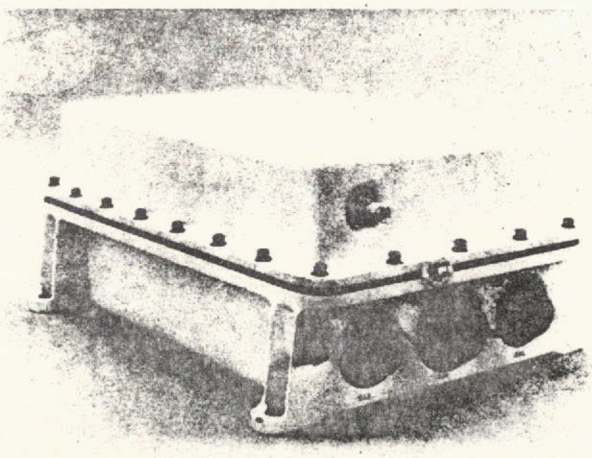
3.4.2.1 Guidance and Navigation. The Centaur inertial measurement group (IMG) consists of the inertial reference unit (IRU) and the system electronic unit. The IMG measures acceleration and provides a time reference for the digital computer unit (DCU) to make the navigation computations. The inertial reference platform is the inner gimbal of a four-gimbal assembly and has unlimited three degrees of freedom movement. Stabilization is achieved with three orthogonally mounted single-degree-of-freedom gyros which drive the gimbals through resolvers and torque motors. Three accelerometers mounted on the platform sense acceleration and generate output pulses at a rate proportional to the sensed accelerations. A crystal oscillator in the IRU serves as the navigation function primary timing reference. All power is supplied to the IRU by the systems electronic unit (SEU) which contains filters, power supplies, and mode control relays. Figures 3-6 and 3-7 show the IRU and SEU hardware.



Weight: 64 lb (29.03 kg)

Size: 18 1/2 × 18 × 13 3/4 in. (46.8 × 45.6 × 34.8 in.)

Figure 3-6. Inertial Reference Unit



Weight: 25 lb (11.34 kg)

Size: 13 3/4 × 8 3/4 × 5 7/8 in. (34.8 × 22.2 × 14.9 cm)

Power: 185 watts (28 Vdc)

Figure 3-7. Systems Electronic Unit

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 axis, and outputs these components via the ac digital to analog converters. Figure 3-8 is a block diagram of the guidance system.

3.4.2.2 Guidance Update. Increased mission times for the Reusable Centaur (36-hour nominal) with the added engine burns and maneuvers over the D-1T will result in larger error dispersions for the guidance function. This will require updating of the initial inertial data. New equipment has been added in the form of a horizon sensor and a sun sensor to assist ground tracking inputs by a three-axis attitude update. This extra capability results in a Level III type autonomy preserving a moderate onboard computer capacity and requiring position and velocity update to be RF uplinked from the ground. Figure 3-9 shows the interrelationship of the various elements that make up the guidance update system.

A more complete trajectory and guidance error analysis is needed for the Reusable Centaur and the longer missions to accurately establish the final update requirements and the best mission update times. Related studies from the STSS study, NAS8-29676, and the Centaur/Shuttle Integration Study, NAS3-16786, have indicated the gross requirement for this subsystem based on typical Centaur guidance and navigation system errors.

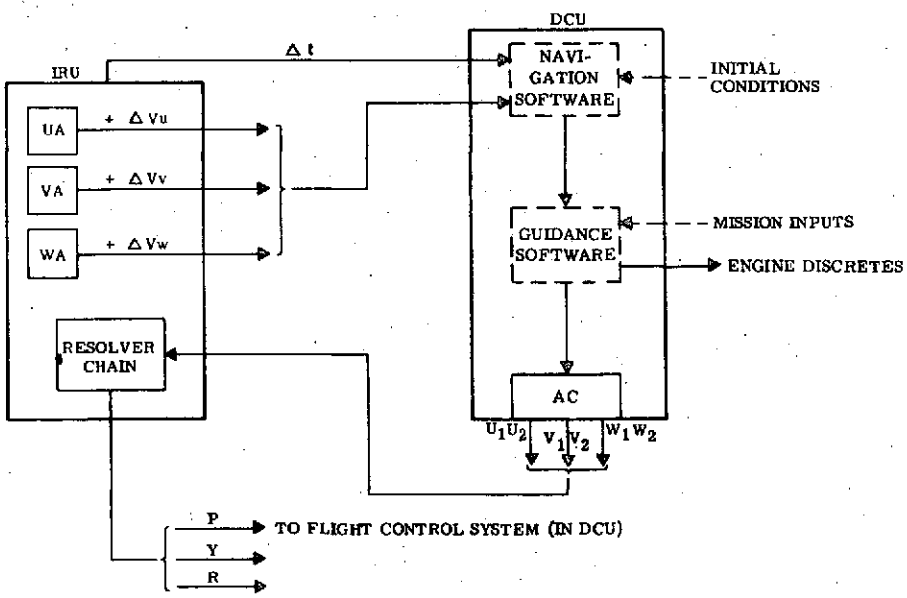


Figure 3-8. Guidance System Block Diagram

3.4.2.3 Flight Control. The DCU receives analog attitude outputs from the inertial measurements group, converts them to digital form, operates on them in accordance with the software instructions, generates output commands, and converts them to analog signals for power flight control and digital commands to the coast phase control system. Figure 3-10 is a functional diagram of the flight control system (primary). The backup attitude system used during retrieval and docking with the Orbiter in the event of primary failure is described at the end of this section.

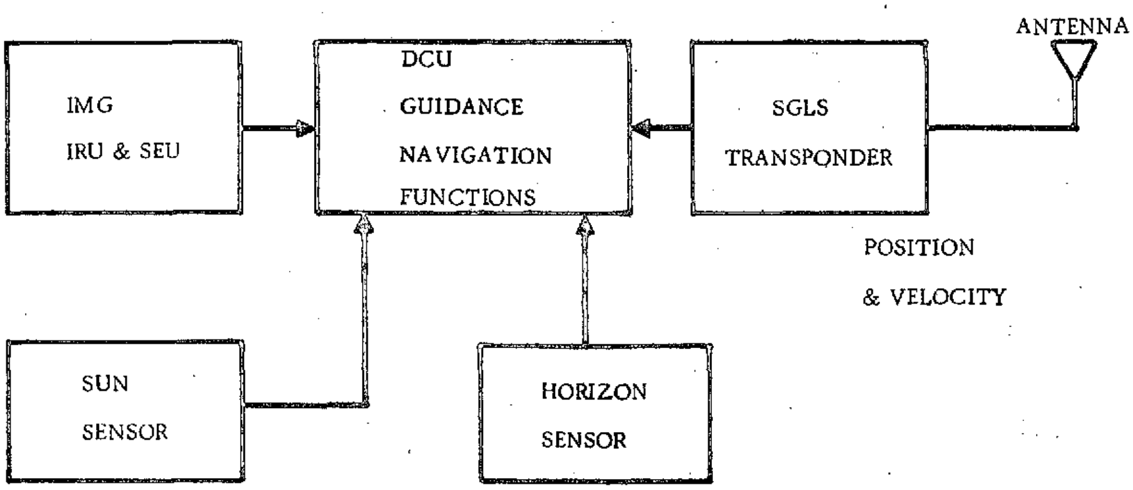


Figure 3-9. Guidance Update

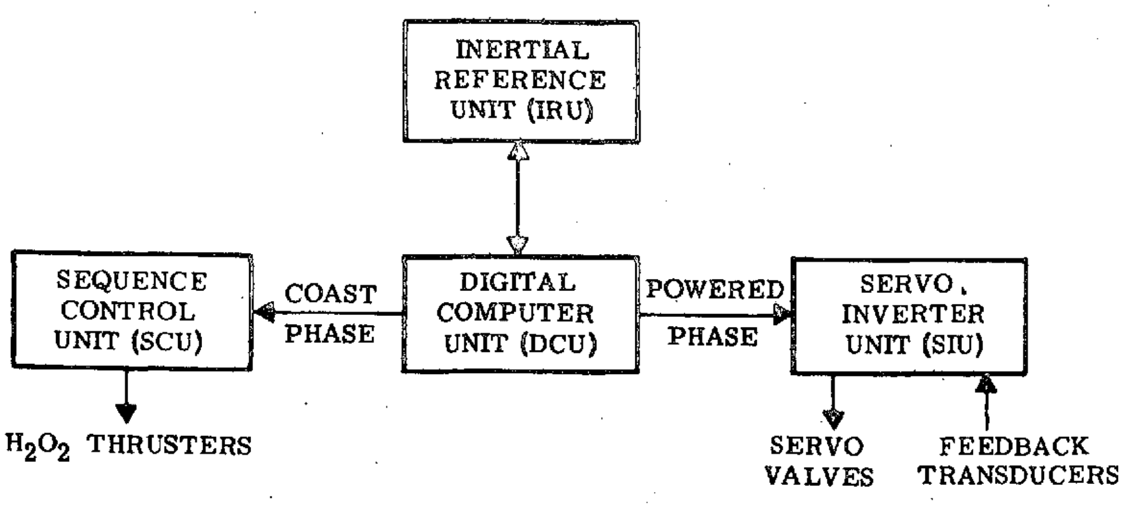


Figure 3-10. Primary Flight Control System

Powered Flight Control. The DCU issues three analog output signals which correspond to a desired engine position to the servo inverter unit (SIU). The mechanical engine control device is a hydraulic actuator controlled by a hydraulic servovalve. An electronic input from the SIU to the valve causes hydraulic fluid flow and engine movement. Engine position signals are provided by a feedback transducer. Each of the main engines has two actuators, one to do pitch movement and one to do yaw.

Coast Phase Control. During coast, vehicle rate and displacement are detected and measured by the IMG, just as in powered flight. These signals are processed by the DCU and translated to vehicle command requirements. Basic control during this phase is provided by monopropellant thrusters mounted in an orthogonal arrangement at the perimeter of the Centaur thrust section. 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.

Docking Control Backup System. Safe retrieval of the Centaur upper stage into the Orbiter bay requires the satisfaction of two principle conditions: (1) the Centaur must be statused via the RF communications link, and (2) no hazardous last minute failures should take place prior to full deactivation of the upper stage subsystems. Stabilization of the stage must be maintained until full control is achieved with the lock-on of the Orbiter manipulator arms. This seems to necessitate continued operation of the attitude control system and the APS thrusters to the point of actual contact.

Dual redundancy in the form of an emergency backup stabilization control has been designed into the flight control subsystem. This backup electronic package is called the docking attitude control unit and consists of an attitude sensor package containing three inertial quality gyros of moderate precision. These gyros have electronic caging to provide rate signals and are activated before the return approach to the Orbiter. The backup attitude gyro signals are compared to the main IRU and if the errors exceed a certain level, with command concurrence from the Orbiter, the backup takes over from the primary system and directly controls the vehicle. Control logic electronics is included to bypass the primary system, and to permit manual override from the Orbiter as needed with visual monitoring by the Mission Specialist.

Estimate and specifications of the complete docking attitude control unit (DAC) are:

Dimensions:	14 × 12 × 10 in. (35.5 × 30.4 × 25.4 cm)
Weight:	40 lb (18.14 kg)
Power:	60 watts, when activated prior to Orbiter approach

Figure 3-11 shows a block diagram of the DAC.

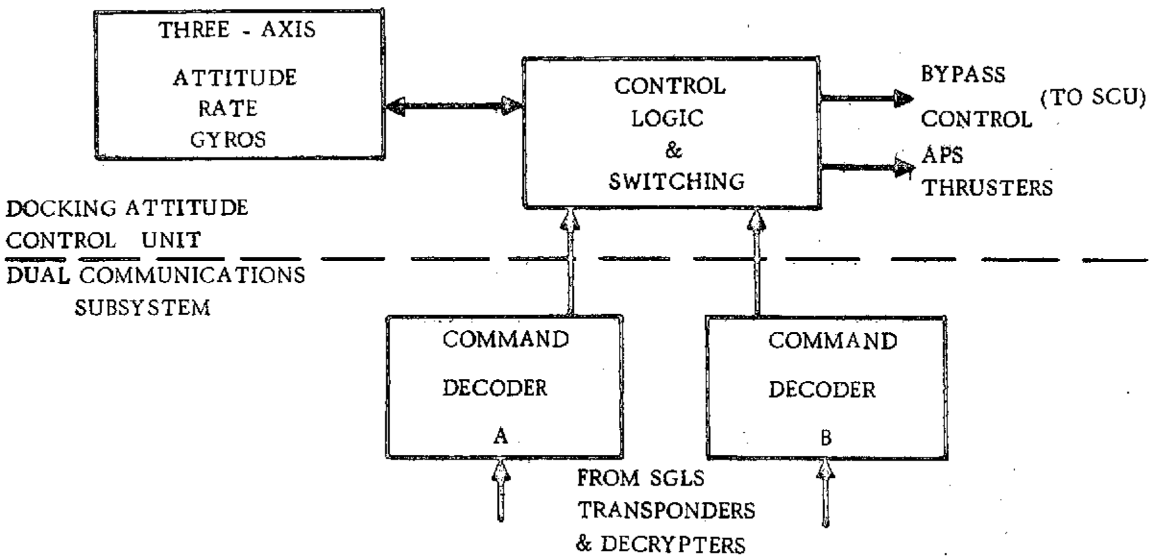


Figure 3-11. Docking Attitude Control Backup System

3.4.2.4 GN&C Options. The foregoing descriptions are for the initial recommendations for subsystem components which are based primarily on minimum development cost and minimum program risk. Several alternatives have been considered for the GN&C subsystem.

Dual Strapdown IMU. Replacement of the D-1T Centaur simplex gimbaled platform with dual skewed axis strapdown gyros has been predesigned at General Dynamics as a potential Centaur improvement, and was proposed to NASA during 1972 This proposal involves two Hamilton Standard strapdown platforms from the Delta inertial guidance system (DIGS) which have flown on Delta launches. The interface with the Centaur central computer (DCU) is fully compatible since the DIGS guidance computer is a smaller memory version of the Centaur Teledyne computer.

The skewed axis arrangement gives the equivalent of triple redundancy failure protection with dual platforms and can significantly improve guidance reliability. Comparable projected mean time between failure (MTBF) values for the Centaur gimbaled platform system and for a single Hamilton Standard strapdown platform are 4000 hours and 10,000 hours, respectively. The strapdown system has slightly less accuracy, but adequately meets the Shuttle upper stage requirements.

Appreciable development cost is required to incorporate the strapdown units if the proposed D-1 Centaur improvement has not been accomplished by NASA prior to the design of a new Shuttle upper stage. Typical equipment values are:

	Usage	Wt	Power
Dual DIGS Platforms	Delta	64 lb (29.03 kg)	240 watts

Star Sensor. Guidance update has been selected so as to need a minimum of additional software for the central computer (DCU). Position and velocity update from ground tracking has been assumed, with attitude update from on-board sensors (Level III type autonomy). The preliminary guidance error analyses performed for the modified D-1T expendable Centaur (NASA study NAS3-16786), with shorter missions, indicate that a horizon sensor alone may sufficiently reduce the dispersions in most orbital situations. For full three-axis update, a second sensor is required as the horizon sensor is limited to two-axis information. The recommendation has been made that a simple sun sensor would be adequate.

An alternative scheme is the use of a star tracker or star sensor with a software star catalog. This introduces a computer software requirement, generally in the range of 2000 to 6000 words, and may require a vehicle maneuver to obtain three-axis update data.

Further study is being done on the desirability of using a strapdown star sensor in lieu of the horizon sensor and sun sensor. Typical equipment values for this option are:

	Usage	Wt	Power
Star Sensor	Small	11 lb (4.99 kg)	5 watts
Ball Brothers CT-401	Astronomy Saturn C		

8 × 8 deg FOV
Estimated Accuracy ± 0.004 deg

Dual Redundancy. A further complete system alternative would be full dual redundancy for the D-1S(R) astrionics, which would increase the GN&C subsystem reliability due to central computer involvement. For the D-1S(R) version of Reusable Centaur with an intermediate size fuel tank, full dual redundancy is considered too costly to meet program objectives and has a higher risk factor than the minor modification of the existing D-1T Centaur.

3.4.3 COMMUNICATIONS. All elements of the communications subsystem are revised from the D-1 Centaur, with the exception of the DCU, so as to be compatible with the Space Shuttle upper stage requirements. The previous C-band tracking RF system and the S-Band telemetry transmitter have been deleted and replaced by dual S-Band transponders which are NASA USB/STDN and DoD SGLS/Satellite Control Facility (SCF) compatible. RF transmission/reception elements are reconfigured to operate in the assigned frequency ranges:

DoD:	Downlink	2200 to 2300 MHz
	Uplink	1750 to 1850 MHz
NASA:	Downlink	2200 to 2300 MHz
	Uplink	2025 to 2120 MHz

Table 3-6 lists the communications equipment. Figure 3-12 is a block diagram of a single channel of the dual redundant system.

Table 3-6. Communications Equipment

	Use/Mfg.	Wt (lb)	Power (W)
SGLS Transponder (2)	Motorola	19 (8.62 kg)	32
Command Decoder (2)	Conic	8 (3.63 kg)	4 avg
Power Amplifier	MSC	1 (0.45 kg)	42 (high data rate only)
RF Switch	Transco	0.6 (0.27 kg)	-
Coupler	Elpac	1 (0.45 kg)	-
Isolator	Wavecom	1.4 (0.64 kg)	-
Antennas (4)	D-1T/GDCA	1.2 (0.54 kg)	-
Encrypter (2)	GFE	11.2 (5.08 kg)	5
Decrypter (2)	GFE	10.6 (4.8 kg)	5

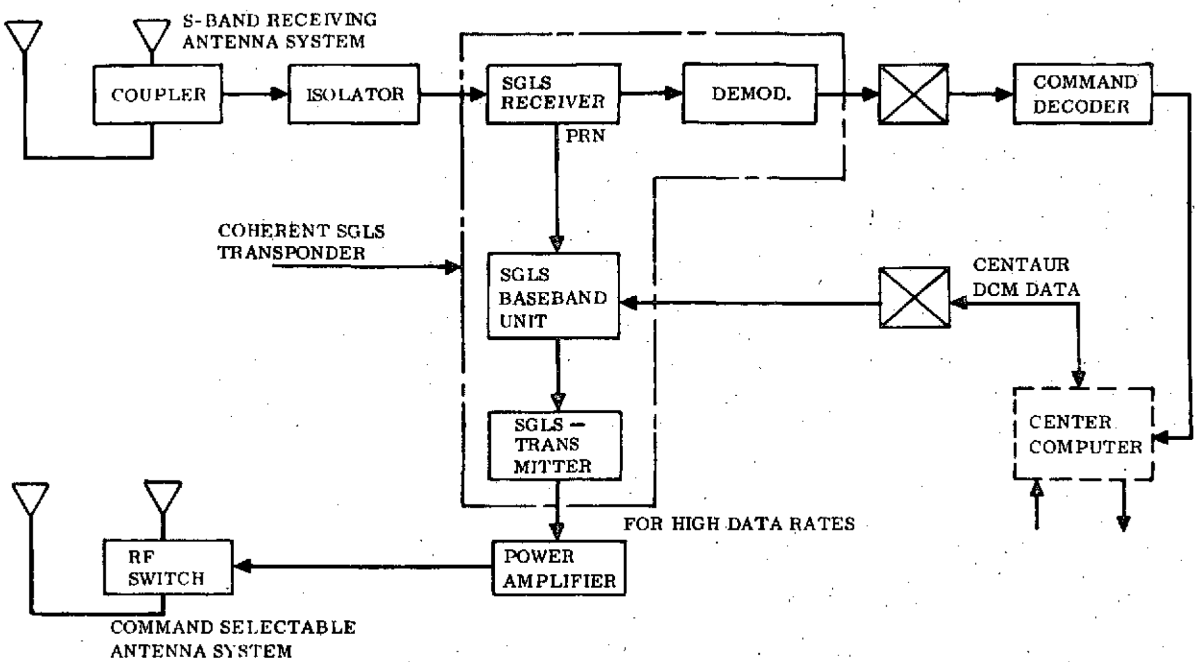


Figure 3-12. Single Channel Functional Diagram - Communications Subsystem (DoD)

The selected USB equipment for the NASA missions was originally developed for the Earth Resources Technology Satellite (ERTS) by Motorola. Some modification of this dual equipment is needed for use as two independent transceivers in the redundant strings desired for application in D-1S(R). Additional units are two command decoders (in a single box), an RF switch, a power amplifier, and dual receiving and dual transmitting antennas. See Table 3-7.

Comparable units for DoD missions are SGLS compatible equipment, which are modifications of the Motorola proposed system for FLTSATCOM. Eighty percent of this SGLS hardware is existing and of proven design. For DoD use, the additional units are two command decoders (in a single box), an RF switch, a power amplifier, dual receiving antennas, dual transmitting antennas, a command decrypter, and a data encrypter.

Figure 3-13 shows the basic RF link requirements for the D-1S(R) communications.

The DCU is used with the communications function and includes a central controller unit (CCU) for formatting the telemetry data. Telemetry measurements are collected by the remote multiplexer units from the various instrumentation transducers and signal conditioners, then digitized. These measurement signals are handled in the bit stream in the order of the form addresses, and the pulse code modulation (PCM) bit

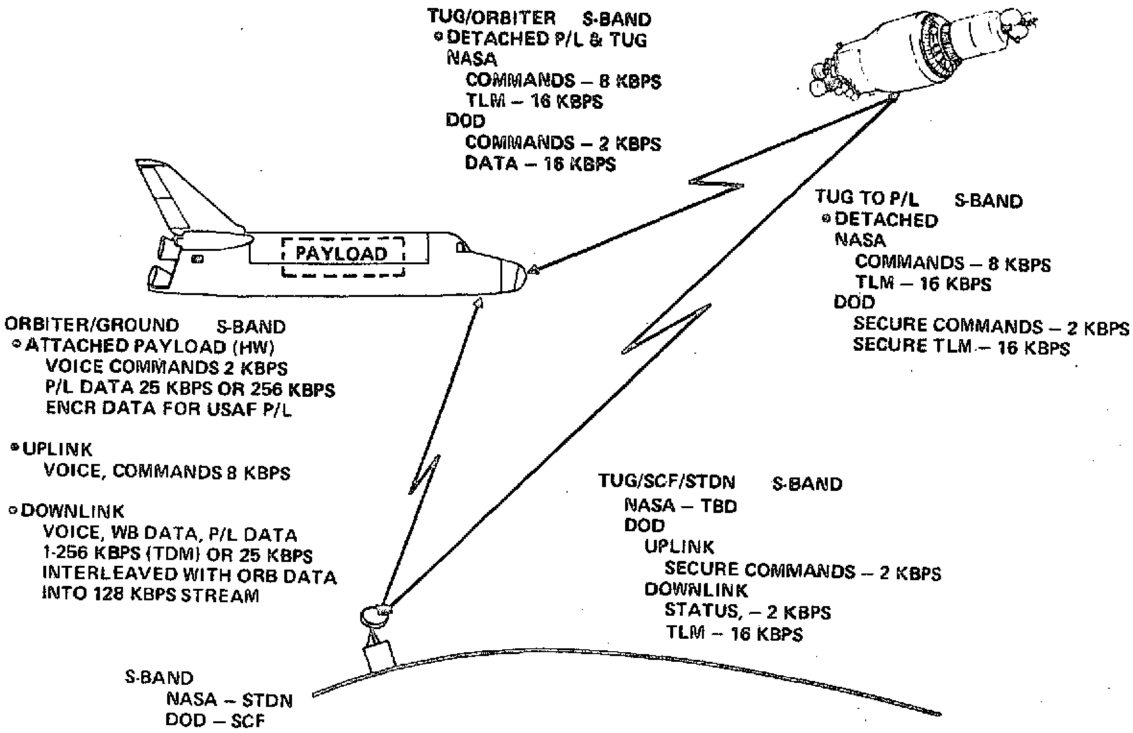


Figure 3-13. Upper Stage Communications Requirements

stream modulates the output of the S-Band downlink transmitter. The maximum capability of the DCU is 267,000 bits per second, which provides an upper limit of 30,000 measurements per second.

3.4.4 INSTRUMENTATION. Vehicle system measurements that are not digital in nature are converted to digital representations and transmitted in a serial format. Measurements made directly or by transducers are collected by the remote multiplexer units (RMUs). Analog signals are scaled by the signal conditioners before reaching the RMUs. The RMUs identify signals by addresses and respond by sending the measurements to the central controller unit (CCU) located in the DCU.

The format sequence of measurement addresses is determined by reading and interpreting a segment of the DCU memory. Formatting is done by programming a dedicated area of DCU memory. Up to four formats may be stored and the one in use is selected by the DCU program being executed. The maximum selectable bit rate is 267,000 bits per second.

The instrumentation 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. Measurement data contained in the PCM bit stream modulates the output of the USB/SGLS communications transmitter while in free flight or is available via coaxial line for Orbiter or ground checkout use.

Table 3-7 presents the estimated measurements summary for D-1S(R) and the equipment weights and power are listed in Table 3-8.

Table 3-7. D-1S(R) Measurements Summary for Normal Flight - Not Including Development Flight Instrumentation

			D-1T Comparison
Discrete Commands (Status)	27 Vehicle 18 Pallet	(106)	109 Umbilical
Discrete Monitors	29 Vehicle 12 Pallet		
Analog Monitors	16 Vehicle 4 Pallet		
PCM Downlink			
Subsystem			
DMS	1		1
G & N	21		12
Guidance Update	6		-
Flight Control	90 Sequence Control		94
	39 Servo		39
Communications	31 Telemetry		32
Power	22		33
Propellant Utilization	12		18
Hydraulic	18		18
Pneumatic	31		31
Propulsion	119		119
Structure	93		119
Range Safety	-		14
Miscellaneous	21		36
Totals	610		675

Table 3-8. Instrumentation Equipment

	Weight (lb)	Power (W)
Forward Instrument Box	17 (7.71 kg)	10 avg
Aft Instrument Box	17 (7.71 kg)	10 avg
Transducers & Signal Conditioner	68 (30.84 kg)	50 avg

The measurements comparison is for a normal operational flight, does not include payload measurements, and does not reflect the increased development flight instrumentation (DFI) typical of a first flight. Centaur has an FM/FM telepak, which is added for DFI instrumented flights, and the number of measurement increases is dependent upon the specific requirements. As noted, the 675 normal measurements can be increased to as many as 1536 without changing the existing DCU capability.

For the D-1S(R), as shown in Table 3-7, the guidance and navigation measurements increase due to the docking attitude backup system, guidance update adds six measurements, the structural measurements decrease due to the absence of the Titan shroud, range safety no longer exists, and some of the miscellaneous items for Titan disappear.

3.4.5 ELECTRICAL POWER/POWER CONVERSION AND DISTRIBUTION

3.4.5.1 Electrical Power. Primary electrical power for Centaur has traditionally been furnished by onboard silver zinc batteries for the relatively short missions of a few hours or less duration. The standard Centaur batteries are a high discharge design with a capacity of between 40 to 55 watt-hours per pound of weight. For longer missions, this results in the carrying of excessive vehicle dry weight.

The D-1S(R) baseline is a revised complement of batteries, mixing standard Centaur types with lightweight batteries. Latest estimates of D-1S(R) electrical power show a requirement of 852 watts average for the 20 hours free flight outside the Orbiter. This is equivalent to 608 ampere hours.

A weight-effective choice for the baseline is envisioned as one 400 A-hr Eagle Picher type 30 battery (10 amperes maximum discharge rate), one 150 A-hr Centaur battery (80 amperes maximum rate), and one 100 A-hr Centaur battery (80 amperes maximum rate) for a weight of (134 + 81 + 66) 281 pounds (127.29 kg) total. This would save 43 pounds (19.5 kg) over all Centaur type batteries and would have 50 A-hr excess capacity. Two standard Centaur batteries are suggested to make up the 250 A-hr high discharge rate capability as they fit without alteration into the existing equipment module design, although an equivalent single case design is available from the manufacturer.

Apparent weight savings could be obtained by using all lightweight low discharge rate batteries. However, the heater and propulsion support requirements for the vehicle could not be met. These loads require peaks for short and medium durations which are about 100% over the average value of 30.4 amperes (up to 63.1 amperes). A mixture of types most easily offers the needed high discharge rates and weight savings. Table 3-9 shows the estimated electrical loads summary.

Table 3-9. Electrical Load Summary

	Average Power (W) (20 hr)	Peak Coast Power (W) (Variable)	Peak Burn Power (W) (0.18 hr)
Subsystem			
Data Management	225	255	255
GN&C	254	388	367
Communications	38	128	128
Instrumentation	70	80	80
Heaters	135	486	-
Propulsion Support			
Main Engine	-	-	504
Other Support	130	196	433
	852	1533	1767

Fuel cells are an excellent choice for the cryogenic D-1S(R) vehicle, and would in this application save about 167 pounds (75.65 kg) of dry weight using a modified Orbiter fuel cell. For this option, there would be a development cost of around five million dollars. With the short mission of 20 hours free flight and the ground rule of minimum Centaur modification, the selected baseline was the mixture of silver zinc batteries as described.

3.4.5.2 Power Conversion and Distribution. The electrical power distribution system uses the batteries to supply dc power to three separate busses so as to isolate equipment that tend to be generators of electromagnetic interference from equipment that may be sensitive. Bus 1 supplies power to the digital computer unit, sequence control unit, multiplexers, and signal conditioners. Bus 2 provides power to the communications subsystem, propellant utilization system, servo inverter unit, and instrumentation. The loads on Bus 1 and Bus 2 are primarily loads that tend to be sensitive to electromagnetic interference. Bus 3 carries loads of a switching nature such as solenoids, relays, and motors, which tend to be generators of electromagnetic interference.

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

Vehicle power is provided from the Orbiter prior to deployment. The power changeover switch is activated prior to deployment and connects the internal power source to the power distribution system. The power changeover switch features a make-before-break contact arrangement to ensure uninterrupted power to the loads during the switching. The electrical system employs a single point ground. Current monitoring for individual system use is provided at the single point ground bus in the sequence control unit.

The changeover switch is a revised Centaur D-1 design capable of carrying 65 amperes per pole continuously at 28 Vdc, with a voltage drop of less than 100 millivolts on each of its multiple poles. The switch also has single pole, single throw break-before-make contacts capable of carrying 7 amperes continuous for each set with less than a 77-millivolt drop. These latter contacts are primarily for signal type circuits. Total maximum transfer time for the switch is 170 milliseconds. The switch is hermetically sealed and pressurized to one atmosphere with 95% dry nitrogen and 5% helium.

3.4.5.3 Electrical System Harnessing. Generally the harnessing consists of three kinds of H-film insulated wires: (1) wires that may be sensitive to electromagnetic interference, (2) wires connecting equipment that tend to generate electromagnetic interference, and (3) wires that do not fall into the above categories. The third group is routed between the other two groups. This routing provides some isolation and minimizes interaction between the various electronic systems.

Estimated weight for the electrical harnessing and connectors for the vehicle is 210 pounds (95.2 kg).

3.5. RLTC ASTRIONIC SUBSYSTEMS

Subsystems for the RLTC astrionics use much of the D-1T Centaur electronic equipment, modified as required for mission reliability goals, reusability, man-rated safety, Shuttle Orbiter integration, and increased vehicle autonomy (Level I). Many of the electronic box descriptions are the same as those for the D-1S(R), or nearly the same.

The astrionic equipment units are mounted to an equipment module ring structure which is larger in diameter than the regular Centaur's (15 feet compared to 10 feet) but similar.

3.5.1 DATA MANAGEMENT SYSTEM (DMS). The DMS has three digital computer units (DCU), two remote multiplexers, and a tape recorder. These units, which provide the required data, reporting, and control functions for the RLTC vehicle, are used in a dual redundant electronics system to meet mission reliability goals and ensure safety relative to the Orbiter and crew.

Computer functions include navigation, guidance, stabilization control, sequencing, venting/pressurization control, propellant utilization control, communications/telemetry formatting and control, instrumentation, and power conversion and distribution control.

3.5.1.1 Digital Computer Unit. This is planned as a modified version of the Centaur TDY-300 computer with increased throughput speed and added main memory capacity. A triple redundant arrangement with simple voting is used to provide computer functional reliability and to keep the software logic from becoming overcomplicated. The basic TDY-300 computer has been developed and produced by Teledyne as it is currently used in the operational Centaurs. A detailed DCU description is contained in the preceding D-1S(R) section.

An alternative selection, which is still under consideration, is the possible use of the Shuttle Orbiter computer the IBM (4 Pi) AP-101 with a 64,000 word memory. For this option, the development of new input/output elements to satisfy the existing Centaur equipment interfaces is a sizeable task, which in the initial tradeoff studies appears to result in higher DDT&E costs for the program as compared to adaptation of the existing design.

Main memory for the DCU has been expanded from the standard TDY-300 size of 16,000 24-bit words to 48,000 words. Requirements as studied under the STSS Contract, NAS 8-29676, for the throughput speed show the possibility of 170,000 operations per second peaks for some mission phases.

3.5.1.2 Flight Software. Estimated programming requirements for the nominal four-hour synchronous equatorial mission are:

Module	Memory Words
Executive	3400
Attitude Control	1600
Communications	7300
Guidance	7000

Module	Memory Words
Guidance Update	8000
Navigation	5000
Onboard Checkout	5600
Sequencing	5000
Thrust Vector Control	2000

3.5.1.3 Tape Recorder. A flight tape recorder is provided to temporarily store vehicle status data to be transmitted to the ground. Initial selection is a Lockheed Electronics MTR 2500 which is being developed for NASA-MSFC on the High Energy Astronomical Observatory program. This recorder is capable of recording and reproduction at a rate of 51,250 bits per second. Total data storage is 4.15×10^8 bits per second for a maximum tape time of 270 minutes.

3.5.1.4 Remote Multiplexer Units. These are revised Centaur designs compatible with the DCU. They collect the inputs to the DCU from the various instrumentation transducers.

3.5.2 GUIDANCE, NAVIGATION AND CONTROL. RLTC has a GN&C subsystem that is revised from the existing D-1T Centaur and the D-1S(R) with redundancy applied to meet reliability and safety goals. Benefit has been derived from the Centaur studies performed in the electronic system redundancy area for Lewis Research Center under NAS3-15314 in 1971 and 1972.

Included in the GN&C subsystem are the functions of:

Guidance and navigation

Guidance update

Flight control

(Note: Rendezvous and docking was omitted as a subsystem since payload retrieval was deleted from the study as a capability requirement.)

3.5.2.1 Guidance and Navigation. The inertial reference unit that has been selected for RLTC is a dual strapdown inertial configuration made up of two DIGS platforms operated in a skewed axis arrangement with a revised external packaging. Design of the D-1T Centaur digital computer unit input/output was originally made so as to be compatible with the DIGS type of platform; the dual arrangement already proposed to NASA as a Centaur change is sufficiently predesigned that the change requirements and interfaces are tied down. It is expected that the basic platform change from the present gimbaled system will be incorporated into Centaur prior to an RLTC go ahead.

3.5.2.2 Guidance Update. New RLTC equipment includes a horizon sensor and a star sensor so as to furnish an autonomous three-axis position, velocity, and attitude update capability. Guidance accuracy requirements for the longer missions (nominal four hours) make the update necessary. Present assumption is that an update will be performed prior to each main engine burn, and this relieves the need for precision inertial capability. The dual strapdown platforms are only slightly less accurate than the existing D-1T Centaur gimbaled IMU, and the accuracy is more than adequate together with the added guidance update feature.

Figure 3-14 shows the functional connection of the elements of the flight control function. The digital computer unit (DCU) from the DMS accepts the sensor inputs from the dual strapdown inertial measurement unit (SDIMU), and from the horizon and star sensor for the determination of three-axis position, velocity, and attitude data. Coordinate transformation and the guidance and navigation computations are performed in the DCU. The DCU is a triple redundant computer for high reliability for critical functions. Voting of the computer outputs is accomplished in the sequence control unit (SCU) which distributes and switches the various power circuits in accordance with the operating outputs from the DCU.

Internal redundancy is incorporated into the SCU, which is revised from the simplex D-1T version employed for the D-1S(R). Voter redundancy is triple, flight critical circuits are at least dual, and other functions are simplex. Interfaces are similar to D-1T Centaur, but are revised for redundancy as controlled by the DCU.

The servo inverter unit (SIU) is identical to the unit used for the present Centaur and the proposed D-1S(R). A dual configuration is used to improve system reliability.

Contained in the Centaur SIU are four printed circuit boards, each of which is a servo amplifier feeding a servo valve, actuator, and feedback transducer so as to form the

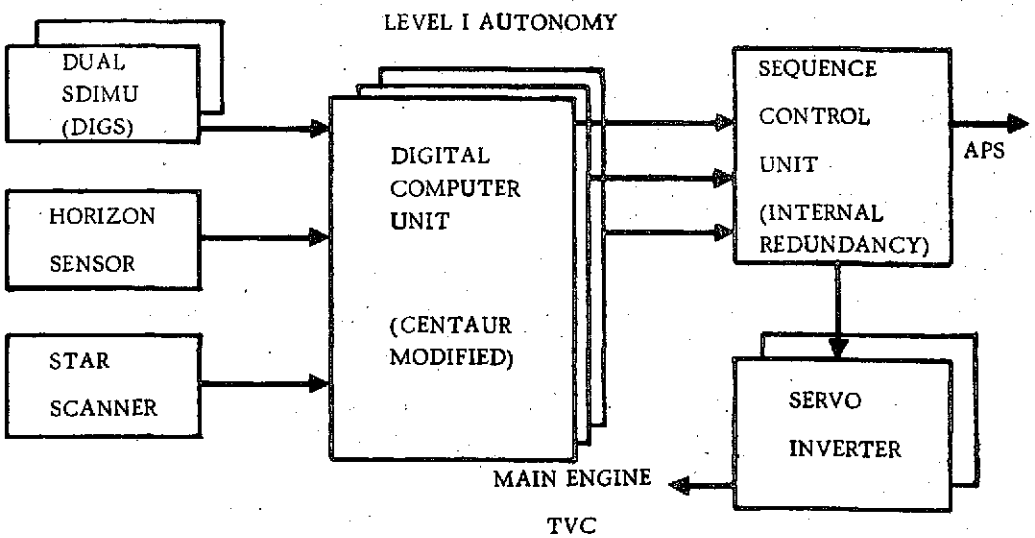


Figure 3-14. Flight Control Subsystem - RLTC

engine positioning servo. From the error-proportional dc input signals, the SIU provides the variable ac power needed for actuating the thrust vector control (engine positioning) system.

3.5.3 COMMUNICATIONS. A dual redundant communication capability is provided with USB/SGLS compatibility. The RLTC communication subsystem is identical to the one described for the D-1S(R) vehicle.

3.5.4 INSTRUMENTATION. The RLTC instrumentation is an extension of the revised D-1T system as applied to the Reusable Centaurs. Compared to the D-1S(R), the RLTC redundancy increases the number of measurements handled internally by the DCU, and can reduce those telemetered to the Orbiter or to the ground. Many of the measurements are used onboard for redundancy management outside of the DCU. Voting is used for the DCU comparison. The total number of measurements are estimated at 970 compared to 610 for D-1S(R).

3.5.5 ELECTRICAL POWER/POWER CONVERSION AND DISTRIBUTION. This subsystem for the RLTC is similar in concept to the distribution busses and harnesses for the battery system of D-1S(R). Necessary revisions will be incorporated to adapt the switching to the fuel cell power source and the backup battery switching. Harnesses between equipments on the vehicle will be redesigned for the redundant system and the additional boxes. Triple redundant astrionics, in combination with increased electrical heating needs for longer missions, increase the RLTC power levels above those of the D-1S(R).

A Pratt & Whitney lightweight fuel cell system provides the primary source of electrical power, with an additional backup battery for emergency. As the fuel cell generates electrical power by the direct catalytic conversion of hydrogen and oxygen, the two waste products of heat and water are rejected to space. Water is rejected by venting stream and the waste heat is used to warm hydrazine and helium in a thermally efficient system.

The advantages of the lightweight versus the Orbiter type fuel cell are a result of how mechanical construction/quality control, contamination/performance degradation and the fluid, pressure and thermal controls support the catalytic fuel cell stack. Beyond its weight advantage, the lightweight fuel cell passive thermal rejection and resulting peripheral equipment support are significantly less than the older designs. The ability to monitor output voltage in relation to power enables venting of contaminants while the unit continues to produce power. Not only can propellant grade reactants be used, but they can contain up to 5% helium. This, in addition to an operating pressure of 16 psia, enables the lightweight fuel cell to use liquid or vaporized propellants directly from the main tanks.

A major portion of the total energy (22%) would be involved in assuring that the hydrazine ACPS system is heated within its operating temperature range, if electrical heaters were used. Using the fuel cell waste heat to heat the hydrazine directly reduced the electrical power requirements to the 1.5 kW range (47.4 kW-hr total). See Table 3-10. Secondary benefits in using the fuel cell waste heat to supply the heat the hydrazine system loses to space is the elimination of 32 separate thermostatically controlled electrical heaters and a separate space radiator.

Another area of performance improvement is the incorporation of a heated reservoir, within the fuel cell heat exchanger, to pass through cryogenically stored helium for tank pressurization. The warmed helium is expanded, requiring less helium to raise the tank pressures before each engine firing. Less storage bottle volume (and weight) is required to store the same weight of helium when it is stored within the hydrogen tank. The energy absorbed from the reservoir in raising the temperature of the helium and cooling the reservoir approaches balancing out the added waste heat (higher energy levels) that occurs during engine ignition, heating the reservoir. The heat exchanger is thus precooled by the helium, anticipating the higher energy that needs to be absorbed during engine ignition. The fuel cell and heat exchanger are shown in Figure 3-15.

Table 3-10. Electrical Load Summary - RLTC with Fuel Cell

Functions	Avg Power (31.5 hr)	Peak Coast Power (Variable)	Peak Burn Power (0.18 hr)
Data Management	525 watts	565 watts	656 watts
GN&C	307	455	425
Communications	38	128	128
Instrumentation	70	80	80
Heaters	394	654	-
Propulsion Support			
Main Engines	-	-	504
Others	131	337	444
	1,465 watts	1,882 watts	2,146 watts
Current levels (28 Vdc)	52.3 amps	67.2 amps	76.6 amps

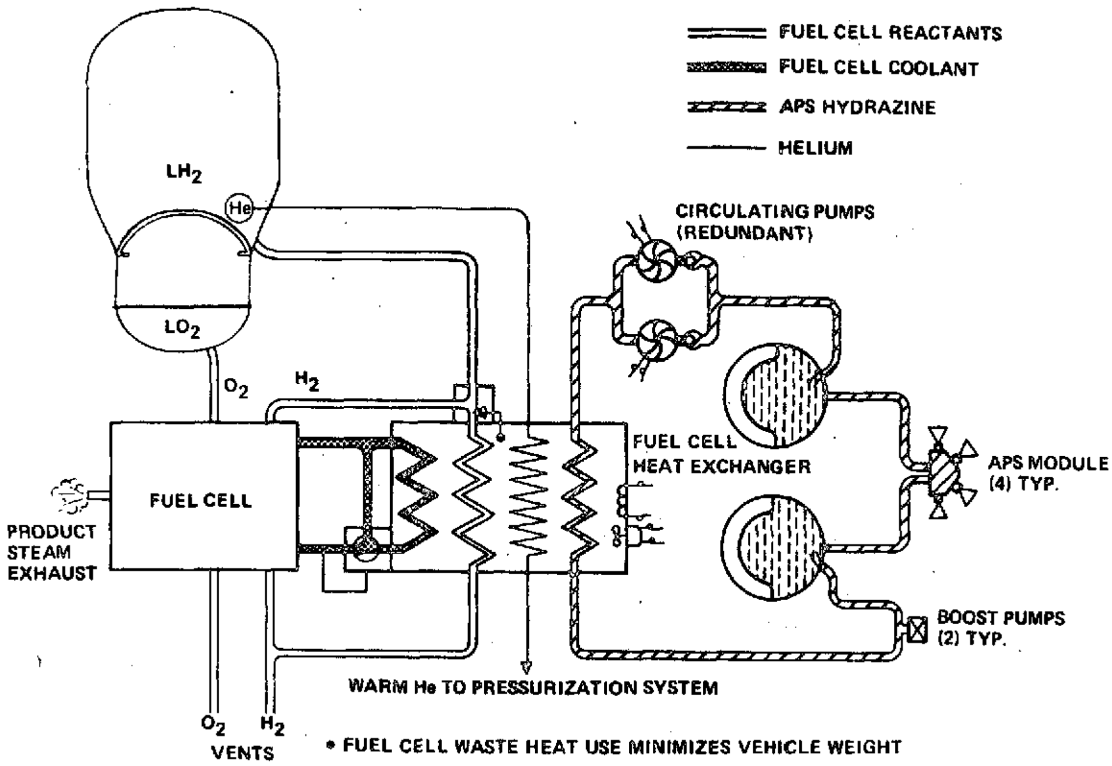


Figure 3-15. Fuel Cell and Heat Exchanger

Thermally interrelating the systems to use the normally rejected fuel cell waste heat to supply the heating needs of the hydrazine APS and the cryogenically stored helium reduces total electrical energy and individual system weights, controls, and complexity. The lightweight fuel cell total power system, including its 36 pound (16 kg) battery, thermal control, and tubing weighs less than 122 pounds (55 kg). Eliminating the thermostatically controlled electric heaters on the APS and boost pump hydrazine systems (22 pounds) reduces the fuel cell systems effective weight to 100 pounds (45 kg).

The thermally interrelated synergic system achieves additional payload delivery over a conventional separate fuel cell system or even more additional payload than could be achieved by adding existing Centaur batteries. See Table 3-11 for battery options.

While the RLTC is not dependent on this advanced fuel cell/fluid system thermal interaction, it is recommended and is not regarded as any more complex than the thermally integrated Orbiter fuel cell/life support system.

3.5.6 <u>EQUIPMENT LIST - RLTC.</u> Table 3-12 shows the astrionic subsystem equipments, usage, weights, and estimated power consumption for the RLTC vehicle.

Table 3-11. RLTC Battery Power Options (31.5 Hours Free Flight)

I. Centaur Type Batteries (150 A-hr, 81 lbs. (36.7 kg) each)

Full Average Load (Including Hydrazine Heaters)	1878 watts, 2100 A-hr
Number of Batteries	2100/150 = 14
Weight	14 x 81 = 1134 lb (514.5 kg)

II. Mixed Lightweight & Centaur Types (150 A-hr & 400 A-hr, 134 lb (60.8 kg))

Peak Burn Current	76.6 amperes
Requires 2 Centaur types at 300 A-hr	
Remaining A-hr 2100 - 300 =	1800 A-hr
Number of Lightweight Batteries	1800/400 = 4.5 (5 batteries)
Weight (2 x 81) + (5 x 134) = 832 lb (377.4 kg)	

Table 3-12. Equipment List Summary - RLTC

Equipment	Usage/Mfg.	Weight (lb)	Power (W)
Data Management			
Digital Computer (3)	D-1T modified	174 (78.93 kg)	480 avg
Remote Multiplexers (2)	D-1T modified	40 (18.14 kg)	30 avg
Tape Recorder	Agena	12 (5.44 kg)	15 avg
GN&C			
Strapdown IMU (2)	DIGS modified	64 (29.03 kg)	240
Star Sensor	Honeywell	6 (2.72 kg)	2 avg
Horizon Sensor	Barnes (13-161)	10 (4.54 kg)	5 avg
Servo Inverter Unit (2)	D-1T as is	100 (45.36 kg)	40 avg

Table 3-12. Equipment List Summary - RLTC, Contd

Equipment	Usage/Mfg.	Weight (lb)	Power (W)
Sequence Control Unit	D-1T modified	78 (35.38 kg)	20 avg
Communications			
USB/SGLS Transponder (2)	Motorola	19 (8.61 kg)	32 avg
Command Decoder (2)	Conic	8 (3.62 kg)	4 avg
Power Amplifier	MSC	1 (0.45 kg)	2 avg
RF Switch, Coupler, Isolator	Transco, Elpac, Wavecom	3 (1.36 kg)	-
Antennas (4)	D-1T/GDCA	4 (1.81 kg)	-
Encrypter (2) (GFE)	USAF	- (11.2 (5.08 kg)	5) -
Decrypter (2) (GFE)	USAF	- (10.6 (4.8 kg)	5) -
Electric Power			
Fuel Cell (2 kW) (LW)	Pratt & Whitney	42 (19.05 kg)	-
F/C Plumbing		44 (19.96 kg)	-
Backup Battery	Eagle-Picher	36 (16.32 kg)	-
Power Conversion & Distribution			
Harnesses & AC Inverter		220 (99.79 kg)	-
Instrumentation			
Instrumentation Boxes		22 (9.97 kg)	20
Transducers & Signal Conditioner		68 (30.84)	50 avg
		951 lb (431.37 kg)	940 watts avg
Astrionics Totals		972.8 lb (441.26 kg)	950 watts avg DoD

SECTION 4

MISSION PERFORMANCE

This section summarizes the mission performance for the D-1S(R) and RLTC configurations. Some estimates are conservative so the performance numbers may be reasonably expected to exceed these estimates.

a. A 10% dry weight contingency was used on all parts, even the existing RL10 engines. On the D-1S(R), which is about 90% existing hardware, alternate weight and performance data are given for a 5% contingency. (10% on new designs; 2% on existing.)

b. A flight performance reserve of 1.73% was used per NASA ground rules. Considering that the Centaur has a PU system, 1% may be possible to achieve.

c. A 145 ft/sec (42.7 m/s) ΔV from the ACS system was assumed, including 50 ft/sec (15.25 m/s) ascent midcourse correction. A total of 40 ft/sec (12.2 m/s) may be all that is required, which would reduce ACS propellants more than one-third.

d. Payload placement within an orbit for RLTC requires a six-burn (rather than a five-burn) mission and maximum RLTC time on orbit prior to payload deployment. This results in additional main engine start/stop losses and maximum propellant venting and ACS usage. Since current payloads have walking capability to place themselves within an orbit, payload placement by RLTC may not be required.

e. Since the D-1S(R) gross weight with pallet, kick stage, and payload is only about 45,000 pounds (20,400 kg) the Orbiter could ascend beyond 160 n.mi. (296 km), increasing the D-1S(R) payload placement capability.

4.1 MISSION REQUIREMENTS

The Reusable Centaur is a propulsion stage carried into low earth orbit by the Space Shuttle. The purpose of this stage is to extend the operating range of the Space Transportation System beyond the capabilities of the Shuttle by transferring payloads to higher orbits with plane changes and on-orbit maneuvers, or injecting payloads on escape trajectories. The Reusable Centaur is a versatile, high performance, LO_2/LH_2 powered stage. Its reusability and multi-payload capability lead to low operating costs.

Reusable Centaur and Orbiter interface equipment shall not exceed 35 feet (10.66 m) in length and shall be dimensionally compatible with the Orbiter cargo bay diameter of 15 feet (4.57 m). The Reusable Centaur shall be compatible with the Orbiter payload capability as shown in Figure 4-1. The Reusable Centaur is delivered to a 160 n.mi. (296 km) circular orbit and recovered from a 170 n.mi. (315 km) circular orbit by the Orbiter. The Orbiter is the active partner during Orbiter-to-Reusable Centaur docking and rendezvous.

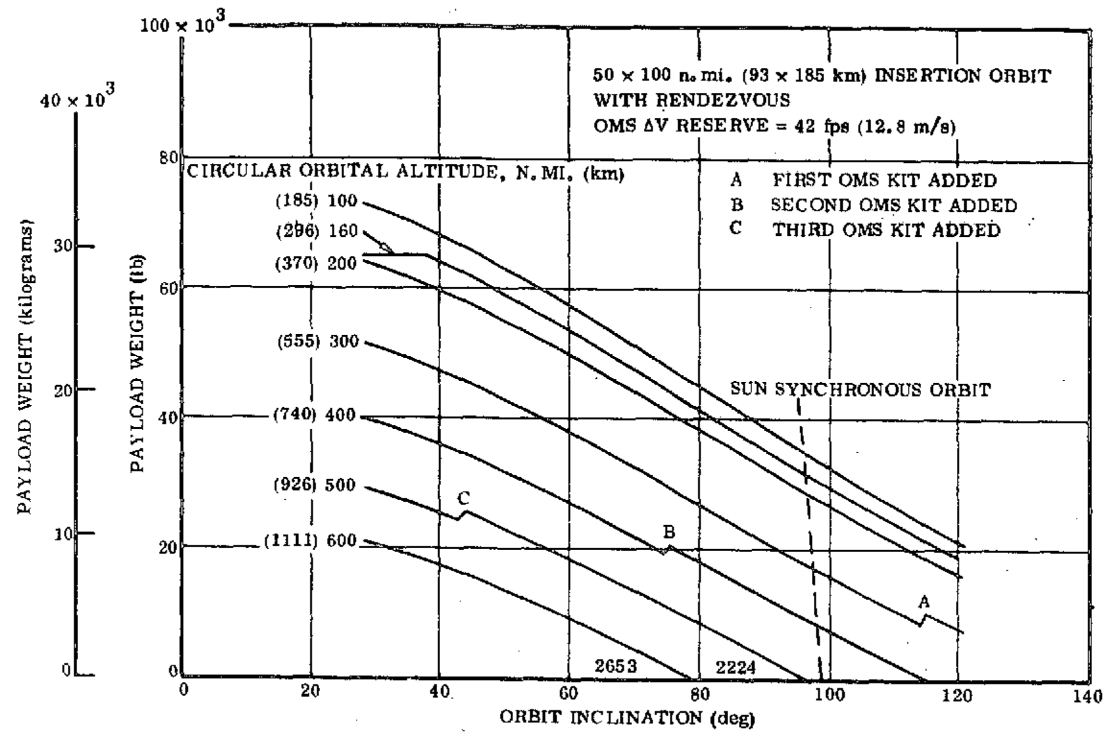


Figure 4-1. Shuttle Payload Capability

The Reusable Centaur configurations accommodate both NASA and DoD payloads, and perform all missions in the mission model. Missions are in the general classes of low earth orbit (mostly high inclination orbits) and synchronous and other high delta-velocity missions, and planetary missions. Payload deployment is the basic mission mode required of D-1S(R) and RLTC. In this mode the Reusable Centaur carries a payload from the Shuttle Orbiter to orbit, deploys the payload, and then returns to the Orbiter for return to the earth. Extremely high energy planetary missions may require that the Centaur be expended.

The RLTC stage deploys synchronous equatorial payloads into the final payload orbit. The D-1S(R) stage has less performance capability; to compensate for this a kick stage (velocity package) is deployed into a synchronous transfer orbit along with the payload. The kick stage then supplies part of the final ΔV for plane change and circularization at synchronous altitude. Kick stages are also used by the D-1S(R) for payload deployment to other high-energy earth orbits; both D-1S(R) and RLTC use kick stages for some planetary missions.

In addition to deployment missions, sortie missions are also accomplished by the Reusable Centaurs, which carry the payloads to orbit. The payload remain attached to Centaur during its on-orbit operational period, after which is then returned by the Reusable Centaur to the Orbiter for earth return.

The Reusable Centaurs are capable of multiple payload delivery with one, two, or three payloads deployed during a single flight. Only single payloads are deployed on planetary missions.

The mission model is defined in NASA-MSFC Memo PD-TUG-M(3-16-73) (the program Option 1 model) and payload orbit requirements are defined in the Space Tug Systems Studies Data Package. One-way delta velocities above the 160 n.mi. (296 km) Orbiter delivery orbits are given in Table 4-1. For programmatic purposes, Reusable Centaur flights are limited to three during 1980 and not more than 21 during 1981.

4.2 MASS PROPERTIES

4.2.1 VEHICLE WEIGHTS — D-1S(R). The D-1S(R) vehicle weight summary is shown in Table 4-2. The weight summary is based on the following vehicle configuration conditions:

Structure

The astrionics module, the forward adapter, and the aft adapter are all made of titanium skin stringers. While the aft adapter is new, the forward structure essentially exists.

The fuel tank was shortened 22 inches (55.8 cm) for a 29-foot (8.8 m) vehicle installed length with propellants ratioed to a 5.8:1 engine mixture ratio. Stainless steel skin thicknesses and ring sizing were based on loads analyses.

The LO_2 tank, including the intermediate bulkhead and the aft bulkhead, are existing, manufactured structures for the Centaur D-1T configuration.

The thrust structure is basic Centaur D-1T design and the equipment mounting structure is Centaur D-1T design with modifications to satisfy vehicle compatibility. The interface equipment for the payload and the EOS docking mechanism is based on vehicle reusability.

Propulsion

The main engine system, including engine actuation and support, is basic existing Centaur D-1T design.

The ACPS system is existing D-1T Centaur design sized to conservative mission requirements. Three H_2O_2 tanks for the system are included, as well as the four thruster modules. Valving and control for the system are included in the system weights.

Table 4-1. Delta Velocities for Capture Analysis

NASA Missions				DoD Missions			
Payload Number	Shuttle Inclination (deg)	One-Way ΔV (fps)	(m/s)	Payload Code	Shuttle Inclination (deg)	One-Way ΔV (fps)	(m/s)
1 to 8	28.5	13,950	4252	2, 3b, 15,	28.5	13,950	4252
9	28.5	14,160	4316	17, 12b, 6, 4a	28.5	13,950	4252
10	28.5	12,910	3935				
11	29.0	13,030	3972	3a	38.0	12,540	3822
				4b	38.0	10,460	3188
12	90.0	2,270	692	10	38.0	10,460	3188
13	90.0	8,520	2597	8	46.0	12,450	3795
14	90.0	3,570	1088				
15	100.0	1,700	518	11a	28.5	13,980	4261
16	99.2	1,110	338	11b	30.0	13,650	4161
				11c	38.0	14,000	4267
17 to 18	28.5	13,170	4014				
19	28.5	16,320	4974	5	90.0	1,850	564
20	28.5	23,500	7163	16	98.3	800	244
22	28.5	24,100	7346	12a	104.0	564	172
23	28.5	18,420	5614				
24	28.5	22,120	6742				

Table 4-2. D-1S(R) Weight Summary

CONCEPT:	MISSION:	CONFIG. NO.
D-1S(R) CENTAUR VERSION (29')	PAYLOAD DELIVERY TO SYNCHRONOUS ORBIT	D-1S(R) (NASA)

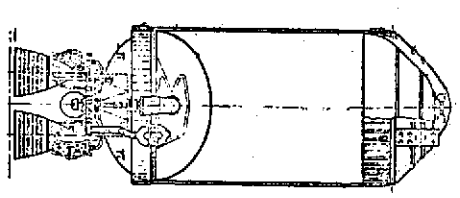
	(lb)	(Kg)
Structures	(1475)	(669.1)
Body Structure	563	255.4
Fuel Tank & Supports	421	191.0
Oxidizer Tank & Supports	318	144.2
Thrust Structure	37	16.8
Equipment Mounting Structures	61	27.7
Meteoroid Structure	-	-
Interface for P/L & Shuttle	50	22.7
Umbilicals	25	11.3
Propulsion	(1830)	(830.1)
Main Engine (ME) Assembly	580	263.1
ME Actuation & Support	153	69.4
ACPS & Support	237	107.5
Pressurization	346	157.0
Fill, Vent & Drain (Incl. Feed)	450	204.1
Purge System	-	-
Propellant Management	64	29.0
Thermal Control	(455)	(206.4)
Fuel Tank Insulation	177	80.3
Oxidizer Tank Insulation	177	80.3
Purge System	95	43.1
Radiators (Inc. Fluid Loop, Heaters, Coatings)	6	2.7
Avionics	(1013)	(459.5)
Rendezvous & Docking	-	-
Data Management	122	55.3
Flight Controls	160	72.6
Guidance & Navigation	89	40.4
Guidance Update	14	6.3
Power	281	127.5
Power Conversion & Distribution	210	95.2
Instrumentation	102	46.3
Communication	35	15.9
STAGE DRY WEIGHT	(4773)	(2165.1)
Contingency Ⓐ	245	111.1
TOTAL DRY WEIGHT	(5018)	(2276.2)
NON-USABLE FLUIDS	(522)	(236.7)
Trapped Propellants - Main	129	58.5
Trapped Propellants - ACPS	12	5.4
Trapped Gases	319	144.7
Trapped Helium	13	5.9
Propellant Leakage	-	-
Propellant Reserves - Main (194)	**	**
Propellant Reserves - ACPS	29	13.1
Propellant Utilization	20	9.1
Trapped Water	-	-
BURNOUT WEIGHT	(5540)	(2512.9)

	(lb)	(Kg)
NON-IMPULSIVE EXPENDABLES	(562)	(254.9)
ME Chilldown & Start	173	79.4
ME Leakage	25	11.3
ME Shutdown	60	27.2
Pressurants - Main	4	1.8
Pressurants - ACPS	-	-
Fuel Cell Reactants	-	-
Fuel Line Chilldown	-	-
Gas Generator	-	-
Thermal Control Fluids	-	-
Boiloff Vented	298	135.2
PROPELLANTS	(29165)	(13229.3)
*Main Engine	(28580)	(12963.9)
Main Propellant - Fuel	4203	1906.1
Main Propellant - Oxidizer	24377	11055.3
Attitude Control Propellants	585	265.4
*FIRST IGNITION WEIGHT	(35267)	15997.1
SHUTTLE INTERFACE ACCOMM	(4112)	(1865.1)
Adapter Structure	(2420)	(1097.7)
Supports	1907	865.0
Attachment Fittings	216	98.0
Deployment Mechanism	297	134.7
Propellant Lines Umbilicals, Tankage	(1277)	(579.2)
Fill, Drain, Dump	310	140.6
Dump Pressurization System	967	438.6
Astrionics Interface	(415)	(188.2)
Shuttle/Payload in Bay	325	147.4
Mission Specialist Station	90	40.8
Bay Purge	-	-
Gases	-	-
Bottles	-	-
Payload Auxiliary Support	-	-
STAGE PAYLOAD	(4096)	(1858.0)
Payload	4096	1858.0
Payload Adapter	-	-
AUXILIARY STAGE	(2812)	(1275.5)
TOTAL WEIGHT IN SHUTTLE	(46287)	(20995.7)

$$\text{*Mass Fraction} = \frac{\text{Main Engine Propellants}}{\text{First Ignition Weight}} = 0.810$$

**Included in Main Propellants

CONFIGURATION SKETCH:



REMARKS:

1) Mission - 5 burn - 24 hours.
2) Engines - RL10-3-3 — 439.8 Isp.
3) ACPS - H_2O_2 (3)
4) Boost Pumps - H_2O_2 driven.
5) Propellant tanks ≈ 30,000 lb cap. Stainless Steel
6) Forward Adapter, Equipment Module, Aft Adapter } Skin-Stringer design
7) Insulation - Goldized Kapton Dimplar
8) Electric Power - Batteries
9) Pressurization - helium (3) ambient.
10) Contingency - 10% new items, 2% existing.

Ⓐ Alternate: 10% on all items

The pressurization system is largely made up of D-1T Centaur components with modifications for plumbing and control. The helium supply includes three large ambient helium bottles.

Existing H_2O_2 boost pump weights are used for the feed system weights. Other components are modified D-1T design.

The fill and drain system includes two 4.75-inch diameter (12.1 cm) LH_2 dump valves and two 4.0-inch diameter (10.2 cm) LO_2 dump valves.

The vent system includes the zero-g vent valves and associated plumbing and hardware, and quad redundant tanking boiloff valves.

Propellant management includes the propellant utilization system and the propellant loading system as presently used on the D-1T Centaur.

Thermal Control System

Insulation on the LH_2 forward dome and in the intermediate bulkhead is existing. Sidewall insulation of the propellant tanks is accomplished by using two goldized Kapton dimplar blankets each 3/4-inch (1.9 cm) thick. Covering this are three radiation shields, the inner one of which also acts as a purge bag. The weights include the insulation, purge bag, purge system, and all supports and seals required. The LO_2 tank insulation includes the fiberglass matting and spring ring bulkhead that form the lower half of the intermediate bulkhead.

Astrionics

Additional computer capability has been included and a tape recorder has been added for the data management part of avionics.

The IRU unit and the servo inverter unit are included. Sensor equipment has been added for guidance, navigation, and control. A backup docking control package is included.

A revised antenna system and component parts have been added as part of a dual communication system.

Electric power is obtained from a battery system sized according to calculated electrical loads.

The existing D-1T electrical distribution system and an instrumentation system have been included.

Contingency

The contingency weight for this vehicle has been defined as 10% of the listed component weight for new items and 2% for existing items, which is reasonable on a vehicle comprised mostly of existing hardware. An alternate allowance of 10% on all parts is also used to be consistent with all Tug studies.

Fluids

All propellant weights are based on mission requirements. Residuals and non-impulsive propellants weights are calculated values.

Shuttle Interface

The weights in this section include all required adapters, attachment fittings, and deployment mechanisms required to release and return the D-1S(R) vehicle to the Shuttle. Also included is a dump pressurization system with 73 pounds (33 kg) of helium and its associated plumbing and controls.

All vent and dumping interface requirements are included.

A weight allowance of 415 pounds (188.2 kg) is included for the astrionics interface.

4.2.2 VEHICLE WEIGHTS – RLTC. The RLTC vehicle weight summary is shown in Table 4-3. The weight summary is based on the following vehicle configuration conditions:

Structure

The avionics module, the forward adapter, and the aft adapter are all manufactured from graphite epoxy with integral parts sizing based on loads analyses.

The fuel tank and the LO_2 tank were sized for 52,500 pounds (23,814 kg) of propellants at 5:1 engine mixture ratio. Stainless steel skin thicknesses and ring sizing were based on loads analyses.

The intermediate bulkhead and the aft bulkhead are existing manufacture for the Centaur D-1T configuration.

The thrust structure is basic Centaur D-1T design and the equipment mounting structure is modified from the Centaur D-1T design. The interface equipment for the payload and the EOS docking mechanism is based on vehicle reusability.

Table 4-3. RLTC Weight Summary

CONCEPT:	MISSION:	CONFIG. NO.
RLTC	PAYLOAD DELIVERY TO SYNCHRONOUS ORBIT	RLTC-2 A

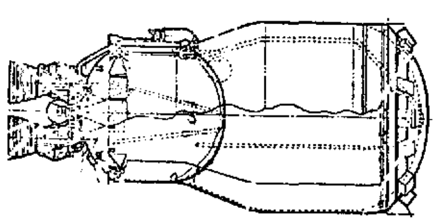
	LB	KG
Structures	(1950)	(884.5)
Body Structure	364	165.1
Fuel Tank & Supports	961	435.9
Oxidizer Tank & Supports	450	204.1
Thrust Structure	37	16.8
Equipment Mounting Structures	76	34.5
Meteoroid Structure	-	-
Interface for P/L & Shuttle	47	21.3
Umbilicals	15	6.8
Propulsion	(1652)	(794.3)
Main Engine (ME) Assembly	580	263.1
ME Actuation & Support	154	69.8
ACPS & Support	253	114.8
Pressurization	142	64.4
Fill, Vent & Drain (Incl. Feed)	459	208.2
Purge System	-	-
Propellant Management	64	29.0
Thermal Control	(508)	(230.4)
Fuel Tank Insulation	213	96.6
Oxidizer Tank Insulation	167	75.8
Purge System	122	55.3
Radiators (Inc. Fluid Loop, Heaters, Coatings)	6	2.7
Avionics	(900)	(408.3)
Rendezvous & Docking	-	-
Data Management	230	104.3
Flight Controls	178	80.8
Guidance & Navigation	64	29.0
Guidance Update	16	7.3
Power	93	42.2
Power Conversion & Distribution	220	99.8
Instrumentation	60	27.2
Communication	39	17.7
STAGE DRY WEIGHT	(5010)	(2272.6)
Contingency	500	226.8
TOTAL DRY WEIGHT	(5510)	(2499.3)
NON-USABLE FLUIDS	(823)	(373.3)
Trapped Propellants - Main	209	94.8
Trapped Propellants - ACPS	13	5.9
Trapped Gases	538	244.0
Trapped Helium	38	17.2
Propellant Leakage	-	-
Propellant Reserves - Main (264)	**	**
Propellant Reserves - ACPS	-	-
Propellant Utilization	24	10.9
Trapped Water	1	0.5
BURNOUT WEIGHT	(6333)	(2872.6)

	LB	KG
NON-IMPULSIVE EXPENDABLES	(1041)	(472.2)
ME Chilldown & Start	210	76.2
ME Leakage	75	19.5
ME Shutdown	72	49.0
Pressurants - Main	8	3.6
Pressurants - ACPS	-	-
Fuel Cell Reactants	53	24.1
Fuel Line Chilldown	-	-
Gas Generator	-	-
Thermal Control Fluids	-	-
Boiloff Vented	661	299.8
PROPELLANTS	(50794)	(23040.2)
*Main Engine	(50117)	(22733.1)
Main Propellant - Fuel	8353	3788.9
Main Propellant - Oxidizer	41764	18944.2
Attitude Control Propellants	677	307.1
*FIRST IGNITION WEIGHT	(58168)	(26385.0)
SHUTTLE INTERFACE ACCOM	(2331)	(1057.3)
Adapter Structure	(736)	333.8
Supports	453	205.4
Attachment Fittings	104	47.2
Deployment Mechanism	179	81.2
Propellant Lines Umbilicals, Tankage	(1235)	(560.2)
Fill, Drain, Dump	207	93.9
Dump Pressurization System	1028	466.3
Astrionics Interface	(360)	(163.3)
Shuttle/Payload in Bay	270	122.5
Mission Specialist Station	90	40.8
Bay Purge	-	-
Gases	-	-
Bottles	-	-
Payload Auxiliary Support	-	-
STAGE PAYLOAD	(4501)	(2041.7)
Payload	4501	2041.7
Payload Adapter	-	-
AUXILIARY STAGE	-	-
TOTAL WEIGHT IN SHUTTLE	(65000)	(29484.0)

$$\text{*Mass Fraction} = \frac{\text{Main Engine Propellants}}{\text{First Ignition Weight}} = 0.862$$

** Included in main propellants

CONFIGURATION SKETCH:



REMARKS:

1) Mission - 6 burn - 47 hours.
2) Engine - RL10-3-3 - 444 Isp.
3) ACPS - Hydrazine
4) Boost Pumps - Hydrazine
5) Propellant Tanks - 52500 lb capacity - Stainless Steel
6) Forward Adapter, Aft Adapter, Equip Module } Graphite Epoxy
7) Insulation - Goldized Kapton Dimplar
8) Electric Power - Fuel Cell
9) Contingency - 10% per NASA Directive

Propulsion

The main engine system, including engine actuation and support, is existing Centaur D-1T design.

The ACPS system is a redesigned, efficient N_2H_4 system based on "off-the-shelf" components and sized to mission conservative requirements. Two N_2H_4 tanks for the system are included, as well as the four thruster modules. Valving and control for the system are included in the system weights.

The pressurization system is largely made up of D-1T Centaur components with modifications for plumbing and control. The helium supply includes the cryo storage tank to increase the quantity of helium.

Hydrazine boost pump weights are used for the feed system weights. Other components are modified D-1T design.

The fill and drain system includes two 6.8-inch diameter (17.3 cm) LH_2 dump valves and two 5.2-inch diameter (13.2 cm) LO_2 dump valves for inflight dump in case of Orbiter abort.

The vent system includes the zero-g vent valves and associated plumbing and hardware, plus quad-redundant tanking boiloff valves.

Propellant management includes an elongated propellant utilization system and the propellant loading system as presently used on the D-1T Centaur.

Thermal Control System

Sidewall insulation of the propellant tanks is accomplished by using two goldized Kapton dimplar blankets each 3/4-inch (1.9 cm) thick. Covering this are three radiation shields, the inner one of which also acts as a purge bag. The weights include the insulation, purge bag, purge system, and all supports and seals required.

Astrionics

Additional computer capability has been included and a tape recorder has been added for data management.

Two strapdown IMU units and two servo inverters are included. Sensor equipment has been added for guidance, navigation, and control.

A revised antenna system and component parts have been added as part of a dual communication system. The DoD version is 22 pounds (10.0 kg) heavier due to encryption devices.

Electric power from a fuel cell has been added including heat exchanger and plumbing required for fuel cell operation.

A typical electrical distribution system and an instrumentation system have been included.

Contingency

The contingency weight for this vehicle has been defined as 10% of the vehicle dry weight. This was an MSFC ground rule for the STSS studies.

Fluids

All propellant weights are based on 6-burn, 47-hour mission requirements. Residuals and non-impulsive propellants weights are calculated values.

Shuttle Interface

The weights in this section include all required adapters, attachment fittings, and deployment mechanisms required to release and return the RLTC vehicle to the Shuttle. Also included is a dump pressurization system with 80 pounds (36.3 kg) of helium and its associated plumbing and controls.

All vent and dumping interface requirements are included.

A weight allowance of 360 pounds (163 kg) has been included for the astrionics interface, including a dedicated panel and computer on the Orbiter's mission specialist panel.

4.2.3 MASS PROPERTIES ANALYSIS. Mass property data for this study indicate no problems with vehicle weights or vehicle centers of gravity when installed in the Shuttle vehicle.

The RLTC and D-1S(R) Centaur vehicles will not exceed the Shuttle capability:

	RLTC	D-1S(R)
Shuttle Capability Weight, lb	65,000 (29,484 kg)	65,000 (29,484 kg)
Shuttle Installation Weight, lb	65,000 (29,484 kg)	46,287 (20,996 kg)

The installed RLTC and the D-1S(R) Centaur vehicle centers of gravity, including the necessary interface accommodations, fall within the stated Shuttle tolerance, as shown in Figure 4-2.

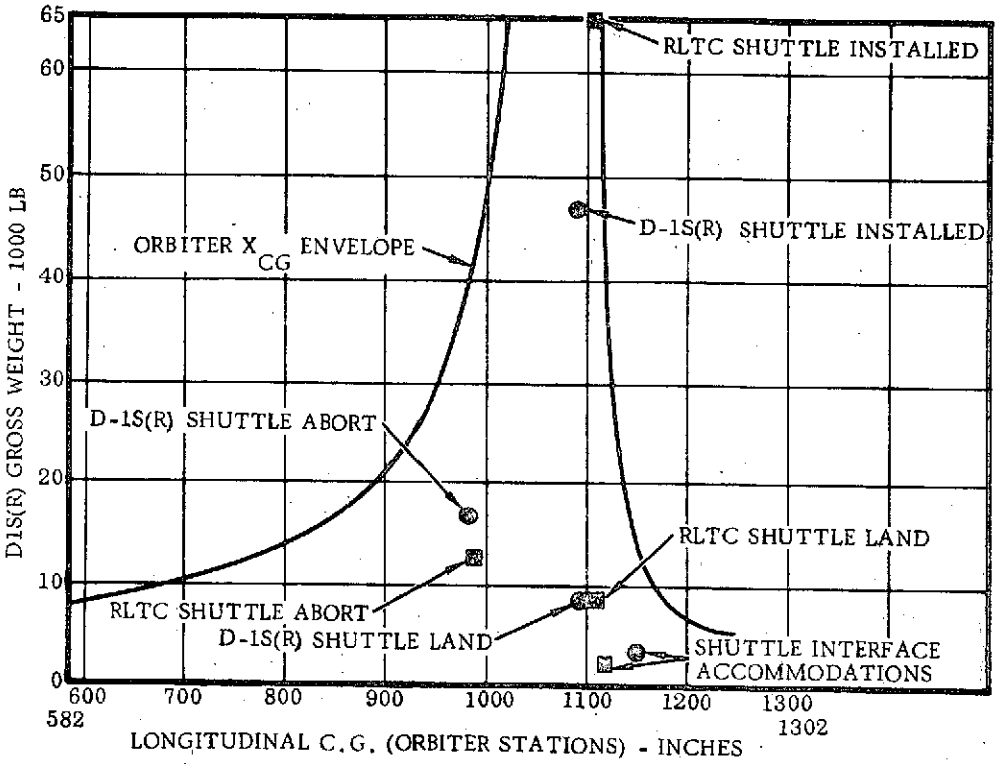


Figure 4-2. Location of Center of Gravity

4.3 PERFORMANCE ANALYSIS

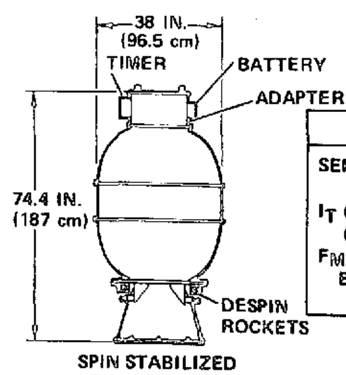
Performance calculations were based on the propulsion characteristics given in Table 4-4.

Table 4-4. Propulsion Characteristics

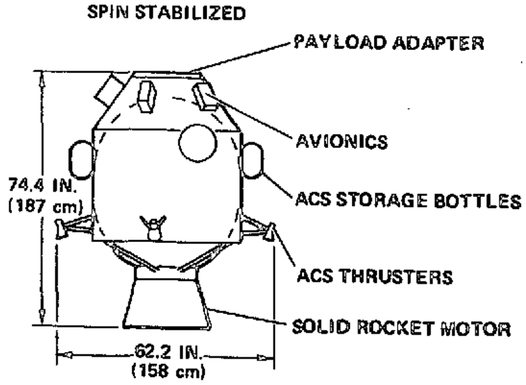
Characteristics	D-1S(R)	RLTC
Main Engine (2)	RL10A-3-3	RL10A-3-3
Thrust (Each Engine)	15,000 lb (66,723 newtons)	15,000 lb (66,723 newtons)
Specific Impulse	439.8 sec (4313 newton-sec/kg)	444 sec (4354 newton-sec/kg)
Mixture Ratio (O_2H_2)	5.8:1	5:1
Main Engine Propellant (Usable)	28,580 lb (12,963.9 kg)	50,117 lb (22,733.1 kg)
ACS Propellant	H_2O_2	N_2H_4 (hydrazine)
ACS Specific Impulse (Steady)	155 sec (1520 newton-sec/kg)	230 sec (2255 newton-sec/kg)
Flight Performance Reserve	1.73% each ΔV	1.73% each ΔV
ACS Propellant Reserve	5% usable H_2O_2	5% usable N_2H_4
Stage Length (Installed)	29 ft (8.8 m)	35 ft (10.7 m)

4.3.1 D-1S(R) PERFORMANCE. The D-1S(R) requires a kick stage velocity package to deploy payloads to synchronous equatorial orbits; D-1S(R) is reusable with the kick stage expended.

The two velocity package or solid rocket motor (SRM) configurations used with the Reusable Centaur Tugs are shown in Figure 4-3. The Thiokol TE-M-364-4 motor is used with both configurations; it is an existing motor flown with Delta and Centaur upper stages. The version used with Centaur is spin stabilized. The three-axis stabilized configuration is an adaptation of the Burner II design. Burner II uses the Thiokol TE-M-364-2 motor; it is a spherical motor with 1440 pounds (653 kg) of propellant. The TE-M-364-4 motor is essentially the same motor case with a 14-inch (35.4 cm) cylindrical section inserted between the hemispherical domes of the -2 motor case. The total propellant weight is 2300 pounds (1043 kg). The maximum thrust shown in Figure 4-3 is for the slow burn propellant grain being developed by Thiokol Corp. The slow burn grain burns for 150 seconds as compared to a burn time of 44 seconds for the current grain. The maximum thrust of the slow burn grain is 4800 pounds (21,351 newtons) resulting in longitudinal load of 1.3 g for a 3500-lb (1588 kg) payload.



ITEM	SPIN	3–AXIS
SEP. WT. (LB)	2,548	2,812
(KG)	(1,156)	(1,276)
I_T (1,000 LB-SEC)	654.5	654.5
(1,000 NEWTON-SEC)	(2911)	(2911)
F_{MAX} (LB) SLOW BURNING GRAIN	4,800	4,800
(NEWTON)	(21,351)	(21,351)



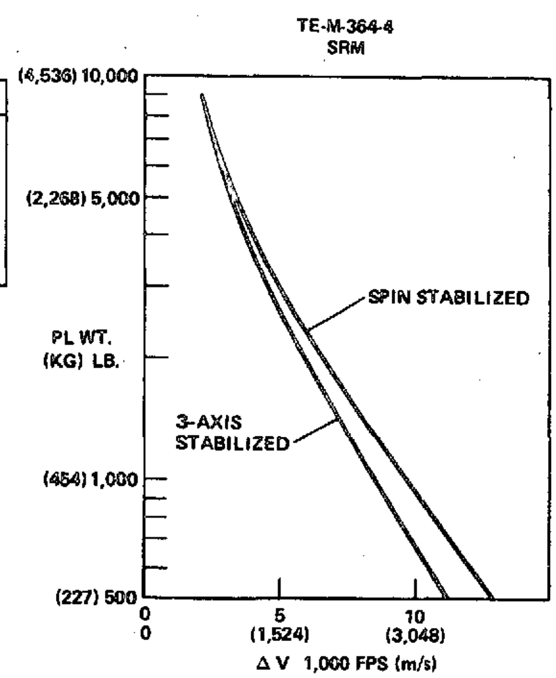
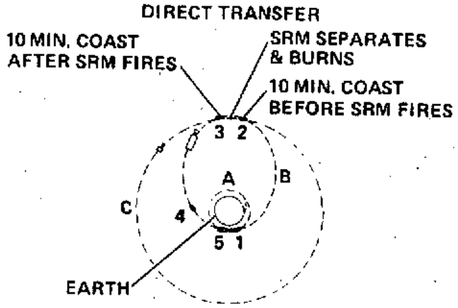


Figure 4-3. Velocity Package Performance

The D-1S(R) uses a trajectory that maximizes the delivery payload capability to synchronous orbit while maintaining minimum mission time as shown in Figure 4-4. A velocity package or solid rocket motor (SRM) is used with the D-1S(R) when payloads are deployed to synchronous orbit. Payloads have "walking" capability to position themselves into final longitudes and to make final orbital corrections. Therefore, no phasing orbit is required after deployment from the Shuttle. The D-1S(R) injects itself immediately into a transfer orbit whose altitude is slightly higher than synchronous orbit altitude. At the crossing of the transfer orbit with synchronous orbit altitude, the D-1S(R) performs its second burn to provide the initial increment of velocity necessary for circularizing the payload into synchronous orbit. The final increment of velocity for mission orbit insertion is provided by the SRM. After releasing the SRM and payload, the D-1S(R) reinserts itself into the original transfer orbit for return to Shuttle. Approximately 90 degrees from the equator, the D-1S(R) makes its fourth burn to align itself with the Shuttle orbit plane and to adjust mission time. At perigee, the D-1S(R) performs its fifth burn to circularize into the Shuttle orbit for recovery by the Orbiter. Total elapsed time from initial burn to the final burn is 10.6 hours. A phasing orbit is not required for the Shuttle to rendezvous with D-1S(R) since the total mission time as adjusted at the fourth burn to 10.6 hours is equal to seven orbits of the Shuttle.

The deployment mission mode is the only mission mode applicable to D-1S(R) geosynchronous equatorial orbits. Maximum payload deployment capability is dependent upon the stabilization requirement of the payload: spin or three-axis. A spin-stabilized

DIRECT TRANSFER



ORBIT SEQUENCE

A. SHUTTLE ORBIT 160 N.MI.
B. TUG TRANSFER ORBIT
C. TERMINAL ORBIT

TUG FLIGHT SEQUENCE

1. FIRST BURN - INJECT INTO TRANSFER ORBIT
2. SECOND BURN - APOGEE BURN, SEPARATE & INITIATE SRM BURN 10 MIN. AFTER BURN NO. 2
3. THIRD BURN - RETURN APOGEE BURN
4. FOURTH BURN - PLANE CHANGE & POSITION - VELOCITY UPDATE
5. FIFTH BURN - RETURN TO SHUTTLE ORBIT IF PHASING ORBIT REQD., SIXTH BURN NECESSARY

SEQ. NO.	EVENT	TIME (HR)		Δ V	
		DURATION	TOTAL	(FPS)	(m/s)
o	SHUTTLE LIFTOFF		0		
o	CIRCULARIZE -160 N.MI.		1.63		
o	COAST & DEPLOY TUG	2.54	4.17		
1	TOI	0.08	4.25	8150	2484
o	COAST	5.02	9.27		
2	Δ MOI	0.01	9.28	2125	648
o	SEPARATE SRM + PL & COMPLETE MOI			(3725)	1135
o	COAST	0.39	9.67		
3	RE-ENTER TO	0.01	9.68	2125	648
o	COAST	4.7	14.38		
4	PRECESSION PLANE CHANGE	–	14.38	700	214
o	COAST	0.4	14.78		
5	COI	0.02	14.8	8060	2457
o	RECOVERY	7.1	21.9		

SUMMARY OF SIGNIFICANT EVENTS
5- OR 6-BURN MISSION
ELAPSED TIME FROM 1 THROUGH 5 = 10.6 HR.
MISSION TIME DEPLOYMENT THROUGH RECOVERY < 24 HR.
SRM INJECTS P/L INTO TERMINAL ORBIT
P/L PERFORMS FINAL ORBIT PLACEMENT

Figure 4-4. D-1S(R) Mission

SRM in conjunction with D-1S(R) will place a maximum payload weight of 4349 pounds (1973 kg) into synchronous orbit, whereas the D-1S(R) plus the three-axis stabilized SRM will place 4096 pounds (1858 kg) into synchronous orbit. These values assume 10% dry weight contingency on new parts and 2% on existing, which comes out to be 245 pounds, or 5% as shown in Table 4-2. Using 10% contingency on the entire dry weight, the performance is 3925 pounds to synchronous equatorial orbit with a three-axis stabilized TE 364-4.

Figure 4-5 shows the D-1S(R) payload capability versus delta-velocity for each of the applicable mission modes. All of the curves presented are based on due-east launch except for polar payload capability which is 90 degrees inclination. In all cases the Shuttle is assumed to have a payload weight capability of 65,000 pounds (29,484 kg) to 160 n.mi. (296 km) circular orbit, due east, and 36,000 pounds (16,330 kg) polar launch to 160 n.mi. (296 km) circular orbit. Weight contingency is 5%.

The curve designated D-1S(R) is the performance capability of the D-1S(R) (only) in the reusable mode. It shows that the maximum velocity capability for a 500-pound (227-kg) payload is 12,300 fps (3749 m/s) or equivalent to a 13,000 n.mi. (24,076 km) circular orbit which is short of synchronous altitude. To achieve synchronous orbit capability a velocity package is required. Performance capability is shown for both a spin stabilized and a three-axis stabilized velocity package. Since the three-axis stabilized velocity package is heavier its performance is less than the spin stabilized unit.

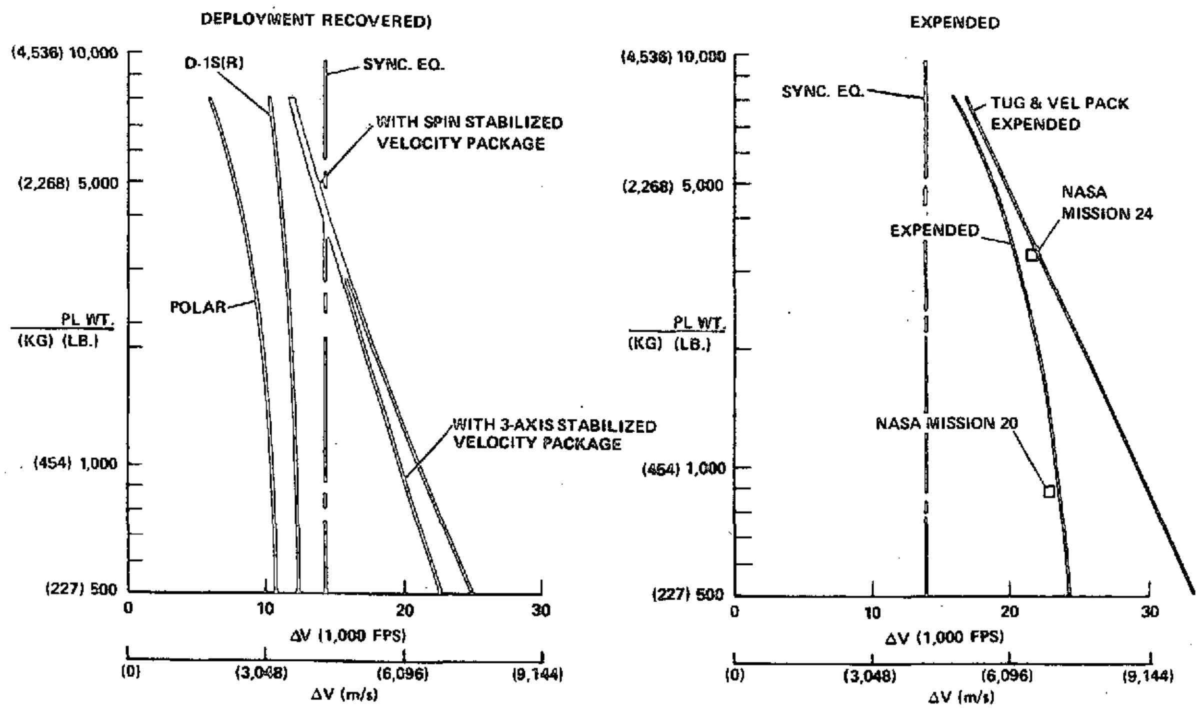


Figure 4-5. D-1S(R) Performance

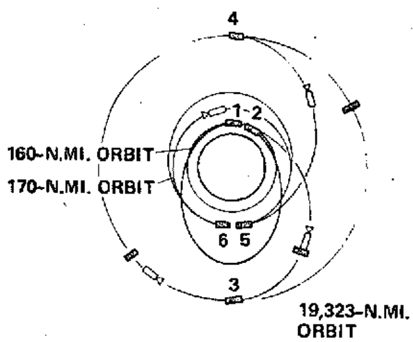
The D-1S(R) capability in the expended mode is shown since it is necessary to expend the D-1S(R) to accomplish some planetary missions. The six-year mission model specifies deployment of NASA Mission Number 20, a Saturn/Jupiter probe, and NASA Mission Number 24, a comet rendezvous. Both of these missions, as shown in Figure 4-5 require expenditure of D-1S(R) with Mission Number 24 requiring the expenditure of a velocity package as well.

4.3.2 <u>RLTC PERFORMANCE</u>. The synchronous equatorial deployment mission was used for sizing the RLTC. The hydrogen and oxygen tanks were sized to be full when deploying a nominal payload weight of 3500 pounds (1588 kg); they are offloaded slightly when carrying the maximum payload.

RLTC deployment of payload to synchronous equatorial orbit and its return for recovery by the Shuttle are presented in Figure 4-6 along with the timeline and the velocity budget. The nominal mission time is approximately 47 hours for the deployment mission from liftoff of the Shuttle to its touchdown at the landing site. This timeline is the maximum synchronous equatorial mission time period expected to be experienced by RLTC, since the worst case phasing requirement of about 14 hours was assumed to exist.

BURN NO.	SEQ. NO.	EVENT	TIME		
			COAST (Hr.)	BURN (MIN.)	TOTAL (Hr.)
		LIFTOFF			0
•	1	ORBITER BURNOUT		9	
•	1-2	COAST			
•	2	CIRCULARIZE (160 N.MI.)			1.63
•	2-3	COAST	13.11*		
1	3	PHASING ORBIT INJECTION (POI)		2.1	14.74
•	3-4	COAST	1.92		
2	4	TRANSFER ORBIT INJECTION (TOI)		5.0	16.66
•	4-5	COAST	5.27		
3	5	MISSION ORBIT INJECTION (MOI)		2.0	21.93
•	5-6	DEPLOY P/L	11.15		
4	6	TOI		1.4	33.08
•	6-7	COAST	5.27		
5	7	POI		0.7	38.35
•	7-8	COAST	3.02**		
6	8	COI		0.5	41.37
•	8-9	COAST RENDEZVOUS	4.53		
•	9	DEORBIT			45.90
•	10	TOUCHDOWN			46.60

* WORST CASE PHASING REQUIREMENT.
** MAXIMUM TIME IN PHASING ORBIT.



VELOCITY BUDGET: BURN	ΔV (FPS)	ΔV (m/s)
1	1832	558
2	6239	1902
3	5849	1783
4	5844	1781
5	3725	1135
6	4310	1314

FLIGHT PERFORMANCE:
RESERVE: 1.73% OF MISSION ΔV

Figure 4-6. Deliver Payload to Synchronous Equatorial Orbit and Return to Orbiter (RLTC)

The mission profile for delivery of payload to synchronous orbit is the standard trajectory employed for the Space Tug. Plane change requirement of 28.5 degrees was accomplished in two steps. A plane change angle of 2.2 degrees was performed at 160 n.mi. (296 km) altitude, distributed between the phasing orbit insertion and transfer orbit insertion burns such that no additional delta velocity is incurred as a result of the phasing orbit. The remainder of the 28.5-degree plane change is incorporated into the mission orbit insertion maneuver at apogee of the transfer orbit.

Figure 4-7 shows RLTC payload capability versus delta velocity for each of the five applicable mission modes. All of the curves are based on due-east launch except for polar payload capability which is 90 degrees inclination. In all cases the Shuttle is assumed to have a payload weight capability of 65,000 pounds (29,484 kg) to 160 n.mi. (296 km), due east, and 36,000 pounds (16,330 kg) polar to 160 n.mi. (296 km) circular orbit. The curves are based on a six-burn flight profile for earth orbit and three burns for the expendable RLTC curve. Maximum payload deployment capability to synchronous orbit is 4501 pounds (2042 kg) with RLTC return for maximum mission time from Shuttle liftoff through touchdown of 47 hours.

Where a velocity package is required, such as NASA Mission 20, the velocity package is expended. The velocity package has a gross weight of 2560 pounds (1161 kg) with a 0.9 mass fraction.

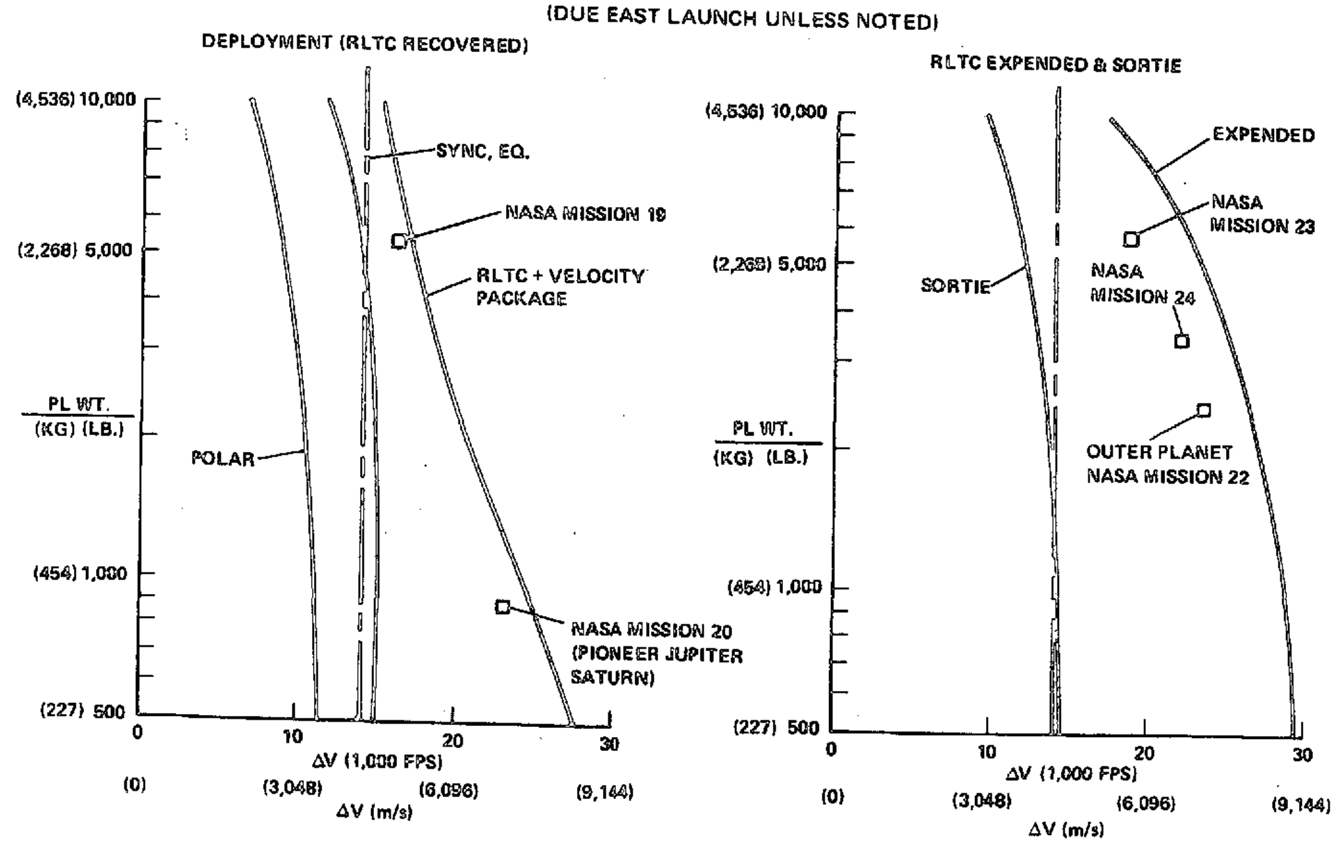


Figure 4-7. RLTC Performance

4.4 MISSION CAPTURE

The buildup in number of payloads deployed over the mission model years from 1980 through 1990 is shown in Figure 4-8. The mission model was defined for Program I in NASA/MSFC Memo PD-TUG-M (316-73) and the Tug flight buildup in NASA/MSFC Memo PD-TUG-M/P. The mission model is limited to deployment missions of NASA current-design payloads and DoD payloads, and the deployment/retrieval of sortie type payloads.

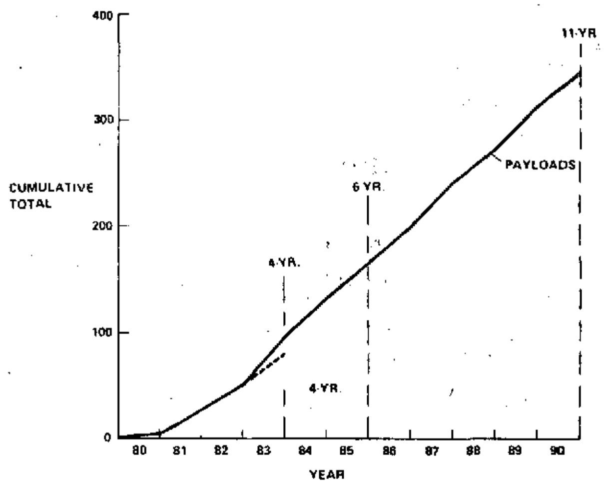


Figure 4-8. Mission Model Traffic

During the first year, Reusable Centaur missions are limited to three single payload flights. The maximum number of flights is limited to 21 during 1981. After 1980, multiple payloads can be deployed during a single flight. Four, six, and eleven year mission model durations are considered for RLTC and for D-1S(R). During 1983 the four-year model payload traffic curve diverges from the curve defined for six and eleven year models. This is the result of not activating WTR as a Shuttle launch site during 1983 for the four-year model, therefore, WTR payloads are not included in the four-year models. For six and eleven year models, WTR launches were included starting in 1983. Mission capture is 100% for both D-1S(R) and RLTC.

4.4.1 D-1S(R) MISSION ACCOMPLISHMENT. D-1S(R) flights for deployment and roundtrip missions are identified in Table 4-5 for the four, six, and eleven year mission models. Flights are subdivided into deployment flight modes: deployment of 1, 2, and 3 payloads during a single flight of a reusable D-1S(R), deployment of a single, planetary payload by an expended D-1S(R). The only type of roundtrip mission used is a sortie type, where a single payload is carried to orbit by the Tug, remains attached to the Tug during its on-orbit operational period, and returns with the Tug to the Shuttle Orbiter. Ground rules covering permissible flight modes are defined in NASA/MSFC PD-TUG-M (316-73).

Flight summary data for the six-year mission model is described as an introduction to the types of information presented for the four, six, and eleven year mission models in Table 4-5. Over the six-year mission model, 111 D-1S(R) flights are used to deploy 165 payloads; 19 of these are single payload flights by a reusable D-1S(R), two payloads are deployed on six flights, and three payloads are deployed on six D-1S(R) flights in

Table 4-5. D-1S(R) Mission Model Flight Summary

Tug Flight Mode	4-Year Model		6-Year Model		11-Year Model	
	Tug Flights	Payloads	Tug Flights	Payloads	Tug Flights	Payloads
Deployment						
o 1 Payload with Tug Return	12	12	19	19	35	35
o 2 Payloads with Tug Return	0	0	6	12	14	28
o 3 Payloads with Tug Return	0	0	6	18	12	36
o Tug Return & Velocity Package	42	65	72	106	146	128
o Expended Tug	2	2	6	6	17	17
Total	56	79	109	161	224	334
Retrieval	0	0	0	0	0	0
Round Trip (Sortie)						
o 1 Payload Up/1 Payload Down	0	0	2	4	4	8
Totals	56	79	111	165	228	342
Average Payloads Per Flight	1.41		1.49		1.5	

the reusable mode. Seventy-two flights are accomplished with velocity packages with D-1S(R) recovered, and six expended D-1S(R) flights are required. There are no retrieval missions and two sortie missions are accomplished (two sortie missions are counted as a total of four payload deployments/retrievals — two deployments and two retrievals).

The six-year mission model (161 deployment payloads, 4 sortie payloads) is accomplished using 111 D-1S(R) and Shuttle flights. Since no Tug-to-Tug or velocity package-to-Tug on-orbit assemblies are required, the number of D-1S(R) and Shuttle flights is identical. The average number of payloads per flight is 1.49.

4.4.2 <u>RLTC MISSION ACCOMPLISHMENT.</u> RLTC flights for deployment, retrieval, and roundtrip missions are identified in Table 4-6 for the four, six, and eleven year mission models. Over the 11-year mission model, 59 RLTC flights deploy two payloads, and 29 flights deploy three payloads, on a single reusable flight. Seven flights are accomplished with velocity packages, and 10 expended RLTC flights are required. There are no retrieval missions and four sortie missions are accomplished (four sortie missions are counted as a total of eight payload deployments/retrievals — four deployments and four retrievals).

The 11-year mission model (334 deployment payloads, 8 sortie payloads) is accomplished using 221 RLTC and Shuttle flights. Since no Tug-to-Tug or velocity package-to-Tug on-orbit assemblies are required, the number of RLTC and Shuttle flights is identical. The average number of payloads per flight is 1.55.

Table 4-6. RLTC Mission Model Flight Summary

TUG FLIGHT MODE	4-YR. MODEL		6-YR. MODEL		11-YR. MODEL	
	TUG FLT	PL	TUG FLT	PL	TUG FLT	PL
DEPLOYMENT						
• 1 PAYLOAD W/TUG RETURN	33	33	56	56	112	112
• 2 PAYLOADS W/TUG RETURN	13	26	27	54	59	118
• 3 PAYLOADS W/TUG RETURN	6	18	15	45	29	87
• TUG RETURN & VEL PACKAGE	2	2	4	4	7	7
• EXPENDED TUG	0	0	2	2	10	10
TOTAL	54	79	104	161	217	334
RETRIEVAL	0	0	0	0	0	0
ROUND TRIP (SORTIE)						
• 1 PAYLOAD UP/1 PAYLOAD DN	0	0	2	4	4	8
TOTALS	54	79	106	165	221	342
AVERAGE PAYLOADS/FLIGHT	1.46		1.56		1.55	

There is a direct relation between the mission model, initial upper stage/OOS capability, and total program costs. The mission model used here is already obsolete, but the inherent high performance of cryogenic propellants will provide high mission capture for Centaur against any mission model.

The performance margins inherent to the Reusable Centaur result in low risk concepts. The performance margin instills a high confidence in meeting performance requirements with ample weight margins to cover unforseen weight increases for reliability improvement; provides backup capability should the Shuttle performance goals be lowered; and has sufficient weight margin to cover added equipment for improved maintenance and checkout procedures.

This performance margin also results in increased mission flexibility. It can be traded for: payload placement in final orbits, increased mission duration, growth to retrieval capability and plane changes for launching polar payloads from KSC. Flight operations complexity is also reduced by the elimination of on-orbit assemblies and the minimal need for multiple stages and auxiliary hardware (kick stages, drop tanks, etc.). Multiple payload capability reduces Shuttle flights by 30 to 40% with the resultant increase in Shuttle flight planning, flexibility and reduction in operations costs.

SECTION 5

FLIGHT OPERATIONS

Results of major flight operations analyses relative to Reusable Centaur are summarized covering ground/on board responsibilities, impact of autonomy, communications requirements, communications coverage, crew and Orbiter interfaces, and deployment/retrieval operations and inflight abort. A single description is utilized where study results apply equally to the RLTC and the D-1S(R); if significant differences exist, separate summaries are furnished.

Based on overall program cost trades, the RLTC was designed to function with Level I autonomy, essentially independent of ground support during upper stage flight except for monitoring and command override. Low operations complexity results from the high performance capability which allows multiple payload delivery with minimal use of velocity packages.

D-1S(R) as a minimum modification to the existing D-1T Centaur retains a simplex astrionics control system and has Level III autonomy requiring some ground support. This reflects a vehicle option of low development cost and proven hardware simplicity. The D-1S(R) with the current size tank has more flight operations complexity than the RLTC due to the limited performance capability.

Exploration of the crew and Orbiter interface with Centaur has resulted in the recommendation of a Centaur monitor and control system (CMACS) to enable direct usage of the existing Centaur computer-controlled launch set (CCLS) technology and software library for lower interface development costs. Implementation of the Centaur/Orbiter interfaces as they will be used for deployment and retrieval of the upper stage is also reviewed.

It is tentatively recommended that both main propellants be dumped from Centaur inflight after an Orbiter abort.

5.1 GROUND/ON BOARD FUNCTIONAL FLOW RESPONSIBILITY

NASA and DoD missions from both ETR and WTR are included. Primary drivers are Orbiter safety and the selected upper stage level. The Air Force will be the executive agent and mission operating agency for the DoD and all DoD users, and will be the launch agency at VAFB. NASA will be the executive agent and mission operating agency for all users other than DoD and will be the launch agency at KSC. However, NASA and DoD will each use both operational sites. Vehicle operations management control will be assumed by the operating agency at hold down release, or start of flight operations, regardless of the launch site.

The basic operating responsibility for flight operations is therefore independent of launch site and dependent on payload user. The following delegation of flight operations responsibilities is applicable for both users. The difference between users is that the NASA flight operations controller is located at the NASA Operations Management Center (OMC) and uses the facilities of the STDN while the Dod flight operations controller is located at the DoD Operations Management Center in the Satellite Test Center Sunnyvale and uses the Air Force Satellite Control Facility network.

Flight operations for each user start at liftoff and end at Shuttle landing. Figures 5-1, 5-2, and 5-3 define the top level functional flow for the flight operations phase. The basic missions included are placement, service, and sortie with Reusable Centaur and placement with expended Centaur. For all missions the Centaur has two operational control modes during flight; Orbiter control by the mission specialist and OMC control by the flight operations controller.

During the time the Centaur is in the Orbiter and adjacent to the Orbiter following Centaur/Orbiter separation and prior to enable/activation the mission specialist in the Orbiter is in control.

The flight operations controller at OMC is available to support the mission specialist during this time if communications permit. This time phase is covered by Functions 1.1 through 1.4 on Figure 5-1. The remainder of the flight, following Centaur enable/activation, to the mission orbit and back to injection into the Shuttle rendezvous orbit is monitored by flight controller at OMC with the launch control facility available for support. This time phase is covered by Functions 1.5 through 1.42 on Figures 5-1, 5-2, and 5-3 for the defined missions. The control of the stage is then transferred back to the mission specialist in the Orbiter for the remainder of the flight until Orbiter landing, Function 1.46.

This assignment of responsibility is based on Orbiter safety requirements, and the level of autonomy of the Centaur. The mission specialist's primary function is to assure the safety of the Orbiter and crew during Shuttle ascent and descent. In addition, the mission specialist performs a final subsystem status check prior to enable/activation. At this point the Centaur becomes autonomous and performs the mission. RLTC Level I autonomy capability as defined on Figure 5-4 results in the Centaur flight controller at OMC being primarily in a monitoring role receiving mission status information. The communications link is therefore primarily for information transfer down. There is onboard contingency planning for reasonable payload alternative action so the ground uplink use would be limited to initiation of a pre-programmed alternate mission or mission termination. Maintenance data is recorded onboard by a mass storage unit for playback following landing in support of ground turnaround activity. Autonomy Level III for D-1S(R) requires OMC to provide position and velocity update information during flight. (See Figure 5-5 for the summary of RLTC and D-1S(R) ground support differences.) Therefore, during D-1S(R) missions the Centaur flight controller and support personnel will provide commands when required.

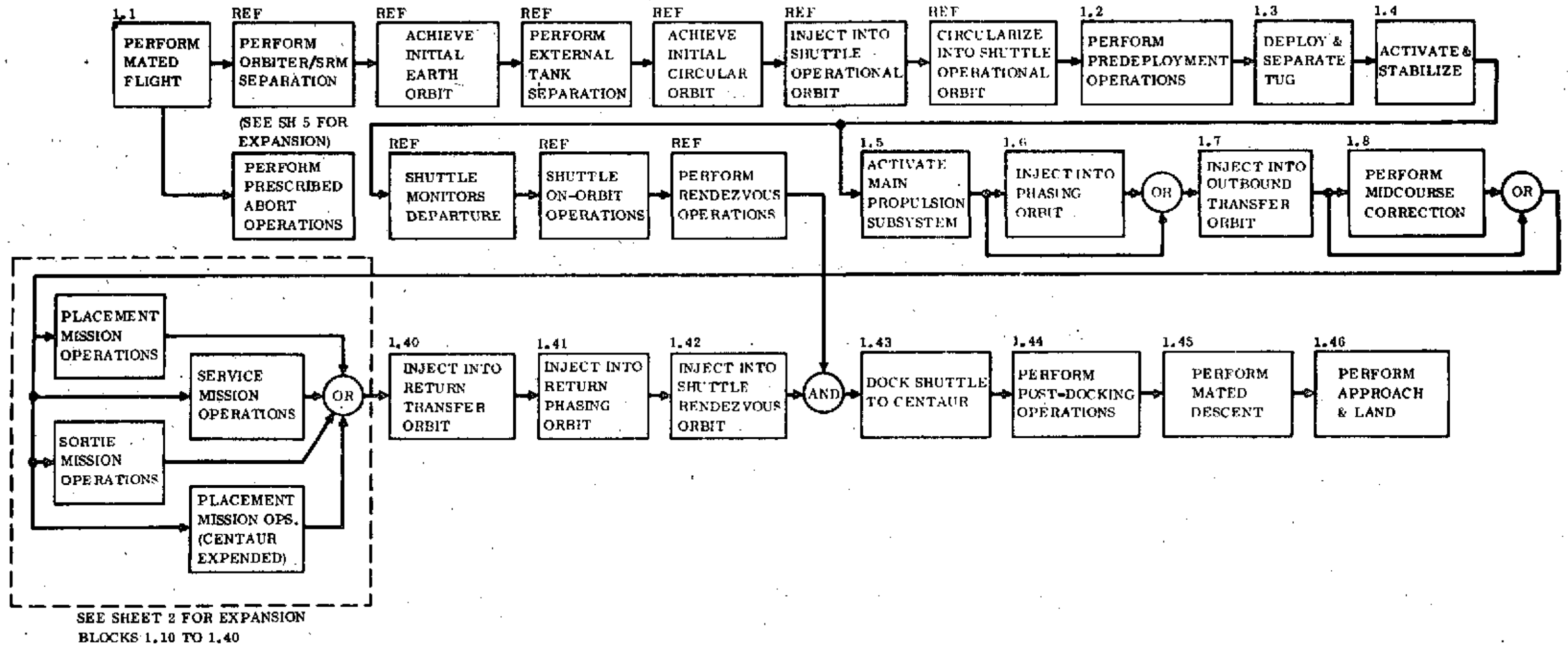


Figure 5-1. Tug Flight Operations

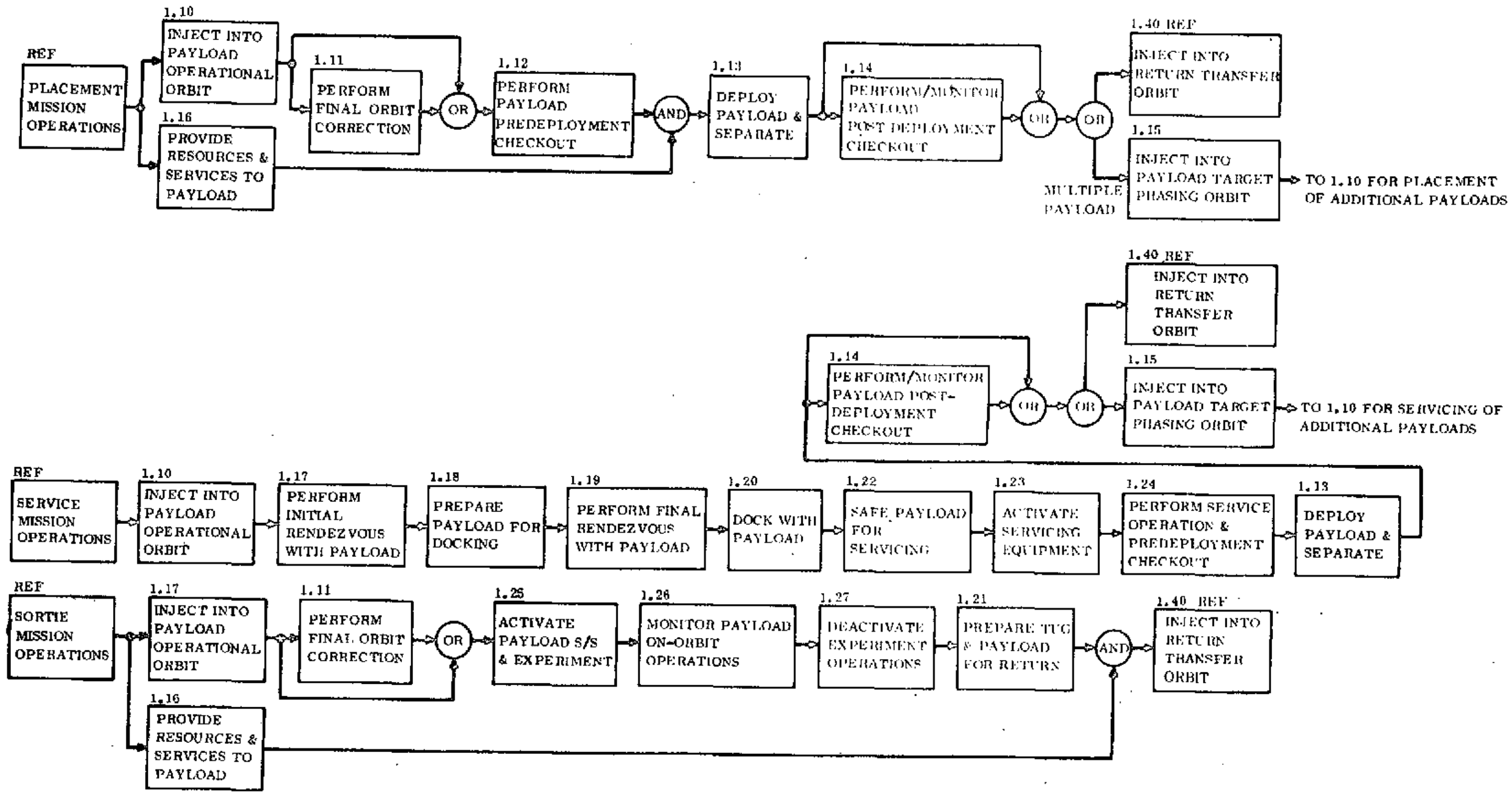


Figure 5-2. Flight Operations - RLTC

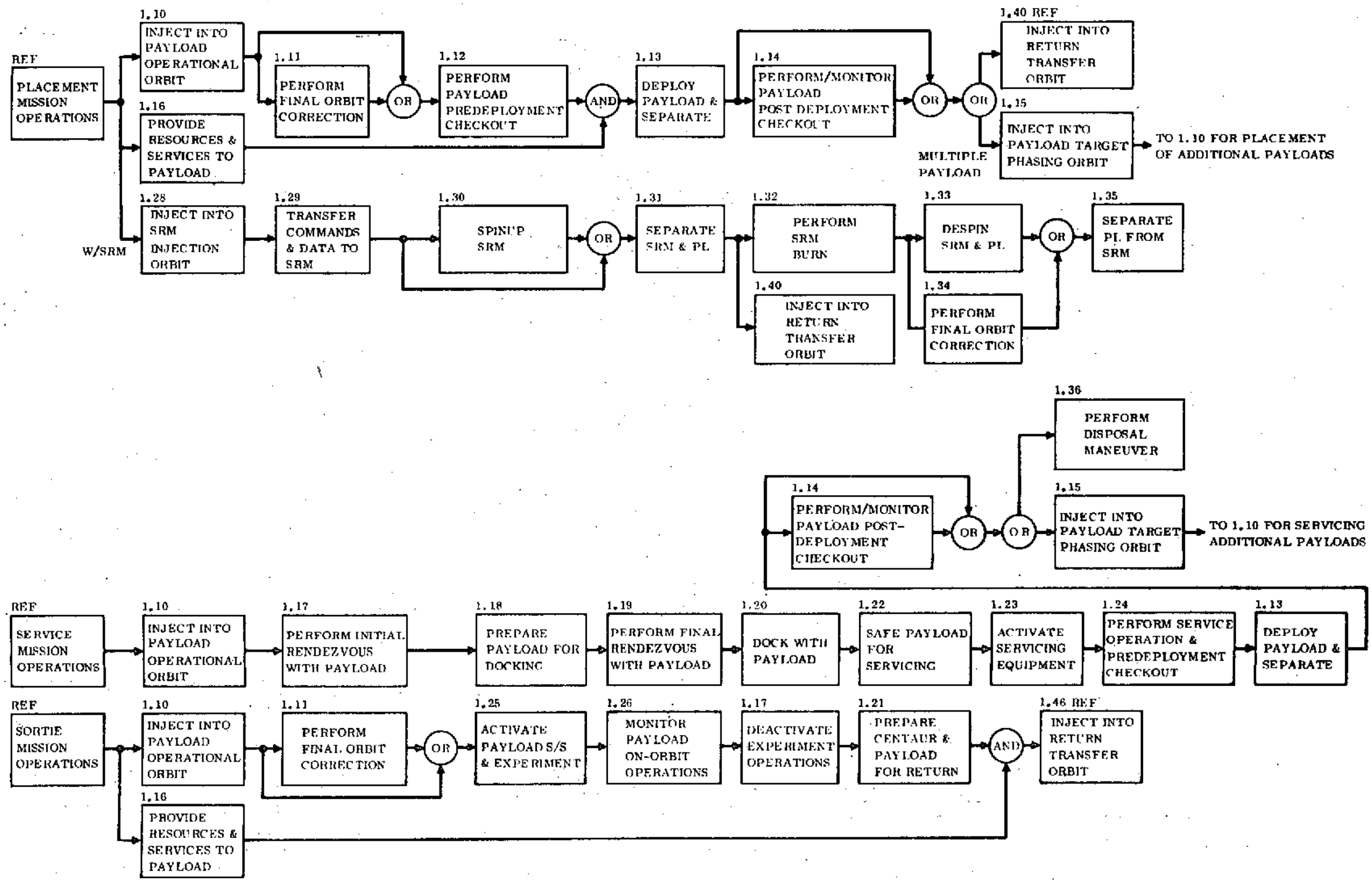


Figure 5-3. Flight Operations - D-1S(R)

	LEVEL IV	LEVEL III	LEVEL II	LEVEL I
DATA MANAGEMENT SUBSYSTEM	ONBOARD CONTROL LIMITED TO PRESET REFERENCE TRAJECTORY	COMPLETE ONBOARD TRAJECTORY CONTROL		
			UPLINK COMMAND REDIRECTION & RETARGETING	
	SOFTWARE: *	SOFTWARE: *	SOFTWARE: *	SOFTWARE: *
GUIDANCE & NAVIGATION SUBSYSTEM	POSITION/VELOCITY UPDATE FROM GROUND		POSITION/VELOCITY DATA FROM NAVIGATION SATELLITE	STATE VECTOR UPDATE FROM GUIDANCE UPDATE SUBSYSTEM
GUIDANCE UPDATE SUBSYSTEM	ATTITUDE UPDATE ONLY			POSITION, VELOCITY & ATTITUDE UPDATE USING ONBOARD SENSORS & COMPUTER CONTROL
COMMUNICATION SUBSYSTEM	MCC EVENTS, BURN DURATIONS ATTITUDE VECTORS DATA FROM GROUND	POSITION/VELOCITY DATA FROM GROUND	POSITION/VELOCITY DATA FROM GROUND IF AVAILABLE	
	COMMAND UPLINK/TELEMETRY DOWNLINK			

REUSABLE CENTAUR GOAL: LEVEL I AUTONOMY

* SOFTWARE IS A FUNCTION OF EACH SYSTEM CONFIGURATION.

Figure 5-4. Autonomy

RLTC CHARACTERISTICS

- LEVEL I AUTONOMY OPERATES INDEPENDENTLY
- MAINTENANCE DATA STORED ON BOARD
- PAYLOAD CHECKOUT NOT REQUIRED

D-1S(R) CHARACTERISTICS

- LEVEL III AUTONOMY DEPENDENT ON GROUND TRACKING
- MAINTENANCE DATA STORED ON BOARD
- PAYLOAD CHECKOUT NOT REQUIRED

CONCLUSIONS

- RLTC
  - NETWORK NOT REQUIRED TO ACCOMPLISH MISSION
  - NETWORK USED ON LOW PRIORITY BASIS FOR STATUS
  - COMMAND UPLINK OVERRIDE CAPABILITY EXISTS
- D-1S(R)
  - NETWORK PROVIDES POSITION & VELOCITY DATA
  - COMMAND UPLINK FOR POSITION & VELOCITY CORRECTION

Figure 5-5. Ground Support During Tug Flight

Payload responsibility is transferred to the payload user after separation from the Centaur. The current mission capture ground rules prevent return of a faulty payload from mission orbit during the placement flight. Mission capture would have to be based on round-trip capability to allow return of faulty payloads on the placement flight. Mission capture is currently based on utilization of the maximum payload placement capability of the Centaur by the use of multiple payload deliveries. This means the payload checkout Functions 1.12 and 1.14 are potentials for deletion from the functional flow on most missions. The high payload capability of the RLTC provides a payload round-trip capability if mission orbit checkout is desired by the payload user.

Following return of the Centaur to the Shuttle rendezvous orbit the control is transferred from the flight controller at OMC back to the mission specialist in the Orbiter for docking activity, system safing and statusing, and return to the launch site.

5.2 COMMUNICATIONS

5.2.1 COMMUNICATIONS VERSUS AUTONOMY. Lesser autonomy than fully independent Level I can offer material savings in onboard vehicle complexity at the price of increased ground support. Vehicle simplicity and lower development costs can be gained by maintaining greater operations support from ground stations. This impacts vehicle design in the areas of onboard checkout provision, redundancy management, redundancy levels, the amount of in flight software, and guidance update hardware.

The RLTC has sufficient dual redundancy and reconfiguration capability, driven by mission duration, reliability goals, and safety, to enable overall program cost savings with a Level I autonomy system. The trade study comparing Level I, II, III, and IV autonomy systems was done in the related Convair STSS study, NAS 8-29676, and showed for Program 1 comparable to RLTC an overall savings of $51.1M from reduced operations costs over the 11-year program.

The D-1S(R) with Level III autonomy represents a minimum modification to the existing D-1T Centaur stage for reusability and preserves the majority of present subsystems. Minimum design changes and development result from extending the current Centaur system capability to the new missions and in providing the needed additional flight support from the ground. This sustains simplicity in the vehicle design, but will increase recurring flight operations costs over RLTC for the same number of flights.

Increased operations costs are principally in communications support for the lower autonomy level and pertain to vehicle tracking, guidance update via RF uplink, added ground station console facilities, added software on the ground, and the provision of global communications coverage for the various DoD and NASA mission orbits. Low altitude polar orbits over the southern hemisphere are troublesome due to limitations in ground station locations. Relay satellites, such as TDRS, are needed for full coverage during some orbits.

D-1S(R) is a compromise striving for simplicity of the vehicle onboard system with reasonable ground flight support and differs from a more costly totally independent vehicle such as RLTC.

5.2.2 COMMUNICATIONS COVERAGE. Representative Reusable Centaur missions were analyzed to determine the availability of communications coverage by the NASA 15-station ground network and of the AF ground network which can support D-1S(R) and RLTC missions. For the NASA missions, two mission models were analyzed for the D-1S(R) and four for the RLTC. RLTC results are similar to those for Program 1 of the STSS Data Dump, September 1973, contract NAS 8-29676, and appear in Section 6.8 of Volume 6 for that study. Autonomy Level I for RLTC and for the Program 1 Tug impose minimal ground communications requirements and minimal coverage needs. For DoD missions and the AF ground network data, the classified information is contained in a secret addendum to STSS NAS 8-29676 as a part of the September Data Dump which included both STSS and Reusable Centaur data (NAS 8-30290) in the single volume. See also Volume 6, Operations, September Data Dump, NAS 8-30290, Appendix C.

Two specific D-1S(R) missions covering combinations of high and low orbital inclinations and altitudes were studied. This analysis supported cost trade studies relating to the degree of automation and the degree of autonomy versus ground support required during a mission. It was determined that the Centaur would be moderately autonomous, Level III. As such, the D-1S(R) does need ground support provided by the NASA or DoD networks to perform the normal missions.

The overall communications problem in simplified form is depicted in Figure 5-6. The functions that communications support are (1) tracking to determine and predict D-1S(R) position and velocity, (2) telemetry to assess the D-1S(R) status, and (3) command which allows override of the on board vehicle functions from external sources. Relating these functions to the overall communications problem limited the coverage analysis to a primary ground domain, and secondary considerations to the Shuttle, tracking data relay satellite (TDRS), and the payload. Rationale for this was simply that all assessment and command emanates from the ground sources and typically the other elements are of a relay nature. This was further supported in the cost area by the considerable costs of the ground elements and the recurring nature.

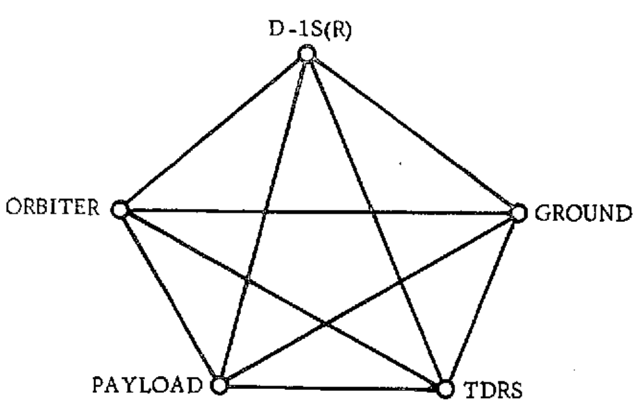


Figure 5-6. Communications Problem

The fifteen ground stations identified by NASA for the study are listed in Table 5-1

Comparable Air Force Satellite Control Facility (AFSCF) network sites are shown in Table 5-2. This network consists of ten stations located at seven locations and the

Table 5-1. NASA Integrated Network Stations

Location	Designator	Chart Code
Merritt Island	MIL	MIL 3
Bermuda	BDA	BDA 3
Canary Island	CYI	CYI 3
Ascension	ASC	ACN 3
Guam	GWM	GWM 3
Hawaii	HAW	HAW 3
Goldstone/Mojave	GDS	GDS 8
Orroral/Honeysuckle	ORR/HSC	ORR 3
Madrid	MAD	MAD 8
Santiago	AGO	SAN 4
Rosman	ROS	R835
Alaska	ULA	ULAS
Tanan Arive	TAN	T403
Johannesberg	BUR	BUR 4
Quito	QUI	Q403

Table 5-2. AFSCF Network Site Locations

Location	Designator	Lat	Long
New Hampshire	NHS	42° 56.9'	288° 22.4'
Vandenberg	VTS	34° 49.4'	239° 29.9'
Hawaii	HTS	21° 34.3'	201° 44.1'
Kodiak	KTS	57° 35.9'	207° 49.3'
Indian Ocean	IOS	-4° 40.0'	55° 28.8'
Guam	GTS	13° 36.8'	144° 51.2'
(Classified)	OL-5	(Classified)	

Satellite Test Center (STC) located at Sunnyvale. Radar tracking is supplied from Air Defense Command support and from the ETR and WTR networks when required.

The presentation of mission segments and the communication contacts within each was on a contact opportunity (CO-OP) chart; see Figure 5-7 for a typical chart. Computerized analysis was used to develop a series of charts for the sample missions. A summary of the communications coverage and gaps for the two D-1S(R) NASA missions is shown in Table 5-3. Dod charts and tables appear in the referenced secret addendum published as part of STSS, NAS 8-29676.

A worst case ground network load could be imposed because of D-1S(R). With both ETR and WTR operational, concurrent missions could occur, thereby requiring simultaneous handling.

With the 15-station NASA network, the syndrome "perigee gap" becomes commonplace. Where perigee is below 3,000 km, the D-1S(R) did not in the analysis have coverage prior to, during, or after a thrusting event. The best tracking data for orbit determination and prediction occurs during a perigee excursion because of good time variant changes in geometry.

Overlap of D-1S(R) and Orbiter communications opportunities were analyzed. The superposition of D-1S(R) events on the Orbiter time line and CO-OP charts showed that very few concurrent communication opportunities would exist. Exception is orbit concurrency during deployment and retrieval of the upper stage by the Orbiter. Orbit concurrency and opportunity overlap also exist for the D-1S(R) and its payload during placement missions.

Table 5-3. NASA Network Coverage Summary, D-1S(R) (Without TDRS)

Mission	Per Cent Coverage	Mean Gap (Min.)	Max Gap (Min.)
Polar 90° Inclination Low Altitude	26.3	29.46	118.71
Synchronous Equatorial 28.5° Inclination	68.6	12.29	39.88

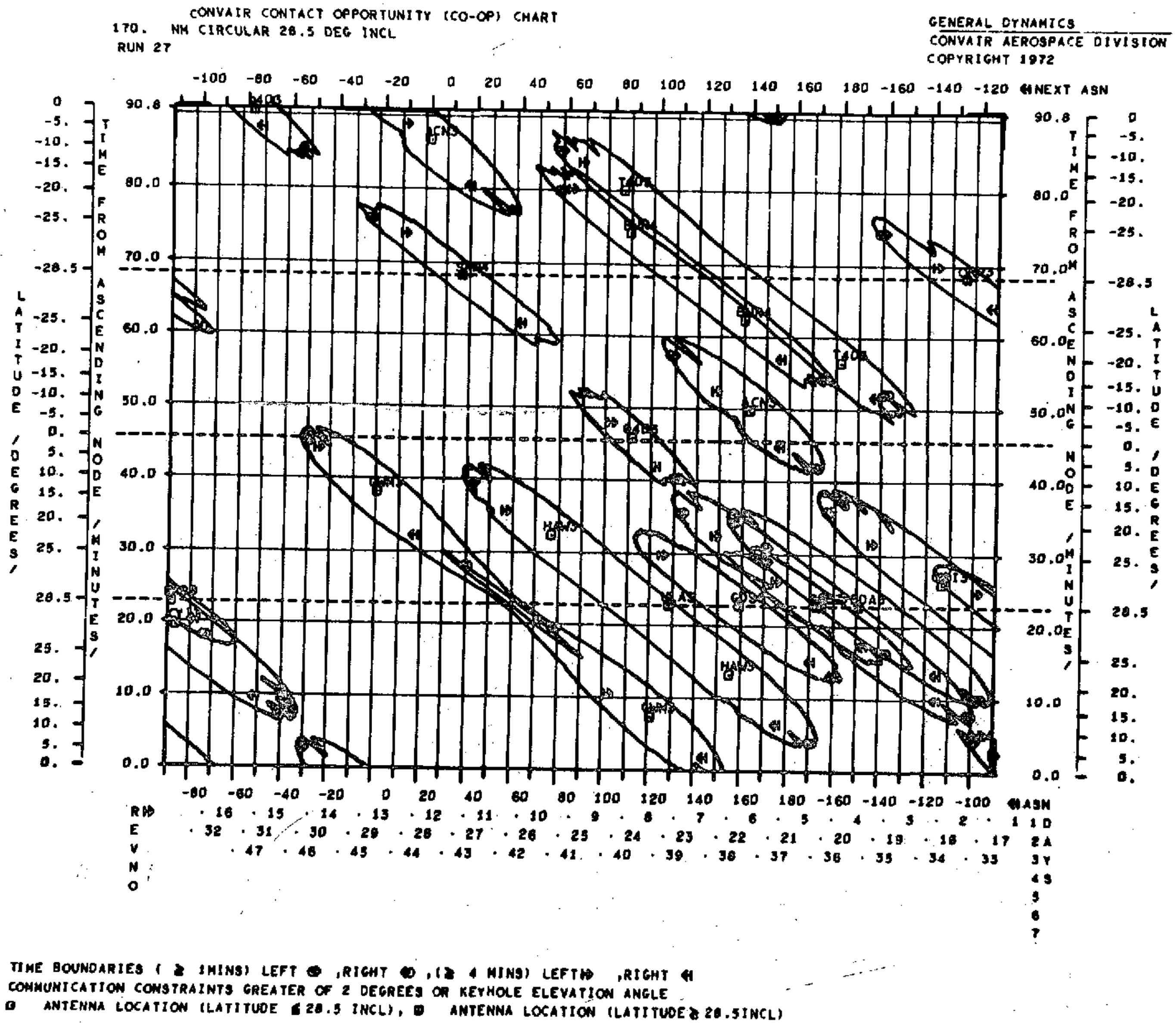


Figure 5-7. Convair Contact Opportunity Chart

With five NASA ground stations and TDRS, 100% coverage can be provided except at altitudes lower than 160 n.mi. (257 km). Above 5000 km communication gaps up to 230 minutes were encountered between the 5000 and the 10,000 km altitudes within inclinations of 20 to 70 degrees and from 110 to 160 degrees. TDRS ground rules also impose an occultation situation in the gap area at 80° E longitude meridian. A low altitude D-1S(R) could be transmitting or receiving at an earth grazing line-of-site vector which would incur the attenuation and distortion of 14 equivalent atmospheres. This latter situation could be avoided by increasing the size of the TDRS gap slightly.

An overview comparison of the combined NASA and AF ground station networks shows overlapping coverage in the lower northern latitudes and equatorial regions. The high northern latitudes are better covered by the AF while the southern hemisphere is partially covered by NASA.

In summary, it is recommended that TDRS network options be used to furnish improved coverage for both Air Force and NASA missions at low altitude and to support the Level III autonomy for the D-1S(R).

5.3 CREW AND ORBITER INTERFACES

The Shuttle Orbiter cabin will accommodate a crew of four: a pilot, copilot, a mission specialist and a payload specialist. The pilot and copilot have overall responsibility for vehicle flight operations, caution and warning monitoring, and critical payload interfaces. The payload specialist has responsibility for vehicle flight operations, caution and warning monitoring, and critical payload interfaces. The payload specialist has responsibility for payload instrument calibration, checkout, monitor, and control. The mission specialist has responsibility for caution and warning monitoring and control of the Orbiter subsystem/payload interfaces. It is assumed during Centaur flights that the mission specialist will be assigned to the Centaur functions and the payload specialist will be assigned to the payload functions.

Centaur functions assigned to the mission specialist include insuring the safety of the crew and Orbiter, verification of mission readiness, and providing control for mission preparation, deployment, separation, and safing following the mission and abort. To do this requires the following major equipment: real time processor, mass storage for software and data, a CRT display and keyboard, and caution and warning displays. Shuttle provision of these devices is shown in Figure 5-8. The Centaur Monitoring and Control System (CMACS) discussed in Reference 1 provides an alternative concept.

Based upon original ground rules for NASA studies of Centaur/Shuttle Integration, NAS 3-16786, an independent Centaur Monitor and Control System (CMACS) was defined for the Orbiter man-machine and data interface. Convair's direct experience with Centaur ground support equipment in the form of the CCLS was adapted for flight usage. This concept of the CMACS configuration and the related costs were carried into the STSS study, NAS 8-29676, and into this Reusable Centaur Study.

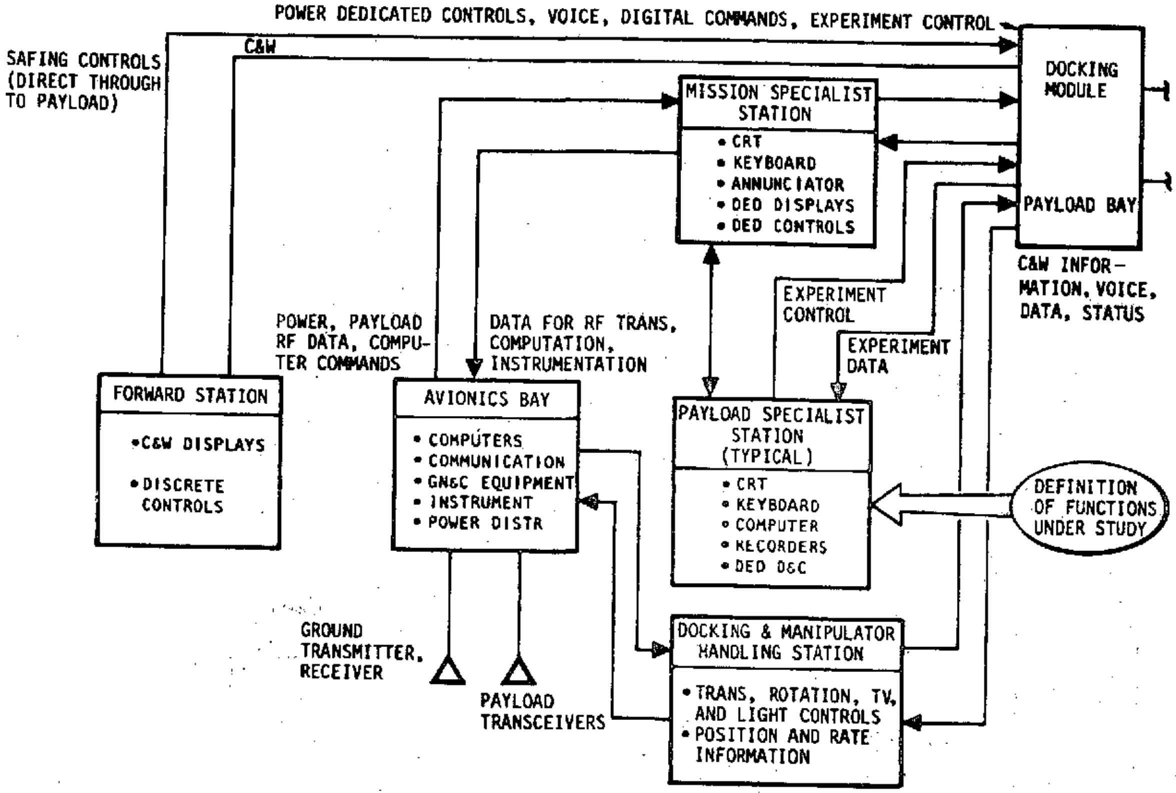


Figure 5-8. Shuttle Specialist Provisions

While the Orbiter is presently planning to provide 10,000 words of program in a shared Orbiter computer, the independent approach enables the definition and accurate costing of the display, keyboard, computer electronics, etc., that are essential for the interfacing of any upper stage as related to the actual support experience for Centaur.

Figure 5-9 shows the functional block diagram for the D-1S(R) pallet/deployment adapter interface. CMACS as shown includes a 16,000-word memory computer which is independent, but which tends to duplicate the shared and assigned 10,000 word Orbiter commitment. A majority of the Centaur CCLS checkout program library is available for direct use, and without major problems in software language which could result from residence in another computer system. The RLTC interface is similar with only minor variations.

Use of the Centaur within the Shuttle requires that the Orbiter crew be provided the capability to ensure the safety of the Shuttle and crew, to verify status for mission readiness, and to control and monitor operations, including mission preparation, deployment, separation, and abort. The baseline design includes all hardware and software functions necessary to perform these functions.

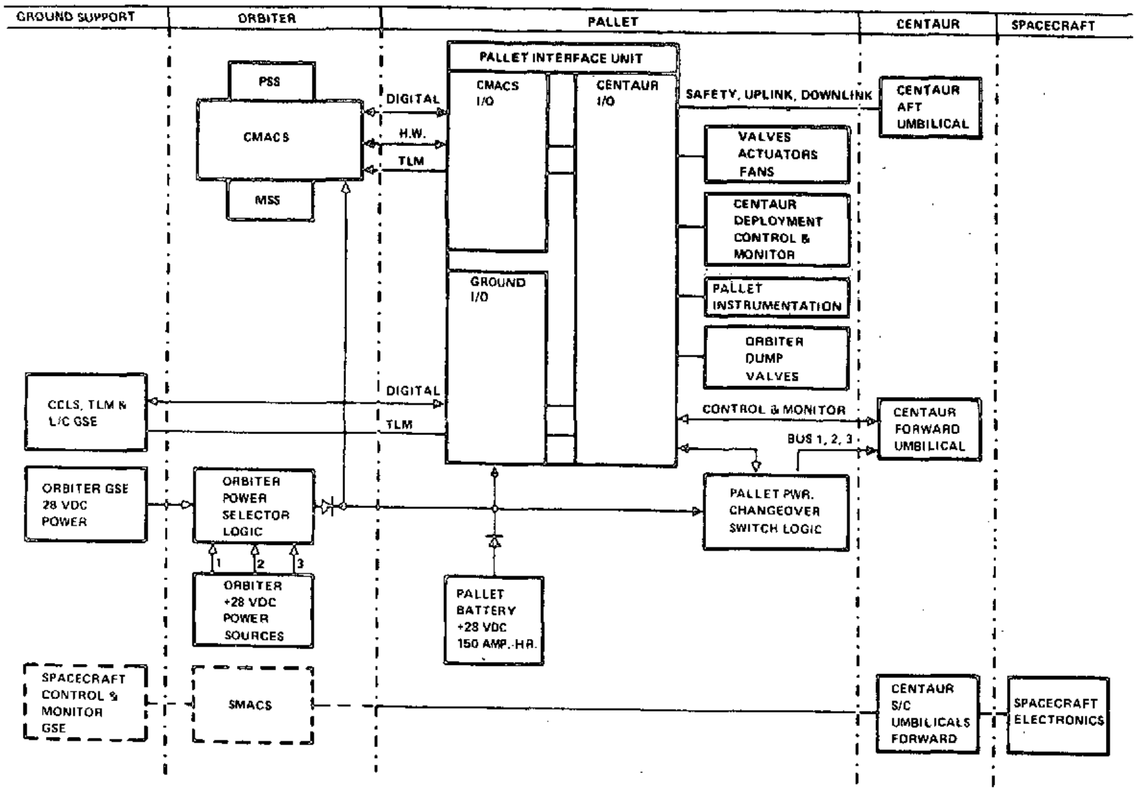


Figure 5-9. Pallet Functional Block Diagram - D-1S(R)

CMACS ensures the safety of the crew and Shuttle by continuously monitoring and displaying critical safety functions, and by providing automatic corrective action sequences when necessary. To verify mission readiness and to ensure deployment of an operational Tug, the CMACS continuously monitors and displays the status of all Centaur subsystems. This monitor and control capability allows the Orbiter crew to perform checkout and deployment or abort operations both rapidly and rigorously for the Shuttle portion of the mission.

The baseline CMACS concept assumes that these Orbiter support requirements are almost totally satisfied by Tug-supplied hardware and associated software which is installed into the spaces provided in the Orbiter payload specialist station for payload supplied support equipment (reference Space Shuttle Payloads Accommodation Document, JSC-07700, Vol. XIV, Baseline issue). If the CMACS functions were performed by the Orbiter payload support equipment or any combination of Orbiter and payload-supplied equipment, these requirements will remain the same. Only the allocation of the actual hardware and software responsibilities will be affected.

Figure 5-10 presents a block diagram of the CMACS hardware configuration. Equipment necessary to perform the CMACS functions may be grouped into three functional categories:

a. Data control

  1. Digital computer

  2. Drum for software storage

  3. Mass memory data logging

    a) Synchronizer and decommutator for PCM telemetry

    b) Multiplexer and A/D converter for analog inputs

b. Communication network

  1. CMACS communicator

  2. Standard input/output terminals

  3. Hardwired signal interface

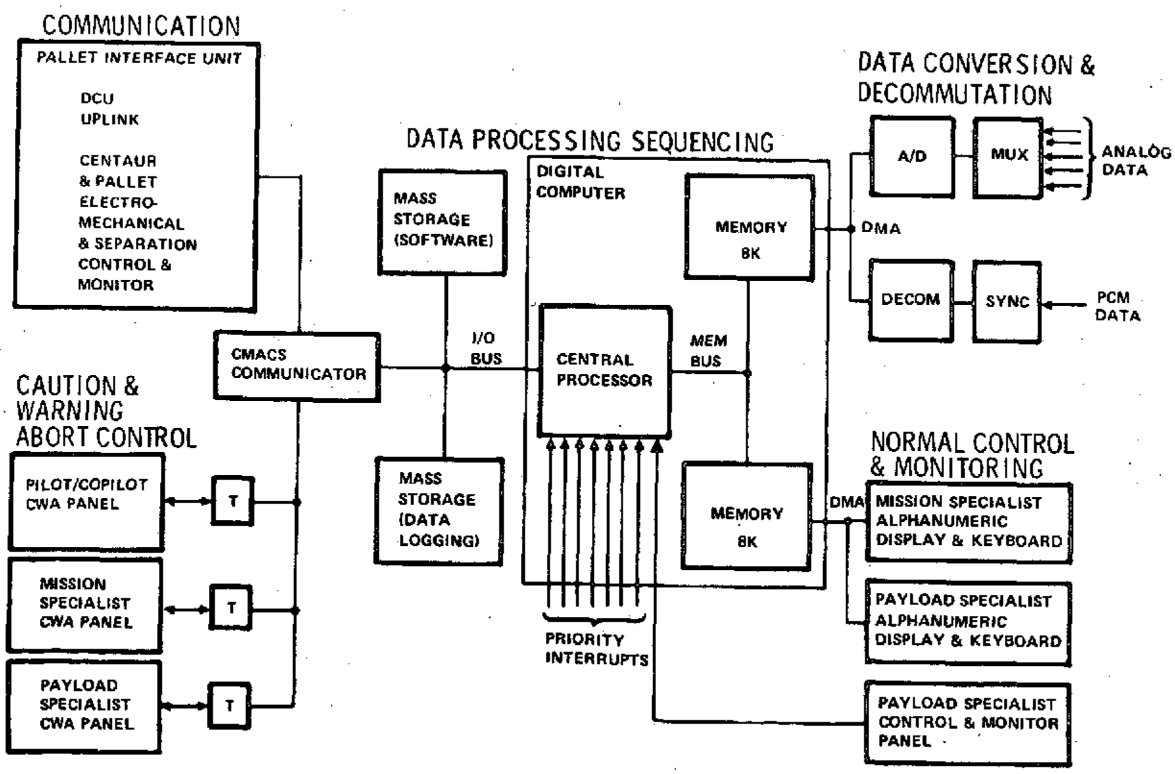


Figure 5-10. CMACS Hardware Block Diagram

c. Crew interface

1. Alphanumeric display and keyboard

2. Control for manual backup capability

3. Caution, warning, and abort panels

5.4 DEPLOYMENT/RETRIEVAL

5.4.1 TIMELINES. Centaur deployment and retrieval are dependent not only on the upper stage requirements and events, but also on Shuttle events which must occur during these operations. The Space Shuttle Upper Stage Conference, 19 June 1973, SSV-73-41, was used as the primary definition of the events and times allocated to stage deployment from the Shuttle. This document's timeline provides 1.75 hours for deployment from Shuttle. The deployment time was provided for all options in PD-TUG-M (315-73) for use in the Space Tug System Studies as follows: include in the mission timeline a minimum of two hours for deployment of the upper stage from the Shuttle and preparation for the first Centaur main engine burn. Based on the ground rule, two hours were used in all mission timelines. Table 5-4 gives the Centaur-Orbiter deployment timelines.

The time required for the Shuttle to rendezvous and dock with the Centaur is Shuttle dependent as the Shuttle is the active element. The "Space Shuttle Upper Stage Conference" did not define this sequence of events and timeline. Therefore the "Tug Operations and Payload Support Study", Vol. 3, Part 3, 5 March 1973, timeline data was used. This data is shown on Table 5-5. Based on the TOPSS results, two

Table 5-4. Centaur-Orbiter Deployment Timeline

Event	Duration (min)	Elapsed Time (min)
Predeployment Payload/Tug Checkout	15	0
Navigation Sightings & IMU Alignment	30	15
Payload/Tug Deployment	5	45
Payload/Tug Checkout	10	50
Payload/Tug Release	5	60
Tug Activation	25	65
Payload/Tug-Orbital Separation	17	90
Tug Initial Burn	-	107

Table 5-5. Centaur-Orbiter Retrieval Timeline

Event	Duration (min)	Elapsed Time (min)
Shuttle Acquisition & Lockup	10	0
Establish Communications Between Vehicles	5	10
Close to Docking Distance	25	15
Expell Excess Main Engine Propellant	20	40
Ready Cargo Bay & Manipulator	2	60
Orbiter Maneuver to Preferred Docking Attitude	6	62
Visually Inspect Tug for Docking Readiness	12	68
Verify all Tug Subsystems Safe for Docking	5	80
Maneuver to Docking Attitude	5	85
Maneuver to Docking Location	1	90
Translate to Final Docking Location	5	91
Inhibit Tug ACPS Thruster Firings	5	96
Attach Manipulator to Tug	<1	97
Verify Physical Attachment Integrity	1	98
Command Tug Retraction/Stowage of Appendages	5	103
Retract/Stow Tug Appendages	2	108
Command Passivation of Tug Subsystems	0.5	110
Verify Tug Safe for Retraction	2	112.5
Passivate Tug Subsystem	5	117.5
Retract Tug into Cargo Bay	5	122.5

hours were used for retrieval by the Shuttle for all mission timelines. For most missions additional time is available due to Orbiter coast time to reentry.

5.4.2 PREDEPLOYMENT AND DEPLOYMENT. Predeployment and deployment operations will include final status checks, DMS inflight software initialization, telemetry bit rate selection, and power changeover to the airborne battery(s) or fuel cell.

Prior to the actual deployment of the Centaur, the CMACS status verification program will continually monitor all signals and sequences required to establish the status of the subsystems. The results are continuously displayed to the operator; if all systems are in a flight ready condition, the deployment may proceed. If any status is no-go, the operator will investigate the anomaly by executing the appropriate utility subroutines via the operator display and keyboard interface. As in the case of the safety monitor program, if the operator determines that the malfunction is not flight critical, he can override the no-go condition and proceed with predeployment checkout and deployment operations. The results of the status verification are especially important during deployment because any improper status will normally result in a decision not to deploy the vehicle.

The actual deployment sequence is controlled automatically by the CMACS deployment program. It is an interactive program requiring mission specialist or payload specialist action to proceed to each major deployment task. The program provides the necessary control and monitor actions to accomplish deployment adapter and Centaur deployment operations and can communicate RF uplink command data to the Orbiter system for radio transmission to the Centaur after release. After verifying that all system flags are "go" (software flags set by status verification program) and the deployment enable command is received from the mission specialist, the program will arm the deployment adapter arm/safe switch. This switch controls execution power to all deployment adapter deployment hardware and prevents premature deployment; a list of these functions follows:

a. Lateral latches command

b. Engine support bladder control

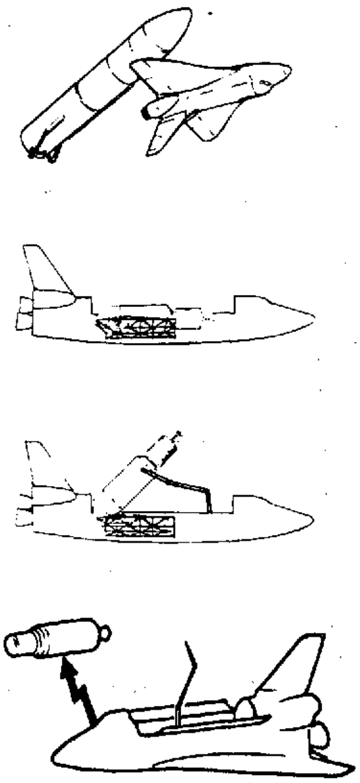
c. Rotary deployment command

d. Aft umbilical panel engage - disengage

e. Centaur separation command

Once the deployment adapter arm/safe switch is armed, this program sequentially performs the operations under the supervision of the mission specialist as summarized in Figure 5-11.

The deployment sequence includes retraction of the forward Centaur/pallet umbilical (containing ground checkout electrical, hydrogen vent, and purge disconnects) (D-1S(R) only), retraction of the lateral latches, and rotation of the Centaur deployment adapter to a 45 degree ($\pi/2$ rad) position relative to the Shuttle centerline. Following this, the Orbiter remote manipulator system (RMS) arm is attached to the Centaur, the aft Centaur umbilical panels are retracted (contains propellant fill and drain, pressurization, electrical, safety functions, etc.) and the Centaur is released from the support adapter. The RMS withdraws the Centaur from the adapter and statically positions the stage parallel to the Orbiter at maximum arm reach.



ASCENT PHASE

- DCU PERFORMS NAVIGATION & CCVAPS TASKS
- CMACS PERFORMS SAFETY MONITORING & STATUS VERIFICATION TASKS
- CMACS PERFORMS THRUST VECTOR AZIMUTH ALIGNMENT

PREDEPLOYMENT TASKS PERFORMED BY CMACS

- FINAL STATUS CHECK
- ARMING PALLET ARM/SAFE SWITCH
- POWER CHANGEOVER TO CENTAUR BATTERIES
- CHANGE PCM TELEMETRY TO INFLIGHT FORMAT & BIT RATE
- INFLIGHT SOFTWARE INITIALIZED

DEPLOYMENT TASKS PERFORMED BY CMACS

- DEFLATE ENGINE SUPPORT BLADDERS
- RETRACT FORWARD UMBILICAL PANEL
- RETRACT LATERAL SUPPORT LATCHES
- ROTATE CENTAUR TO 45-DEG. POSITION
- RETRACT AFT UMBILICAL PANEL
- CENTAUR SEPARATION COMMAND

CENTAUR FLIGHT

- INITIATE RF UPLINK COMMAND TO ENABLE ACS AT RMS RELEASE
- MAIN ENGINE & FLIGHT PROGRAM ENABLE VIA RF UPLINK FROM ORBITER AT 3,000-FT. SEPARATION
- CONTINGENCY CENTAUR SHUTDOWN COMMAND AVAILABLE VIA CMACS ORBITER RF UPLINK

Figure 5-11. CMACS Normal Flight Operation Philosophy

Immediately after disengagement of the RMS end effector, an RF uplink signal is transmitted from the Orbiter to Centaur to close an arm/safe switch and enable the ACPS attitude-hold control mode. This ACPS operating mode is a stable-attitude station-keeping function that maintains Centaur stability while in the vicinity of the Orbiter. Concurrently with transmission of the RF command signal, the Orbiter is backed away from the Centaur with two of the 900-pound (4,000 N) nose ACPS thrusters. Thrust is maintained for 40 seconds to provide a separation velocity of 15 fps (4.6 mps). At approximately 300 feet (91.5 m) of separation, normal Centaur attitude control system operation is enabled, allowing settling thrust initiation for prestart propellant control.

The settling thrust is very low (24 pounds) and does not significantly contribute to the differential separation velocity. A final RF interlock is transmitted at 3,000 feet (915 m) of separation to permit Centaur main engine start.

Centaur is then ready to initiate the remainder of its normal flight sequence. All subsequent functions and sequences performed by the Centaur will be controlled by automatic sequencing within the Centaur and will not depend upon external control from either the Shuttle or the ground. The mission functions will be preprogrammed as is presently the case with Centaur. However, an emergency shutdown command is available to the Orbiter crew via RF uplink, should a Centaur malfunction endanger the Orbiter after main engine start.

5.4.3 UPPER STAGE RETRIEVAL. Shuttle recovery of the Centaur upper stage requires navigation and guidance of the stage to within 10 n. mi. or less of the Orbiter. The upper stage will have the capability to maintain pitch, roll, and yaw attitude hold during the rendezvous and docking to the following attitude and residual velocities:

Longitudinal Velocity	0.1 to 1.0 ft/sec (3.1 to 30.5 cm/sec)
Lateral Velocity	0.5 ft/sec (15.2 cm/sec)
Angular Misalignment	±10 deg
Angular Rate	1.0 deg/sec

After sensor acquisition of the Centaur, the Orbiter will perform the rendezvous, docking and retrieval. The retrieval process tends to be the reverse of the deployment sequence. Detailed retrieval events are listed in Table 5-5.

Safety of the Orbiter and crew is a prime concern, and the provisions to maintain this vary slightly between the D-1S(R) and the RLTC. Dual communications are used with both vehicle types for reliable statusing, and all safety critical functions are either redundant or a backup system is available. In addition to the arm-safe switching, the D-1S(R) has a backup docking attitude control to preclude loss of stage stabilization through failure of the simplex primary system during the last few minutes of docking into the Orbiter. Comparably, the RLTC has fully redundant electronics and control and does not require the added backup.

Active stabilization of the Centaur is maintained until the Orbiter has maneuvered to final docking position and is ready to complete the grappling with the manipulator arms. A few milliseconds prior to final physical contact, the Centaur attitude control system is shut off to allow full Orbiter control of the combination.

For the D-1S(R), if a last minute failure of the primary system occurs, the backup docking attitude control system can take over with the command concurrence of the Orbiter so as to provide automatic stabilization of the upper stage. Manual override via the dual communications link can be accomplished by the Orbiter mission specialist with visual and telemetry monitoring.

RLTC is similar in its sequence except that the redundant inertial reference units and triple redundant digital computer system furnish the necessary backup capability. Manual override can be accomplished via the Orbiter CMACS and the dual communications link when emergency demands.

Docking of the Centaur involves guiding the vehicle into the deployment adapter and securing the latch fittings. The physical guiding/alignment occurs in four phases following retrieval:

a. Initial alignment is accomplished with the RMS and manual control of the Centaur ACPS. The first physical clearance to be maintained is when the engine nozzles enter the forward ring of the deployment adapter. The engine nozzles have an allowable centerline miss distance tolerance of ±10 inches (±25 cm) in the Y direction or ±14 inches (±35 cm) in the Centaur Z direction. The Y direction alignment is directly observable from the mission specialist's station, (assuming a window is in the Orbiter) or by TV monitor. Centaur angular positioning of up to $\pi/32$ radians is adequate in this position.

b. The engine nozzles enter the nozzle support guides at about 35 inches (90 cm) from the docked position. The nozzles will enter the guides from a radial miss distance of 8 inches (20 cm). Pneumatically inflated pads on the nozzle support prevent engine damage. In this position, the Centaur aft adapter must be located on the center of the deployment adapter to within 6 inches (15 cm) radial. Angularity is allowed to meet a combination of the nozzle and adapter misalignments.

c. Probe/drogue type alignment of the aft adapter and the deployment adapter flanges begins when the flanges are separated 4 inches (10 cm) in the Z direction and up to 6 inches (15 cm) radially. Angular alignments around all three axes are maintained by the engine nozzle guides. Shock absorbers in the drogue mechanisms arrest the Z direction velocity. Catches in the drogues hold the probes upon contact and stabilize the Centaur 4 inches (10 cm) from being fully docked.

d. Geared electric actuators in the drogues are retracted to final mate the Centaur to the adapter. Tapered end shear pins engage for the final alignment and shear load path between the adapters. The 16 latch fittings are then actuated to lock the Centaur in the deployment adapter.

The fluid and electrical disconnect panels are retracted during docking, thus preventing damage or binding. The disconnect panels in the deployment adapter have alignment pins which guide the panels during engagement. The pins accommodate radial misalignment of up to 1.2 inches (3 cm). Additionally, each disconnect/connector floats in the panel for an additional 0.08 inches (0.2 cm). The panels are reconnected with electric ball screw actuators to establish the hardline communications and pressure and vent lines.

5.5 INFLIGHT ABORT PROPELLANT DUMP

If the Shuttle aborts during ascent, what should be done with the payload, in particular what should be done with the main propellants? The basic options are to land fully tanked or to dump some or all propellants in flight. While it is structurally feasible for the Centaur to land full, or partly full of propellants, it is recommended that inflight propellant dump be used to minimize special postlanding operations. It seems preferable for the Orbiter to land with less than full payload to reduce landing speed, tire loads, and required runway length. The post abort operation of the Centaur is very dependent upon many Shuttle variables, so the final selected operational mode must be compatible with the final Shuttle data. The several options are safe and feasible and the final selection depends on the Shuttle requirements. The current Reusable Centaur configurations provide simultaneous inflight dumping of both propellants in 250 seconds using oversized dump systems and a large capacity helium pressurization.

5.5.1 ABORT RESTRICTIONS. Reusable Centaur abort operations are constrained by the following requirements:

a. Orbiter safety must not be degraded while performing intact abort.

b. During Orbiter atmospheric glide the c.g. of the Tug and its payload must be within the prescribed envelope.

c. Avoid combustion of dumped propellants outside the Orbiter.

d. Avoid hazardous contamination of the payload bay and other Orbiter compartments by reingestion of dumped propellants.

e. While the Orbiter specification requires that it be able to land with a 65,000-pound (29,500-kg) payload, some improved margins will be achieved at lighter loads, such as 32,000 pounds (14,500 kg) maximum.

5.5.2 SAFE DUMP SEQUENCE. The critical Shuttle abort mode is Return to Launch Site (RTLS) since it has the minimum time available prior to landing. By Shuttle ground rule, abort will not be attempted during solid rocket motor burn. Dumping during SSME powered flight for the critical RTLS abort mode allows use of the normal fill and drain plumbing, appropriately oversized to meet the dump time constraints.

With the Orbiter "flying backwards," as shown in Figure 5-12, dumped propellant could enshroud the vehicle (along with SSME exhaust gases) and infiltrate into the payload bay and the other Orbiter compartments. After return to the atmosphere, these ingested vapors might be a combustion hazard. The pressure at which hydrogen and oxygen will not burn is <0.1 psi (0.007 kg/sqm). This pressure limit is attained at altitudes of 110,000 feet (33,528 m). Further increases in altitude, of course, result in additional decreases in atmospheric pressure with consequent increases in the safety margin.

All possible combustion problems, either external or internal to the Orbiter, can be avoided by completing the dump in 250 seconds at altitudes above the 220,000 feet (67 km). The minor amount of hydrogen vapor that might be ingested above that altitude, would be too dilute to burn during reentry. Additional safety assurance can be achieved by incorporation of active vents on the Orbiter. These vents would be closed during propellant dump to minimize ingestion of propellant vapors. Pressurization of the interstitial spaces during abort dump coupled with active Orbiter vents, would completely preclude propellant vapor ingestion. The nontoxic, noncorrosive nature of these propellants will have no adverse effect on Orbiter surfaces.

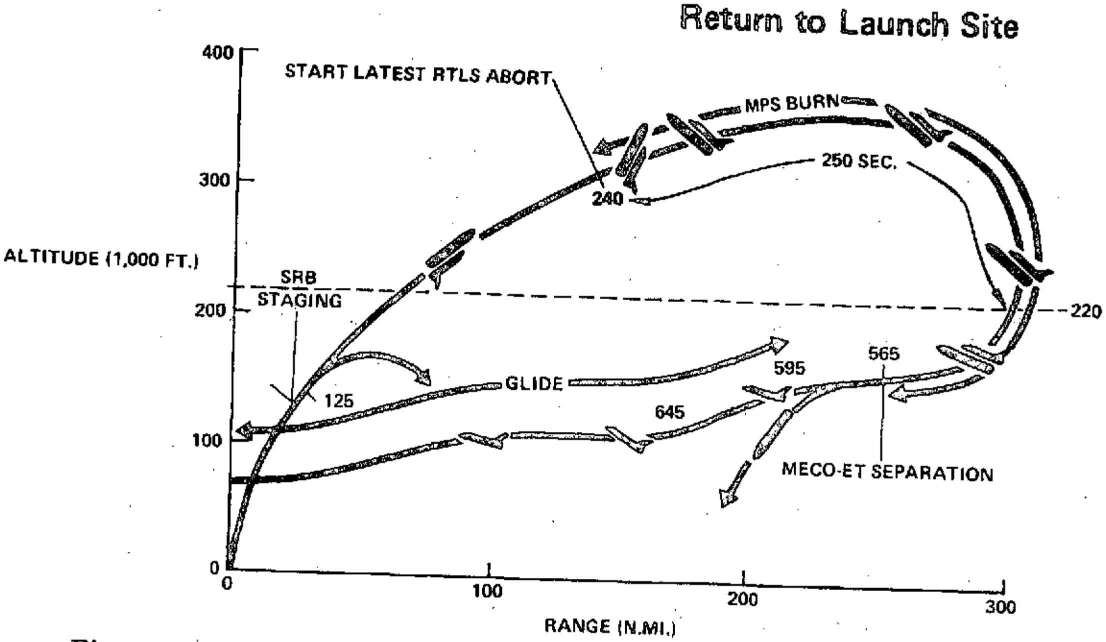


Figure 5-12. Critical Abort Mode, Due East Launch, 65k Payload

Simultaneous dump of LH_2 and LO_2 is not an external combustion problem at altitudes above 110,000 ft (33.5 km) since with a 300 inch (762 cm) dump port separation distance the pressure at the intersection of the dump plumes is less than 0.1 psia (0.69 kN/m^2) and will not support combustion.

The recommended dump approach for the RTLS (return-to-launch-site) abort mode was simultaneous dump of both propellants in 250 seconds. This precludes a hazardous H_2 concentration in or near the Orbiter, reduces structural loads at landing, and minimizes special post abort ground operations.

For the Shuttle abort to orbit mode, propellant orientation in orbit must be provided by either the dumped propellants or OMS settling. In no case is there a requirement or a condition where the Tug will return with fully loaded propellant tanks. The use of redundant dump valves and the techniques shown are designed to always provide the in-flight dump capability under all abort modes.

5.5.3 FURTHER ABORT/DUMP TRADE STUDIES. Although there is not a firm program requirement for in-flight propellant dump, it would provide operational and safety advantages for both the Shuttle and its initial upper stage. Therefore, in-flight dump of both propellants is recommended and the current Reusable Centaur configurations provide for it at a considerable weight penalty. For prelaunch operations, Centaur fill and drain lines are normally about 2.5 inches (6.35 cm) diameter, with single shutoff valves. In order to dump inflight in 250 seconds, this system is much enlarged as described in Section 2.4.6.

The lines are routed from the tank through dual shutoff valves to separate disconnect panels. The lines continue in the Deployment Adapter, with two more shutoff valves (quad redundancy for safety and reliability) through pivot or rotation features to the Orbiter interface panel. The Orbiter routes the fill and dump lines to the launch pad umbilicals on the aft fuselage sides. A large amount of helium is required to expel the main propellants in less time than during a hot firing. The helium storage system is carried in the Deployment Adapter, totaling over 1000 pounds (454 kg).

Obviously if only oxygen were dumped inflight, considerable helium and hydrogen dump system weight could be saved. When more complete Orbiter and initial upper stage data are available, the study should be rerun to optimize the abort solution from the viewpoint of both the Orbiter and payload. Simultaneous in-flight propellant dump, which currently seems like the best safety and operational solution, may be unnecessarily conservative.

SECTION 6

SAFETY AND RELIABILITY

Safety is a major consideration for any Reusable Upper stage that is flown in conjunction with the Shuttle. The Centaur must be essentially fail safe from the time the stage is installed in the Shuttle until it has separated a sufficient distance from the Orbiter to preclude any possibility of its imposing a safety hazard. During this study, a detailed safety analysis was conducted to identify potential design, operational, and interface hazards and to implement design features that will eliminate those hazards. One of the most important of these design features is the complete control of any cryogenic propellant leaks while in the Orbiter by utilizing tank isolation valves and a vented purge bag concept.

The safety features designed into the Centaur were not considered tradeable and the more than 200 pounds (91 kg) weight associated with implementation of these safety features is included in the weight statement. The result has been a Centaur design that is considered to be fully compatible with manned Shuttle operations. Additionally, the basic Atlas/Centaur pressurized stabilized tank design concept was "man rated" on the Mercury program. There have been no vehicle losses attributable to structural failure in more than 440 Atlas and Centaur Flights.

In many instances, the incorporation of safety features also tended to enhance Centaur reliability. Trade studies were performed to determine further reliability improvements that could be realized through systematic implementation of Centaur redundant items versus the consequent weight penalties. The subsystem/component redundancies designed into the D-1S(R) and the RLTC have resulted in a mission reliability that approaches the 0.97 goal for the D-1S(R) with kick stage, and that exceeds the 0.97 goal for the RLTC.

6.1 SAFETY CRITERIA AND APPROACH

To provide a fundamental set of safety requirements for use in this study, the following general safety criteria were used for the Centaur:

a. No single component failure in the Centaur, or Centaur peripheral equipment, shall cause the loss of the Orbiter or its crew.

b. Structural elements, tanks and lines are excluded from the above redundancy requirements. Sufficient margins of safety are used in the cases so as to reduce their probability of failure to negligible levels. MIL HDBK 505 safety factors used by the Shuttle are imposed on the Centaur while in the Shuttle vicinity. Load factors and safety margins established for use on the Centaur while it is in or near the Shuttle are consistent with these requirements.

c. Provisions shall be made for Orbiter crew command override capability of Centaur safety safety critical functions.

d. Centaur shall not preclude Orbiter from intact abort.

The methodology employed in critically examining the proposed and/or existing designs in terms of the established safety criteria were the fault tree analysis and the mission hazard analysis.

The fault tree analysis (Figure 6-1) was developed at the start of the Reusable Centaur study. Its main purpose was to (1) systematically identify those hazards that are associated with use of cryogenic propellants, and (2) generate design alternatives that are capable of countering each of the identified hazards. In addition to the development of the propellant hazard analyses, other potential hazards were identified in the fault tree that also required resolution. Each of these additional hazards was treated in detail in the mission hazard analysis.

The mission hazard analysis was used in the identification of potential hazards that could occur during any mission phase. It has as its basis the fault tree analysis and uses operational block diagrams and time lines in the systematic evaluation of hazards that can occur during ground or flight operations. The detailed mission hazard analysis for the Centaur Tug is contained in Volume 7 of Reference 1. Every Centaur Tug Vehicle hazard identified in the mission hazard analysis and the fault tree analysis has been countered with implementation of specific safety features into the Centaur vehicle design. These safety features are discussed in the following paragraphs.

6.2 CRYOGENIC VEHICLE SAFETY CONSIDERATIONS

The source of the cryogenic vehicle hazards is the Centaur liquid hydrogen and liquid oxygen main propellants. These propellants are cold and flammable and constitute a safety hazard when not properly handled. As used in the Centaur, the propellants are contained within the pressure-stabilized Centaur structure and the fuel and oxidizer are separated by a redundant intermediate bulkhead. As indicated on the fault tree, the principal hazards associated with the cryogenic propellants in the flight vehicle are those associated with leakage, pressurization, and abort dump. Design features are included in the Reusable Centaur concept to safeguard against these potential hazards.

In-flight propellant dump, after an Orbiter abort, which involves both operational and safety interactions, are discussed in Section 5.

6.2.1 <u>PROPELLANT LEAKAGE</u>. The propellant safety features implemented into the Centaur design that counter each of the propellant hazards identified in the fault tree are presented in Figure 6-2. As indicated in this figure, each of the main propellant tanks, and its outlet valve, is individually enclosed in a purge bag. The purge

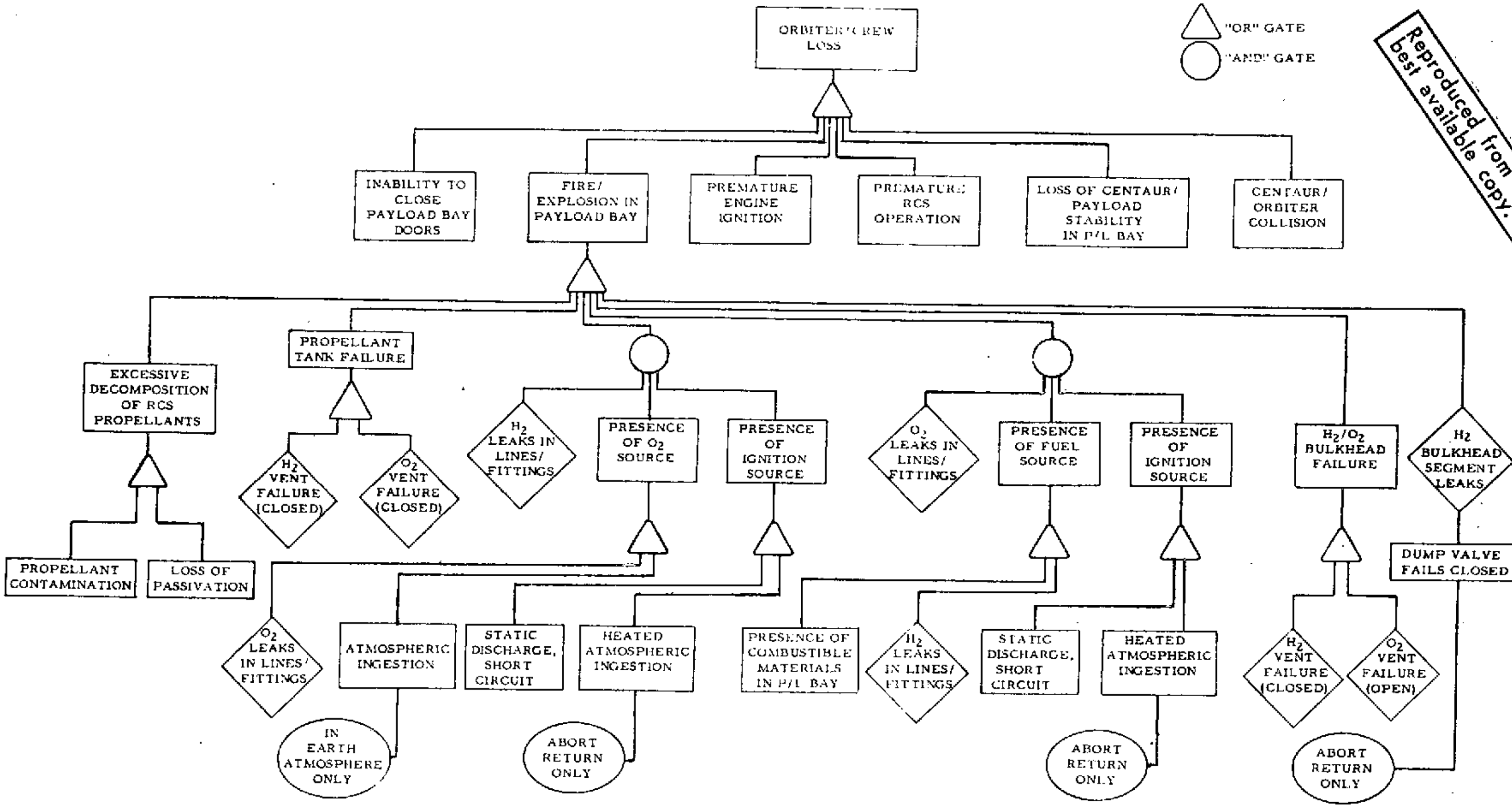


Figure 6-1. Fault Tree Analysis, Cryogenic Propellants

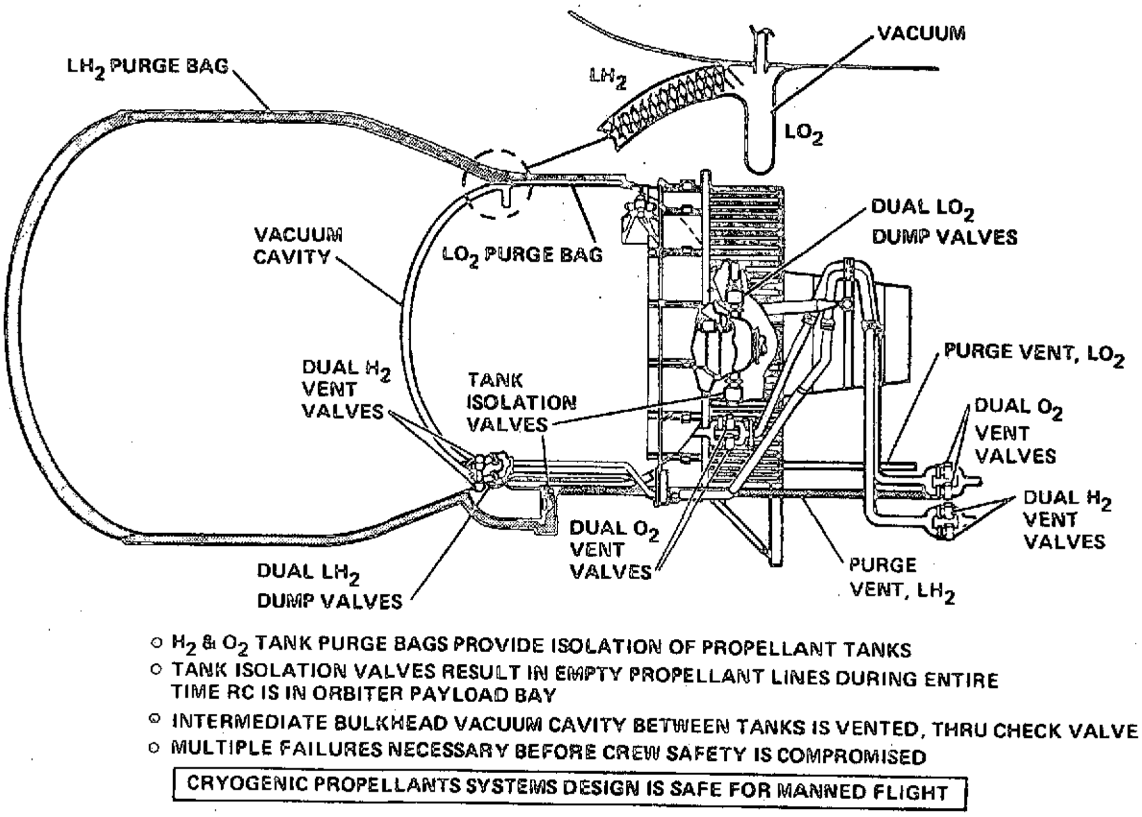


Figure 6-2. Propellant Safety Features

bag is an integral part of the LH_2 ground hold insulation blankets. The Deployment Adapter forms the aft portion of the LOX tank purge bag. These closures are purged with helium and are individually vented external to the Orbiter. In a leak should develop in a propellant tank, the leaking propellant (up to 1×10^6 scims or 16×10^6 cu cm min) will be safely carried overboard. Tank isolation valves are used to maintain the propellant lines in an empty condition during the entire time that the Centaur is in the Orbiter payload bay, thus eliminating the chance of a propellant leak from lines/fittings and the main engine during Orbiter flight. Dual redundant dump valves are installed on each of the propellant tanks to assure propellant dump capability in event of an aborted flight. These dump valves are backed up by redundant dump valves installed on the adapter. This arrangement effectively results in quad redundancy for the dump valves.

To assure that hazards associated with hydrogen leakage are controlled, even in the presence of multiple equipment failures, the following design/operational modifications were implemented.

a. The Shuttle payload bay is continuously purged with GN_2 during tanking, terminal countdown, and postflight safing operations.

b. No ignition source, i.e., spark or 644°F (340°C) temperature, is present in any Centaur equipment exposed to payload bay atmosphere.

c. All exposed flammable materials have been removed from Centaur and its ancillary equipment. For instance, mylar has been replaced with Kapton.

d. The joints in the propellant fill and drain lines and the GH_2 vent line are purged with He.

e. Both propellant fill and drain lines receive a He blowdown purge after each use and before launch.

f. The LO_2 and LH_2 hardline panels are separated by the maximum distance allowed by Orbiter design.

6.2.2 TANKING, VENTING AND PRESSURE CONTROL. The Centaur, like any other Tug, must not exceed the propellant tank pressurization limits consistent with Shuttle manned compatibility safety factors. However, since the Centaur's tanks are pressure stabilized, a minimum tank pressure must also be maintained. Thus, venting and pressure control is a vital function and the Centaur, with its adapter and peripheral equipment, has been designed to provide multiple backup pressure control capability. The redundancies implemented result from both the fault tree analysis and failure modes and effects analysis. The design features that were implemented as a result of these analyses are summarized as follows:

a. Parallel redundant GO_2 and GH_2 vent valves were added to the airborne system (Figure 6-2).

b. Parallel redundant vent valves were also incorporated in the adapter design for both GO_2 and GH_2. This results in quad redundancy in the vent system (Figure 6-2).

c. The propellant tank pressure regulating vent valves require power to lockup. Therefore, even with loss of vehicle power, the pressurization/vent system will still function properly when backed up by the CMACS/adapter systems, which incorporate a crew override capability.

d. Both the 800 psi and 50 psi regulators on the adapter are parallel redundant. Isolation valves and relief valves provide fail-safe/fail-operational capability.

e. Shutoff valves are provided upstream of all adapter regulators used in flight or terminal countdown to control fail open or over-pressure modes. CMACS control will still allow a burp operational mode in emergency or abort situations.

f. Careful attention has been given to separation of vent functions to preclude existence of flammable mixtures at vent outlets. Also, check valves have been provided where backfilling could occur which would be potentially hazardous or degrade system operation.

g. Any credible leak in either tank occurring during boost phase can be made up by the adapter helium supplies and normal propellant boiloff until abort altitude is attained. The dedicated helium supply will then be available to dump remaining propellants.

In addition to the redundant pressurization and vent systems described above, a pneumatic stretch mechanism can be used to maintain structural integrity of the Centaur if all normal systems and their backups should fail. The stretch mechanism would normally be inactive during flight and would place the Centaur in stretch only during an emergency when all pressurization controls and their backups become nonfunctional. Such stretch devices are currently standard in Centaur manufacturing docks.

The Centaur must, of course, maintain pressure integrity during the return flight. Any Tug vehicle that is taken to near orbit altitudes, and has its propellant tanks emptied, must be repressurized prior to being resubjected to atmospheric pressures. Failure to maintain pressure for any of these vehicles would result in catastrophic crushing of the tanks.

6.2.3 SAFETY OF INTERMEDIATE BULKHEAD. The intermediate bulkhead is composed of two entirely independent steel bulkheads (Figure 6-2). The space between the bulkheads is insulated (which also serves as a spacer between the bulkheads) and is cryogenically pumped to a pressure below 50 microns. The space between the bulkheads is connected to a check valve and is vented external to the Orbiter. If a hydrogen leak should develop across the hydrogen bulkhead, the remaining bulkhead will continue to maintain separation of the propellants. Hydrogen in the vacuum space will, however, absorb heat energy through the oxygen bulkhead. The leaking hydrogen will turn to gas and will be vented through the check valve. If, on the other hand, a leak should develop through the oxygen bulkhead, the leaking oxygen will either freeze or remain liquid. Heat energy transmitted through the bulkhead to the liquid hydrogen will increase the boiloff rate of the hydrogen. The hydrogen vent valves, however, are capable of handling the increased boiloff.

In-flight intermediate bulkhead reversal would require multiple failures in the redundant pressurization system, the series-parallel vent system, and the Centaur monitor and control system. In the worst case (series vent valves in the oxygen tank simultaneously fail open at launch and the crew command override capability is nonfunctional), the pressure decay in the oxygen tank will be sufficiently slow as to allow 10 minutes for hydrogen dumping (hydrogen dumping takes less than 6 minutes) and the Orbiter will be able to safely complete its flight to orbit.

The only likely cause of bulkhead reversal is human error during ground operations. To preclude the chance that a bulkhead reversal will take place during ground operations, a positive ΔP control, such as a differential piston, can be installed between the hydrogen and oxygen tanks, both of which are inert gas pressurized. The piston is biased toward the hydrogen tank pressure; i.e., the area on the hydrogen tank side of the piston is greater than that of the oxygen side of the piston. If the hydrogen tank pressure starts to approach the oxygen tank pressure, the piston is automatically displaced. Piston displacement uncovers the vent port and the pressure in the hydrogen tank is reduced to a safe differential with respect to the oxygen tank. This ΔP control will be used in addition to the continuous pressure recording devices that are normally used to monitor tank pressures on Centaur vehicles. Breakwires can be used as an additional indicator for assuring bulkhead integrity.

Inspection of the intermediate bulkhead can be accomplished by (1) visual inspection via the access doors, (2) continuity/resistance checks of the breakwires, (3) use of acoustic emission monitors at points of concentrated stress, and (4) pressure testing. In pressure testing the intermediate bulkhead, the oxygen and hydrogen tanks are pressurized to their operating pressure with helium. The interbulkhead space is vacuum pumped to achieve a pressure differential of 25 to 30 psi. A helium mass spectrometer, installed in the vacuum line, will detect any leak that might occur across either bulkhead.

6.3 SAFETY FEATURES OF CENTAUR VEHICLE SUBSYSTEMS

In addition to the subsystems associated with main propellant safety, the following Centaur flight and support equipment were also evaluated in terms of the Shuttle safety criteria.

6.3.1 AUXILIARY PROPULSION SYSTEM. The hydrogen peroxide APS used in the D-1S(R) vehicle and the hydrazine APS used in the RLTC (see Section 2.5) incorporate the following safety features to counter the hazards identified in the fault tree analysis.

a. Isolation valves are used between the storable propellant supply and each of the propellant users. These isolation valves are maintained in a closed condition while the Centaur is in the payload bay. If any APS component valve should fail open, the isolation valves will preclude operation of that component.

b. The isolation valves are commanded open (or closed) by crew command via RF link. During deployment operations, the APS thrusters are enabled just after release of the Centaur from the Orbiter manipulator. On recovery of the Centaur, the APS thrusters are isolated just prior to engagement of the Centaur by the manipulator.

c. APS propellant tanks incorporate overboard vent/dump capability to protect against tank overpressures that may result from inadvertent propellant decomposition.

d. Continued safe operation of the Centaur APS, particularly during the critical docking/retrieval maneuvers where the Orbiter is in close proximity to the Centaur, is assured by the fail operational/fail safe capability of the reaction control system. If any thruster should fail open, the condition is sensed by the Centaur astrionics and the appropriate isolation valve is closed to isolate the thruster cluster. Centaur stability will continue to be maintained by the remaining clusters. If multiple RCS failures should occur, the RCS function can be immediately shut down by Orbiter crew command via the RF link.

6.3.2 HYDRAULIC SYSTEMS. The remaining fluid system not previously discussed is the hydraulic system (see Section 2.4) which provides actuator power for gimbaling and controlling the position of the Centaur main engines. Each hydraulic system (one for each engine) consists principally of a recirculation pump, a reservoir, two actuators, an engine-driven high pressure pump, and the necessary interconnecting lines and hoses. Each system is closed and is disconnected from ground hydraulics subsequent to cryo tanking and flight events demonstration testing. The system is checked for leaks subsequent to all high pressure tests and when on standby or recirculation (temperature control, engine start positioning, etc.), operates at only 10% of the normal operating pressure. Therefore, although the hydraulic fluid is flammable, it is considered to be an acceptable risk when in the payload bay; the pressures are low with regard to previously demonstrated operating flight pressures, and the amount of fluid is small (less than one gallon or 3.8 liters per system).

6.3.3 POWER AND ELECTRONICS SYSTEMS. With the D-1T Centaur used as a baseline, all electrical/mechanical systems, including GSE, requiring power or electrical control were examined for Shuttle manned compatibility. This investigation provided the data for implementation of the safety features necessary to make these systems compatible with Shuttle operations. These safety features are:

a. During ascent, the safety of the Shuttle and crew is ensured by three CMACS functions. First, CMACS continuously monitors and displays data related to the following critical safety functions:

1. LO_2 tank pressure/venting
2. LH_2 tank pressure/venting
3. H_2O_2 bottle temperature/hydrazine temperature
4. H_2O_2 bottle pressure/hydrazine pressure
5. Centaur Bus 1 voltage
6. Umbilical disconnect status
7. APS status
8. Intermediate bulkhead status

9. GN&C status

10. Data management status

Second, CMACS (via the caution warning and abort panel) provides an alert to the crew when a safety function approaches a limit and executes corrective action sequences automatically. Third, CMACS provides an alert to the crew when there is an out-of-tolerance condition for a safety function and executes abort sequences automatically when enabled by the crew.

b. The resident safety monitor program, which performs the continuous monitor task, continuously displays the values of the safety measurements on the CRT display screens. Each safety signal is input to the CMACS via both a digital (PCM) and an analog path. If a major discrepancy between the two values occurs, it will be detected; the safety monitor will inform the operator of the discrepancy and then proceed as if the worst case signal were correct. The operator then uses utility and self-validation subroutines to determine which signal is incorrect. If he can determine which is incorrect, he can call up another utility option to set flags in the safety monitor program that will disable the monitor for that type of data for that critical function.

When the resident safety monitor detects a critical function approaching its red-line limit, a light identifies the corresponding function on the caution, warning, abort (CWA) panel and a message appears on the mission specialist's display. The corresponding corrective action sequence is automatically loaded and executed in an attempt to correct the malfunction. If an out-of-tolerance condition is detected, the CWA panel shows a warning light, a message identifying the condition is displayed on the mission specialist's CWA panel, and the abort sequence is automatically loaded and begins executing as soon as an abort execution signal is received from the CWA panel.

c. Two arm/safe switches in the SCU control the functions related to the safety of the Orbiter and crew while Centaur is aboard. One SCU switch will be controlled by the RF uplink command decoder. It will control equipment that must be disabled while the Centaur is in the Orbiter payload bay and will be enabled after the Centaur is deployed for flight. The other switch will control the vent and pressurization valves so that the ground GSE may control these functions during prelaunch operations and the DCU may control these functions (CCVAPS) during flight.

d. A switch in the RF uplink command decoder is used to enable the Centaur attitude control system after deployment from the Orbiter.

e. An arm/safe switch on the adapter prevents any inadvertent adapter or Centaur deployment operations during the Shuttle ascent or descent flight operations. The functions controlled by the adapter and Centaur arm/safe switches and the attitude control enable are discussed in Section 6.4.

f. Backup hardwire control is provided between CMACS and the Centaur adapter for safety-critical functions. These controls are supplemented by warning/action readouts to alert crew that a hazardous situation is developing.

g. For the D-1S(R), a mechanical drive crossconnect was incorporated in both fore and aft umbilical panel retract mechanisms so that a signal to either drive motor of the forward or aft panel will engage or disengage that panel (only the aft panels are used in the RLTC).

h. A backup avionics package was added to the D-1S(R) that allows continued Centaur attitude control during docking maneuvers in event of a primary system failure. The RLTC incorporates dual string astrionics that provides backup capability during the docking maneuver.

i. Centaur software shall be structured to include at least two separate sections: (1) preflight software and (2) flight software. This technique will prohibit premature or erroneous commands or responses from the DMS during any launch operational phase, i.e., during prelaunch of predeployment operations, the Centaur engines will be prevented from inadvertently firing because the DMS software modules that control this function will be locked out via software/hardware control techniques. A software/hardware interlock system shall prevent the DMS from going to the flight mode prior to a "go to flight" command.

j. An RF uplink will communicate at least two commands to the Centaur after separation from the Orbiter. These are:

   1. Go to (or continue) flight mode

   2. Stop (safe Centaur)

   The stop command may be initiated by the Orbiter mission specialist if he observes that the Centaur is malfunctioning after the Centaur engine start sequence has begun. This command will cause the Centaur to shut down the main engines and reset all systems to a safe position.

6.4 CENTAUR DEPLOYMENT/RETRIEVAL SAFING

In addition to the inherent safety existing in the astrionics because of dual- and triple-redundant subsystems, the Centaur arm/safe switching provides the interlocking of functions to preclude potential critical failures that could occur during critical deployment or retrieval operations. Areas protected this way are those related to Centaur deployment main engine start, engine positioning, and attitude control enable.

The major deployment/retrieval events are shown in Figure 6-3. They start with the predeployment checkout when arm/safe switch No. 1 is activated to the arm condition. The digital computer unit (DCU) and sequence control unit (SCU) are turned on, the IMU is energized, and the CMACS continues with the further checkout and monitoring of Centaur functions. At the beginning of deployment, the guidance initialization of the

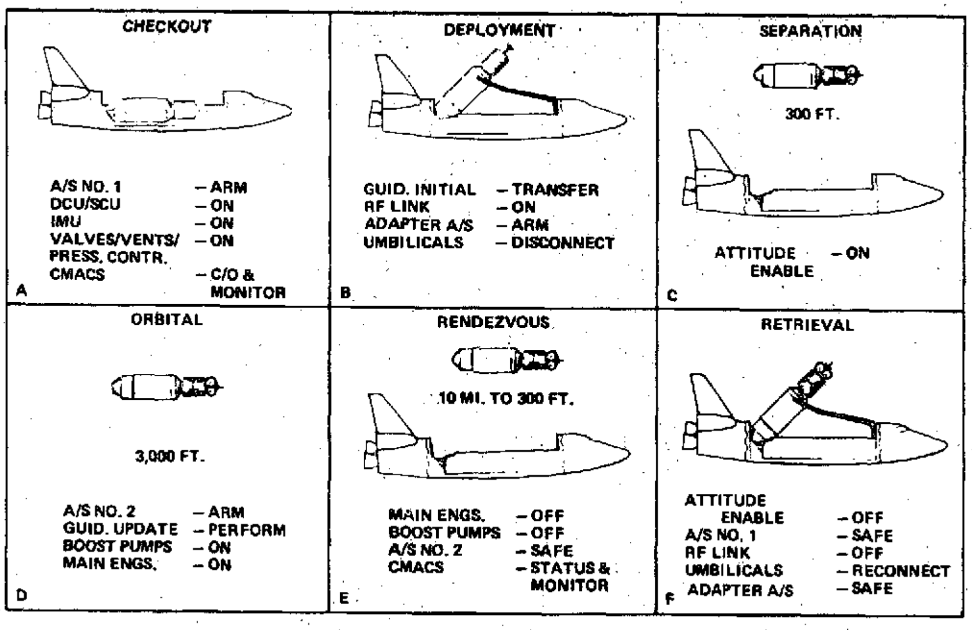


Figure 6-3. Events Sequence - Orbiter/Centaur Safing

IMU is transferred from the Orbiter reference system. The aft support arm/safe switch is placed in the ARM position, the fuel cell is energized, and the external Orbiter power removed. Communication is sustained by operation of the RF uplink and the umbilicals are disconnected. On disengagement of the Orbiter manipulator, the attitude enable signal energizes the auxiliary propulsion system, and vehicle stabilization is achieved. The Orbiter increases the separation distance out to 3,000 feet (915m) or more, and arm/safe switch No. 2 is operated to ARM. A first guidance update is performed with the Horizon and Star Sensors, the boost pumps are turned on, and the main engines fired. After subsequent burns, the completion of independent Centaur orbital operations, and navigation back into the Orbiter rendezvous region (10 n. mi. to 3,000 feet); the main engines are shut off, the boost pumps shut off, and arm/safe switch No. 2 is returned to SAFE. CMACS in the Orbiter then provides safety status via the RF link of Centaur condition before retrieval.

For the D-1S(R), a backup avionics package provides redundancy of those avionics functions required to provide vehicle stability during docking operations. This backup package will be turned-on approximately one hour prior to docking to assure adequate warm-up time for the gyros. During the docking maneuver, the output from the primary guidance and control astrionics is compared to the output of the backup docking package via telemetry. A discrepancy between outputs, coupled with visual observation of abnormal Centaur movement, would indicate a failure of the primary avionics.

In this case, the mission specialist would command the Centaur to switch to the backup docking package. A discrepancy between outputs, but without abnormal movement of the Centaur, would indicate that the primary system is functional but that the backup system had failed.

The RLTC utilizes dual string astrionics with triply redundant computers. For this system the failure sensing and switching functions are autonomous and no action is required on the part of the mission specialist other than monitoring of the safety critical systems.

After verification of Centaur safety, the Orbiter is moved to within manipulator engagement distance of the Centaur, using cooperative ranging and velocity data. Just prior to manipulator engagement, the attitude enable is deenergized. On manipulator engagement and attachment of the Centaur to the deployment adapter, arm/safe switch No. 1 is returned to SAFE and the RF link is shut off. The Centaur is then rotated to its stowed configuration and a final check is made of all safety critical components, via CMACS, prior to de-orbit operations.

6.5 SAFETY SUMMARY

The main thrust of the GD/CA safety effort has been the assurance of Centaur manned compatibility with the Space Shuttle. Table 6-1 summarizes the principal design

Table 6-1. Designed-In Safety

POTENTIAL FLIGHT HAZARDS (FROM FAULT TREE ANALYSIS/ MISSION HAZARD ANALYSIS)	DESIGN FEATURES	Δ WEIGHT (LB.) (SAFETY ONLY)
• PROPELLANT LEAKS – PROPELLANT TANK LEAKS – LEAKS IN LINES/FITTINGS	• PURGE BAG CONTAINMENT, SEPARATELY VENTED • MAIN TANK ISOLATION VALVES • BACKUP CONTAINMENT FITTINGS ON ALL RC/ ORBITER INTERFACES	40 30 20
• PROPELLANT TANK OVER PRESSURE OR UNDER PRESSURE	• QUAD VENT VALVES USED FOR EACH PROPELLANT TANK (NO VENT VALVE FAILURE IN OPEN OR CLOSED MODE WILL COMPROMISE ORBITER SAFETY); REDUNDANT PRESSURE CONTROL	24 – RC
• PREMATURE MAIN ENGINE OPERATION • RC/ORBITER COLLISION	• TANK ISOLATION VALVES (CREW OVERRIDE) • DUAL COMMUNICATIONS • FAIL OPERATE/FAIL SAFE APS • BACK-UP ASTRIONICS DOCKING PACKAGE*	(ABOVE) 17 4 40
• PREMATURE APS OPERATION • RC/ORBITER COLLISION • INADVERTENT DECOMPOSITION OF APS PROPELLANT	• APS ISOLATION VALVES (CREW OVERRIDE) • OVERBOARD VENT AT APS PROPELLANT TANK OUTLET	6 4
• LOSS OF RC STABILITY IN PAYLOAD BAY • INABILITY TO CLOSE PAYLOAD BAY DOORS	• 14 OF 18 LATCHES WILL SECURE RC THROUGH CRASH LANDING LOADS • EMERGENCY MANUAL RELEASE (EVA) AVAILABLE IF RC HANGS UP PARTLY DEPLOYED	20
• H_2 BULKHEAD SEGMENT LEAK INTO INTERMEDIATE BULKHEAD CAVITY	• PRESSURE RELIEF CAPABILITY IN INTERMEDIATE BULKHEAD CAVITY	0 (EXISTING CENTAUR SYSTEM)
• INFLIGHT ABORT DUMP	• QUAD REDUNDANT DUMP VALVES FOR MAIN TANKS INCREASED He. INCREASED LINE SIZES	45 – RC

* D-15(R) ONLY; RLTC HAS DUAL STRUNG ASTRIONICS

features that have been incorporated into Reusable Centaur design. The basis for each of the safety features is the systematic evaluation of potential hazards through fault tree analyses, mission hazard analyses, and implementation of Shuttle safety criteria.

As indicated in this table, each of the potential flight hazards identified has been countered by a specific design feature. The additional weight required to incorporate these safety features is also listed. The resulting weight penalty for safety implementation is >200 pounds (91 kg) for the D-1S(R) and the RLTC. The weights listed are on the Centaur itself. There is over 1000 pounds (454 kg) of safety-related equipment in the deployment adapter or pallet, such as two of the quad redundant vent and drain valves, oversize dump lines, and storage for about 80 pounds (36 kg) of helium for in-flight dump. The resulting Centaur design is considered to be fully compatible with manned Shuttle operations.

6.6 RELIABILITY

The overall D-1S(R) vehicle reliability characteristics, including the redundancy levels for each subsystem, are summarized in Table 6-2. The reliability characteristics for the RLTC are described in Table 6-3. The limited redundancy levels and the resulting reliabilities for the D-1S(R) subsystems result in a vehicle reliability of 0.968 for a 24-hour mission. Actually the Centaur solos away from the Orbiter only 10.6 hours.

In addition to the D-1S(R) vehicle, a velocity package is required for most missions. For missions that require a velocity package and a non-complex spin system (similar to that used in the AEC Pioneer G programs; Ref. Report No. GDCA-BKM-72-003) the reliability is the product of the basic vehicle reliability (0.968) and the kick stage reliability (0.984) which results in a mission reliability of 0.953. Some missions may require a more complex kick stage that incorporates an inertial reference system and attitude control similar to that used on Burner II. The reliability for this velocity package is 0.982. The mission reliability when this kick stage is used is $0.968 \times 0.982 = 0.951$.

It should also be noted that the reliability of the D-1S(R) is sensitive to mission duration (see Figure 6-4). Missions that are shorter than 24 hours result in a significant improvement in reliability. The reliability of the RLTC, on the other hand, is not quite as sensitive to variations in mission duration. This is due to the implementation of dual redundance into most of the RLTC subsystems.

Table 6-3 summarizes the redundancy levels that have been incorporated into the RLTC. These redundancy levels are wholly consistent with the redundancy implementation analysis; i.e., all candidates for redundancy are essentially dual redundant. The resulting RLTC mission reliability is 0.975 for non-kick stage missions. The kick stage missions occur approximately once for each year of RLTC operations.

Table 6-2. D-1S(R) Reliability Characteristics

Subsystem	Reliability	Redundancy Level	Design Mission Time (hr)	Remarks
Structure	0.9999	---	24	Based on structural factors of safety designed into structure
Propulsion	0.9857			
- Main Engine System	0.9900	Component	0.12	Redundant valves
- Thrust Vector Control	0.9986		0.12	
- Pressurization, Vent, Fill, Drain	0.9974	Component	24	Redundant valves
- PU, Purge	0.9998	Component	0.12	Redundant valves
- ACPS	0.9999	Component	18	Redundant engines, valves
Astrionics	0.9813			
- Data Management	0.9920		24	
- Guidance & Navigation	0.9921		18	
- Flight Control	0.9990		18	
- Guidance Update	0.9998		18	
- Communications	0.9999	Component	18	Redundant transceiver, antenna
- Electrical Power & Distribution	0.9984		24	
Interface Systems	0.9999		5	
Kick Stage (Pioneer G)	0.984			Based on AEC Pioneer G Safety Study
Kick Stage (Burner II)	0.982			Based on Boeing analysis
Vehicle Reliability	0.9675			Without kick stage

Table 6-3. RLTC Reliability Characteristics

Subsystem	Reliability	Redundancy Level	Design Mission Time (hr)	Remarks
Structure	0.9999	---	47	Based on structural factors of safety designed into structure
Propulsion	0.9806			
- Main Engine System	0.9875	Component	0.21	Redundant valves
- Thrust Vector Control	0.9977		0.21	
- Pressurization, Vent, Fill, Drain	0.9952	Component	47	Redundant valves
- PU, Purge	0.9995	Component	0.21	
- ACPS	0.9999	Component	35	Redundant engines, valves
Astrionics	0.9950			
- Data Management	0.9994	Component	47	Triple redundant computers
- Guidance & Navigation	0.9999	Component	35	Dual redundant IMU
- Flight Control	0.9989	Component	35	Dual redundant servo-inverter
- Guidance Update	0.9996		35	
- Communications	0.9999	Component	35	Redundant receiver, decoder, antennas
- Electrical Power & Distribution	0.9973	Component	47	Redundant switches, backup transient battery
Interface Systems	0.9999	Component	13	
Kick Stage	0.984			Based on Boeing analysis
Vehicle Reliability	0.9751			Without kick stage

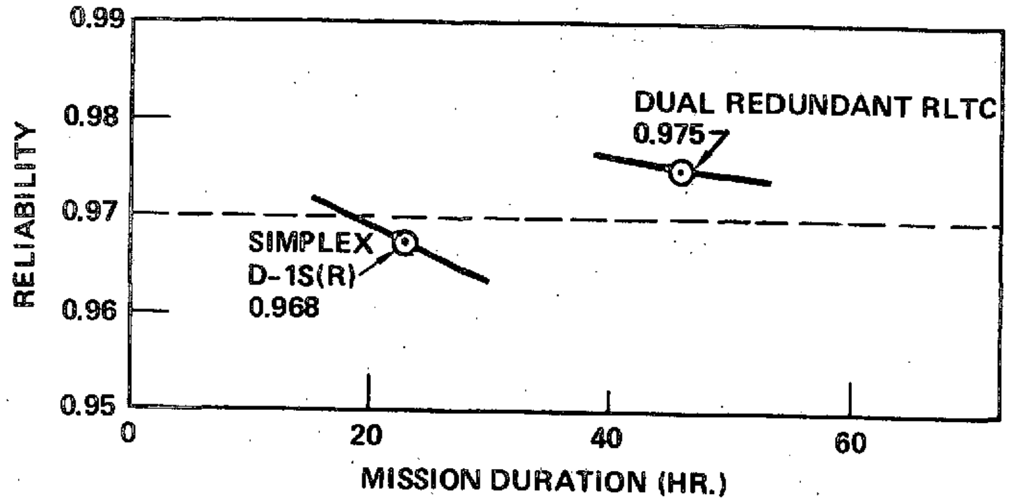


Figure 6-4. Mission Duration Sensitivity

For these missions, the RLTC only operates for 24 hours and the vehicle reliability consequently increases to 0.983. When the reliability of the kick stage is coupled with RLTC reliability, the overall kick stage mission reliability is approximately 0.967.

While the D-1S(R) reliability is based on the redundancies contained in an existing vehicle, the RLTC was reconfigured to achieve a reliability of 0.97 for a 47-hour mission. The redundancy levels required to attain this goal were determined through a redundancy/weight optimization computer program.

The input to the program was initiated with (1) a "single string" system, (2) the mean time between failure (MTBF) for each component, and (3) the weight increase that would result from incorporation of component redundancy. The computer program automatically tests the ratio of reliability increase to weight increase for all candidates for redundancy. (The main engine, main engine propellant lines, boost pumps, etc., were excluded as redundancy candidates. The reliability of these elements thus sets the upper limit on RC reliability.) The component that yields the best ratio of reliability increase to weight increase is the first component selected. After implementation of the first redundant element, the computer repeats the process to determine the next best ratio of reliability increase to weight increase. This process is continued until the incremental change in reliability is close to zero. The MTBF data used in the analysis is from Centaur data, MSFC data (Saturn Component Failure

Rates and Failure Rate Modifiers) and supplier data. The MTBF data was adjusted to reflect the severity of the flight environment according to the following schedule:

Mission Phase	MTBF Divided By
SRB/Orbiter Boost	32
Orbiter Coast	1
RC Burn	20
RC Coast	1
Entry	3

The Shuttle environment is not yet fully defined, but was assumed about equal to Titan. Although some Centaur components are inactive during boost, the severe acoustic and vibration environment should not be overlooked.

The results of the redundancy/weight analysis are presented in Figure 6-5. It should be noted in this figure that implementation of dual redundancy, in most instances, results in attaining maximum reliability. Once these redundancy levels have been attained, continued application of redundancy will only result in very small incremental increases in reliability at comparatively large increases in vehicle weight and total program costs.

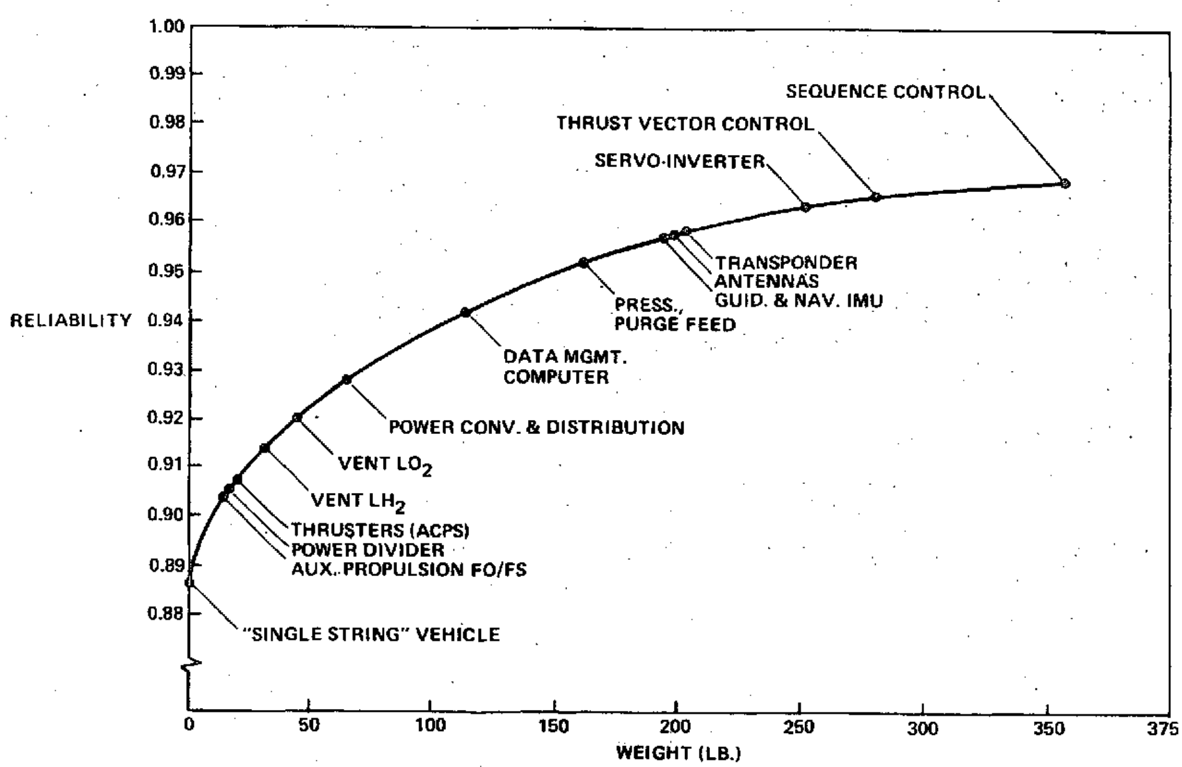


Figure 6-5. Redundancy/Weight Optimization

SECTION 7

GROUND OPERATIONS

For a Shuttle reusable upper stage vehicle, ground operations covers the time span from initial delivery of the new vehicle at the launch range through test and checkout, spacecraft mating, Orbiter mating and checkout, pre-launch operations, launch, landing, safing, removal from the Orbiter, maintenance and refurbishment, and storage or preparation for the next mission. In planning ground operations for the reusable Centaur vehicles, STS ground rules were followed to assure upper stage compatibility with Shuttle operations and turnaround timelines. Planning includes transportation, maintenance, refurbishment, checkout, GSE and facility requirements, utilization of existing resources, and logistics, integrated to assure smooth operational flow and low cost.

In this section the basic Centaur/Shuttle operational concept is discussed with respect to upper stage test and checkout philosophy, followed by a definition and discussion of each phase of the ground operations plan for each vehicle with respect to all of the above elements except logistics. Logistics, fleet size, and costs are treated separately in Programmatics and Cost sections.

7.1 TEST AND CHECKOUT CONCEPT

Ground operation plans for the Centaur vehicles have been developed to accomplish turnaround operations within a reasonable time with the least expenditure of support resources, and to integrate with Shuttle operations on a non-interference basis. As shown in the Orbiter turnaround timeline, Figure 7-1, only 18 hours are allowed in the normal operational Orbiter turnaround schedule for payload installation and checkout at the Orbiter Maintenance and Checkout Facility (MCF), and 6 hours for combined Shuttle checkout at the launch pad. The ground operations plans therefore require that all payload preparation, checkout, and testing be complete prior to installation in the Orbiter, to the maximum extent possible. Only mandatory Orbiter/payload composite testing and interface validation can be allowed in the installation and checkout periods shown at the MCF and launch pad. To satisfy these objectives, test and checkout philosophy for the Centaur vehicles is planned on the same basis as that for the Tug, i.e.:

a. Test and checkout for first flight of each vehicle based on complete test and checkout to attain maximum reliability, maximum probability of mission success, and assurance of vehicle return. This is the "test and retest" philosophy that has achieved proven results on existing fly-right-the-first-time launch vehicles such as Centaur D-1.

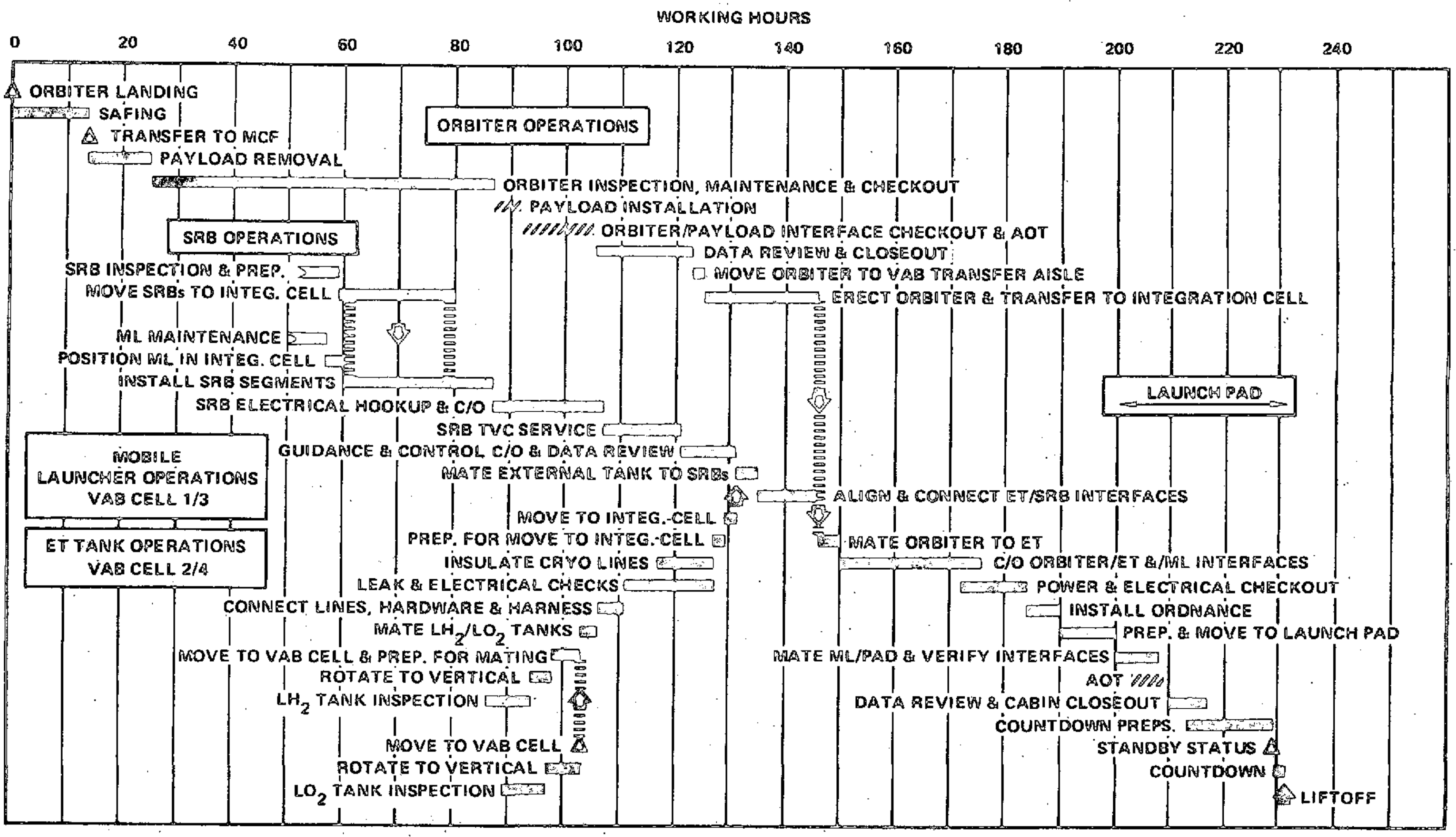


Figure 7-1. Integrated Centaur/Shuttle Ground Operations Timeline

b. Test and checkout for remaining flights of each vehicle based on condition-monitored maintenance with preflight testing (CMM_{PF}); in other words, the preceding flight data is the best test and checkout for the next flight.

When first delivered to Cape Kennedy, each new Centaur vehicle will undergo complete mechanical, fluid, and electrical systems checkout at ETR Complex 36 prior to its first flight in an Orbiter. Checkout will include cryogenic tanking, ground hold, and terminal countdown demonstration, proofing all systems for operation with flight propellants, and demonstrating launch readiness.

The vehicle, in its pallet or deployment adapter, will be installed in a simulated Orbiter payload bay at Complex 36 to verify Orbiter interfaces and disconnect functions, then tested. The pallet and deployment adapter include all Centaur in-flight support systems and equipment, deployment and separation mechanisms, and separation disconnects. Mated to Centaur, the pallet and deployment adapter allow complete testing of these in-flight functions on the ground before the unit is installed in the Orbiter. Hard connections are used between the Orbiter interface panels in the payload bay and the Centaur pallet or adapter hardline panels, obviating the need for Orbiter/Centaur disengagement testing. The pallet (or deployment adapter) remains in the Orbiter payload bay after Centaur deployment, is removed after Orbiter landing, and is recycled to the Centaur handling facility (Hangar J) at KSC for refurbishment and preparation for the next flight.

Upon completion of test and checkout at Complex 36, Centaur will be transferred to a designated spacecraft preparation and mating facility for spacecraft mating. All interface connections and combined Centaur/pallet/spacecraft systems will be checked out here, prior to moving to the Orbiter mating area, to assure that the Shuttle payload is in all respects ready for installation in the Orbiter, with minimal possibility of interference with the Shuttle turnaround schedule.

7.2 D-1S(R) GROUND OPERATIONS

All D-1S(R) vehicles and flight pallets will be fabricated at existing factory facilities in San Diego. A test vehicle and checkout pallet will be fabricated first to satisfy pre-IOC requirements. In the operational phase, the checkout pallet will be used in San Diego for test and checkout of production vehicles. Depending on the mission model and length of program, three or more flight pallets will be shipped to ETR and two to WTR to support flight operations of the reusable vehicles.

7.2.1 TRANSPORTATION. For ground transport, Centaur D-1S(R) will be moved on existing D-1 transport pallets, supported by existing shipping adapters, or on its flight pallet which has its own transport pallet. In either case, the transport pallets can be moved locally on their own detachable wheels, or on transport trailers.

For air transport, Centaur can be shipped by C-5A long range cargo aircraft as is presently done; flight pallets will be shipped in the larger NASA Super Guppy cargo aircraft.

All production vehicles will be shipped to the Cape for pre-first-flight testing before assignment to ETR or WTR for Shuttle operations. After checkout at the Cape, vehicles assigned to WTR operations will be flown to the west coast in C-5A aircraft and will enter the normal flow of turnaround operations at the Payload Processing Facility (PPF).

7.2.2 RECEIVING AND INSPECTION. After landing and unloading at the CKAFS skid strip, (Figure 7-2), Centaur and its ancillary equipment are transported to Hangar J for receiving-inspection. Centaur, flight pallets when new, and adapters are given a brief electrical inspection, and a visual check for transportation damage, corrosion, and overall completeness. All components are prepared for mating, and transferred to Complex 36 for mating and checkout.

7.2.3 PRE-FIRST FLIGHT CHECKOUT AND TESTING. At Complex 36, Centaur will be lifted to level 8 in the service tower for installation of sidewall insulation and radiation shields, in a protected environment, Figure 7-3. The vehicle will then be lowered, mated to its flight pallet, and the vehicle/pallet unit installed in the payload bay simulator on the launcher. The simulator is a truss structure simulating the

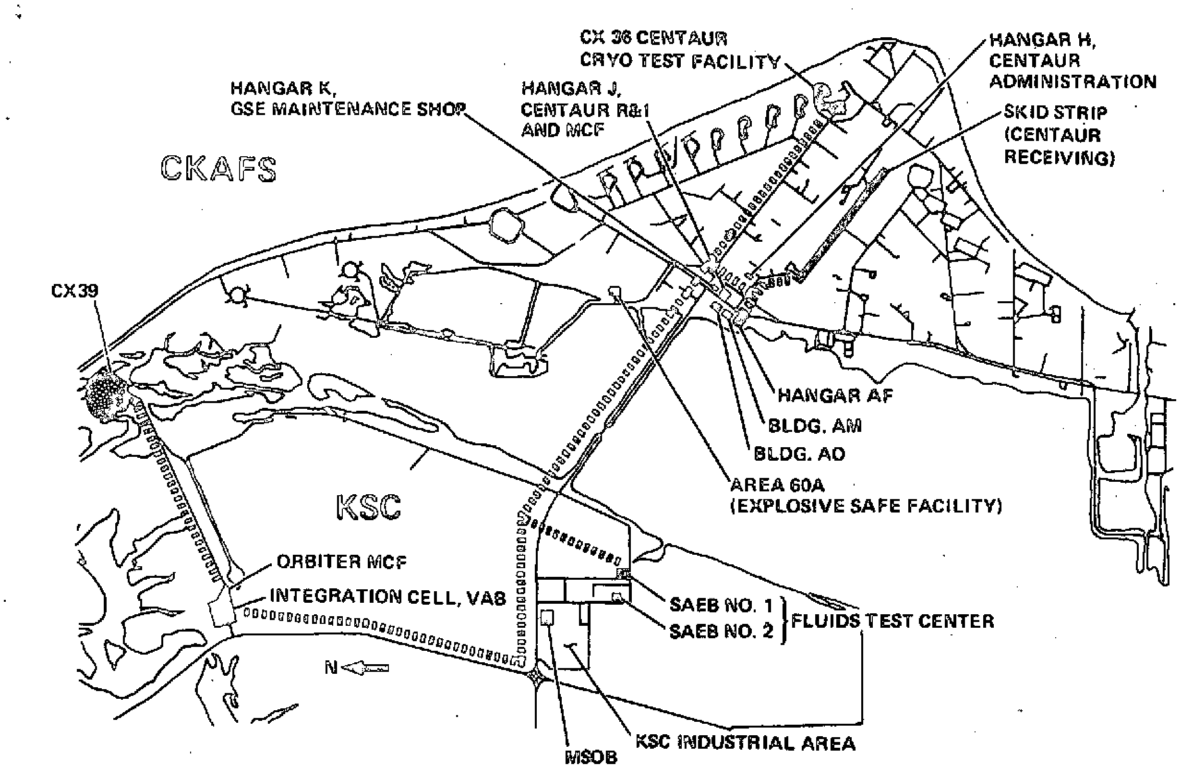


Figure 7-2. Cape Kennedy Operations Facilities

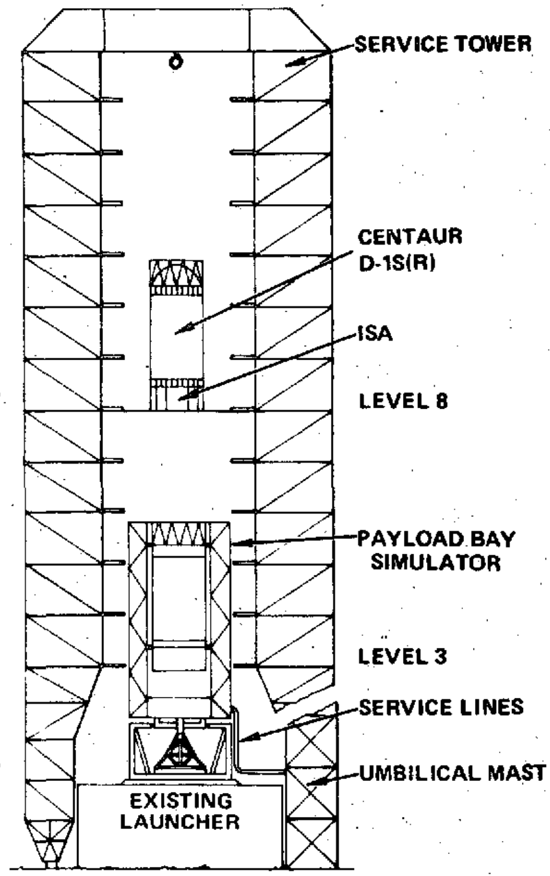


Figure 7-3. Centaur Pre-First-Flight Cryogenic Tanking Test and Checkout Facility, ETR Complex 36

Orbiter payload bay with respect to payload support fittings and hardline panel services for Centaur fluid, gas, and electrical lines. The simulator will mount directly on the existing Complex 36A launcher, to maintain a dual D-1S(R) checkout/Atlas-Centaur launch capability. Service lines will run from the simulator hardline panels down the launcher to the base of the existing umbilical tower, and tee into existing Centaur service lines. Work platform modifications in the Service Tower will only be required at levels 3, 4, 5, and 6, to accommodate the width of the payload bay simulator.

After installation checkout, the complete Centaur/flight pallet unit will undergo fit and function checks, combined electrical tests, a terminal countdown demonstration, flight events demonstration, and a combined electrical readiness test. Functional testing will include peroxide system loading, reaction control system engine firing, and Centaur cyro-tanking. This will be the first and only time that the propulsion systems of each new vehicle will be exposed to propellants and proofed as complete systems, before final loading and launch at Shuttle Complex 39. Individual subsystems and components (main engines, vent valves) will have been exposed to cryogenic temperatures and functional tests, however, during individual acceptance tests (IAT), prior to vehicle final assembly at San Diego.

7.2.4 CENTAUR/SPACECRAFT MATING. Following checkout and tanking tests in the Payload Bay Simulator at Complex 36A, the mated Centaur/flight pallet unit will be transported to a NASA-designated spacecraft mating facility for mating with a spacecraft, entering the normal Shuttle upper stage turnaround cycle at functional block 2.2, Figure 7-4.

Spacecraft mating with Centaur can be done with Centaur horizontal in its flight pallet, on its transport pallet and trailer, or it can be done with Centaur vertical. Horizontal mating requires a 22-foot bridge crane hook height; vertical mating

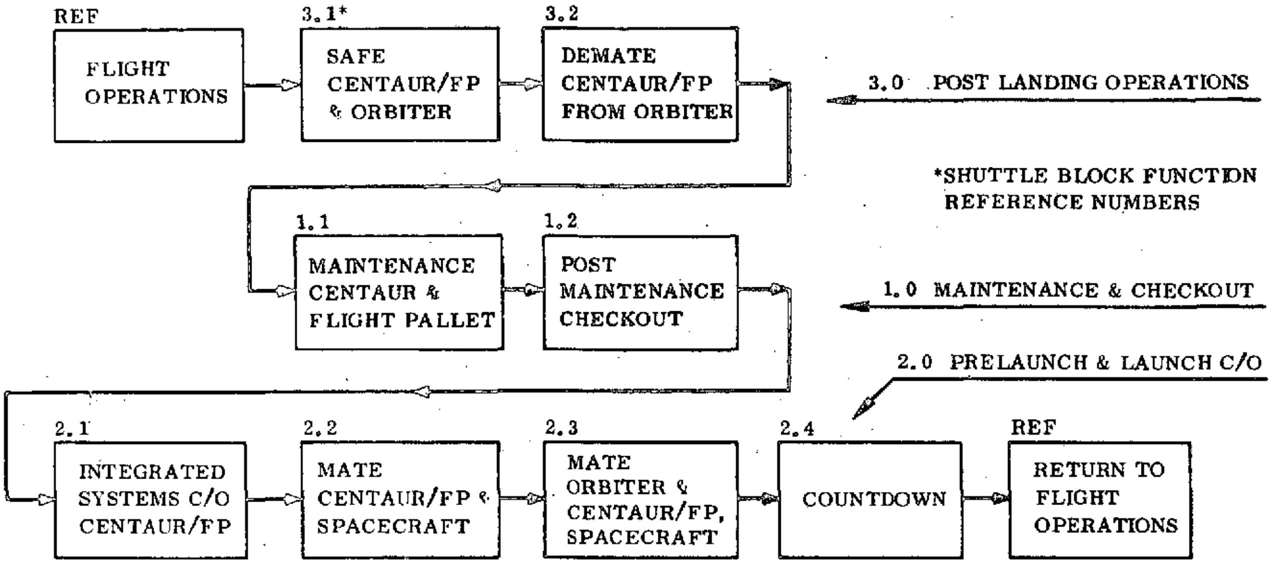


Figure 7-4. Centaur D-1S(R) Top Level Functional Flow Diagram

requires a hook height of 65 to 70 feet. NASA maintains three spacecraft facilities at KSC, west of the Banana River in the NASA Industrial Complex on Merritt Island: Spacecraft Assembly and Encapsulation Buildings No. 1 and No. 2 (SAEB No. 1, SAEB No. 2), and the Spacecraft Operations & Checkout Building, M7-355, referred to also as the Manned Spacecraft Operations Building (MSOB). All three facilities are air-conditioned to Class 100,000 clean room rating, and have floor area, door clearance, crane capacity, and hook height adequate to handle any payload for Centaur.

In the clean room of the spacecraft mating facility, the Centaur protective cover will be removed and the stub adapter interface prepared for spacecraft mating. After mating with the spacecraft, all interface connections and combined Centaur/pallet/spacecraft systems will be checked out, prior to moving to the Orbiter mating area, to assure that the Shuttle payload is in all respects ready for installation in the Orbiter, with minimal possibility of interference with the Shuttle turnaround schedule. Checkout will include system compatibility verification of functional readiness for flight sequence operations.

Electrical and electronic systems will be run through an abbreviated countdown culminating in transfer to internal power, arming, and initiation of a simulated flight sequence. The spacecraft will be monitored to verify receipt of Centaur generated signals at the proper times with sufficient current and voltage levels where applicable. Following the simulated flight sequence, the vehicle will be safed and returned to GSE power for shutdown. This test will be similar to the Complex 36 electrical checkout except for the abbreviated sequence and omission of Centaur pyrotechnic monitoring.

Upon completion of Centaur/spacecraft compatibility checkout and correction of any discrepancies, transport covers will be installed on the Centaur/spacecraft assembly and the complete Shuttle payload transported to the Orbiter MCF. In the case of contamination-sensitive spacecraft such as interplanetary probes, space telescopes, and others involving optics and sensitive surfaces, however, existing Orbiter payload bay cleanliness requirements for Class 100,000 purge gas are not compatible with the spacecraft surface cleanliness criteria, and spacecraft sponsors may provide either a lightweight or structural shroud to protect their craft, rather than a temporary transport protective cover. Such shrouds will be installed immediately after Centaur/ spacecraft mating, and will not be jettisoned until Centaur has been deployed from the Orbiter in space.

"Payload" from this point on is used in reference to the complete Centaur/pallet/ spacecraft unit, as a single integrated Orbiter payload.

7.2.5 PAYLOAD/ORBITER MATING.

Shuttle Preps. Two and a half to three weeks before launch, while Spacecraft and Centaur preps are in progress, Shuttle preps begin with Orbiter landing, safing, and post-flight activities (Figure 7-1). After completion of Orbiter inspection, maintenance and refurb, mission peculiar equipment is installed in the Orbiter in preparation for payload mating. While Orbiter payload support services are not as yet defined, a good possibility exists that many service lines for a cryogenic upper stage payload will be included as permanent Orbiter equipment, running from hardline panels in the payload bay to the Orbiter/ground disconnect panels. These services will include LH_2 and LO_2 fill and drain lines, propellant vent lines, pressurization charge lines, purge lines, and possibly others. Any service lines required by Centaur which are not included as permanent Orbiter equipment will be installed in the Orbiter during the period of preparation for payload mating.

The single major item of mission-peculiar equipment to be installed in Orbiter in preparation for payload mating will be the Payload specialist station system, in support of the Centaur monitor and control system (CMACS). Once the CMACS capability has been installed in Orbiter and checked out, together with any non-permanent TBD mission-peculiar service lines, the Orbiter will be ready for payload installation.

Centaur Preps. On arrival at the MCF nine work days (11 to 13 calendar days) before Shuttle launch, Centaur final preps for Orbiter mating will be performed, including removal of all desiccant plugs, installation of flight batteries and pyrotechnic squibs, and retraction of the two aft payload bay hardline panels to provide installation clearance.

Payload/Orbiter Mating and Checkout. When mating preps have been completed, the spacecraft/Centaur/flight pallet will be lifted from the transporter as a unit, installed in the Orbiter payload bay, and mated.

Test and checkout at the MCF will consist of Orbiter/flight pallet interface checkout and verification, followed by an avionics operations test planned and scheduled by Shuttle Operations Planning to assure integrated Shuttle/payload functional compatibility. This test will include the essentials of a simplified CERT in the safe mode.

a. Orbiter/Flight Pallet Composite Tests. These tests are run for the sole purpose of validating the Orbiter/flight pallet interfaces for continuity and compatibility. No disconnect functions will be involved. The tests will include:

   1. Pressure test and leak check of all fluid and gas interface connections at the Orbiter payload bay/flight pallet hardline panels.

   2. Continuity, compatibility, and function of the digital and analog interface links between the CMACS and the Centaur, pallet, and Orbiter.

b. Avionics Operations Test (AOT). The AOT is undefined as yet. Presumably it will be run to provide an electrical and electronic verification of all Orbiter/Centaur/spacecraft related electrical systems, before Orbiter mating to the Shuttle external tanks and mobile launcher. The test will be run with Centaur in its flight configuration except for propellants and gases, and the spacecraft will be in a complete launch ready configuration including propellants and gases. All vehicle and spacecraft systems will be operated open-loop with their associated ground stations. Landline instrumentation, launch control GSE, and telemetry will be used for event monitoring; control and monitoring will also be performed from the payload station in the Orbiter.

   Electrical and electronic systems will be run through an abbreviated countdown of approximately 45 minutes duration. Centaur deployment, separation, prestart, main engine burn, payload separation, and blowdown phases will be monitored for proper generation of command signals and associated vehicle responses. The test will be terminated after completion of the Centaur programmer cycle, by interruption of power to the vehicle.

After AOT data review and discrepancy correction, Orbiter will be moved to the VAB, erected to the vertical in the transfer aisle, lifted, and mated to the external tank on the mobile launcher, in one of the two VAB integration cells. During the next three to five days, all Orbiter/mobile launcher and Orbiter/external tank interfaces are connected, fluid and electrical systems checked out, integrated electrical interference tests run, ordnance installed, and the Shuttle/mobile launcher prepared for transfer to launch complex 39.

7.2.6 LAUNCH PAD OPERATIONS. Two work days or 37 hours before launch, the mobile launcher and Shuttle are transferred to Complex 39 on the transport crawler, arriving 31 working hours before launch, Figure 7-5. During the first 9 hours at the launch pad, all mobile launcher/pad interfaces are connected and verified, Centaur Monitor & Control System (CMACS) activated, Shuttle powered on, and a final overall combined AOT run. The next 20 hours, until 2 hours before launch, are taken up in data review, cabin closeout and payload servicing, and countdown preps.

Centaur preparations for launch are reasonably flexible and can be accomplished at almost any time during the 20-hour period before standby status is attained at T-2 hours, convenient to Shuttle scheduling.

As shown in Figure 7-5, Centaur hydrogen peroxide loading will be started at T-18 hours (initiation of countdown preparations) and will be completed in 4 to 8 hours, depending on the number of bottles aboard Centaur. Pyrotechnic circuits will be connected at T-12 hours, but not armed, followed by initial helium pressurization of flight pallet and vehicle pressurization and purge storage bottles to 1500 psig. Centaur guidance will be warmed up at T-7 hours, and calibrated and aligned starting at T-5 hours. At T-5 hours, any remaining electrical, pneumatic, and propellant system preps will also be completed. Spacecraft and Centaur environmental conditioning will be initiated at T-3 hours, one hour before start of propellant loading. Standby status will be attained at T-2 hours, and launch countdown initiated at any time after that.

7.2.7 COUNTDOWN AND LAUNCH. A Centaur/Shuttle countdown sequence from T-2 hours to liftoff is shown in Table 7-1, to show integration of major Centaur activities with the Shuttle sequence.

The most recent Orbiter propellant loading sequence shows a total load time of 60 minutes from initiation of countdown at T-2 hours to fast fill cutoff of both LO_2 and LH_2 at T-1 hour. At this point, the external hydrogen and oxygen tanks are 98.5% full, and crew ingress begins while the tanks are being brought up to 100% with the propellant replenish systems.

The current Centaur loading sequence in use today is adaptable to this 60-minute Orbiter load time. As indicated on the combined Shuttle/Centaur loading sequence, Centaur loading will begin at T-110 minutes, after a 10-minute chilldown, and will finish at T-60 minutes with both LO_2 and LH_2 tanks full to the 100% levels.

Centaur propellant loading starts with cold gas chilldown of the LO_2 tank as the facility transfer line is chilled and filled. LO_2 tank fill to 55% and intermediate bulkhead chill follows, at a load rate monitored and adjusted to maintain a saturated LO_2 temperature of -286°F in the vehicle tank. LO_2 loading is secured at 55%, and the hydrogen sequence started.

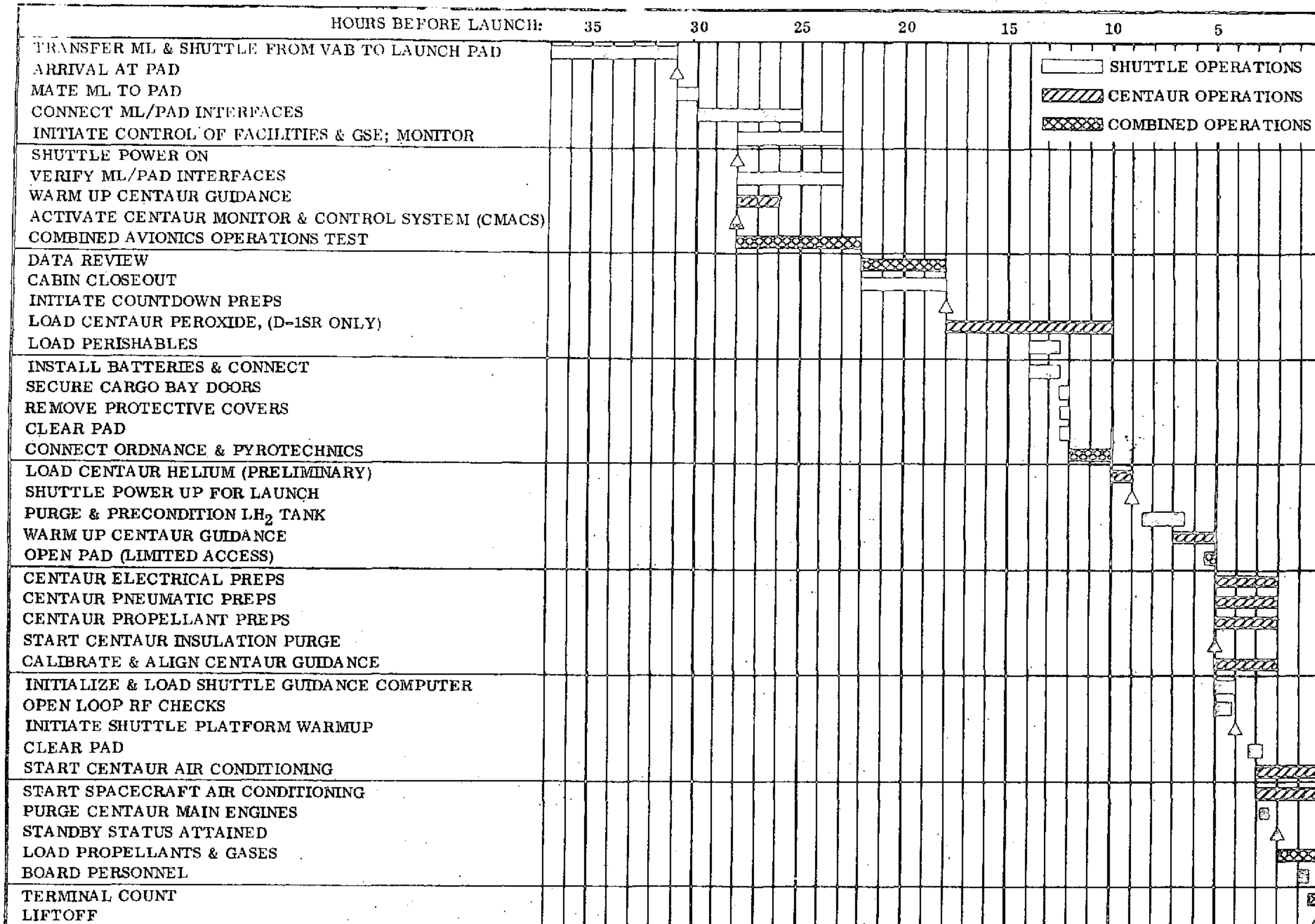


Figure 7-5. Centaur/Shuttle Launch Pad Operations

Table 7-1. Centaur/Shuttle Countdown From T-120 Minutes

Time Minutes	Function
T-120:00.0	Standby status; start LH_2 and LO_2 facility chilldown
120:00.0	Start high rate Centaur insulation and engine purge
114:00.0	Start external tank (ET) LH_2 fill
110:00.0	Start Centaur and ET LO_2 fill
100:00.0	Stop Centaur LO_2 fill; start Centaur LH_2 tank chill
90:00.0	Start flight control countdown preparations; start Centaur final guidance tests and telemetry and RF systems warmup
80:00.0	Start Centaur LH_2 fill
70:00.0	Load Centaur helium, final
68:00.0	Restart Centaur LO_2 fill
60:00.0	Centaur LO_2 and LH_2 fill complete; start replenishing
60:00.0	Crew ingress; start terminal countdown
55:00.0	Shuttle LO_2 and LH_2 fill complete; start replenishing
50:00.0	Start Centaur guidance ready preps
5:00.0	Final status check
1:15.0	Secure cyro propellants
:32.0	Initiate automatic launch sequence
:28.0	Arm Centaur SCU; start DCU count; arm flight pallet
:21.5	Test solid rocket booster (SRB) ignition; steering test
:03.0	Arm SRB igniters
:01.0	Start Shuttle programmer
:00.5	Lock out all holds
:00.0	SRB & Orbiter engine ignition; liftoff

A 20-minute chilldown of the Centaur hydrogen tank is used to properly condition the structure before filling begins. Hydrogen is loaded at 700 gpm until the tank is approximately half full, then slowed to obtain a propellant utilization mixture ratio check between 50% and 60%. After the PU check is made, loading continues at 700 gpm to the 100% level.

LO_2 loading is reinitiated when the PU check is made at about the 55% LH_2 level, and continues along with LH_2 loading until both tanks reach the 100% level at T-60 minutes.

The 180-gpm LO_2 fast fill rate shown is less than the 300-gpm maximum allowable rate for Centaur loading, and the timeline shown here is therefore conservative. The load rate is set to maintain LO_2 saturation temperature and can be increased to compress the loading schedule if desired, by suitable insulation design of the transfer line.

The 700 gpm LH_2 fast fill rate is less than the 750 gpm now used on Centaur loading at Complex 36. The rate can be increased somewhat to further reduce Centaur loading time, if required.

At T-120 minutes, Centaur engine and insulation purges are increased to high rate, concurrent with initiation of chilldown for propellant loading. Propellant loading is completed in one hour and crew ingress and initiation of terminal count start at T-60 minutes. During the loading period, Centaur guidance, flight control, telemetry and RF systems are checked, and helium storage system brought to final pressurization at 2800 psig.

Cryogenic propellant replenishment is terminated 75 seconds before liftoff, and automatic launch sequence initiated at T-32 seconds, followed by arming of Centaur pyrotechnics, Shuttle ignition, and liftoff.

7.2.8 POST-LANDING OPERATIONS (SAFING). Centaur will be involved in Orbiter post-landing operations under two circumstances: normal mission completion, and an aborted mission landing. Both will require Centaur safing activities.

a. Normal Mission Completion. Following Centaur docking with Orbiter in orbit, the Orbiter will return to the Shuttle home airfield (SHA), land, and proceed to the safing area (Figure 7-6) where Orbiter cooldown and safing procedures will take place.

   No safing procedures are required for the Centaur flight pallet other than crew monitoring and checking to assure that the abort helium bottles were blown down in flight to 1500 psi to meet ground handling safety requirements.

   Safing procedures for Centaur are similar to those for Orbiter, each vehicle returning with empty liquid oxygen and liquid hydrogen systems. On landing, ground purge nitrogen will be connected to the Centaur LH_2 and LO_2 fill and

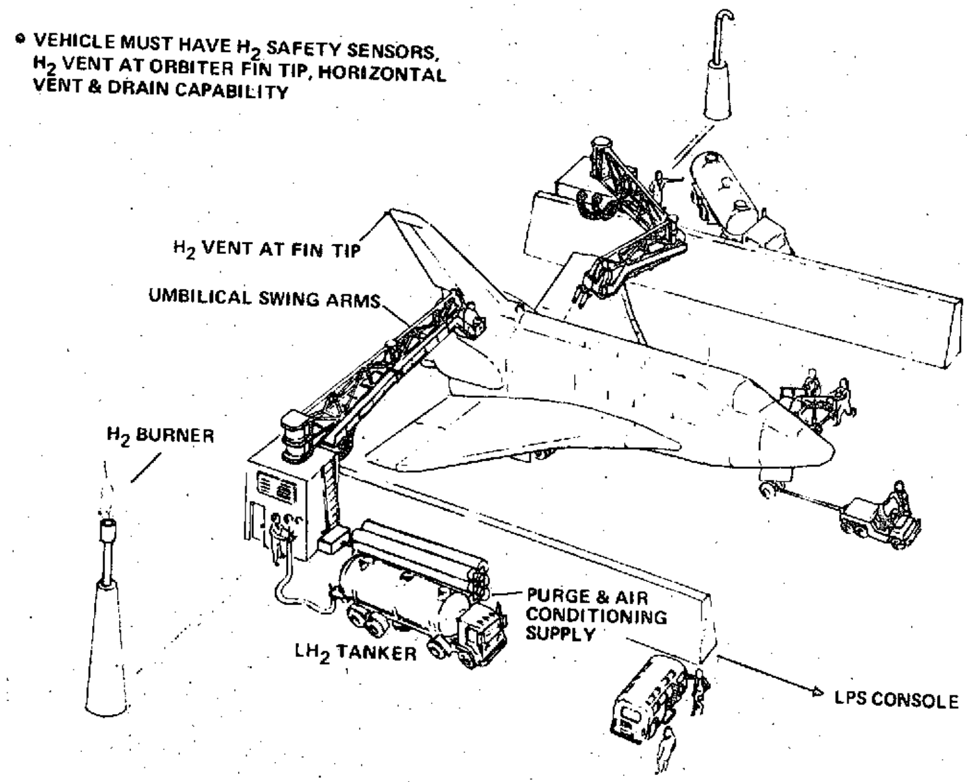


Figure 7-6. Concept of Orbiter Safing Area at Shuttle Home Airfield

drain disconnects on the Orbiter/ground umbilical panel, and vehicle tank vent valves will be unlocked to permit normal "ground hold" tank pressure regulation. Vent discharge at the start of the purge procedure will be mostly helium (which remains as tank pressurant after the inflight propellant dump sequence), mixed with gaseous oxygen or hydrogen. As residuals boil off and are displaced, the effluent from the two tank vent lines will approach 100% GN_2. Vent discharge from the vehicle tanks can be ducted to atmosphere. There is no internal hazard in the vehicle tanks from contained gas mixtures; the primary safety concern is inadvertent application of an ignition source to a gaseous leak from the fuel tank at some later time if the tank has not been properly purged.

Safing procedures for Centaur will also include removal of pyrotechnic squibs, blowdown to safe ground handling limits of all gas storage bottles, and monitoring of all systems before removal of Centaur on its flight pallet and transport to the maintenance facility.

b. Aborted Mission Landing. Of the several possible abort modes other than abort to orbit, the shortest inflight time is that for the "latest decision" return-to-launch-site abort, as defined in the latest NAR abort guidelines. Flight time in this mode is approximately 25 minutes, with 5 minutes available for third stage vehicle propellant dump. Centaur D-1S(R) is therefore presently designed to dump propellants in flight for any abort mode, and to land with empty tanks. At the completion of abort propellant dump, dump lines are purged and Centaur tanks repressurized with pallet-supplied helium. Propellant residuals will not be present in the dump lines at landing, since the Centaur propellants are cryogenic, vaporize quickly, and disperse rapidly.

After landing, safing procedures will duplicate those used for a normal mission completion landing.

7.2.9 PAYLOAD CHANGEOUT AT THE LAUNCH PAD. Payload installation in the Orbiter can be done in the VAB, or at the launch complex. If done in the VAB, facility modification will be required to provide adequate cleanliness control and security for classified payloads.

Payload installation at the launch complex, necessitated by the 10-hour contingency payload changeout requirement, is being considered as a normal installation procedure for some payloads. The procedure takes advantage of changeout equipment required at the pad, eliminates duplicate facility requirements and GSE at the VAB, provides a greater time span for upper stage operational activities outside the Orbiter payload bay, greatly decreases the time required for upper stage/spacecraft support in the Orbiter payload bay, and minimizes the time span between final spacecraft preparation and launch.

Payload installation and changeout capability at the KSC launch pad will require tower structure modifications and addition of launch complex facilities and GSE. A recent concept for vertical installation of payloads at the launch complex is shown in Figure 7-7, with Launch Complex 39 umbilical tower modified to include a swinging payload changeout cleanroom. The concept employs a lightweight, environmentally controlled and protected changeout enclosure that swings over the launch deck to attach to and seal the Orbiter payload bay.

The Orbiter payload is brought to the base of the changeout tower on a ground transport trailer, transferred to an erector, raised to a vertical attitude and lifted by changeout tower hammerhead crane to the vestibule of the changeout room. Entry to the vestibule can be up through the floor, through rear doors, or down through an overhead door. The payload is positioned in a payload changeout unit and moved to the cleanroom. Inside the cleanroom, with the payload bay doors open, the payload is installed in the Orbiter with the payload changeout unit. With umbilical mating completed and work platforms withdrawn, the payload bay doors are closed, seals released, and the cleanroom withdrawn to launch position with blast screens in place.

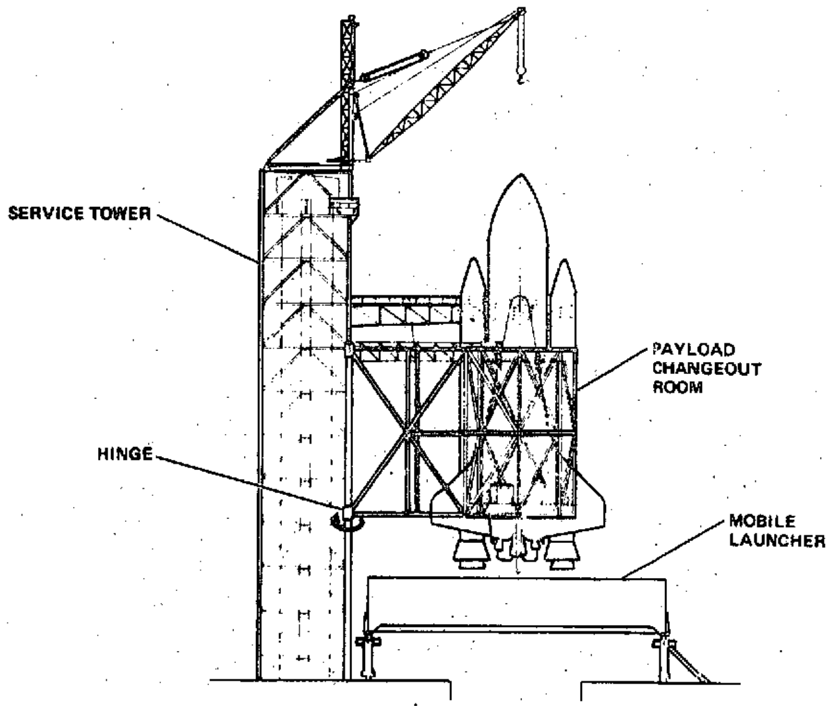


Figure 7-7. Orbiter Payload Vertical Changeout at the Launch Pad

Major items of concern here are cleanliness control during transfer and installation in the Orbiter, functional design of the erector and payload changeout unit, and adequacy of upper stage attach points for horizontal/vertical holding and transfer from one item of equipment to another.

7.2.10 MAINTENANCE AND REFURBISHMENT. Restoration of a Centaur vehicle returning from a Shuttle mission is based on results of the refurbishment and cost analyses performed for the Space Tug Launch Site Service Interface Study, GDCA-BNZ73-003. For Centaur, refurbishment operations require that the vehicle be restored to a state of readiness, at practicable cost, with two objectives:

a. Maintenance of the required level of safety for the Orbiter crew.

b. Restoration to a baseline reliability of 0.97.

Maintenance and refurbishment will be performed using the condition monitored maintenance philosophy proven by the airlines industry to achieve high safety and reliability at low cost. This maintenance method achieves its goals through performance monitoring and inspections of equipment rather than by "hard time" replacement of components, or by test and retest of components. It makes extensive use of operational flight instrumentation (OFI), and advanced methods of conducting internal and external inspections, component testing, leak testing, and non-destructive testing.

The condition monitored maintenance philosophy now used by most major commercial airlines relies solely on technical analysis of flight data and crew reports as a basis for committing to subsequent flights, with no planned testing between flights. Application of this philosophy to an interim reusable upper stage for Shuttle, without any qualification, would require an exceptionally high degree of confidence in vehicle performance and analysis of performance data, for a program of this magnitude in investment and objectives. Reusable Centaur maintenance will therefore be based on the airlines condition monitored maintenance philosophy, supplemented by an integrated systems preflight checkout (CMM_{PF}).

The CMM_{PF} philosophy for Centaur maintenance is based on the assumption that mission performance is the ultimate in system tests. System performance will be monitored during the mission to detect data trends toward parameter limits. The data, combined with the integrated systems preflight test and checkout, will provide the basis for committing to the next mission. This maintenance philosophy requires appreciable onboard performance monitoring capability (CMACS), but results in reliable, fast, and low-cost turnaround.

Remove and replace action will be based upon engineering analysis of the flight data, permitting preplanned maintenance.

Turnaround maintenance, refurbishment, test and checkout of the Centaur vehicle commences immediately following landing and safing of the Orbiter and removal of Centaur from the payload bay. Flight data tapes are removed and submitted to data analysis as a prerequisite to formulation of a preliminary maintenance plan. Following post-flight checkout of the vehicle, the maintenance plan is established, and the vehicle undergoes concurrent scheduled and unscheduled maintenance, followed by post-maintenance checkout, fault correction, correction verification, and preparation of the vehicle for spacecraft mating, or for storage if there is no immediate mission assignment.

Scheduled maintenance is comprised of tasks or actions to be accomplished at specific intervals, in order to assure retention of the inherent design level of reliability. Such tasks or actions include analysis of flight data, inspection, checkout, calibration, adjustment, servicing, repair, removal and replacement.

Unscheduled maintenance is essentially corrective action resulting from scheduled tasks and condition monitoring.

Maintenance and refurbishment of Centaur is accomplished at three levels:

Level I: Maintenance activities performed directly on installed hardware, including fault detection, isolation, correction and prevention.

Level II: Maintenance activities performed in direct support of Level I, consisting of repair and/or disposition of hardware removed during Level I maintenance. Level II maintenance will be performed at maintenance shops located at the launch site.

Level III: Maintenance activities performed in direct support of Levels I and II, performed at off-site locations such as contractor or vendor facilities, or government facilities equipped with the special skills, facilities and equipment required.

Centaur maintenance will be accomplished at Hangar J in the Industrial Complex of the Cape Kennedy Air Force Station (CKAFS), Eastern Test Range. Details of this plan are included in Section 7.5.

7.2.11 CENTAUR REUSABILITY. Centaur D-1T subsystems for propulsion, propellant management, and attitude control require very little modification to provide functional capabilities needed for Reusable Centaur missions. Space Transportation System missions require longer space residency for Centaur and impose more stringent thermodynamic conditions on the systems than originally designed for. Multiple engine starts are required to establish orbits, to return and rendezvous with the Orbiter, and the longer flight time affects propellant heating, venting, repressurization, settling, and attitude control requirements.

Basically, most of the subsystem components remain unchanged and are inherently suitable for longer service life and Shuttle flight environment. These requirements have been considered on an individual basis of component testing in estimating the cost of the Reusable Centaur program.

Funds have been allocated for the component test requirements, including requalification of existing, unchanged or modified components, and new components.

Vibration testing philosophy for the Reusable Centaur will remain the same as that applied to the existing Centaur D-1A and D-1T, differing only in magnitude and extent of requirements. The predicted Shuttle environment is actually very close to that encountered by Centaur when boosted on Titan/Centaur missions, and qualification testing for Reusable Centaur will therefore be primarily concentrated on extension of testing for required life cycles. The Shuttle environment is described in Figure 2-3.

A review of current Centaur hardware specifications, requirements, and qualification test data indicates that in most cases, Centaur D-1T components have received sufficient testing to confirm their suitability for 10 to 20 Reusable Centaur missions, in terms of operational cycles, thermal cycles, pressure cycles, acceleration, and shock. Component fatigue (vibration) life qualification testing may require demonstration in some cases. Typical examples of major subsystem component reusability are discussed below.

Centaur tanks are constructed of very tough, thin, stainless steel with overlapped weld seams and spot-welded doublers and brackets. A decade of background data has been collected, documenting design and quality assurance on welded specimens of tank construction.

The fatigue test curves for Centaur tank specimens are shown in Figure 2-9. A factor of 5 is applied to the 225 cycle data to allow for biaxial skin stress versus uniaxial testing of specimens, stress buildup at tank bosses, thermal cycling, and tank pressure fluctuations in flight. The result is a predicted tank life of 45 reusable Centaur missions. Program costs include full size tank cycle tests to confirm predicted tank life.

The Reusable Centaur uses two RL10A-3-3 main engines with boost pumps, as does the current D-1T Centaur. The only modifications planned for the engines are adjusted bleed valves, resequencing during thermal conditioning, and readjusting the mixture ratio (if required).

As shown in Section 2.3, Pratt & Whitney has indicated complete confidence that the engines could be scheduled for 10 missions without overhaul, and a service life equivalent of 18 RLTC missions has been demonstrated at the P&W West Palm Beach Facility.

Estimates of reusability of Centaur astrionics components are based on the assumption that component design life is equal to the component's MTBF. While the maximum mission duration is 24 hours for D-1S(R) and 47 hours for the RLTC, severity factors for ascent and engine burn give an equivalent flight time of 33.5 and 50.5 hours for the two vehicles. An allowance of 24 hours is made for ground testing during maintenance and prelaunch checks, establishing mission-cycle requirements of approximately 58 and 75 hours for the two vehicles. Design life estimates indicate that the worst case for astrionics components provides a minimum reusability of more than 45 missions for Centaur, with most components indicating a life expectancy greater than 100 missions. This indicates that mechanical systems will limit Centaur reuse life, not astrionics.

In general, Centaur in a reusable mode has a minimum predicted life expectancy of 10 missions, with very little life testing required, and may be extended to 20 missions or more with additional redesign and life demonstration tests. Such extended service life tests may be performed after initial development.

7.2.12 STORAGE. Centaur vehicles not actively engaged in turnaround activities or in preparation for operational activation, will be held in storage at Hangar H, CKAFS, in close proximity to the Centaur MCF (Hangar J), and other spacecraft and Centaur facilities at the Cape. Storage space is adequate for all vehicles required for the 4-, 6-, or 11-year D-1S(R) programs.

Vehicles will be stored on storage/transport pallets in the following condition:

a. In stretch. (Existing procedure.)

b. 0 psig in the fuel tank, connected to atmosphere through a desiccant/filter breather. (Existing hardware and procedure.)

c. 9 psig in the oxidizer tank, locked up. (Existing procedure.)

d. Tank pressures monitored and recorded on 24-hour spring-powered drum recorders on each pallet; oxidizer tank pressures checked daily; pressure increased by bleed-in of GN_2 if required. (Existing procedure.)

e. Pallets have GN_2 K-bottle supply and regulated pressurization systems to both tanks for transport (when tanks are pressurized and not in stretch). These pressurization systems are locked out while the vehicle is in storage. (Existing procedure.)

f. Vehicles covered with vinyl-coated nylon transport/storage covers. (Existing hardware and procedures.)

g. No air-conditioning requirements. (Existing procedure.)

h. No service requirements except GN_2 hose supply for occasional oxidizer tank pressurization. (Existing procedure.)

i. Sidewall insulation stored in separate container with each vehicle, connected to atmosphere through a desiccant/filter breather. (Requirement for D-1S(R) new sidewall insulation.)

Barring natural disaster or deliberate sabotage, Centaur storage under the conditions outlined provides the following advantages: no possibility of tank collapse; no possibility of bulkhead reversal; no possibility of overpressurization or overstress; positive history of tank pressures, providing documented proof that the bulkhead has not been reversed nor the tanks overpressurized; external vehicle cleanliness (covered, no introduction of purge gas contamination over an extended period); no reliance on power or pneumatic supply; no "tank watch" required, with associated manpower costs; no investment or operating costs for air-conditioning.

Upon removal from storage, Centaur will re-enter the turnaround cycle at the Maintenance and Checkout Facility, Hangar J, for reinstallation of the sidewall insulation and radiation shields, final checkout, and preparation for spacecraft mating.

7.3 RLTC GROUND OPERATIONS

Plans for the manufacture, checkout, pre-flight and post-flight activities of the reusable large tank Centaur follow the same pattern as the D-1S(R) operations plans. Details of the two plans differ primarily because of the difference in basic concept of the Centaur/Orbiter interface. D-1S(R) interfaces with its pallet only and is deployed or recovered with the pallet deployment adapter. RLTC interfaces with the Orbiter directly at the forward attach points, and through a separate deployment adapter aft. The RLTC operations plan differs from the D-1S(R) as follows:

a. Manufacture. A checkout deployment adapter and flight deployment adapters will be manufactured for RLTC, rather than the flight pallets.

b. Handling. RLTC will be handled on its own transport pallet only, supported by simulated Orbiter attach point(s) at the stub adapter, forward, and by either a deployment adapter or shipping adapter aft, mating with simulated Orbiter attach points on the transport pallet. D-1S(R) is handled on its own transport pallet, with forward and aft shipping adapters, or in the flight pallet, supported by the stub adapter forward, and by the latch skirt/deployment adapter, aft.

c. Air Transport. RLTC will be shipped in the NASA Super Guppy or equivalent, on its transport pallet. D-1S(R) will be shipped by C-5A, on its transport pallet. D-1S(R) flight pallets will be shipped by NASA Super Guppy.

d. Checkout at ETR. Both RLTC and D-1S(R) vehicles will be given pre-first-flight tanking and checkout at ETR Complex 36, in an Orbiter payload bay simulator. D-1S(R) will be checked out in its flight pallet; RLTC will be checked out in its flight deployment adapter.

e. Centaur/Orbiter Mating. The RLTC will be installed in and removed from the Orbiter payload bay with the deployment adapter considered as a part of the vehicle. Centaur/deployment adapter mating and demating will be done in the Centaur maintenance facility. D-1S(R) will be installed in and removed from the Orbiter payload bay with the flight pallet considered as part of the vehicle. D-1S(R)/flight pallet mating and demating will be done at the Centaur maintenance facility, or at Complex 36, as appropriate.

Turnaround operations for RLTC from Orbiter landing to subsequent launch are based on the top level functional flow diagram shown in Figure 7-8, applicable at both KSC and WTR.

No other major differences exist between the D-1S(R) and RLTC ground operations plans.

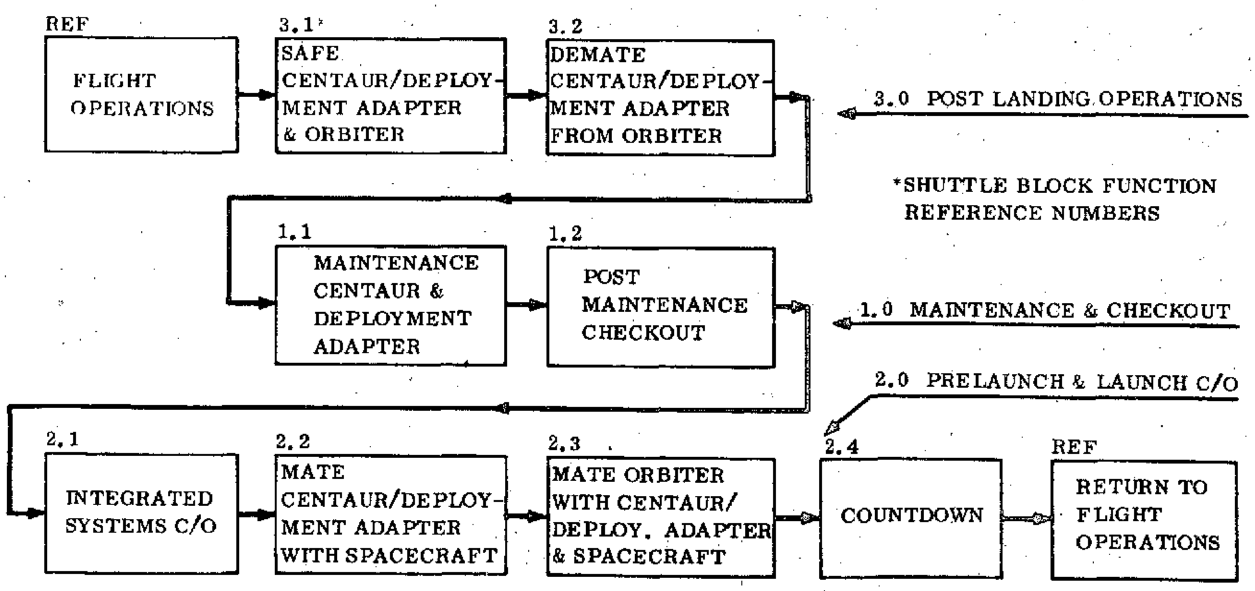


Figure 7-8. RLTC Top Level Functional Flow Diagram

7.4 CENTAUR TURNAROUND TIME

An overall ground turnaround summary timeline for Centaur is given in Figure 7-9. As shown, 280 working hours are required for vehicle turnaround, from landing in one Orbiter to launch in another. Detailed analyses completed during the Space Tug Launch Site Service Interface Study and the present Space Tug Systems Study indicate the 280 hours as a conservative figure. Vehicles in this category have shown an analytical requirement of 265 to 270 hours, in general.

With 24-hour D-1S(R) and 48-hour RLTC missions, both vehicles have a capability of flying 13 to 14 missions per year on a 5-day week, 2-shift per day basis. This capability is far in excess of that required by the mission model, due to the number of expendable vehicle missions that will be flown (see Section 8.6). The turnaround timeline is therefore not pacing and is of importance only in regard to task and scheduling functions. Of the 280 working hours scheduled for Centaur turnaround, 157 hours are constrained by Orbiter activities, as shown. Only 123 hours are Centaur ground operations dependent, including transport to and from Shuttle facilities.

Of the ground tasks involved in the 123 hours of turnaround time which are independent of Orbiter preparations, only maintenance and post-maintenance checkout functions in Block 1 are sensitive to the vehicle configuration. Subsystem and system differences are reflected in slightly different task requirements, task times, and spares,

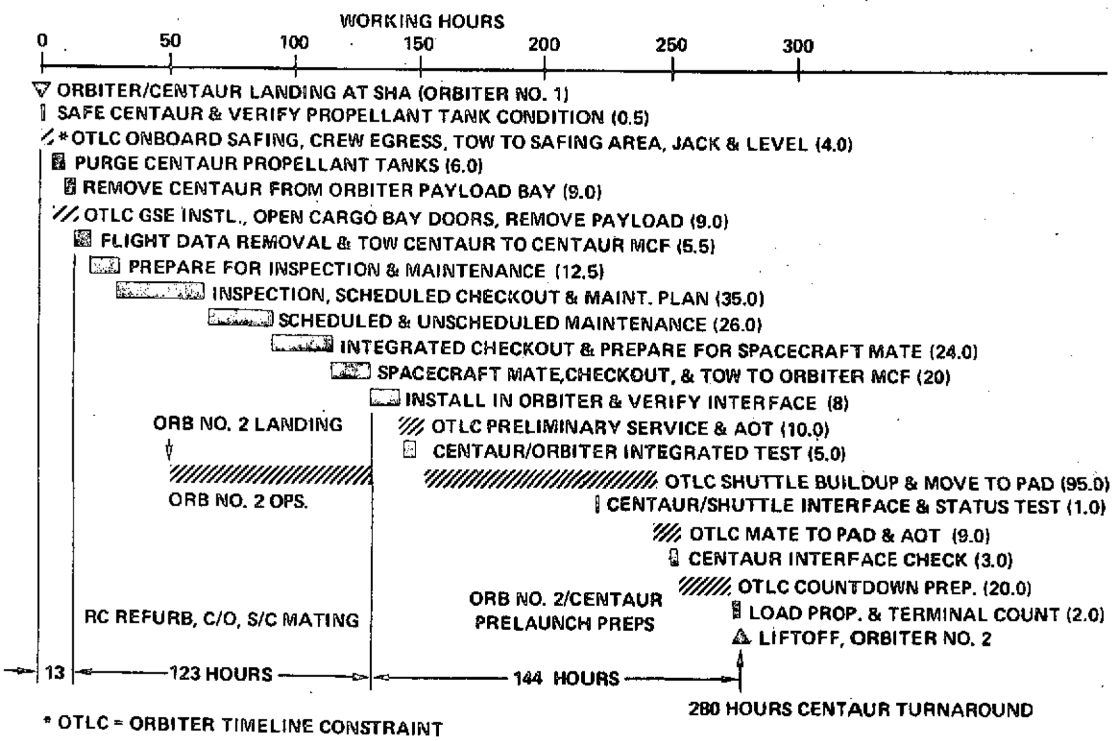


Figure 7-9. Centaur Ground Turnaround Timeline Summary, WTR and KSC

resulting primarily from the differences in the ACPS systems and inclusion of the flight pallet in D-1S(R) maintenance. In Block 2 functions, Centaur/spacecraft mating (2.2) is largely a handling task independent of the vehicle configuration. Centaur systems integrated checkout (2.1) does not differ appreciably between the two vehicles from a time and manpower standpoint; the primary difference is in software requirements to accomplish a largely automated task. The remaining tasks in the ground turnaround cycle are Orbiter timeline constrained.

As shown in Figure 7-9, the first 13 hours and the last 144 hours of the Centaur ground turnaround cycle are Orbiter constrained, per the KSC Preliminary Planning Operations Flow Plan, dated 6 March 1973. Centaur turnaround is based on removal from the Orbiter payload bay at the safing area and transfer to the Centaur maintenance area within 18.5 hours after Orbiter touchdown. If this time increases because of a longer period required for Orbiter cooldown and thermal protection system tasks, or because final Orbiter plans call for payload removal at the Orbiter MCF rather than at the safing area, Centaur turnaround time will increase proportionately.

In the last 144 hours of Centaur turnaround time, from installation in Orbiter to liftoff, Centaur and its spacecraft are dormant from an activity or task standpoint. Centaur systems are compatible with this planned inactivity, but benefit from an increase in maintenance availability if the dormant period is reduced. Payload installation at the launch pad as a routine operation rather than a contingency operation would reduce the dormant period from 144 to approximately 40 hours, and would benefit both Centaur and time-critical spacecraft such as bio-experiments.

Overall, a reduction in Centaur ground turnaround time or an increase will have no effect on planned active fleet sizes for either D-1S(R) or RLTC for the 4-, 6-, or 11-year programs. Present scheduling for these vehicles to accomplish the Shuttle mission model requires no more than eight flights within a year (Section 8.6). Since the 280-working-hour turnaround cycle provides a 14-flight-per-year capability, fleet size and scheduling become independent of turnaround cycle time, within reason.

7.5 FACILITIES, GROUND OPERATIONS

Facility requirements to support Tug ground operations at both KSC and WTR were defined in the Space Tug Launch Site Service Interface Study, Contract NAS 10-8031. Of the Tug ground operations facilities listed, Centaur program requirements duplicate those for the Tug at the Orbiter MCF, VAB, launch pad, and SHA safing and ramp areas. Centaur does not require a vertical maintenance facility, since the propellant tanks are never separated. The remaining facility requirements defined in the study are shown in Table 7-2. Predominant is the Tug Maintenance and Checkout Facility (TMF), with a maintenance, assembly, and test floor area of 29,400 square feet, surrounded by support shops, storage, service and administrative facilities. Total area requirement for the TMF is estimated at 82,000 square feet.

Table 7-2. Ground Turnaround Operations Facility Requirements

Tug (From Launch Site Service Interface Study)		Reusable Centaur	
Tug Maintenance Facility			
Maintenance, Assembly & Test Area:	29,400 sq. ft.	CKAFS Hangar J:	41,300 sq. ft.
Engine Preparation & Service Area:	2,000		
Adapter Service Area:	3,500		
Engineering:	1,500		
Feeder Shops:	9,500	CKAFS Hangar K:	41,300
Receiving & Inspection:	3,200	CKAFS Hangar H:	41,300
Storage (2 Vehicles):	3,300		
Administration:	3,600		
Misc. Services:	6,000		
	82,000		124,000
Pre-First Flight Cyro Test			
Off-Site NASA Facility		CKAFS Complex 36	
Spacecraft Mating			
Spacecraft Assembly & Encapsulation Bldgs 1 & 2		CKAFS Bldg AO, AM, Area 60A; SAEB 1 AND 2	

For the reusable Centaur programs, it appears advantageous to use existing Centaur facilities at KSC for storage, maintenance, and administrative services, deferring the initial investment cost associated with a new Tug maintenance facility until Tug is phased in at a later date. This is particularly applicable to the four-year Centaur program, with no WTR operations planned. Centaur facilities at KSC could provide upper stage support for the first four years of the Shuttle program; during the latter part of this period the Tug MCF could be phased in, either at both sites or at a single centralized facility. Centaur facilities for existing Centaur operations are concentrated east of the Banana River, Cape Kennedy Air Force Station (CKAFS), as shown in Figure 7-2. The vehicles are air-shipped from San Diego, landing at the CKAFS skid strip as shown, taken to Hangar J for receiving and inspection, then to Complex 36 for checkout and launch on Atlas, or to Complex 41 for checkout and launch on Titan. Machine shop services, GSE maintenance, Centaur airborne systems labs, and administrative support are provided at Centaur Hangars H and K.

For the Shuttle program, new Centaur vehicles arriving at the skid strip from San Diego will be taken to Hangar J for receiving and inspection, then to Complex 36 for cryogenics tanking and checkout. After checkout, Centaur will be transferred from

Complex 36 to a NASA-designated payload facility for spacecraft mating, then to Complex 39 Orbiter MCF for installation and subsequent launch at the pad.

Centaur vehicles returning on Orbiter flights, after removal from the Orbiter payload bay, will be transferred to Hangar J for refurbishment and checkout, but will not go to Complex 36 again for tanking during their life span unless major rework has been done on the tank structure. After refurbishment, a vehicle will re-enter the turnaround cycle for reflight, or go to storage in Hangar H.

The following (existing) Centaur facilities at KSC are proposed for Centaur support:

Hangar J: Centaur Maintenance & Checkout Facility (MCF); and R & I.

Hangar K: Machine shop service, GSE maintenance & storage.

Hangar H: Administrative offices, Centaur storage, labs.

Complex 36: Pre-first flight cryogenics tanking & checkout facility.

7.5.1 HANGAR J. Hangar J at KSC (Figure 7-10) has a floor area half that recommended for the Tug program. Shop, storage, and administrative support, however, will be taken care of at nearby Hangars H and K (Table 7-2), already in partial use for Centaur operations, realizing a cost effectivity for combined program operations. Hangar J has existing air-conditioned gyro, telemetry, and battery labs, office space, trenches for pneumatics and cabling, 120 psi shop air, distribution for 2200 psi helium and GN_2 service, 25 ton overhead bridge cranes capable of transfer from one maintenance bay to the other with 24 feet 6 inches of hook height, 200 KVA single phase 120/208 power, and 800 KVA three-phase 480-volt power supply.

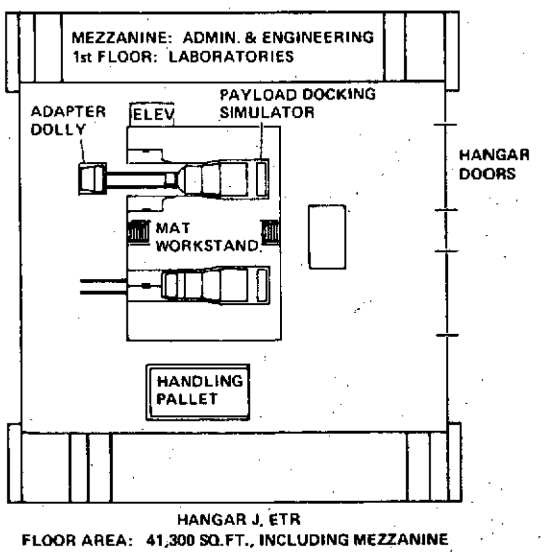


Figure 7-10. Centaur Maintenance Facility

Conversion of Hangar J for either D-1S(R) or RLTC maintenance and refurbishment would involve installation of MAT workstands as shown in Figure 7-10, and installation of air-conditioning in the maintenance area for class 100,000 clean room rating.

The mezzanine area of the hangar is adequate for engineering offices; the main floor area is adequate for aft support adapter, pallet, and main engine service areas. Level II shops are available at existing base facilities and hangar K.

7.5.2 HANGAR K (CENTAUR AND GSE SERVICE SHOP). This facility has two bays, each approximately 60 feet by 160 feet. The South Bay is presently in use as a machine shop for Centaur and GSE; the North Bay is used as a maintenance work area for transport trailers and other equipment. No major change is planned for this facility to provide support for the reusable Centaur programs.

7.5.3 HANGAR H (CENTAUR ADMINISTRATION AND VEHICLE STORAGE). At present, Hangar H is used for Centaur administrative offices, and for a few small airborne system labs. The main high bay area is used for storage space for another KSC launch vehicle program. For the Shuttle program, the main high bay area is proposed as a Centaur vehicle storage facility. Available space is 120 feet by 160 feet, adequate for storage of up to 15 vehicles on transport/storage pallets. For the 4-, 6-, and 11-year programs, a maximum of 5, 7, and 11 vehicles will be operating at KSC; with vehicles in operational and turnaround cycles, the maximum storage requirement is expected to be no greater than 3, 5, and 9 vehicles at any time.

7.5.4 CENTAUR PRE-FIRST-FLIGHT CHECKOUT AND CRYOGENICS TANKING FACILITY (COMPLEX 36). Based upon the test and retest philosophy for first flight, vehicles (whether launched by expendable boosters or Space Shuttle) require an integrated vehicle/flight support equipment systems test under tanked conditions before launch. The test could be performed in the Shuttle before launch, or at a separate facility before Centaur is installed in the Shuttle. The present Shuttle turnaround schedule leaves insufficient time for an integrated tanking test, resulting in the need for a suitable cryogenics tanking/checkout facility. Atlas-Centaur launch complex 36 at ETR is an obvious first choice, since propellant storage and transfer equipment suitable to Centaur needs is available. In addition, the Centaur D-1S(R) ground checkout and launch control capability is accomplished by CCLS, located at Complex 36. The use of Complex 36 for checkout and cryogenic tanking is consistent with the goal of maximum use of existing facilities and minimum operating costs. At Complex 36, launch pad B will remain as the principal Atlas-Centaur launch pad; Complex 36A will be modified slightly to perform the dual function of being the Atlas-Centaur launch pad and the Centaur (for Shuttle) cryogenic test facility.

An additional benefit to be realized by using the existing Centaur facilities is the proximity of Area 60A (Explosive Safe Area) for spacecraft propellant loading when needed, and NASA-operated buildings AO and AM for spacecraft preparation and mating, if desired.

7.6 GROUND SUPPORT EQUIPMENT

The inventory of GSE required to support any launch vehicle in use today covers a wide variety of equipment, from major items such as transport vehicles, shipping pallets, handling adapters, electronic checkout systems, propellant loading valve skids, and engine service units, down to the normal list of minor items such as hand tools, tie-downs, dollies, fixtures and covers.

For the Reusable Centaur vehicles, much of this inventory is already in existence in support of Centaur D-1A and D-1T operations, particularly in the transport and handling categories, and in the category of minor items. Most of the existing equipment is in sufficient quantity to partially if not wholly support concurrent D-1A, D-1T, and Centaur for Shuttle operations. Some items of existing equipment will require minor modification to accommodate the D-1S(R) or RLTC, and some new equipment will be required.

New, modified, and major related items of GSE required to support D-1S(R) and RLTC operations are listed in Tables 7-3 and 7-4. The list includes those ground operations items also used in manufacturing, to give a total GSE requirement summary with allocation and numbers required. Most of the test and checkout items of support equipment are self-explanatory and need no discussion. Other items are fully described in Reference 1, Volume 6. The tables provide inventory requirements for the 4-, 6-, and 11-year programs, existing hardware identification, and net purchase requirements for the programs.

Table 7-3. Centaur D-1S(R) Major Ground Support Equipment

Item No.	Name	Design Status	Program Requirement 4-Year SD	4-Year KSC	4-Year Tot	6-Year SD	6-Year KSC	6-Year WTR	6-Year Tot	11-Year SD	11-Year KSC	11-Year WTR	11-Year Tot	Existing Hardware	Purchase 4-Yr.	Purchase 6-Yr.	Purchase 11-Yr.
1201	Centaur Transport Pallet	Exis	2	2	4	2	6	1	9	2	10	1	13	2	2	7	11
1202	Aft Support Adapter	Mod	2	2	4	2	6	1	9	2	10	1	13	-	4	9	13
1203	Fwd Conical Handling Adapter	Exis	2	2	4	2	6	1	9	2	10	1	13	2	2	7	11
1204	Transport Trailer	Exis	2	2	4	2	2	1	5	2	2	1	5	4	0	1	1
1205	Transport Cover, Centaur	Exis	2	3	5	2	6	1	9	2	10	1	13	0	5	9	13
1206	Transport Cover, Centaur/FP	New	1	4	5	1	4	2	7	1	4	2	7	-	5	7	7
1207	Lift Sling, Centaur Transport Pallet	Exis	2	2	4	2	2	1	5	2	2	1	5	3	1	2	2
1208	Flight Pallet Transport Pallet	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1209	Lift Sling, FP Transport Pallet	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1210	FP Leveling & Stabilizing Kit	New	1	1	2	1	1	1	3	1	1	1	3	-	2	3	3
1211	Insulation Workstands	Exis	1	1	2	1	1	0	2	1	1	0	2	1	1	1	1
1212	Level 8 Workstand	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1213	Payload Bay Simulator	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1214	Payload Bay Sim. Workstands	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1215	Hydrogen Peroxide Transfer Kit	Exis	0	1	1	0	1	0	1	0	1	0	1	1	0	0	0
1216	H_2O_2 Vacuum Drying Unit	Exis	0	1	1	0	1	0	1	0	1	0	1	1	0	0	0
1217	H_2O_2 Hot Firing Kit	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1218	H_2O_2 Transfer Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1219	LH_2 Fill & Drain Skid	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1220	LO_2 Fill & Drain Skid	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1221	CCLS	Exis	1	2	3	1	2	1	4	1	2	1	4	3	0	1	1
1222	FAP	Exis	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1223	GDIE	New	1	1	2	1	1	1	3	1	1	1	3	-	2	3	3

Table 7-3. Centaur D-1S(R) Major Ground Support Equipment (Continued)

Item No.	Name	Design Status	Program Requirement											Existing Hardware	Purchase		
			4-Year			6-Year				11-Year							
			SD	KSC	Tot	SD	KSC	WTR	Tot	SD	KSC	WTR	Tot		4-Yr.	6-Yr.	11-Yr.
1224	Launch Control GSE	Mod	0	1	1	0	1	1	2	0	1	1	2	1	0	1	1
1225	Factory Checkout Trailer	Exis	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1226	Telemetry Ground Station	Mod	1	1	2	1	1	1	3	1	1	1	3	2	0	1	1
1227	Telemetry Data Station	Mod	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1228	CMACS Simulator	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1229	Subsystem Support Equipment (SE) DCU/RMU SE, IMG SE, CCTE, and SCU, SIU, SC Test Sets	Exis	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1230	Maint, Assy & Test Workstand	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1231	Deployment Adapter Dolly	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1232	Pneumatic Test Panel	Exis	1	1	2	1	1	1	3	1	1	1	3	2	0	1	1
1233	Ultrasonic Scan Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1234	IMU Rate Table	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1235	Acoustic Leak Detection Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1236	Radiography Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1237	Mass Spec Leak Detection Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1238	MLI Purge Metering Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1239	Borescope & Fiber Optics	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1240	Main Engine Test Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	1	0	1	1
1241	Engine Handling Dolly	Exis	0	2	2	0	2	2	4	0	2	2	4	4	0	0	0
1242	Spacecraft Docking Simulator	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1243	GH_2 Vent Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1244	GO_2 Vent Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2

Table 7-3. Centaur D-1S(R) Major Ground Support Equipment (Continued)

Item No.	Name	Design Status	Program Requirement 4-Year SD	4-Year KSC	4-Year Tot	6-Year SD	6-Year KSC	6-Year WTR	6-Year Tot	11-Year SD	11-Year KSC	11-Year WTR	11-Year Tot	Existing Hardware	Purchase 4-Yr.	6-Yr.	11-Yr.
1245	Purge Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1246	Star Simulator	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1247	Horizon Scanner Test Target	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1248	Velocity Pkg Transport Dolly	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1249	Velocity Pkg Transport Cover	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1250	Velocity Pkg Handling Sling	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1251	Velocity Pkg Mobile A/C Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1252	ACPS Engine Test Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	2	0	0	0
1253	Propellant Utilization Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	2	0	0	0
1254	Pallet Tractor	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0
1255	Tow Tractor	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0
1256	Escort Vehicles	Exis	2	2	4	2	2	2	6	2	2	2	6	6	0	0	0
1257	C-5A Cargo Aircraft	Exis			1				1				1	Yes	0	0	0
1258	Super Guppy Cargo Aircraft	Exis			1				1				1	2	0	0	0
1259	Cargo Lift Trailer	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0

Table 7-4. RLTC Major Ground Support Equipment

Item No.	Name	Design Status	Program Requirement 4-Year SD	4-Year KSC	4-Year Tot	6-Year SD	6-Year KSC	6-Year WTR	6-Year Tot	11-Year SD	11-Year KSC	11-Year WTR	11-Year Tot	Existing Hardware	Purchase 4-Yr.	6-Yr.	11-Yr.
1301	Centaur Transport Pallet	New	2	2	4	2	3	2	7	2	8	2	12	-	4	7	12
1302	Aft Support Adapter	New	2	2	4	2	2	1	5	2	5	1	9	-	4	5	9
1304	Transport Trailer	Mod	2	2	4	2	2	1	5	2	2	1	5	-	4	5	5
1305	Transport Cover, Centaur	New	2	2	4	2	2	1	5	2	7	2	11	-	4	5	11
1307	Lift Sling, Centaur Trans Plt	New	2	2	4	2	2	1	5	2	2	1	5	-	4	5	5
1308	Deployment Adapter Trans Plt	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1309	Dep. Adapter Pallet Lift Sling	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1313	Payload Bay Simulator	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1314	Payload Bay Sim. Workstands	New	0	1	1	0	1	0	1	0	1	0	1	-	1	1	1
1319	LH_2 Fill & Drain Skid	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1320	LO_2 Fill & Drain Skid	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1321	CCLS (Comp Contr'd Launch Set)	Exis	1	2	3	1	2	1	4	1	2	1	4	3	0	1	1
1322	FAP (Flight Accel. Profile)	Exis	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1323	GDIE (Grd. Dig. Inter. Electr)	New	1	1	2	1	1	1	3	1	1	1	3	-	2	3	3
1324	Launch Control GSE	Mod	0	1	1	0	1	1	2	0	1	1	2	1	0	1	1
1325	Factory Checkout Trailer	Mod	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1326	TLM Ground Station	Mod	1	1	2	1	1	1	3	1	1	1	3	2	0	1	1
1327	TLM Data Station	Mod	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1328	CMACS Simulator	New	1	2	3	1	2	1	4	1	2	1	4	-	3	4	4
1329	Subsystem Support Equipment DCU/RMU SE, IMG SE, CCTE, SCU, SIU, SC Test Sets	Mod	1	0	1	1	0	0	1	1	0	0	1	1	0	0	0
1330	Maint, Assy & Test Workstand	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3

Table 7-4. RLTC Major Ground Support Equipment (Continued)

Item No.	Name	Design Status	Program Requirement: 4-Year SD	4-Year KSC	4-Year Tot	6-Year SD	6-Year KSC	6-Year WTR	6-Year Tot	11-Year SD	11-Year KSC	11-Year WTR	11-Year Tot	Existing Hardware	Purchase 4-Yr.	Purchase 6-Yr.	Purchase 11-Yr.
1331	Deployment Adapter Dolly	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1332	Pneumatic Test Panel	Exis	1	1	2	1	1	1	3	1	1	1	3	2	0	1	1
1333	Ultrasonic Scan Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1334	IMU Rate Table	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1335	Acoustic Leak Detection Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1336	Radiography Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1337	Mass Spec Leak Detection Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1338	MLI Purge Metering Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1339	Borescope & Fiber Optics	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1340	Main Engine Test Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	1	0	1	1
1341	Engine Handling Dolly	Exis	0	2	2	0	2	2	4	0	2	2	4	4	0	0	0
1342	Spacecraft Docking Simulator	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1343	GH_2 Vent Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1344	GO_2 Vent Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1345	Purge Lines	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1346	Star Simulator	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1347	Horizon Scanner Test Target	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1348	Velocity Pkg Transport Dolly	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1349	Velocity Pkg Transport Cover	New	0	2	2	0	2	1	3	0	2	1	3	-	2	3	3
1350	Velocity Pkg Handling Sling	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1351	Velocity Pkg Mobile A/C Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1352	ACPS Engine Test Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	2	0	0	0
1353	Propellant Utilization Kit	Exis	0	1	1	0	1	1	2	0	1	1	2	2	0	0	0

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Table 7-4. RLTC Major Ground Support Equipment (Continued)

Item No.	Name	Design Status	Program Requirement 4-Year SD	KSC	Tot	6-Year SD	KSC	WTR	Tot	11-Year SD	KSC	WTR	Tot	Existing Hardware	Purchase 4-Yr.	6-Yr.	11-Yr.
1354	Pallet Tractor	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0
1355	Tow Tractor	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0
1356	Escort Vehicles	Exis	2	2	4	2	2	2	6	2	2	2	6	6	0	0	0
1357	C-5A Cargo Aircraft	Exis			1				1				1	Yes	0	0	0
1358	Super Guppy Cargo Aircraft	Exis			1				1				1	2	0	0	0
1359	Cargo Lift Trailer	Exis	1	1	2	1	1	1	3	1	1	1	3	3	0	0	0
1360	Fuel Cell Test & Checkout Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2
1361	Hydrazine Service Unit	New	0	1	1	0	1	1	2	0	1	1	2	-	1	2	2

SECTION 8

PROGRAMMATICS

Programmatics, encompassing the plans, resources and scheduling required to support a program, is a significant factor in the ultimate economic and technical success of a program. Careful attention to programmatic analysis details is particularly essential if realistic evaluation is to be made of program options and alternatives.

Included in this section are the plans, resources, and schedules required for design, development, testing and manufacture of Centaur as a Shuttle reusable third stage. Two configurations are considered: the Reusable Large Tank Centaur (RLTC) and a modified version of the existing D-1 Centaur, designated the D-1S(R). The configuration, performance, and operational capabilities of both Reusable Centaur stages have already been described and are the basis for deriving the specific development programs summarized in this section. The development philosophies for both programs are similar. Specific differences are noted under each subsequent programmatic subsection.

8.1 PROGRAM SCHEDULES

The summary program schedule for both the D-1S(R) and the RLTC versions of the Centaur reflect a 51-month development phase leading to an IOC on 31 December 1979. This development span is the same as proposed for an interim direct developed Tug such as the Program 1 vehicle covered in the Space Tug Systems study. The 51-month period is minimum for the Program 1 Tug development, defining the critical path to IOC with little or no provision for contingency. The same 51-month period for the RLTC program provides some contingency for unscheduled events. The D-1S(R) has considerable contingency in the same time since it is the most like the existing Centaur. A minimum development program is established to meet Shuttle safety and third-stage reusability requirements. Both vehicles will have a payload deployment only capability, and emphasis on a low-cost approach to development. They also make use of existing D-1 hardware design, or modifications thereof, wherever applicable and compatible with the Shuttle safety and reusability requirements.

Figure 8-1 is the program summary schedule for the D-1S(R) as developed for a Shuttle third-stage vehicle. Development of the D-1S(R) vehicle is straightforward, inasmuch as most of its systems and components are either off-the-shelf hardware or represent current state-of-the-art development. The system development and integration activities shown are considered sufficient to support the initial IOC flight and place an operational payload in its required orbit, with a high confidence level. This initial vehicle will carry development flight instrumentation (DFI) and, following deployment of the

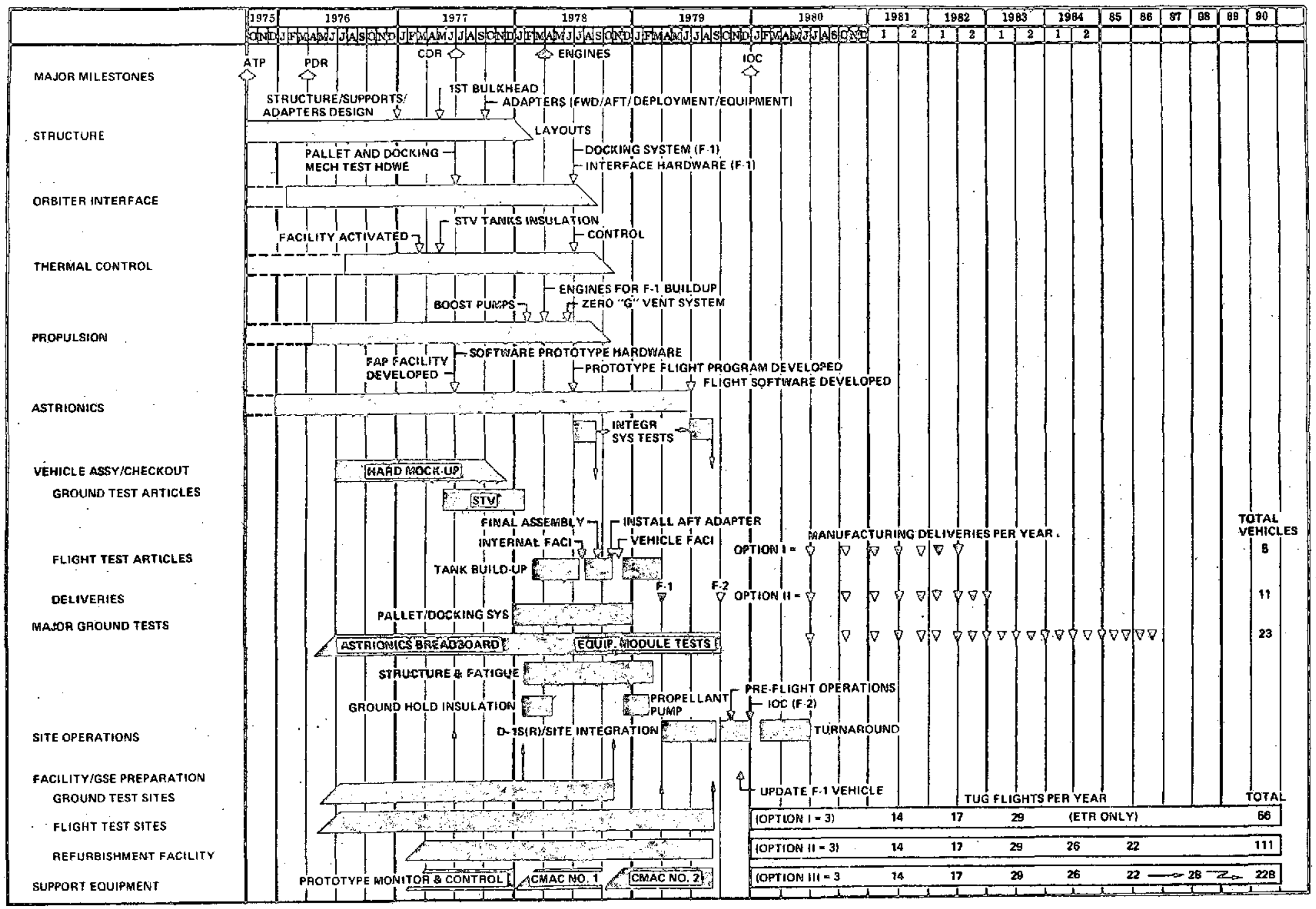


Figure 8-1. D-1S(R) Program Summary

designated payload, will be exercised in various modes to attain verification of flight data not available from the payload placement flight profile.

Major ground tests on the D-1S(R) program are minimal, since supporting data is available from prior D-1A and D-1T flight programs. Consequently, there is only one major ground test article in the DDT&E program; it is designated for structural loads testing and fatigue and life cycle evaluations. Other major tests include the deployment adapter and the astrionics module tests.

There are a total of 5, 11, and 23 D-1S(R) vehicles required for support of the 4-, 6- and 11-year program options, respectively. (See Section 8.6.) The initial flight vehicle will be at the operational site some six months prior to start of pre-flight operations for vehicle/site integration, compatibility, and verification of operating procedures (including checkout of Centaur Hangar J and Pad 36). Each production D-1S(R) will be checked out at Hangar J following delivery, and run through initial tanking checks at Pad 36 prior to payload mating and Orbiter integration.

Ample contingency time is included in the D-1S(R) development schedule to account for any reasonable unscheduled events, including slippage of long lead elements such as avionics hardware and software development.

Figure 8-2 is the program summary schedule for the RLTC as developed for a Shuttle third-stage vehicle. The development phase for this vehicle is similar to that outlined for the D-1S(R) inasmuch as the ground test program is minimized and there are no dedicated flight test vehicles. The first flight for the RLTC is also designated as the IOC. This initial RLTC flight vehicle carries DFI for post-payload deployment, flight-mode evaluations. The DDT&E activities leading up to IOC are only slightly more constrained (timewise) for the RLTC than for the D-1S(R), due to additional advanced hardware such as fuel cells and other astrionic equipment. The major ground test programs are also similar except that the RLTC uses two structural test articles to satisfy the structural loads testing and life-cycle evaluations (including the ground hold insulation and propellant dumping tests).

As discussed in Section 8.6, a total of 4, 7, and 16 RLTC vehicles are required for support of the 4-, 6-, and 11-year program options, respectively. The first flight vehicle will be at the operational site six months prior to start of initial preflight operations. During this six-month period, the RLTC will be integrated with the operational site in much the same manner as was the D-1S(R), including Centaur Pad 36 and Hangar J wringout. Both the initial pre-flight operations and post-flight turnaround activities are scheduled for three-month periods to assure ample contingency time. Post-flight maintenance and refurbishment operations will eventually decrease to a nominal span of 3.2 weeks after about 12 to 15 flights.

The RLTC development schedule reflects a nominal, low-cost program approach leading to an early IOC, but with somewhat less inherent contingency time than was

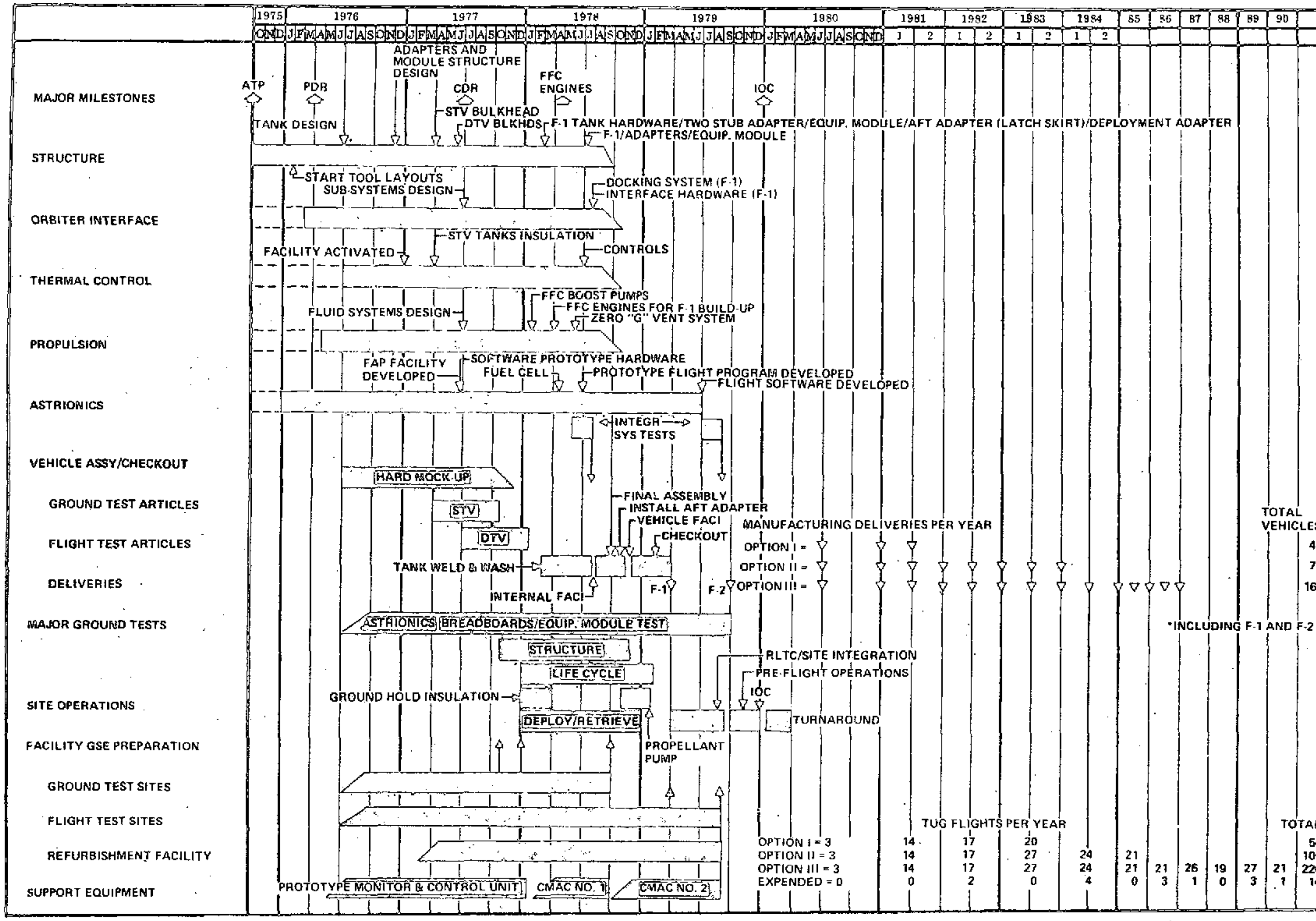


Figure 8-2. RLTC Program Summary

attainable with the D-1S(R) program (for the reasons discussed above). The potentially pacing elements with the RLTC program include flight software development and critical astrionics hardware.

8.2 DEVELOPMENT TESTING

The test programs summarized herein are based on an assessment of planned Reusable Centaur subsystem designs. Key factors considered were determination of those items which are critical to crew safety, the degree of technology required for new design, and the items for which existing Centaur design will be used. Since the majority of the reusable Centaur subsystems are of existing design with proven technology, the total magnitude of the test program is relatively small. Table 8-1 presents an overall summary of the test programs required to support development of both the D-1S(R) and RLTC vehicles. Note that for the D-1S(R), a single test vehicle is planned to satisfy both the structural loads and the life-cycle testing, whereas the RLTC program uses two test articles for this purpose. Current Centaur test philosophy and criteria along with Centaur D-1A and D-1T test experience were used to determine the unique reusable Centaur test requirements. Analyses indicate that the major test requirements for both the D-1S(R) and RLTC are within current state-of-the-art techniques. Consequently, low cost and high confidence were primary considerations in formulating these overall test programs.

Table 8-1. Test Program Summary

TEST ACTIVITY	TEST SITE	TEST ARTICLE	TOTAL COST ($M) (INCLUDING TEST ARTICLE)		REMARKS
			D-1S(R)	RLTC	
COMPONENT QUALIFICATION	CONVAIR VENDOR LABS	PREPRODUCTION COMPONENTS	2.4	6.8	ALL NEW HARDWARE REQUIRES QUALIFICATION.
COMPONENT EXTENDED LIFE.	CONVAIR LABS	PRODUCTION COMPONENTS	1.0	0.6	SOME EXISTING CENTAUR COMPONENTS SHOULD BE TESTED FOR LONGER VIBRATION LIFE.
PROPULSION TESTS	P&W (E5)	ENGINES AND FEED SYSTEM HARDWARE	3.3	3.6	ENGINES RETURNED TO INVENTORY AFTER TEST
STRUCTURAL TESTS	MSFC-4650	LH_2/LO_2 TANKS, OTHER SIGNIFICANT STRUCTURE,		3.1	STRUCTURAL LOADS TESTS REQ'D ON RLTC ONLY
LIFE CYCLE TESTS	MSFC-4557	INSULATION, & PROPELLANT SYS HDWR	3.2	4.2	INCL. FUNCT. TESTS OF INSULATION, PROPELLANT LOADING & RAPID DUMP SYS.
ASTRIONICS SYSTEMS	CONVAIR JSC (SIL) KSC	ASTRIONICS SYSTEM & SUPPORT STRUCT.	3.3	5.1	HARDWARE, SOFTWARE & GSE INTEGRATION
DEPLOYMENT/RETRIEVAL SYSTEM TESTS	CONVAIR	ORBITER INTERFACE HARDWARE	2.9	1.6	DEMONSTRATES FUNCTIONAL CAPABILITY
ON-ORBIT TESTS		1ST FLT. ART.	1.1	1.5	"TESTING" PERFORMED AFTER PAYLOAD DEPLOYMENT
TOTAL PROGRAM COST ($M)			17.2	26.5	

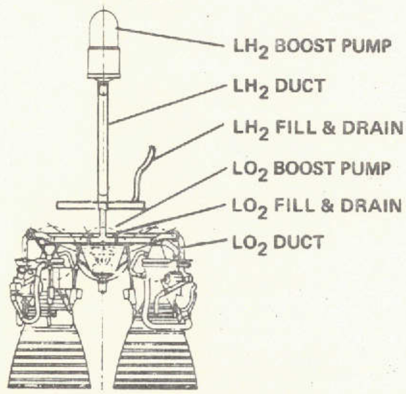
The majority of existing design components to be used on the Reusable Centaur configurations have been qualified to environments at least as severe as those to be encountered in the Reusable Centaur mission mode. New components will be subjected to complete qualification testing and existing hardware will, as required, be subjected to additional testing to qualify under reusability criteria. Flight pallet (D-1S(R)) and deployment adapter (RLTC) components are basically new and will be subjected to complete qualification tests. Component and subsystem tests will be minimized whenever higher level major ground test results will provide the required data. All component test data will also serve to support flight worthiness certification and reliability, reusability, and maintainability assessments.

The RL10A-3-3 engines proposed for Reusable Centaur applications are identical to the engines in current use on Centaur. Changes in the technique of thermal conditioning of the propulsion system require verification testing of the total propulsion feed system. Minor engine adjustment and subsequent integrated testing with support subsystem hardware can be economically accomplished by the engine contractor at his facilities. Altitude simulation is not a requirement for these tests. Engine support subsystem hardware will be integrated with the basic engine testing at the engine contractor's site, as reflected in Figure 8-3.

Structural load tests of the D-1S(R) propellant tank structure will not be necessary. This structure is nearly identical to that of the present Centaur for which an adequate

- CHANGE IN MODE OF CHILLDOWN NECESSITATES RETEST
- ALTITUDE FACILITY NOT REQUIRED
- BUILDS ON ENGINE DATA BASE COMPILED DURING CENTAUR DEVELOPMENT
- MOST COST EFFECTIVE FACILITY TO USE IS P&W/A TEST STAND E-5
- CONVAIR WILL SUPPLY PROPELLANT FEED, FILL & DRAIN HARDWARE
- P&W WILL CONDUCT TESTS
- GFP ENGINES RETURNED TO INVENTORY

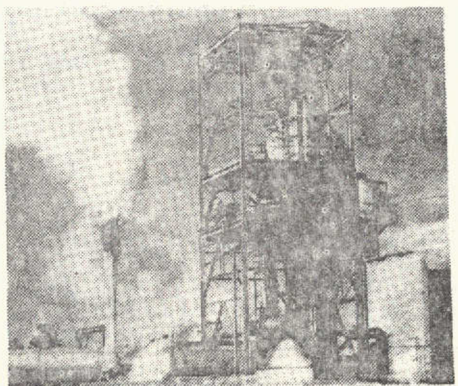
TEST ARTICLE



TEST OBJECTIVES

- ESTABLISH & VERIFY CHILLDOWN
- ESTABLISH & VERIFY START SEQUENCE
- CONFIRM MIXTURE RATIO ADJUSTMENT

TEST FACILITY



P&W/A TEST STAND E-5

- BATTLESHIP TANKS
  LH_2 = 2,000 GAL.
  LO_2 = 1,000 GAL.
- SUPERSONIC DIFFUSER & STEAM EJECTOR SYSTEM FOR STARTS.

Figure 8-3. Propulsion Testing

amount of test data is available. However, load tests of the RLTC propellant tank structure will be required to qualify the changes incorporated in this design. Structural extended life testing for reusability verification will be required for both approaches. Structural tests for the D-1S(R) and RLTC support structures (adapters, equipment modules, etc.) are to determine load paths and interaction effects under various loading conditions and to verify design and fabrication of these structures. The D-1S(R) equipment module and stub adapter are exempt from these tests since sufficient data is available from numerous basic Centaur development tests conducted on similar items. Structural loads applied to the RLTC structural vehicle include ultimate axial, lateral, and torsional design loads to verify design strength adequacy of the large tank structure. Figures 8-4 and 8-5 reflect the major structural tests for the D-1S(R) and RLTC vehicles, respectively.

The life-cycle test programs for both D-1S(R) and RLTC are divided into three phases, each conducted at the same test facility and on a single test article for either program. Phase 1 will be a thermal evaluation of Centaur vehicle ground hold insulation, Phase 2 will demonstrate adequate safe-life characteristics of the tank structure dictated by reusability requirements, and Phase 3 will investigate the vehicle propellant rapid dump system qualities. Structural testing to failure is not considered a requirement in this program since sufficient empirical data is available from previous Centaur test tank development testing to support actual margins of safety and safe life of the structure.

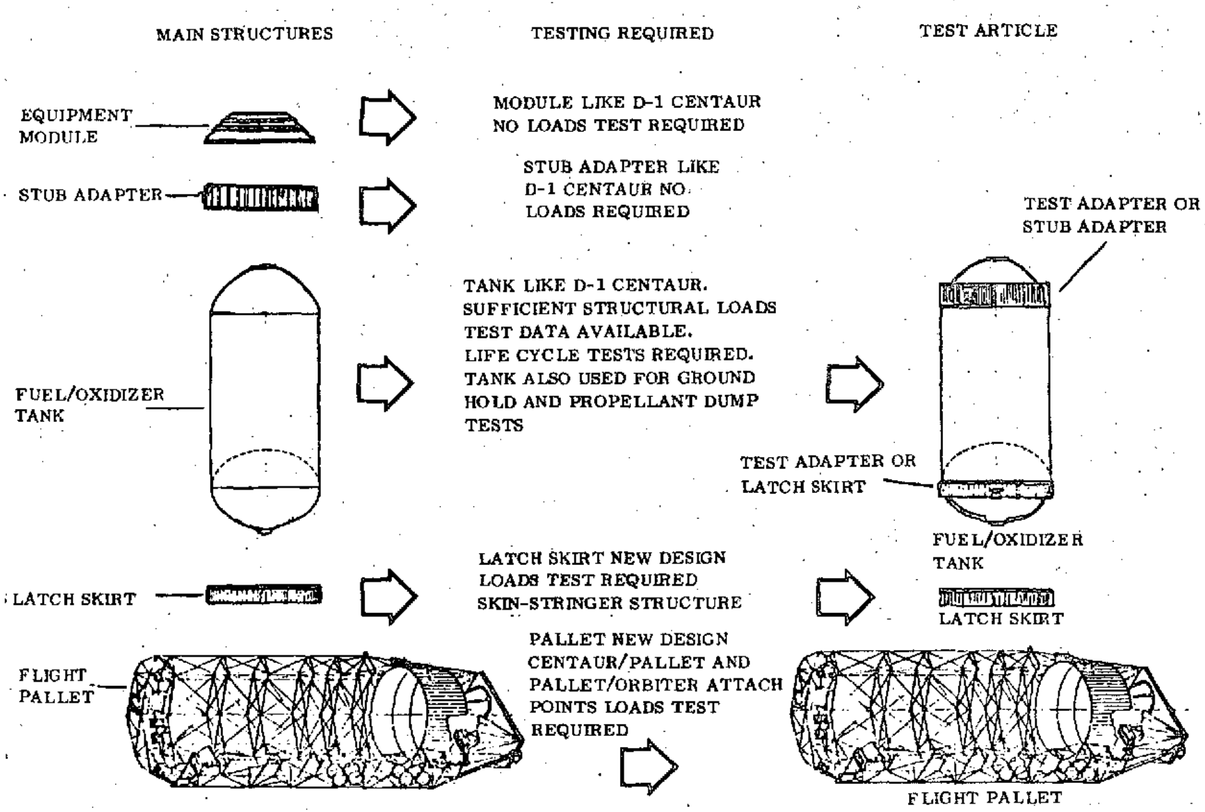


Figure 8-4. Major Structural Tests - D-1S(R)

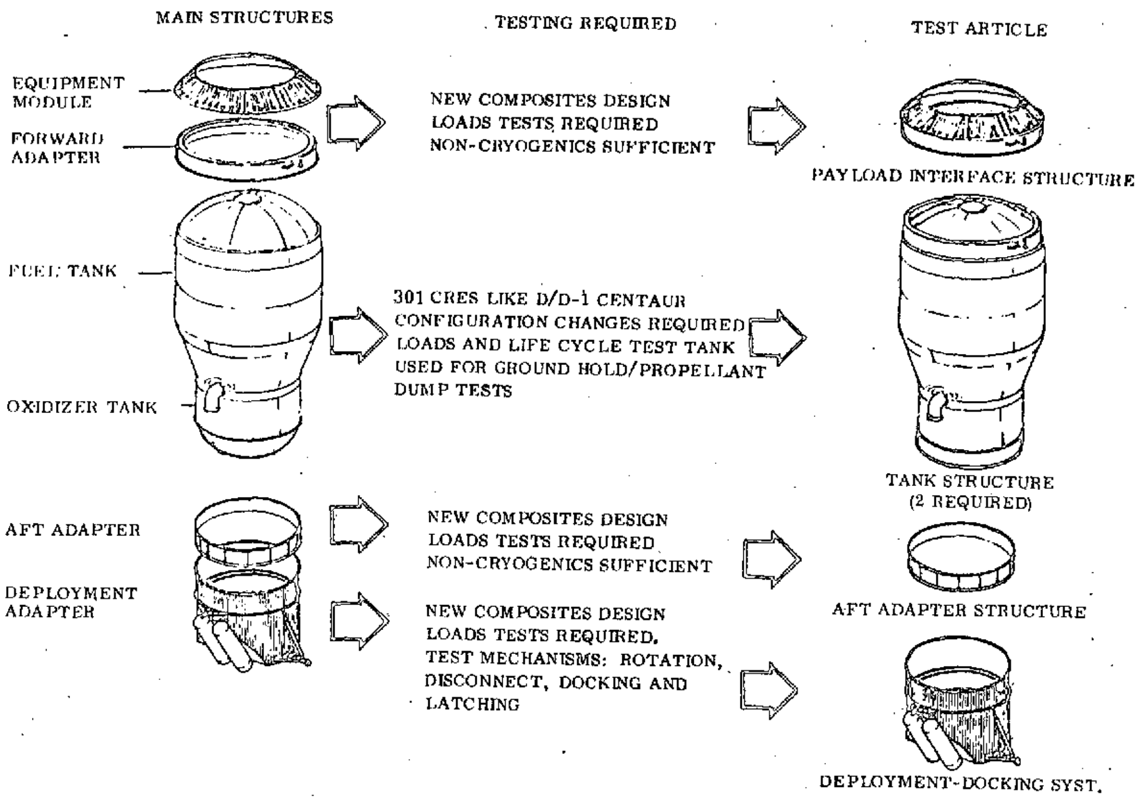


Figure 8-5. Major Structural Tests - RLTC

Satisfactory performance of Reusable Centaur deployment and retrieval systems will be demonstrated by a test program. The test article will contain the actual docking and hardware retrieval and interfacing hardware. All other hardware will be simulated to provide the correct weight and center of gravity. The D-1S(R) and RLTC deployment/retrieval systems are similar in concept and will be tested in a similar manner. The tests will consist of repeated deployment and retrieval of the test article from a deployment adapter under simulated operational conditions of low temperature, approach angles, and maneuvering loads.

Propulsion testing by the main engine contractor will include verification of engine modifications for slow thermal conditioning. The boost pump changes to accomplish the engine modification will be verified at the Sunstrand test facility, where units will demonstrate the low flow rates, and by similarity with additional long idle mode operations incorporated. A verification of total system operation will be accomplished as part of the thermal conditioning testing at the P&W E-5 dual engine test stand.

Extensive P&W testing and analytical work formed the basis for orifice sizing against extended D-1T expected environmental analysis and testing. Testing to verify the STS thermal conditioning times and propellant losses will be accomplished on P&W's E-6 single engine test stand. Operation at 5.8:1 mixture ratio (if used) will be demonstrated on a single engine stand. From there, a complete Reusable Centaur propulsion system, including boost pumps, isolation valves, production ducting, and dual engines will be tested to verify the previous work. Tolerances in the analytical work have been conservative, with testing to narrow the expected system tolerances. System test, including the full maximum expected environmental temperature induced variables, will verify the slow thermal conditioning sequence with one hundred five-second tests.

The main engines for either the D-1S(R) or RLTC vehicles will not require altitude chamber testing of the revised chilldown sequencing. These tests, as planned for the P&W E-5 test stand, are similar to those used to derive current Centaur inflight chilldown sequencing. Thermal conditions and engine interface conditions in these tests are close to flight conditions, and provide assurance that the sequencing derived through successful tests will be adequate, with margin, for flight. Further, highly successful results from previous E-5 testing for D-1, D-1A, and D-1T Centaur have shown that testing in an altitude chamber is not required and would result in a propellant savings of only 25 to 45 pounds, certainly not worth the cost of altitude chamber testing.

Thermal vacuum testing at the vehicle level is also considered unnecessary and inadvisable in light of Centaur experience. Considerable thermal and functional data relative to a space environment has already been accrued from previous Centaur development tests and operational flights and will continue to be accrued on D-1T Centaur flights. The multi-layer ground insulation blankets will be tested during a ground-hold test planned for the flight article at ETR Launch Complex 36A. The two-layer perforated shields for space-environmental tank thermal control will be demonstrated on D-1T flights. System and component installations and design are geared to thermal isolation from the cryo propellant tanks. Resulting mutual thermal decoupling renders each item a separate thermal control task which is uniquely appropriate for system and component-level testing rather than vehicle-level testing.

Astrionic hardware interfaces, procedures and computer control software for testing of electronic systems on the D-1S(R) and RLTC will be demonstrated during subsystem level tests while installed on astrionics modules in a flight configuration. These Astrionics Modules, with associated AGE, will be used with the Shuttle SIL at JSC, for Orbiter compatibility testing. They will also be transported to KSC Hangar J for compatibility checks with the facility equipment. In the VAB, the Modules will be positioned at a point simulating final installation to checkout GSE compatibility, and module function checks will be performed to verify interfaces. At Complex 39, the Astrionics Modules will be installed at simulated launch positions in the MST and a simulated prelaunch test program and launch operation performed. The module will eventually be returned to the vehicle contractor for use in further software development and verification based on mission peculiar requirements.

Operational site facility integration with Reusable Centaur vehicles is accomplished through use of the initial production flight vehicle and the astrionics module used in the contractor's systems integration testing. These units are used for a complete run-through of all planned pre-flight activities to validate or verify vehicle/launch site facility and AGE interfaces, to proof handling and operational procedures, and to train operational crews.

Similarity of the Reusable Centaur to the existing Centaur makes a dedicated flight test unnecessary. It will be desirable however, to obtain data to substantiate the validity of ground testing and to gain confidence in Reusable Centaur performance capabilities. This will be accomplished by incorporating additional instrumentation on the first flight article and conducting on-orbit tests on a non-interference basis with its operational mission. These tests will impose effects of zero-g and other space environmental conditions which cannot totally be duplicated in ground testing. The Reusable Centaur's inherent capability for relatively long orbital stay times permits an extensive amount of testing to be performed during a single flight mission. A typical flight, such as reflected in Figure 8-6, could be divided into phases where progressively more stringent operating conditions would be separated by periods of data evaluation. An operational payload could be placed into a low performance orbit, after which the remaining performance reserve of the Reusable Centaur could be used for flight verification testing into the higher risk operating regimes as shown in the figure.

8.3 VEHICLE MANUFACTURING

The manufacturing plans required to produce both the RLTC and D-1S(R) Reusable Centaur vehicles are based on existing resources and plans used in the production of the

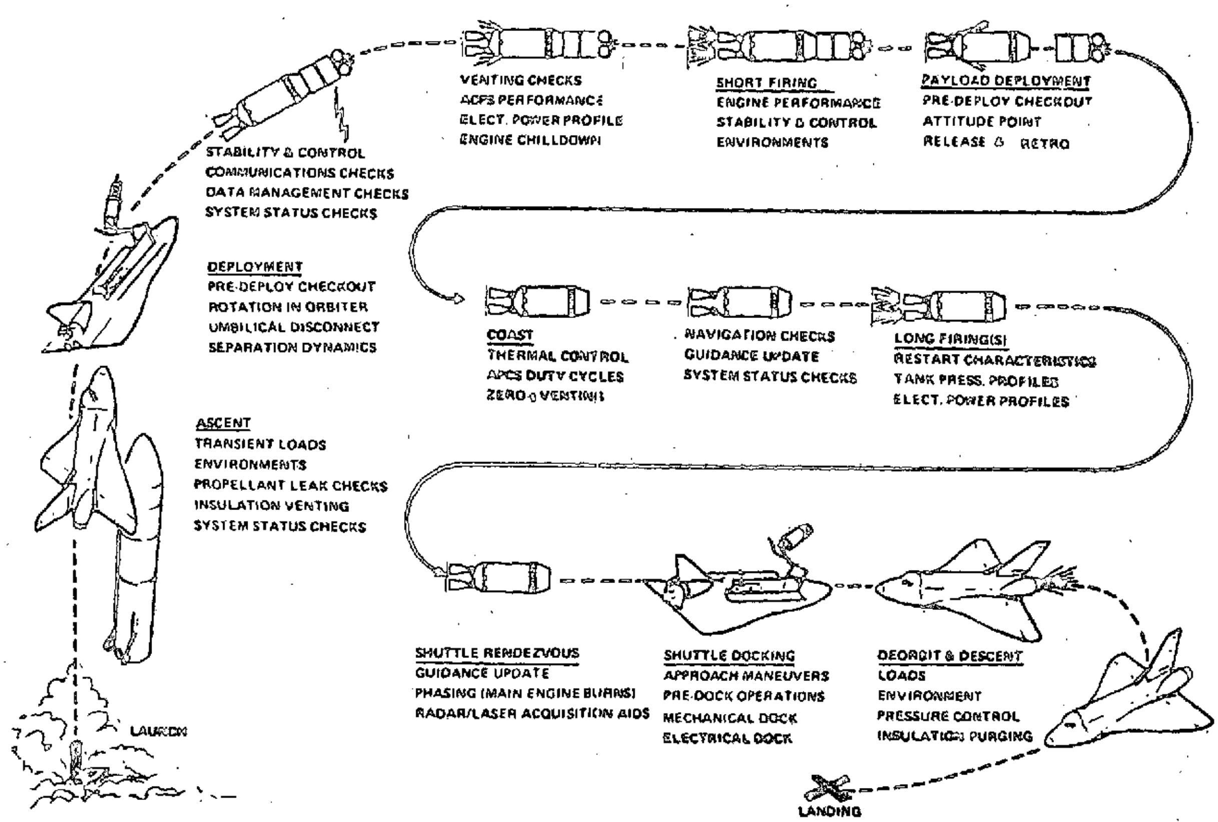


Figure 8-6. Typical On-Orbit Testing

current D-1A and D-1T Centaur vehicles. Wherever possible, existing tools, facilities and operations have been incorporated in the Reusable Centaur manufacturing plans with a minimum of modification, to assure low costs and minimum scheduling risk. The materials and fabrication operations needed for the Reusable Centaurs are those now in use on the existing Centaur production vehicle or on other products manufactured at Convair's San Diego plants. No special requirements or critical processes are involved.

An improved tank welding technique is planned for the RLTC vehicle, based on recent testing programs initiated to improve the fatigue life of welded stainless steel tanks. The operation will use the newly developed (proven) laser welding equipment. Benefits derived from laser welded joints include greatly reduced heat affected zone at the weld, giving increased joint efficiency and fatigue life, and a significant increase in welding speed with resulting lower labor costs. This improved technique is not a requirement.

The manufacturing sequences and assembly flow diagrams for the D-1S(R) and RLTC vehicles, including their deployment adapters and pallets, are diagrammatically detailed in Volume 8 of the September 26 Data Dump. These diagrams illustrate each major component and the general manufacturing indentures required to achieve the progressive build-up from the component levels.

Individual systems checkout of Centaur D-1S(R) will use the same equipment and procedures now used for D-1A and D-1T, employing the computer-controlled launch set, PCM telemetry, and flight acceleration profile equipment available in the computer and ground station facility area at Plant 71. These systems are connected through remote digital interface equipment to the factory checkout trailer, located at the final assembly docks in Building 5. All electronic systems on Centaur are checked out with this equipment, which is also used to run system functional checks, calibration, and DCU software flight acceleration profile tests. A flight events demonstration (FED) is normally run on Centaur D-1A and D-1T after completion of the individual systems checkout. For Centaur D-1S(R), the FED will not be run until the vehicle and its flight pallet are mated and checked out, to verify Centaur/flight pallet as a unit payload for the Orbiter.

The RLTC manufacturing sequence and assembly flow plan duplicates that for the D-1S(R) vehicle with the exception of those functions relating to the flight pallet, and is therefore quite similar to the manufacturing plan for current Centaur vehicles. After fabrication, assembly and individual systems checkout in the final assembly dock, a FED is run to check out all RLTC combined systems. Sidewall insulation blankets will be installed to fit check, then removed and shipped to ETR for final installation before the vehicle undergoes cryogenic tanking tests. The RLTC will be shipped to KSC with the forward stub adapter, equipment module, and aft launch skirt mated. Manufacture of the RLTC vehicle will require some new procedures with respect to the large diameter LH_2 tank, and the composite structure of the adapters. Major fabrication operations associated with new tooling are outlined below.

In general, the Reusable Centaur flight pallets and adapters will follow the same manufacturing, assembly, and checkout procedures as the vehicle itself, with the same equipment, tooling, and facilities that are used for Centaur at Plants 19 and 71, to minimize production costs.

There are no major new tooling requirements for the D-1S(R) program. The tank is identical to the D-1T tank now being produced at the San Diego facility of Convair and no changes in tooling are planned. The titanium skin-stringer and aluminum ring latch skirt will be fabricated using the same procedures and modified tooling used for the currently produced stub adapter. Fabrication of the deployment adapter and flight pallet will require some new tooling, but will for the most part use existing tooling or tooling similar to that now in use for existing truss adapters and interstage adapters.

Tooling required for the RLTC includes new or modified designs to support fabrication and/or assembly of the larger LH_2 tank bulkheads, cylindrical sections, and transition sections from the 170-inch diameter to the 120-inch diameter. The existing Centaur intermediate bulkhead detail and assembly tooling will be used much as it is for this program, although it is intended to modify the trimming and welding operations to install laser equipment. Other existing tooling will be modified for the insulation bulkhead, aft bulkhead, and thrust cylinder. The LO_2 tank cylinder tooling is unchanged.

The current Centaur tooling inventory contains two nearly identical tank major weld fixtures. Fabrication of the 170-inch diameter fuel tank of the RLTC will necessitate modification of one of these D-1 Centaur tank major weld fixtures. This modification will include adaptation of the laser beam weld equipment, a new forward bulkhead positioning and rotation tool, a removable structure to support and rotate the 10-foot diameter LO_2 tank section, and an extension to the base of the existing aft bulkhead locating tool.

8.4 FACILITIES AND GSE

8.4.1 <u>MANUFACTURING FACILITIES</u>. In keeping with the low development cost approach to the STS program, General Dynamics proposes the use of existing manufacturing facilities at San Diego for manufacture of the Reusable Centaur vehicles. Neither the D-1S(R) nor the RLTC configuration require any extraordinary size, weight, or materials that preclude the use of these facilities. Both vehicles are capable of air cargo transport to KSC; the geographical location of the San Diego industrial facilities is therefore no problem.

The Centaur manufacturing facilities are located at the Kearny Mesa plant of General Dynamics/Convair Aerospace in San Diego.

Convair's inventory of machine tools provides sufficient capacity and versatility to support both the manufacture of production tooling and fabrication of Centaur detail hardware. No new facilities will be required at San Diego for fabrication and checkout of either the Centaur D-1S(R) and its flight pallet or the RLTC; existing facilities are adequate for concurrent production of the D-1A, D-1T, and either the D-1S(R) or RLTC. Modification to existing facilities will be required to accommodate the D-1S(R) flight pallet and the vehicle pallet assembly, or to accommodate the increase in the RLTC hydrogen tank diameter from 120 to 170 inches.

The vertical assembly tower (VAT) currently used for Centaur vertical mate checks, will not need modification to handle the D-1S(R), and its flight pallet for mating and functional checkout.

Specific facility modifications required in support of both the D-1S(R) and the RLTC are discussed in some detail in Volume 8, Programmatics and Cost, of the 26 September Data Dump on the Reusable Centaur Study, Contract NAS8-30290.

8.4.2 TEST FACILITIES. An examination of the test facilities required to support the Reusable Centaur (D-1S(R) or RLTC) test program reveals that no new major test facilities are required. The complement of existing government and contractor facilities built during the past decade for space vehicle development is adequate for all presently conceived Reusable Centaur testing. In addition, because of the relatively modest size of the Reusable Centaurs and their general similarity to vehicles like the current Centaur and S-IVB stage, only minor modifications should be required to most of these facilities. The basic facility problem is not the availability of facilities or to the modifications required, but rather the cost effective use of existing facilities. Here, the important factors for consideration are the relative operating costs of existing competing facilities and the cost of reactivating facilities that have been mothballed for lack of current test activity.

This study did not attempt to arrive at final definitive conclusions regarding the relative economics of all possible test facilities for each test; instead, test facilities with which the study contractor is familiar were selected based on capability to perform the required tests; the more significant ones include:

a. The obvious site for tests of the RL10A-3-3 engines used in the D-1S(R) or RLTC programs is the engine contractor's facilities (Pratt & Whitney). However, performing these tests at MSFC in the S-IVB static test stand is an alternative. No altitude environment is required for these propulsion systems tests.

b. The structural loads test proposed for the RLTC test vehicle could be conducted at MSFC exclusively, as an extension of the external tank structural tests in Test Facility 4550. Required personnel and equipment are available from the external tank structural tests and the Reusable Centaur testing schedule meshes very well with the external tank testing schedule.

c. The facility (4557) used for the Saturn IB dynamic tests is suitable for the life cycle testing of either the D-1S(R) or RLTC vehicles. The testing concept consists of successive fill, pressurization, and drain of propellants. The cryogenics (LH_2 and LN_2) required for chilldown and pressurization during the life cycle tests can be provided by this facility.

d. The deployment/retrieval (Orbiter interface hardware) could be run at the contractor's facility with the Load Test Annex (Building 4619) at MSFC as an optional location.

Facilities for ground support operations of the reusable Centaur vehicles are discussed in Section 7.

8.4.3 GSE

8.4.3.1 D-1S(R). New, modified, and major related items of existing GSE required to support the D-1S(R) factory operations are listed in Table 8-2.

The aft support adapter will need to be modified to provide a detachable 15-inch wide section, permitting it to be mated to D-1A and D-1T at Station 2240.78, and to D-1S(R) at the latch skirt interface, Station 225.78. Two PCM telemetry ground stations are in existence, one at ETR Complex 36 and one at Plant 71 in San Diego. Both stations will need minor modifications in support of D-1S(R) to allow them to interface with the ground digital interface electronics (GDIE). A telemetry data station also exists at Plant 71 in San Diego and needs minor modifications in support of the D-1S(R) application.

A new transport pallet will be required to handle the Centaur D-1S(R) flight pallet. This transport pallet will duplicate the Orbiter half of the Orbiter/flight pallet attach points, and will provide a configuration interface check for the flight pallets when they are first fabricated. Construction of the transport pallet will be similar to the existing Centaur transport pallet with respect to the basic materials and strength members interfacing with the transport pallet. To lift the flight pallet transport pallet onto the transport trailer, a new sling is required, similar to the existing Centaur transport pallet sling. Similar slings with slightly greater lift capability will be required at the Cape to handle the Centaur/flight pallet assembly with mated spacecraft (ground operations GSE).

For D-1S(R), a leveling and stabilizing kit will be installed in the existing VAT at Plant 71 to align and level the flight pallet, and provide stable support for Centaur mating.

The GDIE provides a seven-wire, triplex redundant digital interface to connect the CCLS and launch control GSE to the D-1S(R) pallet at ETR. A prototype of the ETR GDIE will be used in San Diego for the Centaur/pallet factory checkout operation. The GDIE is also a new piece of equipment which is similar in operation and philosophy

Table 8-2. D-1S(R) Manufacturing GSE

Item No.	Name	Hardware Status	No. Req'd
1201	Centaur Transport Pallet	Existing	2
1202	Aft Support Adapter	Modified	2
1203	Forward Conical Handling Adapter	Existing	2
1204	Transport Trailer	Existing	1
1207	Centaur Transport Pallet Lift Sling	Existing	1
1208	Flight Pallet Transport Pallet	New	1
1209	Flight Pallet Transport Pallet Lift Sling	New	1
1210	Flight Pallet Leveling & Stabilizing Kit	New	1
1211	Insulation Workstands, VAT	Existing	2
1221	Computer Controlled Launch Set	Existing	1
1222	Flight Acceleration Profile Unit	Existing	1
1223	Ground Digital Interface Electronics	New	1
1225	Factory Checkout Trailer	Existing	1
1226	Telemetry Ground Station	Modified	1
1227	Telemetry Data Station	Modified	1
1228	CMACS Simulator	New	1
1229	Subsystem Support Equipment (SE) (DCU/RMU SE; IMG SE; CCTE; SCU, SIU, SC Test Sets)	Existing	1

to the present local digital interface electronics associated with the existing CCLS checkout operation.

The CMACS simulator is the only other new GSE required at San Diego to support checkout of the CMACS, pallet, and the Centaur/Orbiter interfaces.

8.4.3.2 RLTC. New, modified, and major related items of existing GSE required to support RLTC factory operations are listed in Table 8-3. Existing GSE to be modified for use with the RLTC include: existing low-boy trailers modified somewhat to accept the RLTC transport pallet and fitted with bolt-on and turnbuckle supports to secure the pallet during transport; some minor modifications to an existing factory checkout trailer involving some control panels, meters, switches, etc.; telemetry ground and data stations as noted under the D-1S(R) but modified for the RLTC; and some subsystem support equipment modifications since the RLTC uses a different DCU and IMG than the D-1S(R) (which in turn affects such equipment as the SCU, SIU and SC).

There are few more new GSE items required for the RLTC than for the D-1S(R). For instance, a new transport pallet will be required to handle the RLTC. The structural design of this pallet will be similar to that of the existing Centaur transport pallet,

Table 8-3. RLTC Manufacturing GSE

Item No.	Name	Hardware Status	No. Req'd
1301	Transport Pallet	New	2
1302	Aft Support Adapter	New	2
1304	Transport Trailer	Modified	1
1307	Transport Pallet Lift Sling	New	1
1308	Deployment Adapter Transport Pallet	New	1
1309	Deployment Adapter Lift Sling	New	1
1321	Computer Controlled Launch Set	Existing	1
1322	Flight Acceleration Profile Unit	Existing	1
1323	Ground Digital Interface Electronics	New	1
1325	Factory-Checkout Trailer	Modified	1
1326	Telemetry Ground Station	Modified	1
1327	Telemetry Data Station	Modified	1
1328	CMACS Simulator	New	1
1329	Subsystem Support Equipment (SE) (DCU/RMU SE; IMG SE; CCTE; SCU, SIU, SC Test Sets)	Modified	1

with an increase in structure width and depth to accommodate the increase in tank diameter from 120 to 170 inches and a four-point pallet support installation. Pallet supports at the engine end will duplicate the Orbiter trunnion supports for the deployment adapter, and at the forward end will duplicate (on opposite sides) the single Orbiter forward attach point at the stub adapter forward mating ring.

A new aft support adapter will duplicate the configuration of the existing adapter, with two modifications: the new adapter will be 15 inches shorter, and its trunnions will be identical to the deployment adapter trunnions, mating with the transport pallet supports and the Orbiter payload bay deployment adapter supports. A new lift sling with allowance for the larger tank diameter and increased weight will be required to handle the transport pallet for the large tank Centaur. A deployment adapter pallet will be required for factory handling, shipment, and operational functions at the two launch ranges. The pallet will be patterned after existing shroud and adapter pallets now in use for Centaur operations. Likewise, a simple lift sling will be required to handle the deployment adapter pallet, similar to those now in use for Centaur adapter pallets. Existing slings may be adequate, depending on the final deployment adapter pallet design. The GDIE and the CMACS simulator are new equipment requirements, the same as for the D-1S(R).

8.5 LOGISTICS

8.5.1 SPARES. The objective of the spares analysis is to provide a high probability of spare parts availability at low investment cost. The spares analysis is an integral part of the overall refurbishment and maintenance (R&M) plan which establishes scheduled and unscheduled requirements. The inherent reusability and expected life of major Centaur components are discussed in Sections 2 and 7.

The Reusable Centaur vehicles will have a high percentage of equipment already in the NASA inventory for current production D-1A and D-1T Centaur vehicles. Spares inventory for the reusable vehicles can therefore be maintained at a reduced level. Reusable Centaur hardware can be identified as either (1) identical to existing hardware (with some minor modifications), or (2) major modified and new hardware.

Category 1 hardware benefits from the initial spares pipeline being partially filled, and scheduled production of both vehicles can be used to support immediate spares, with replacements made from later shipsets.

Category 2 hardware is subject to higher replacement rates and must be planned accordingly.

8.5.2 TRAINING. During the development phase, training plans will be developed and implemented to prepare or maintain personnel at skill levels required to operate and maintain program equipment at its specified standard of performance. The training function will include a training requirements analysis to identify types of training, personnel quantities to be trained, equipment necessary to conduct training, schedules for training, technical data for training and program operation, and training interfaces with other STS and Centaur program elements.

Preliminary estimates of the number of personnel to be trained were made for current program costing analyses. Training costs were estimated for the average individual and multiplied for the crew sizes identified in the ground and flight operations analyses. Retraining requirements were based on a 25% annual turnover in WTR crews and a 15% annual turnover in KSC crews. The difference was based on the assumption that WTR crews may be largely comprised of military personnel with greater turnover rates as compared with NASA or contractor crews at KSC.

8.5.3 TRANSPORTATION AND HANDLING. During the development phase, a transportation and handling plan will be developed to accomplish early assessment of hardware transportation and handling requirements for test and operational phases of the program. Identified transportation and handling requirements for each major end item will be documented in the transportation plan. Hardware items below the major end item level, such as spares, will be controlled by NASA Specification NHB 6000.1(1A). The plan will identify transportation support which NASA will provide versus that to be provided by the contractor.

Regulations will include the following baseline government and NASA policies within their transportation, handling, and lifting requirements.

a. Air Force Manual (AFM) 71-4, entitled "Packaging and Handling of Dangerous Materials for Transportation by Military Aircraft," which establishes requirements for shipment of explosives and other dangerous materials by Government agencies and contractors using military aircraft.

b. Department of Transportation Tariff 25 (Graziano's) which establishes requirements for shipment of explosives and other dangerous articles by land, highways, rail service, or water.

c. Civil Aeronautics Board (CAB) Tariff 6D which will govern shipment of dangerous articles by commercial air carriers.

d. NHB 6000.1 (1A), entitled, "Requirements for Packaging, Handling, and Transportation of Aeronautical and Space Systems, Equipment and Associated Components" (NHA 6000.1 should be interpreted to include as a reference MIL-STD-1366 and MIL-STD-1367), establishes requirements for packaging, handling, and transportation of all Shuttle hardware, excluding explosives and other dangerous articles.

e. NASA Policy Directive (NPD) 5300.8 which establishes requirements for lifting program hardware.

Pending development of the complete Reusable Centaur Program Transportation and Handling Plan, costs were estimated based on current Centaur experience.

8.6 FLEET SIZE

The number of reusable third stage vehicles needed to satisfy the STS mission model, annually and for the entire program, has a significant influence on DDT&E, investment, and operational costs, as well as on scheduling and planning. Determination of optimum values for the active (annual) fleet size and for the total program fleet size must be based on proper evaluation of all programmatic factors applied to numerical analysis. Both the active fleet size, or number of vehicles required each year to capture the mission model, and the total fleet size for an entire program are dependent on the following factors:

a. Payload mission model

b. Vehicle capability for multiple payloads

c. Number of expendable vehicles required

d. Attrition estimate

e. Ground turnaround time in hours or days

f. Launch Center Scheduling (time between launches)

g. Production rate efficiency

h. Vehicle life (reusability) in terms of total allowable flights

In determining D-1S(R) and RLTC fleet sizes, basic annual vehicle requirements based on factors 1, 2 and 3 were first obtained from the mission capture analysis of Section 4 and summarized on mission capture charts for the 4-, 6-, and 11-year programs. A 1% attrition (factor No. 4) was added to the basic vehicle requirements and applied as an early (first flight) attrition loss, normal (mid-life) loss, and late (last scheduled flight) loss, plotting curves of total flights or mission duty cycles (TMDC) per vehicle versus total fleet size as a variable, for each attrition case. These curves provide visibility in judging early, normal and late attrition risks, reusability requirements for minimum fleet size, and effects of reusability limitations on maximum fleet size.

TMDC curves for D-1S(R) 4-, 6-, and 11-year programs are shown in Figure 8-7 as an example. If reusability (factor No. 8) is limited to 10 missions, then fleet size for the three programs will be 7, 13, and 25 vehicles as shown. If reusability is not considered as a limiting factor, then the fleet sizes will be 5, 11, and 23, and the maximum number of flights any vehicle will make will be 14, 13 and 12 respectively, for the three programs.

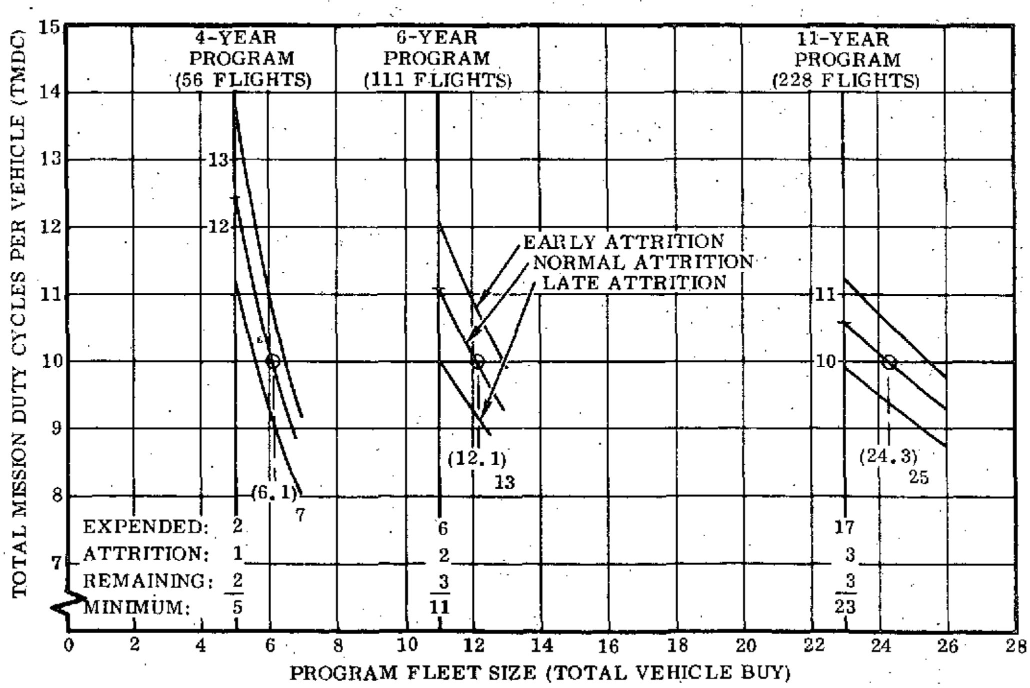


Figure 8-7. Centaur D-1S(R) Fleet Size vs Mission Duty Cycles - Four-, Six-, and Ten-Year Programs

Assuming fleet sizes on a preliminary basis as the minimum 5, 11, and 23 vehicles, fleet utilization and requirement schedules were developed, detailing mission assignments (to provide balanced TMDCs for all vehicles in a program), vehicle need dates, production schedules (factor No. 7), mission schedules for each vehicle, ultimate vehicle disposition, and active annual fleet size. Typical schedules are shown in Table 8-4 for the six-year program, considering both early and late attrition. As indicated in the late or no attrition case, vehicle life is three to five years, averaging about three flights per year, never more than six in any one year, with a total life of no more than 11 flights maximum. Considering the worst case of early attrition, vehicle life is one to four years, averaging about four flights per year, never more than nine in any one year, with a total life of no more than 13 flights maximum.

With an annual capability of 14 flights per year based on a 280-hour turnaround time (factor No. 5), the utilization schedules provide contingency for uneven launch centers (factor No. 6) and unforeseen scheduling problems. In effect, ground turnaround time is non-critical because of the early availability of vehicles established by efficient production rates.

A summary of the D-1S(R) and RLTC fleet requirements is given in Table 8-5, comparing operational characteristics of the two vehicles for the 4-, 6-, and 11-year programs. The following points are of particular interest:

a. In the three programs considered, D-1S(R) will require about 4% more flights to complete the mission models than will the RLTC.

b. D-1S(R) will require more expended vehicles to complete the mission models than will the RLTC, resulting in a higher initial investment for the 4-, 6-, and 11-year programs of 1, 4, and 7 vehicles respectively.

c. RLTC vehicles will fly approximately 50% more missions in their life spans than will the D-1S(R) vehicles, resulting in higher vehicle costs for designed reusability factors.

d. TMDC values (reusability, total flights per vehicle) for both the D-1S(R) and RLTC will remain fairly low (less than 25) as long as mission models require expenditure of vehicles.

8.7 MANPOWER REQUIREMENTS

Centaur ground turnaround crew requirements at KSC were developed by determining manpower requirements on a task basis, and plotting these requirements against detailed turnaround timelines on an hourly basis. Hourly variations were then averaged by shift, and again across the total turnaround cycle to arrive at total direct support manpower quantity and type requirements as shown in Table 8-6. Engineering support and Level II shop support manpower requirements were then added based on projected workload, numbers of direct support personnel, and anticipated Level II support workloads derived in conjunction with failure rate data in the spares analysis tasks. This

Table 8-4. Centaur D-1S(R) Fleet Requirement and Utilization Schedule, Six-Year Program

Active Fleet				Annual Traffic (No Attrition)						
Veh. No.	Need Date	Deliv. Date	Dispo-sition	1980	1981	1982	1983	1984	1985	Total
1	Jan 80	Nov 79	Expended	1	5	4△				10
2	May 80	Dec 79	Expended	1	5	4△				10
3	Sep 80	Jun 80	Expended	1	1	1	4	4△		11
4	Jan 81	Oct 80	Expended		1	1	4	4△		10
5	May 81	Feb 81	Expended		1	1	4	4△		10
6	Sep 81	Jun 81	Expended		1	1	4	4△		10
7	Jan 82	Oct 81	Inv'tory			1	2	2	6	11
8	May 82	Feb 82	Inv'tory			2⇨	4*	1*	2*	9
9	Sep 82	Jun 82	Inv'tory			2⇨	3*	1*	3*	9
10	Jan 83	Oct 82	Inv'tory				2	3	6	11
11	May 83	Feb 83	Inv'tory				2	3	5	10
KSC Flights				3	14	17	22	24	17	97
*WTR Flights							7*	2*	5*	14*
Total Flights				3	14	17	29	26	22	111
Active Fleet, Including Expended				3	6	9	9	9	5	
With Early Attrition										
1	Jan 80	Nov 79	Attri'on	1						1
2	May 80	Dec 79	Attri'on	1						1
3	Sep 80	Jun 80	Expended	1	6	6△				13
4	Jan 81	Oct 80	Expended		6	6△				12
5	May 81	Feb 81	Expended		1	1	5	5△		12
6	Sep 81	Jun 81	Expended		1	1	5	5△		12
7	Jan 82	Oct 81	Expended			1	5	6△		12
8	May 82	Feb 82	Expended			1	5	6△		12
9	Sep 82	Jun 82	Inv'tory			1	2	1	8	12
10	Jan 83	Oct 82	Inv'tory				2*	⇨ 1	9	12
11	May 83	Feb 83	Inv'tory				5*	2*	5*	12
KSC Flights				3	14	17	22	24	17	97
*WTR Flights							7*	2*	5*	14*
Total Flights				3	14	17	29	26	22	111
Active Fleet, Including Expended				3	4	7	7	7	3	

Total Program Fleet Size: 11

△: Expended

⇨: Transfer Launch Sites

Table 8-5. Centaur D-1S(R) & RLTC Fleet Size & TMDC Program Summary

	D-1S(R)			RLTC		
	4-Year	6-Year	11-Year	4-Year	6-Year	11-Year
Total Flights	56	111	228	54	106	220
Fleet Size	5	11	23	4	7	16
Attrition	-1	-2	-3	-1	-2	-3
Expended Vehicles	-2	-6	-17	-0	-2	-10
Left in Inventory	2	3	3	3	3	3
Average TMDC						
Early Attrition	13.8	12.1	11.2	17.7	20.8	16.7
Normal Attrition	12.4	11.1	10.6	15.4	17.7	15.2
Late Attrition	11.2	10.1	9.9	13.5	15.2	13.7
Maximum TMDC						
Early Attrition	14	13	12	18	21	17
Normal Attrition	13	12	11	16	18	16
Late Attrition	12	11	10	14	16	14

is the basic crew required to accomplish and support ground turnaround operations on a single vehicle within the previously determined 280-hour turnaround time span.

The basic WTR manpower requirements are shown in Table 8-7. Here, low annual launch rates and long intervals between launch centers offer an opportunity to eliminate the second shift and accomplish total vehicle turnaround with single shifts. Level II shop support personnel have also been eliminated at WTR to illustrate potential attainable manning levels assuming Centaur Level II shop support being accomplished in joint Centaur-Orbiter and/or Payload Level II shops.

When the number of launches increases in accordance with the traffic model and mission capture analysis, it is necessary to augment the basic turnaround crew with additional personnel to accommodate concurrent operations on two or more vehicles and/or increased inventory and administrative support for additional active vehicles in the fleet at a given launch site. This augmentation is reflected in the annual ground turnaround crew requirements shown in Table 8-8.

Crew sizes shown in Tables 8-6, 8-7, and 8-8 are actual on-site administrative, maintenance, and launch support personnel; systems engineering and integration activity is a San Diego-controlled operation and most of the personnel engaged in SE&I functions are accounted for accordingly in the cost breakdown.

Table 8-6. KSC Launch Site Manpower Requirements, Reusable Centaur

Manpower Skills	Turnaround Support		Total Required
	Shift 1	Shift 2	
1. Astrionics Technician	8	6	14
Propulsion - Fluids Technician	8	6	14
Structures - Handling Technician	6	4	10
Quality Control	2	2	4
Safety	1	1	2
Engineer	3	2	5
Other	4	2	6
Total Direct Support	32	23	55
2. Engineering Support			
Logistics & Admin. Support	13	5	18
Design - Systems Engineering	6	2	8
Scheduling	2	1	3
Procedures - Test Requirements	4	-	4
Data Analysis	4	-	4
Programming	2	-	2
Total Engineering Support	31	8	39
3. Level II Shops			
Metal	2	-	2
Paint	2	-	2
Cleaning	-	-	-
Plastics - MLI	2	1	3
Pneumatics	1	1	2
GSE Maintenance	-	-	-
Avionics	1	1	2
Standards - Inspection	1	1	2
Total - Level II Shops	9	4	13
Grand Total - Centaur Support	72	35	107

There is no formal provision for crew augmentation to support a learning curve in this analysis. As shown in Table 8-8, the flight buildup in the first four years of the program at KSC, of 3, 14, 17 and 20 flights, provides a built-in learning period, obviating the need for an oversize crew at the beginning of the program. At WTR, the number of flights per year is low, and manning will benefit from three years of experience at KSC, requiring no additional manpower at WTR for learning.

The manning levels shown result in a number of direct support personnel who are not working on direct turnaround tasks 100% of the time during a Centaur turnaround cycle. This is particularly true during the long Orbiter host time following installation in the payload bay, when only a two-man monitor crew is required. During these slack times, direct support personnel will be used to assist in Level II shop support tasks and accomplish GSE maintenance.

Table 8-7. Centaur Launch Site Manpower Requirements, WTR

Manpower Skills	Turnaround Support	Total Required
	Shift 1	
1. Astrionics Technician	10	10
Propulsion - Fluids Technician	10	10
Structures - Handling Technician	8	8
Quality Control	2	2
Safety	1	1
Engineer	3	3
Other	4	4
Total Direct Support	38	38
2. Engineering Support		
Logistics & Administrative Support	10	10
Design - Systems Engineering	5	5
Scheduling	2	2
Procedures - Test Requirements	3	3
Programming	2	2
Total Engineering Support	25	25
Grand Total - Centaur Support	63	63

Table 8-8. Crew Size Vs Annual Launch Rate, Reusable Centaur

		Calendar Year											Total
		80	81	82	83	84	85	86	87	88	89	90	
Centaur Flights	KSC	3	14	17	20	22	16	18	22	17	20	19	188
	WTR				7	2	5	3	4	2	7	2	32
	Total	3	14	17	27	24	21	21	26	19	27	21	220
	(Expended Vehicles, KSC)					(2)		(3)	(1)		(3)	(1)	(10)
Ground Turn-Around Crew	KSC	107	107	107	113	113	107	107	113	107	113	113	
	WTR				63	63	63	63	63	63	63	63	
	Total	107	107	107	176	176	170	170	176	170	176	176	

SECTION 9

COST ANALYSIS

This section describes the work breakdown structure (WBS) provided by NASA for organizing cost estimates, the methodology followed in estimating costs, and the resulting program costs estimated for the Reusable Centaur configurations.

The estimates represent total predicted cost to the Government and, therefore, include allowances for cost growth due to such things as changes in requirements, currently undefined tasks, schedule realignments, and uncertainties in technology. The cost estimating techniques and ground rules are the same as those applied to the STSS study for MSFC. This approach produces realistic growth allowance for a new design Space Tug, but the results are conservative for modifications to an existing stage.

These estimates for the D-1S(R) and RLTC should not be directly compared with those in Section 10 for an RC because of programmatic and estimating differences. For instance, the mission models and allocation of the flight tests are not the same.

A comparison of program costs for the two configurations at various operational program durations is illustrated in Figure 9-1. Shown are the development, investment, and operations costs at the Tug Project level. The D-1S(R) configuration has a lower development and investment cost than the RLTC. Operations costs for the D-1S(R) are higher due to the greater number of auxiliary stages required. This results in very little difference between the configurations on a total program cost basis.

9.1 GROUND RULES AND ASSUMPTIONS

The D-1S(R) and RLTC program cost estimates were prepared using the following ground rules and assumptions:

a. Costs presented are for budgetary and planning purposes only. These costs represent the estimated total program cost to the government at program completion. As such they include allowances for cost growth over the program life cycle which are not normally included in bid type estimates; therefore, these cost estimates do not represent a commitment on the part of General Dynamics Corporation.

b. Costs are expressed in 1973 dollars, without prime contractor fee/profit.

c. Costs include GFE items such as engines, guidance, and propellants.

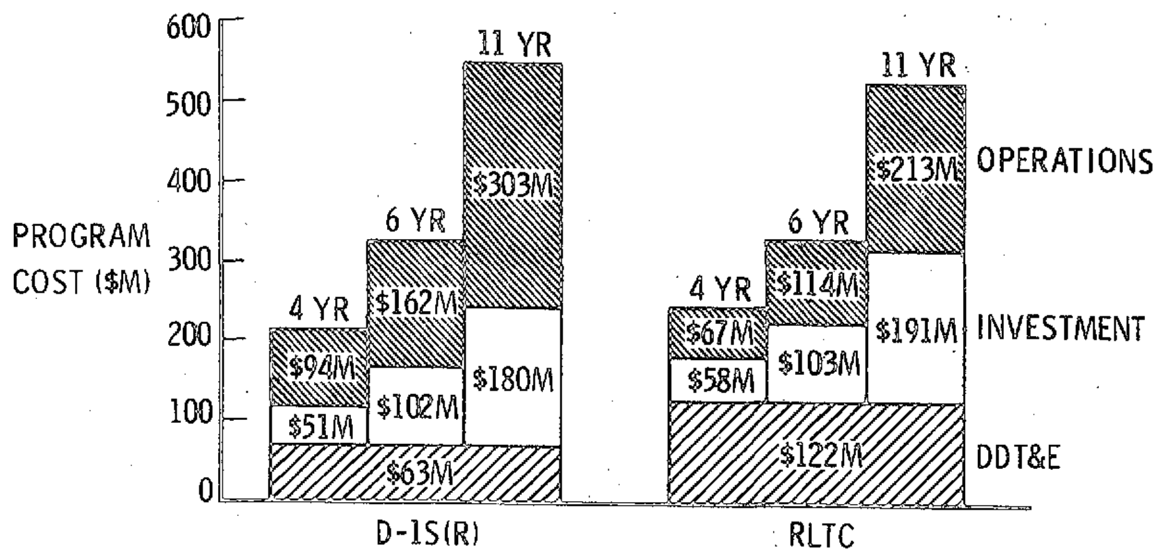


Figure 9-1. Total Program Cost Comparison

d. Cost estimates are predicated on making maximum use of existing government and contractor facilities and equipment.

e. Cost estimates for the following items were provided by NASA: Space Shuttle recurring launch cost and main engine production cost.

f. The four-year operational program options were estimated assuming ETR operations only, including launching WTR flights from ETR. For the six- and eleven-year programs, the cost of both ETR and WTR activation and operations was included.

g. A dedicated flight test vehicle was not required for either Reusable Centaur development program; therefore, no flight test hardware costs are included. The cost of flight test support and evaluation is included for analyzing data received from the first operational flights.

9.2 WORK BREAKDOWN STRUCTURE

The WBS used to organize Tug cost estimates is shown in Figures 9-2 and 9-3. This structure is essentially in accord with the NASA MSFC-supplied WBS Dictionary,

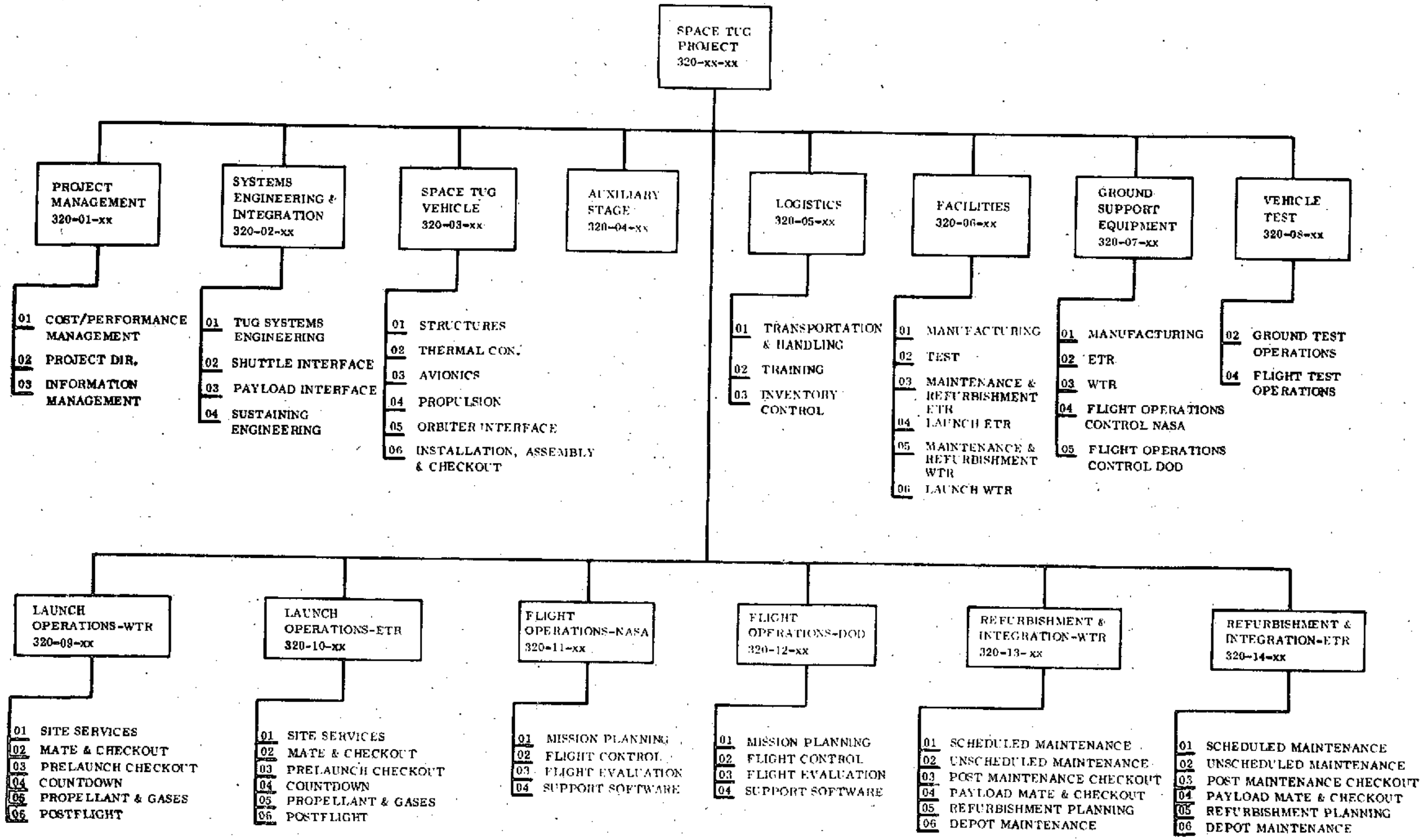


Figure 9-2. Space Tug Program Work Breakdown Structure

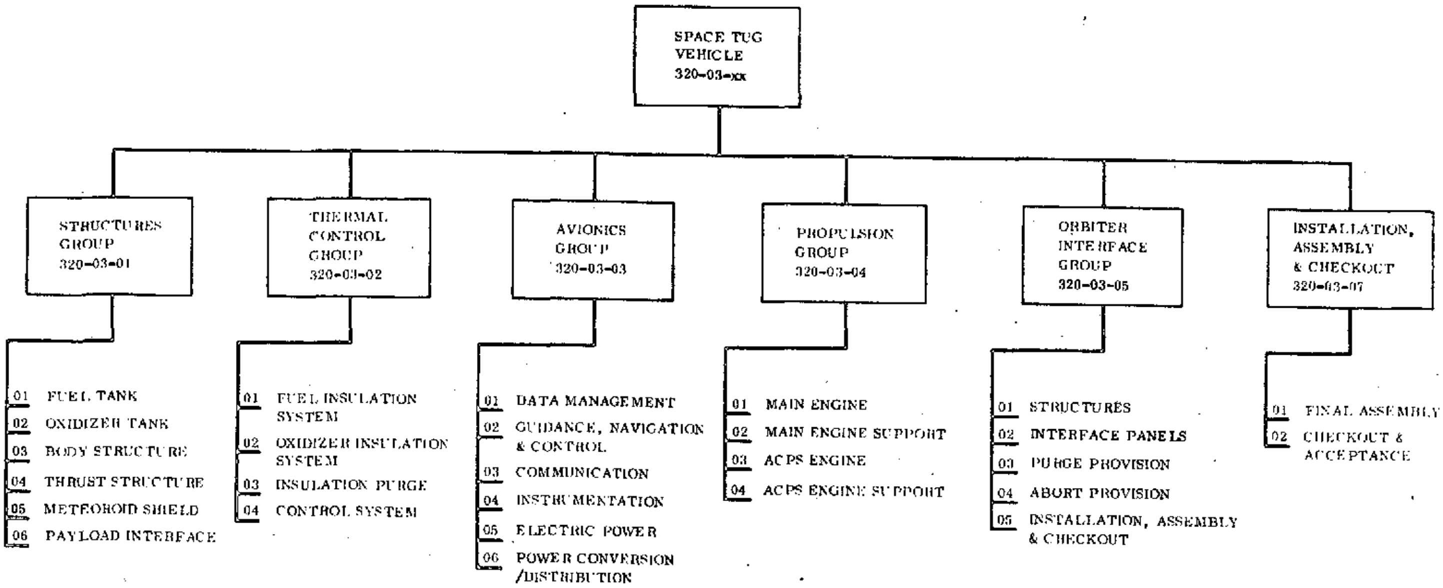


Figure 9-3. Tug Vehicle Work Breakdown Structure

document number PD-TUG-P, dated 5-7-73 as amended by the following NASA revision memos:

PD-TUG-P (006-74), dated 7-9-73

PD-TUG-P (024-74), dated 8-17-73

PD-TUG-P (029-74), dated 9-4-73

Most of the WBS categories are self-explanatory and are defined in detail in the NASA WBS Dictionary. Some discussion of the WBS cost categories may be appropriate here. The cost of ground and flight test operations is included under the Vehicle Test category, however the cost of the test hardware consumed during the test program is included in the subsystem DDT&E costs under the Tug vehicle WBS element. Test hardware costs can be readily identified in the Tug cost model output forms contained in Section 8.13.5 of the September Data Dump document. Ground test hardware costs and flight test hardware costs are individually estimated for each subsystem and are displayed in separate columns on the cost model output forms. Development, evaluation, and qualification at the component level are included in the engineering design and development cost estimate for each subsystem as are the tool design and initial tooling costs. Each individual subsystem DDT&E cost includes the estimated cost of engineering design and development, tooling, and test hardware for that subsystem.

DDT&E program costs include vehicle and ground support equipment (GSE) development; simulators design and procurement; initial crew training; facilities construction, GSE procurement, and site activation at ETR; initial software development; and associated management and integration activities.

Investment costs include production of the operational Tug fleet; initial (inventory and pipeline) spares; facilities construction, GSE, and site activation at WTR; and project management and integration.

The recurring operations costs include the cost of launch and refurbishment operations crews at both launch sites, NASA and DoD flight operations crews, mission and flight peculiar software, facility and GSE maintenance, project management and integration activities, the procurement and launch services cost of auxiliary stages expended during operations, and the cost of Tug vehicle replenishment spares. The Tug spares costs were estimated at the subsystem level and appear as Tug vehicle operations costs in the cost model output forms.

9.3 COST ESTIMATING DATA BASE

Convair Aerospace has an extensive data base from which the Reusable Centaur costs were derived.

Recent experience with the improved Centaur D-1A for Atlas and the Centaur D-1T for Titan III provides a good "modification type" cost base. Cost data from D-1A and D-1T development and Titan/Centaur integration, which commenced in mid-1969 and will conclude with the Titan/Centaur proof flight in February 1974, provides a comprehensive and relevant cost data base from which to estimate the D-1S(R) and Shuttle/Centaur integration costs. As a result of the close resemblance between Centaur D-1T and D-1S(R) and the availability of extensive current D-1T cost data, the D-1S(R) cost estimate is considered to be more realistic than would normally be the case in a Phase A study. In addition to the cost data base used for D-1S(R), data from a previously proposed Growth Tank (GT) Centaur was used in developing the RLTC cost estimates. Again, the close resemblance between Centaur RLTC and the GT Centaur Program provides a more credible cost estimate than is typical of a Phase A study.

Other important data is available from the Launch Site Services Study (KSC), Centaur/Shuttle Integration Study (LeRC), and the Space Tug Systems Study (Cryogenic) being performed for MSFC and the Air Force. The Centaur Programmatic, Technical and Cost Study, which has recently been completed for MSFC, contains Centaur program historical cost data from the inception of the program through the flight of Atlas/Centaur-15.

9.4 COST ESTIMATING METHODOLOGY

Development of Reusable Centaur program costs involved application of three basic estimating techniques: parametric estimates (CERs); vendor quotations plus "wrap around" factors; and detail cost build-ups, primarily in the form of man loading by task. Application of these techniques was a function of the specific tasks involved.

The estimating philosophy for the Reusable Centaur was predicated on the concept of projecting the total cost to the Government. Because of the derivative nature of both vehicles, extensive use was made of historical data from various phases of the Centaur project. When parametric estimating techniques were employed for development costs, adjustments were applied to account for commonality with existing Centaur systems. In many cases the development cost represents an allowance for incorporation of reusability features and integration tasks associated with the new application of existing Centaur subsystems. Recurring production costs were estimated assuming a first unit cost base which was conservative, particularly in the case of the D-1S(R) version. Where vendor quoted costs were used, added allowances were included for Convair tasks. These Convair task costs were based on studies of similar subsystems experience on Centaur and other programs. This methodology maintains a consistent pattern of cost reporting for comparison with the Space Tug Systems Study results.

In order to predict total costs to the Government, it is necessary to make allowances for cost growth. The amount of these growth allowances should, if possible, be determined on an individual subsystem basis, dependent upon the state of the art, state of development (off-the-shelf or new), and extent of modification. Convair Aerospace studies of cost growth indicate costs could increase as much as 70% even on well planned programs if new technologies are involved. This cost growth generally results from changes in requirements, technical problems and/or schedule changes. Even off-the-shelf subsystems may experience cost growth due to interface impacts of other subsystems, schedule changes, etc. Because of the modification nature of the Reusable Centaur configurations, cost growth allowances included in the study range generally between 10% and 30% and are about 20% overall.

Both versions of the Reusable Centaur represent modifications to an existing launch vehicle stage, the differences being in the degree of modification involved. Therefore, the primary data base for development of cost estimating relationships (CERs) and "wrap around" factors was Centaur project history. Particular attention was given to the D-1 program which was also a modification type development task. Data from other programs such as Atlas, Saturn, Apollo, and Gemini were used, where applicable, to validate and check CERs.

In order to improve the confidence in Reusable Centaur costs, two independent D-1S(R) estimates were developed for comparison. One of these estimates was primarily based upon a parametric approach with some detailed cost build-ups. The other used the detailed cost build-up, based on Convair Aerospace experience, particularly in the Centaur D-1T (Centaur for Titan) program. This latter estimate also drew heavily on the Expendable Centaur (D-1S) Study recently performed for Lewis Research Center. The two estimates were in close agreement.

9.5 COST ANALYSIS RESULTS

The cost data contained in this section summarizes the D-1S(R) and RLTC development (DDT&E) costs and the RLTC investment and operations costs previously presented in the September 1973 Data Dump (Reference 1). Since submittal of the September data dump, a review of the D-1S(R) operational mode has indicated that significant improvements in the program cost could be achieved by utilizing the vehicle's multiple payload capability with subsequent reduction in the required Tug fleet size and number of launches:

	D-1S(R) Fleet Size			D-1S(R) Launches		
	(4 yr)	(6 yr)	(11 yr)	(4 yr)	(6 yr)	(11 yr)
September Data Dump	9	17	34	77	152	317
Final Report	5	11	23	56	111	228

The D-1S(R) investment and operations costs summarized in this final report reflect the smaller fleet size and number of launches and are therefore somewhat lower than those costs reported in the September Data Dump.

During the preparation of the final report it was discovered that $4.1M of WTR facilities cost for the RLTC 6- and 11-year options was inadvertently left out of the September data dump. The omission has been corrected in this final report.

Summary schedule charts and funding curves at the Tug Project level for the D-1S(R) and RLTC four, six, and eleven year options are shown in Figures 9-4 through 9-9. This data is presented in the Tug Cost Summary format specified in NASA Memo PD-TUG-P (025-74) dated August 20, 1973 in accordance with Section 1.5.2 of MF 003M. The figures present Tug Program technical characteristics, schedules, annual funding, and cumulative funding requirements for each program option. The annual funding curves indicate a peak funding requirement for the D-1S(R) of approximately $35M in the 1979-1980 fiscal year period for the 4-year program option, a $45M peak in the fiscal 1981-1982 period for the 6-year program option, and a peak of roughly $55M over the fiscal 1981-1984 time period for the 11-year program option. By comparison, the RLTC annual funding curves display a peak of over $50M in fiscal 1978 for all three program options. Therefore, the D-1S(R) configuration has the advantage of both delaying and reducing the peak year funding over that required by the RLTC configuration.

Table 9-1 summarizes programmatic and cost data for comparison between the configurations and various program options. Because of performance differences between the two configurations, the D-1S(R) requires a slightly larger fleet size and a few more launches than the RLTC. The D-1S(R) also requires auxiliary stages for over 70% of its launches to accomplish the same mission model as the RLTC with less than 4% of its launches aided by auxiliary stages.

SPACE TUG
TECHNICAL CHARACTERISTICS

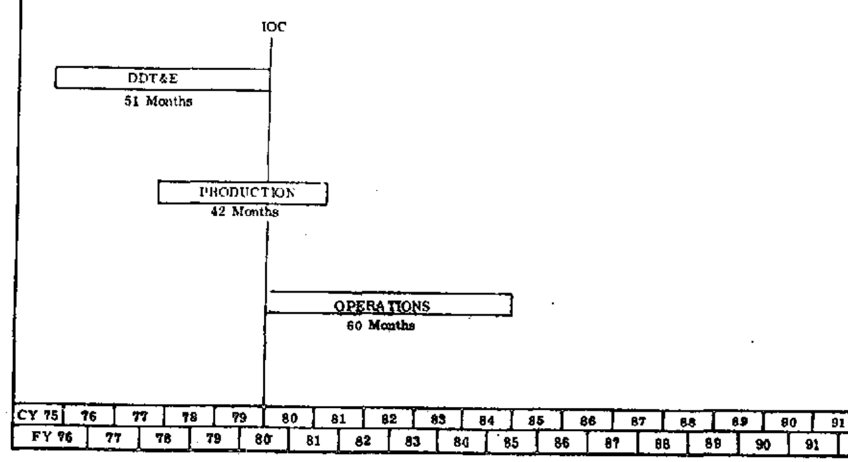
WBS 320-00 SPACE TUG PROJECT [CENTAUR D-1S(R)]

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the Centaur D-1S(R) Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	5	11	23
Tugs Expended	2	8	17
Auxiliary Stages Required	44	78	156
Total Tug/Shuttle Flights	56	111	228

SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 Centaur (D-1S(R)) Option I



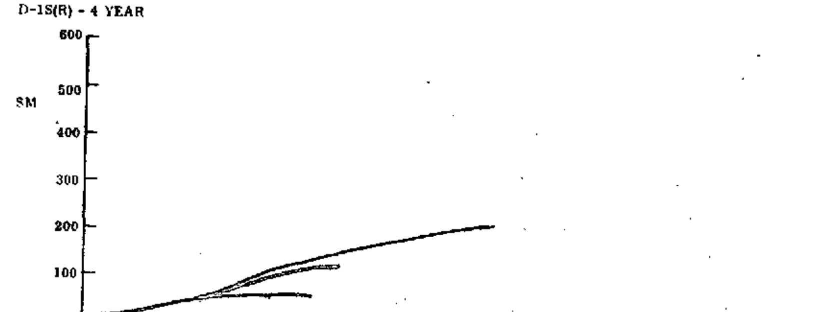
PROGRAM ANNUAL FUNDING

D-1S(R) - 4 YEAR

FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	20.3	26.7	12.1	1.3											
PRODUCTION			6.7	19.8	18.7	5.8										
OPERATION				3.6	14.2	21.2	23.6	10.6								
TOTAL	2.7	20.3	33.4	35.5	34.2	27.0	23.6	10.6								

PROGRAM CUMULATIVE FUNDING

D-1S(R) - 4 YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	23.0	49.7	61.8	63.3											
PRODUCTION			6.7	26.5	45.2	50.5										
OPERATION				3.0	18.2	39.4	63.0	84.5	93.8							
TOTAL	2.7	23.0	56.4	91.9	126.5	153.5	177.1	198.6	207.6							

Figure 9-4. Tug Cost Summary, D-1S(R), Option 1

SPACE TUG
TECHNICAL CHARACTERISTICS

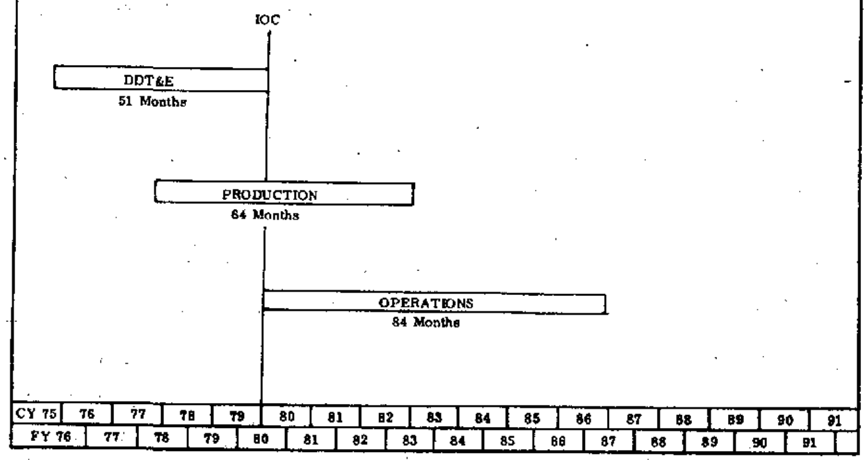
WBS 320-00 SPACE TUG PROJECT [CENTAUR D-1S(R)]

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the Centaur D-1S(R) Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	5	11	23
Tugs Expended	2	6	17
Auxiliary Stages Required	44	78	156
Total Tug/Shuttle Flights	56	111	228

SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 Centaur D-1S(R) Option II



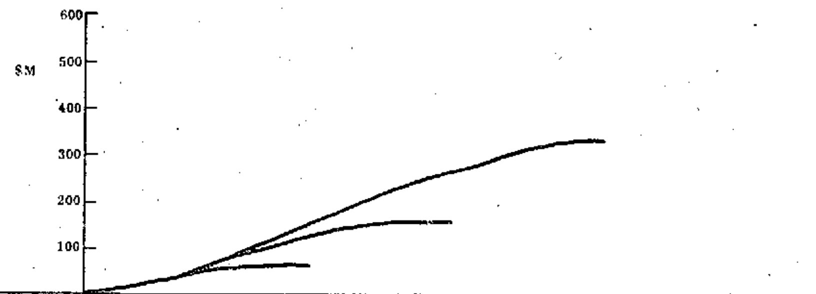
PROGRAM ANNUAL FUNDING

D-1S(R) - 6 YEAR

FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	20.3	26.7	12.1	1.3											
PRODUCTION			6.4	22.2	25.0	23.3	19.9	5.5								
OPERATION				4.0	15.6	22.4	25.3	27.2	28.1	25.8	12.7					
TOTAL	2.7	20.3	33.1	38.3	41.9	46.2	45.2	32.7	28.1	25.8	12.7					

PROGRAM CUMULATIVE FUNDING

D-1S(R) - 6 YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	23.0	49.7	61.8	63.3											
PRODUCTION			6.4	28.6	53.6	76.9	96.8	102.3								
OPERATION				4.0	19.6	42.5	67.8	95.0	123.1	148.9	161.6					
TOTAL	2.7	23.0	56.1	94.4	136.3	182.5	227.7	260.4	288.5	314.3	327.2					

Figure 9-5. Tug Cost Summary, D-1S(R), Option 2

SPACE TUG
TECHNICAL CHARACTERISTICS

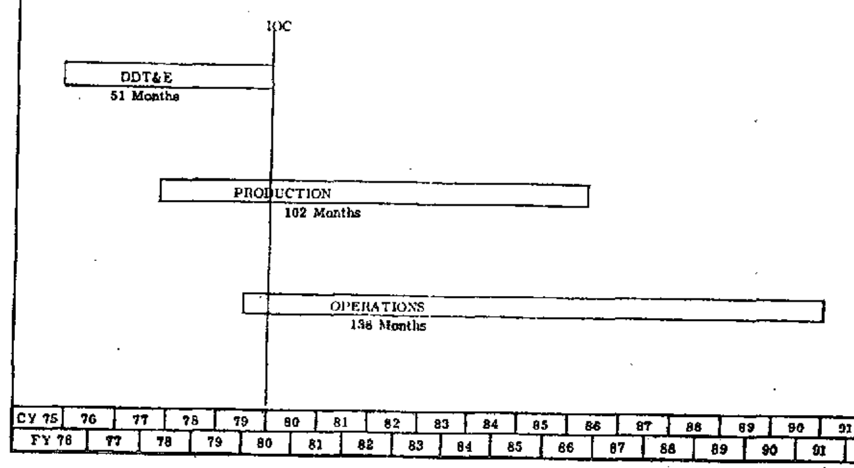
WBS 320-00 SPACE TUG PROJECT [CENTAUR D-1S(R)]

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the Centaur D-1S(R) Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

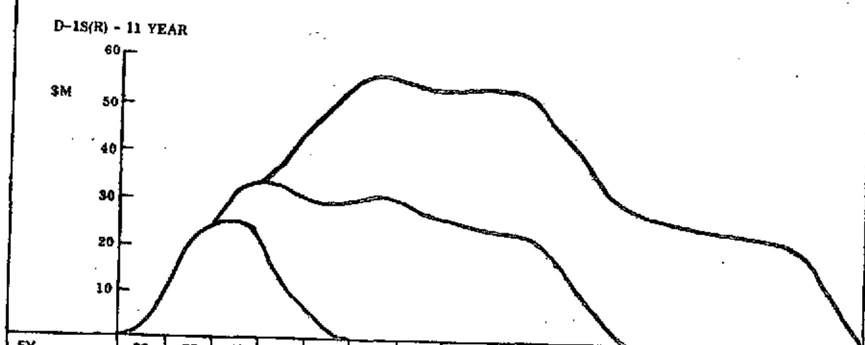
OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	5	11	23
Tugs Expended	2	6	17
Auxiliary Stages Required	44	78	156
Total Tug/Shuttle Flights	56	111	228

SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 (Centaur D-1S(R) Option III

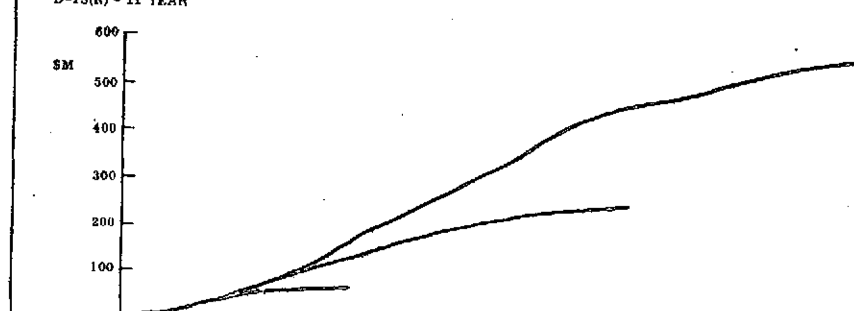


PROGRAM ANNUAL FUNDING



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	20.3	26.7	12.1	1.3											
PRODUCTION			5.9	20.6	28.0	31.5	27.3	25.7	23.4	14.8	2.5					
OPERATION				5.4	18.5	25.1	26.6	28.6	29.6	29.6	29.6	28.0	26.5	26.5	21.1	7.7
TOTAL	2.7	20.3	32.6	38.1	47.8	56.6	53.9	54.3	53.0	44.4	32.1	28.0	26.5	26.5	21.1	7.7

PROGRAM CUMULATIVE FUNDING



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	2.7	23.0	49.7	61.8	63.3											
PRODUCTION			5.9	26.5	54.5	86.0	113.3	139.0	162.4	177.2	179.5					
OPERATION				5.4	23.9	49.0	75.6	104.2	133.8	163.4	193.0	221.0	247.5	274.0	295.1	303.0
TOTAL	2.7	23.0	55.6	93.7	141.6	198.2	252.1	306.4	359.4	403.8	435.9	463.9	490.4	516.9	538.0	545.8

Figure 9-6. Tug Cost Summary, D-1S(R), Option 3

SPACE TUG
TECHNICAL CHARACTERISTICS

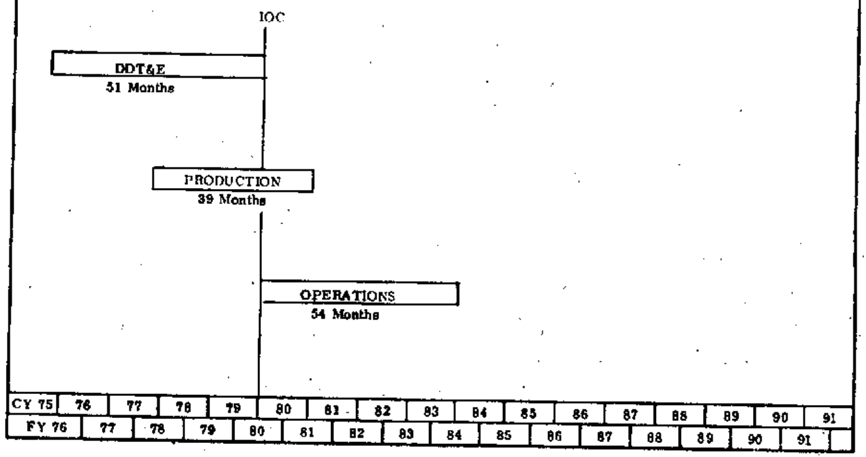
WBS 320-00 Space Tug Project (Centaur RLTC)

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the RLTC Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	4	7	16
Tugs Expended	0	2	10
Auxiliary Stages Required	2	4	7
Total Tug/Shuttle Flights	54	106	220

SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 (Centaur RLTC) Option I



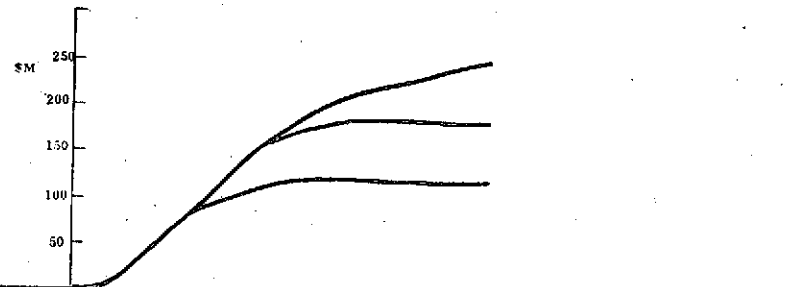
PROGRAM ANNUAL FUNDING

RLTC 4 YEAR

FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90
DDT&E	7.2	46.2	45.5	19.7	2.6	1.0									
PRODUCTION			10.5	26.9	18.9	1.9									
OPERATION					10.6	16.9	16.0	15.8	7.7						
TOTAL	7.2	46.2	56.0	46.6	32.1	19.8	16.0	15.8	7.7						

PROGRAM CUMULATIVE FUNDING

RLTC - 4 YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	7.2	53.4	98.9	118.6	121.2	122.2										
PRODUCTION			10.5	37.4	56.3	58.2										
OPERATION					10.6	27.5	43.5	59.3	67.0							
TOTAL	7.2	53.4	109.4	156.0	188.1	207.9	223.9	239.7	247.4							

Figure 9-7. Tug Cost Summary, RLTC, Option 1

SPACE TUG
TECHNICAL CHARACTERISTICS

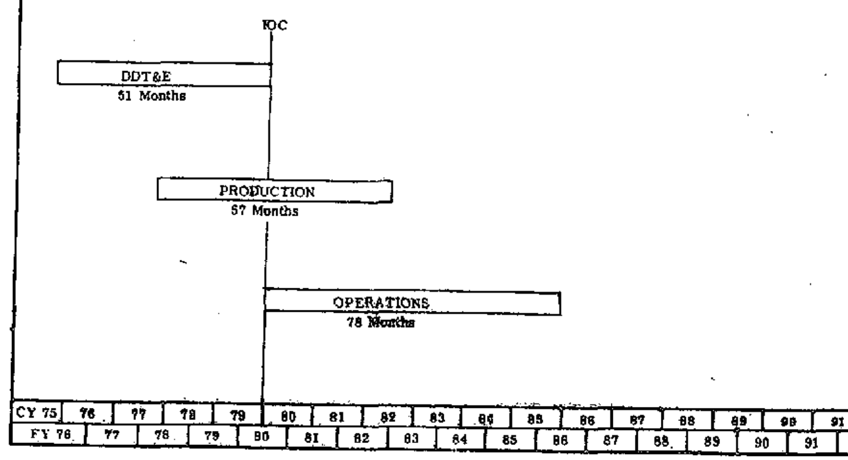
WBS 320-00 Space Tug Project (Centaur RLTC)

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the RLTC Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	4	7	18
Tugs Expended	0	2	10
Auxiliary Stages Required	2	4	7
Total Tug/Shuttle Flights	59	106	220

SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 (Centaur RLTC) Option II



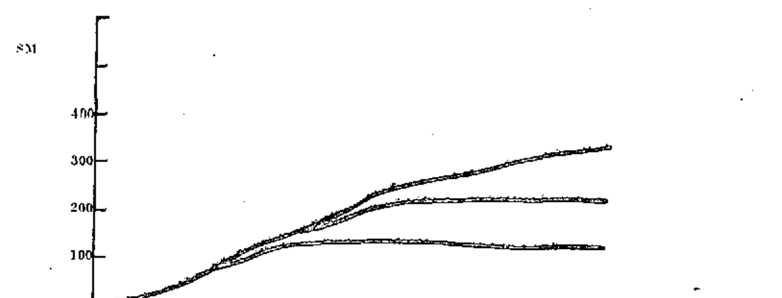
PROGRAM ANNUAL FUNDING

RLTC - 6 YEAR

FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90
DDT&E	7.2	46.2	45.5	19.7	2.7	1.0									
PRODUCTION			8.6	27.3	31.8	29.0	6.7								
OPERATION					10.1	18.5	20.3	17.9	19.1	19.0	9.1				
TOTAL	7.2	46.2	54.1	47.0	44.6	48.5	27.0	17.9	19.1	19.0	9.1				

PROGRAM CUMULATIVE FUNDING

RLTC - 6 YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90
DDT&E	7.2	53.4	98.9	118.6	121.3	122.2									
PRODUCTION			8.6	35.9	67.7	98.7	103.4								
OPERATION					10.1	28.6	48.9	66.8	85.9	104.9	114.0				
TOTAL	7.2	53.4	107.5	154.5	199.1	247.6	274.6	292.5	311.6	330.6	339.6				

Figure 9-8. Tug Cost Summary, RLTC, Option 2

SPACE TUG
TECHNICAL CHARACTERISTICS

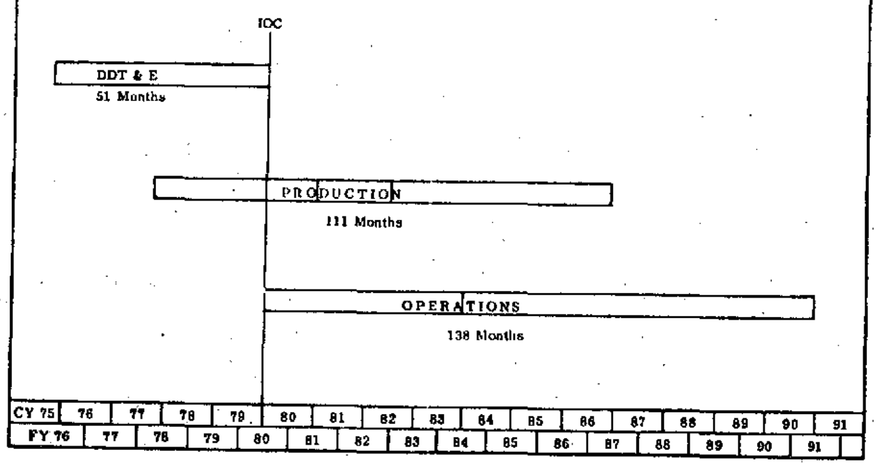
WBS 320-00 Space Tug Project (Centaur RLTC)

Summarizes the effort to provide hardware, software, services and facilities required to develop, produce, operate, support, and maintain the RLTC Space Tug. Included are all contractor and government tasks associated with management, integration, logistics, facilities, GSE, launch and flight operations including auxiliary stages and Orbiter interface equipment, which are necessary to support the prescribed mission model over the designated time period.

OPTION	I	II	III
Number of Flight years	4	6	11
Tugs Manufactured (Operational)	4	7	16
Tugs Expended	0	2	10
Auxiliary Stages Required	2	4	7
Total Tug/Shuttle Flights	54	106	220

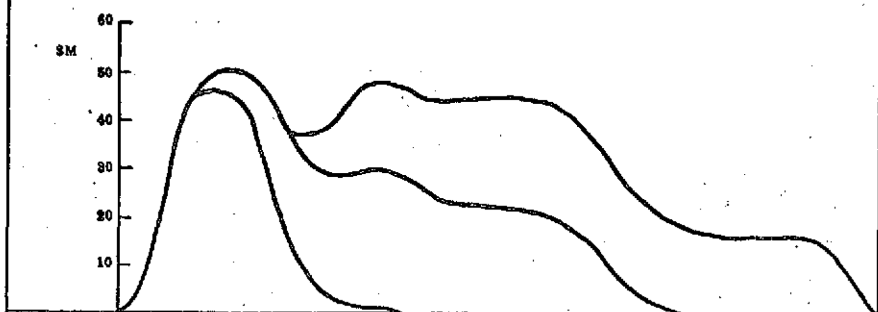
SPACE TUG
MAJOR SCHEDULE MILESTONES

WBS 320-00 (Centaur RLTC) OPTION III



PROGRAM ANNUAL FUNDING

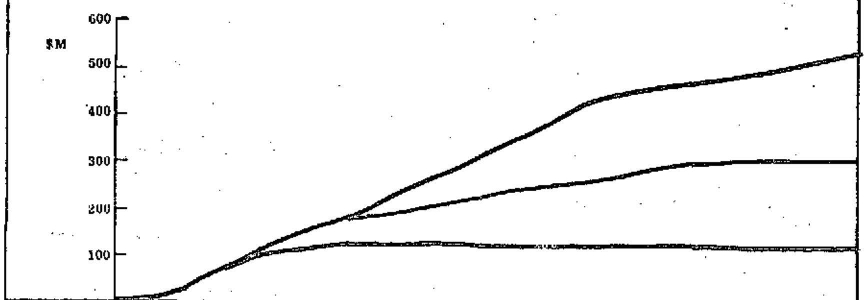
RLTC - 11YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	7.2	46.2	45.5	19.7	2.6	1.0										
PRODUCTION			5.7	21.3	29.1	31.6	26.4	24.6	23.6	19.5	8.9	.9				
OPERATION					9.7	16.7	18.0	20.5	21.6	21.8	21.8	20.0	18.0	17.9	17.8	8.6
TOTAL	7.2	46.2	51.2	41.0	41.2	49.3	44.4	45.1	45.4	41.3	30.7	20.9	18.0	17.9	17.9	8.6

PROGRAM CUMULATIVE FUNDING

RLTC - 11 YEAR



FY	76	77	78	79	80	81	82	83	84	85	86	87	88	89	90	91
DDT&E	7.2	53.4	98.9	118.6	121.2	122.2										
PRODUCTION			5.7	27.0	56.1	87.7	114.1	138.7	162.3	181.3	190.7	191.5				
OPERATION					9.5	26.2	44.2	64.7	86.5	108.3	130.1	150.1	168.1	186.0	203.8	212.4
TOTAL	7.2	53.4	104.6	145.6	186.8	236.1	280.5	325.6	371.0	412.3	443.0	463.9	481.9	499.8	517.6	526.1

Figure 9-9. Tug Cost Summary, RLTC, Option 3

Table 9-1. Reusable Centaur Program Cost Summary
(all costs in 1973 $M)

	D-1S(R)			RLTC		
Programmatic Data						
Operational Years	4	6	11	4	6	11
Tug Production	5	11	23	4	7	16
Auxiliary Stages	44	78	156	2	4	7
Total Launches	56	111	228	54	106	220
Unit Cost Data ($M)						
Average Unit Cost	8.1	7.5	6.9	11.9	11.3	10.4
Average Cost/Flight	1.7	1.5	1.3	1.2	1.1	1.0
Tug Program Costs ($M)						
DDT&E Cost	63	63	63	122	122	122
Investment Cost	51	102	180	58	103	191
Tug Operations Cost	64	109	198	66	113	210
Auxiliary Stage Cost	30	53	105	1	2	3
Tug Program Cost	208	327	546	247	340	526
Shuttle Operations Cost ($M)	588	1166	2394	567	1113	2310

In comparing costs between the programs, the DDT&E cost of the RLTC is approximately twice the cost of the D-1S(R). A more detailed discussion of the DDT&E costs is included later in this section. The investment costs are roughly the same for both configurations, because although more D-1S(R) Tugs are produced, the RLTC has a higher unit cost. Since both configurations have almost the same number of launches and require roughly the same size of operating crews, the tug operations cost is approximately the same for both. The difference in program operations costs is due to the relatively large number of auxiliary stages used with the D-1S(R). These auxiliary stages are the cause of the higher D-1S(R) cost per flight. Higher operations costs for the D-1S(R) partially offset the lower development costs, but result in a total Tug program cost that is slightly lower for the D-1S(R) 4- and 6-year options. RLTC operations costs are lower than D-1S(R) for all options and the RLTC program cost is slightly lower for the 11-year option.

The last cost item shown in Table 9-1 is the Shuttle operations cost (at $10.5M per launch). It should be noted that the Shuttle operations costs approach an order of magnitude greater than the Tug operations costs and are two to four times greater than the total tug program costs. This has the effect of Tug/Shuttle costs being very sensitive to the number of launches required. Therefore, the greater the Tug vehicle performance and multiple payload capability, the lower the number of launches, Shuttle flights, and program cost.

A breakdown of the D-1S(R) and RLTC program costs by major project WBS elements is shown in Tables 9-2 and 9-3. The costs presented include "cost growth" allowances at the subsystem level which result in an overall program allowance of approximately 20% in order to represent a "most likely" program cost.

Estimates were developed using Convair's recent experience in the development of the current D-1 Centaur by projecting the costs associated with modifications required for the Reusable Centaur Tug configurations. The Centaur D-1S(R) requires little modification from the current Centaur D-1T configuration and appears to place no new demands on technology. Many of the development costs have already been borne by preceding Centaur D-1A and D-1T programs and, therefore, Centaur D-1S(R) would become an incremental step in Centaur's evolutionary development. The Centaur RLTC requires more modifications to the current Centaur than D-1S(R), but like D-1S(R) there are no new demands on technology.

As shown in Tables 9-2 and 9-3, Tug vehicle costs account for more than one-third of the D-1S(R) and half of the RLTC development program costs. The Orbiter interface package provides the mechanical and avionics link between Orbiter and Tug. Auxiliary stages are modifications of existing TE 364-4 and Burner II velocity packages for the added performance required by certain missions. Logistics consist of personnel training and some software modifications required for the KSC simulator. Facilities costs include electrical and air-conditioning modifications to the ETR Hangar K maintenance facility. The vehicle test category includes the cost of personnel and services to conduct major ground and flight tests, but excludes the cost of test articles. Ground test hardware is included in the individual subsystem development costs. Systems engineering and integration and program management costs were derived from recent Centaur experience data on the D-1A/D-1T program as they relate to the vehicle engineering design and development effort.

Table 9-4 provides a breakdown of the elements for the Tug vehicle DDT&E costs. The D-1A/D-1T development task included modifications that closely resemble the D-1S(R) structural development tasks and the RLTC tank structure closely resembles the proposed GT Centaur configuration. Therefore Centaur D-1A/D-1T structures development cost data and the Growth Tank Centaur proposal cost estimates were used in estimating the D-1S(R) and RLTC structures costs. The RLTC body structure incorporates the use of composites for the adapters. Convair's experience in the use of composites on other programs was used in estimating these costs.

Table 9-2. D-1S(R) Program Costs (in 1973 $M)

WBS Elements	Program DDT&E	4 Year Program		6 Year Program		11 Year Program	
		Invest.	Opns	Invest.	Opns	Invest.	Opns
-01 Project Management	1.7	0.9	0.1	1.7	0.2	3.1	0.4
-02 Systems Engineering & Integration	3.5	6.0	0.7	10.9	1.4	19.9	2.8
-03 Vehicle							
Tug	23.6	36.8	4.9	72.7	9.8	139.1	20.1
Orbiter Interface	8.3	6.3	0.1	6.3	0.1	6.3	0.2
-04 Auxiliary Stage	1.9	--	29.6	--	52.4	--	104.8
-05 Logistics	3.5	0.4	2.5	2.1	6.0	2.4	11.6
-06 Facilities	0.7	--	0.8	4.1	1.7	4.1	3.5
-07 Ground Support Equipment	6.6	--	0.3	4.5	0.6	4.5	1.3
-08 Vehicle Test	3.7	--	--	--	--	--	--
-09 Launch Operations - WTR	--	--	--	--	7.8	--	20.8
-10 Launch Operations - ETR	--	--	12.3	--	19.3	--	35.4
-11 Flight Operations - NASA	9.7	--	20.9	--	29.1	--	45.2
-12 Flight Operations - DOD	--	--	15.8	--	20.6	--	30.9
-13 Refurb & Integration - WTR	--	--	--	--	3.1	--	8.2
-14 Refurb & Integration - ETR	--	--	5.9	--	9.4	--	17.7
Program Phase Subtotals	63.2	50.5	93.8	102.3	161.6	179.5	303.0
Tug Program Totals (DDT&E + Invest. + Opns)		207.6		327.2		545.8	

Table 9-3. RLTC Program Costs (in 1973 $M)

WBS Elements	Program DDT&E	4 Year Program		6 Year Program		11 Year Program	
		Invest.	Opns	Invest.	Opns	Invest.	Opns
-01 Project Management	3.6	1.1	0.1	1.7	0.3	3.4	0.6
-02 Systems Engineering & Integration	7.4	7.1	0.9	11.0	1.8	21.8	3.7
-03 Vehicle							
Tug	61.9	44.2	6.1	71.2	12.0	146.4	24.9
Orbiter Interface	7.2	5.2	0.1	5.2	0.2	5.2	0.4
-04 Auxiliary Stage	1.7	--	0.7	--	1.4	--	2.4
-05 Logistics	4.1	0.4	2.4	2.0	5.9	2.3	11.5
-06 Facilities	0.7	--	0.8	4.1	1.7	4.1	3.5
-07 Ground Support Equipment	15.0	--	0.7	8.2	1.3	8.2	2.7
-08 Vehicle Test	5.8	--	--	--	--	--	--
-09 Launch Operations - WTR	--	--	--	--	7.8	--	20.9
-10 Launch Operations - ETR	--	--	12.2	--	18.7	--	34.7
-11 Flight Operations - NASA	14.9	--	20.8	--	28.8	--	47.6
-12 Flight Operations - DOD	--	--	15.6	--	20.3	--	30.6
-13 Refurb & Integration - WTR	--	--	--	--	3.4	--	8.8
-14 Refurb & Integration - ETR	--	--	6.6	--	10.5	--	20.3
Program Phase Subtotals	122.2	58.1	67.1	103.4	114.0	191.4	212.5
Tug Program Totals (DDT&E + Invest. + Opns)		247.4		339.6		526.1	

Table 9-4. Reusable Centaur Vehicle DDT&E Costs (1973 $M)

	D-1S(R)	RLTC
Structure	(2.4)	(14.6)
Tankage	1.3	7.3
Body	0.9	5.7
Payload Interface	0.2	1.6
Propulsion	(6.7)	(14.1)
Main Engine	3.3	3.6
Support	2.1	5.2
ACPS	1.3	5.3
Avionics	(12.5)	(27.9)
Data Management	2.7	5.9
GN&C	4.5	8.6
Communications	3.9	3.9
Instrumentation	0.7	1.3
Electrical	0.7	8.1
Thermal Control	(1.4)	(1.5)
Installation, Assembly, & Checkout	(0.7)	(3.8)
Total Vehicle	23.6	61.9

Propulsion subsystem costs include validation of the current Centaur RL10A-3-3 engines for slow chill capability. These costs are supported by an estimate from the engine manufacturer. Subsequent information indicates that added main engine testing is required for D-1S(R) to demonstrate the existing engine operation at a 5.8:1 mixture ratio. This added testing would increase the reported estimate by approximately $1.0M. The RLTC propulsion subsystem also includes a hydrazine drive system for the boost pumps. The RLTC ACPS is a N_2H_4 system utilizing components from existing programs. The D-1S(R) ACPS is a modification of the Centaur H_2O_2 system.

Avionics subsystem costs were derived from current Centaur data where applicable and also from vendor cost data. The D-1S(R) subsystems are the same as current Centaur with added DCU memory, sun and horizon sensors for guidance update, and USB and SGLS communications systems. The RLTC incorporates a larger version of the D-1S(R) computer in a triple redundant configuration; dual redundant guidance platforms (DIGS), which have been proposed for the current Centaur; and a lightweight fuel cell. Costs for the above estimates are supported by data from the fuel cell and guidance system manufacturers.

Thermal control costs were based on the Centaur D-1 multi-layer insulation system which covers the forward bulkhead. The Reusable Centaur insulation system is essentially the same system with Kapton replacing mylar and the metallized film changed to gold.

Installation, assembly, and checkout cost estimates were derived from Centaur historical cost data. Since the D-1S(R) represents a small departure from the existing Centaur operation, a relatively low initial cost would be incurred. The RLTC would require a larger initial cost but, because of the similarity to the existing Centaur operation, it is still less than a new Tug configuration.

9.7 CONCLUSIONS

In summary, the D-1S(R) configuration has the advantage of low development program expenditures during the critical 1975 to 1978 time period with a peak funding requirement of \$35M to \$55M occurring in the early 1980's. The RLTC configuration's higher development cost, which causes a peak funding of over \$50M in fiscal 1978, is offset by lower operating costs. These lower operating costs are primarily due to better performance, resulting in lower requirements for auxiliary stages.

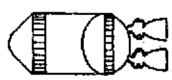
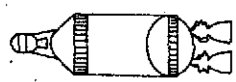
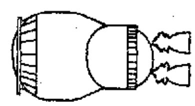
In terms of total program cost, the RLTC is 19% higher than the D-1S(R) for the four year option and the two configurations are within 4% of each other for the 6- and 11-year options. Therefore, the D-1S(R) has a program cost advantage for the 4-year option, while the RLTC has program cost advantages for the 11-year program and an operations cost advantage for all three options.

SECTION 10

ADDITIONAL VERSIONS CONSIDERING LONG DoD PAYLOADS

As part of the overall OOS study, Convair developed additional data on a short version of Centaur considering long DoD payloads. This information is summarized separately to avoid any confusion that might occur because of the different mission model and other variations in requirements and ground rules.

There are a range of Centaur candidates to satisfy DoD missions. Until the DoD program requirements and study ground rules are firmly established, no attempt will be made to optimize. Figure 10-1 compares three possible solutions. An expendable Centaur, based on the Centaur/Shuttle Integration Study for NASA/LeRC, is a viable approach with very high performance and low development cost. Reusable D-1S(R) versions with kick stages can perform the missions with the lower recurring costs inherent in reusability. A higher payload capability version is a short, wide tank configuration called Reusable Centaur (RC), which can perform the DoD mission model without kick stages. This section summarizes data on this last version.

	 SHORT D-1S EXPENDABLE	 D-1S(R) REUSABLE	 RC REUSABLE
INSTALLED LENGTH	28 FEET	35 FEET (WITH KICK STAGE)	28 FEET
PERFORMANCE			
SYNC. EQ. EXP.	7,800	–	–
SYNC. EQ. REUSE.	–	4,096	4,581
12 HR. EXP.	4,000 (20 FT. LENGTH)	–	–
12 HR. REUSE.	–	6,690 (25-FT. LENGTH)	17,600 (22-FT. LENGTH)
DEVEL. COST TOTAL	$42.5M	$62.3M	$77.2M
INCREMENT FOR REUSE (Δ FROM EXPENDABLE)		$19M	$35M
RECURRING OPS COST			
EXPENDABLE	$6.7M	$8.8M	$9.3M
REUSABLE		1.4 (INCL. KS)	0.8
LAUNCHES TO AMORTIZE DEVEL. INCREMENT FOR REUSE	–	6	8

REUSABILITY IS WORTH THE ADDED DEVELOPMENT COST

Figure 10-1. Centaur Versions for Initial Upper Stage

A basic concern is whether or not reusability is worth the added development cost. In most instances the cost is not associated with longer life components, but rather with higher performance systems capable of return to the Orbiter. Therefore the comparison is not really between two versions of one configuration, but rather two configurations. Figure 10-1 indicates that the increment cost for reusability of about $19M to $35M can be recovered in less than a year of operation due to a saving of approximately $5M per flight because the vehicle is not expended. Reusability is therefore worth the added development cost.

10.1 DoD REQUIREMENTS

The most basic DoD requirements for an OOS are:

a. Accommodates DoD payloads up to 35 feet in length.

b. Performance capability exceeds placement of 3500 pounds in synchronous equatorial orbit.

c. Total development costs are below $100M.

The NASA and DoD mission model provided during the Reusable Centaur study showed nothing longer than 25 feet. While NASA had placed great emphasis on low development cost, the DoD limit was apparently lower. The following ground rules were used for the DoD long payloads portion of this report:

a. ETR launches only; OOS can dogleg for Polar flight.

b. DoD and NASA traffic similar to STSS, but ignore three flight limit for the first year.

c. Some DoD payloads up to 35 feet long, others 32 feet long.

d. Payloads can walk to final position.

e. Low development cost is a primary requirement.

f. A dedicated flight test.

g. Fail safe so as not to jeopardize Orbiter and crew.

h. Multiple payload placement is desirable.

i. 0.97 probability of mission success is a goal.

j. Weight contingency of 2% for existing hardware, 10% for new.

k. 1% flight performance reserve.

l. Total velocity change of 50 feet/sec from APS.

The short duration, company-funded study summarized here concentrated on three areas:

a. Weight extrapolated from existing Centaur actuals.

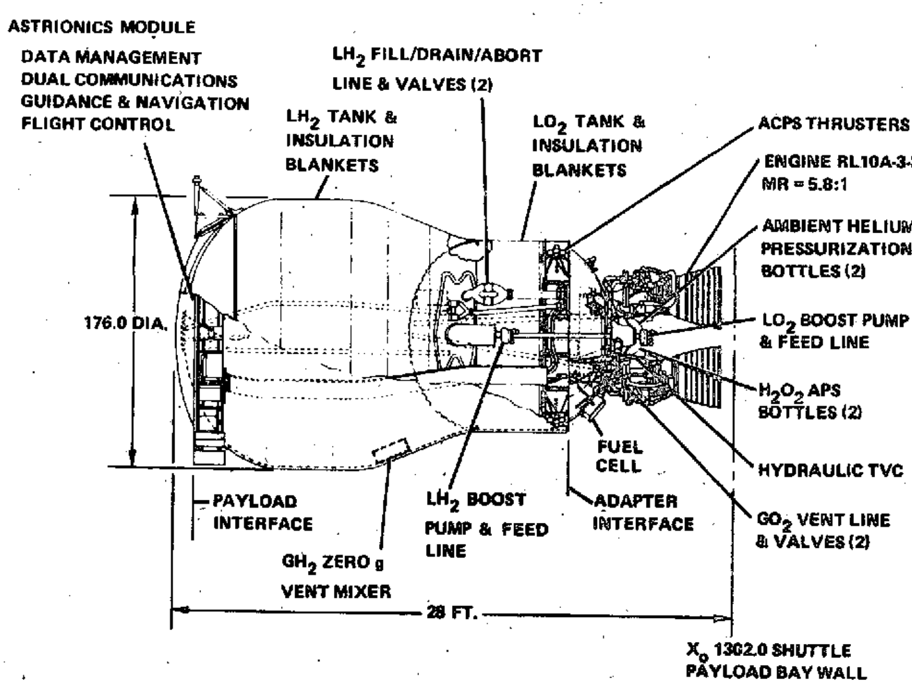
b. Development costs derived from the ongoing Titan-Centaur integration program actual costs.

c. Development tests from component qualification to vehicle flight test.

The following data emphasizes the differences from the NASA study results already covered in this report; where the data is equally applicable it is not repeated. A minimum of cost data is published here due to its proprietary nature. Whether DoD or NASA eventually becomes the developing agency for an OOS or Initial Upper Stage, some Reusable Centaur version is a leading candidate.

10.2 MECHANICAL SYSTEMS CONFIGURATION

The general configuration of Reusable Centaur is shown in Figure 10-2. Like D-1S(R) and RLTC, a large portion of the hardware and components selected for RC are existing Centaur D-1T hardware or off-the-shelf-type hardware. The remainder is well within the current state-of-the-art technology.



WEIGHTS (LB.)	
STRUCTURES	1,723
PROPULSION	1,717
THERMAL CONTROL	488
ASTRIONICS	787
CONTINGENCY	200
TOTAL DRY WEIGHT =	4,995

Figure 10-2. Reusable Centaur Configuration

The engines are existing D-1T RL10A-3-3 operating at 5.8:1 mixture ratio and with the cooldown valves adjusted to minimize propellant chilldown losses. The boost pumps are also D-1T hardware with an idle mode added to conserve chilldown propellants. The auxiliary propulsion system is the D-1T hydrogen peroxide system with forward facing thrusters and added valves for increased safety and operational capability. The tanks are pressure stabilized 301 CRES (like D-1T) with an enlarged diameter LH_2 tank and lengthened LO_2 tank to provide greater propellant capacity. The tank insulation system (similar to the blanket and radiation shield design for D-1T) has multilayer goldized Kapton blankets for ground operations, and three-layer radiation shielding for deep space protection. Tank vent systems use D-1T hardware plus new zero-g thermodynamic vent and mixers (prototype tested at Convair). The adapters are of aluminum and titanium skin-stringer construction. The propellant tank pressurization system uses D-1T hardware and stores helium at ambient temperature. The propellant fill and drain system also provides the capability for in-flight propellant dumping in the event of an abort. The thrust vector control system is the same as the D-1T hydraulic system with the addition of reservoir heaters for thermal control.

While the propellant tanks are enlarged, so that the RC looks like an RLTC, most of the subsystems are current type like the D-1S(R).

Note that this Reusable Centaur is 28 feet long installed in the Orbiter payload bay. This length will satisfy all but one of the NASA and DoD payloads currently reported. For the special 35-foot DoD payload in the 12-hour orbit, a second extra short version is required. A 22 to 25 foot Reusable Centaur can be built without the cylindrical sections in the hydrogen or both tanks. The hydrogen PU probe has to be reshaped, and several lines and cables and the insulation blankets shortened. This is a minor task, based on experience with several Atlas versions with tank length (and forward bulkhead contour) variations.

10.2.1 STRUCTURE. The Reusable Centaur is a modification of the existing D-1T Centaur with increased propellant capacity and changes to enable use in the Space Shuttle Orbiter. The propellant capacity has been increased about 50% by adding a cylindrical section to the LO_2 tank and by increasing the LH_2 tank diameter from 120 to 172 inches (3.05 to 4.37 m). The other changes involve the relocation of fluid disconnects from the locations of the present expendable launch vehicles to locations suitable for reuse in the Orbiter. The structural adapters are modified to enable introduction of the four-point loads for mounting in the Orbiter payload bay for both vertical launch and horizontal landing. The general configuration shown in Figure 10-3 identifies the structural components.

Main Propellant Tanks. The Reusable Centaur propellant tanks are constructed the same as the RLTC tank described in Section 2.2.1, except they are shorter to accommodate long DoD payloads. Hat shaped rings are welded to the inside of the 60-inch transition and the forward bulkhead to prevent buckling of the skin when

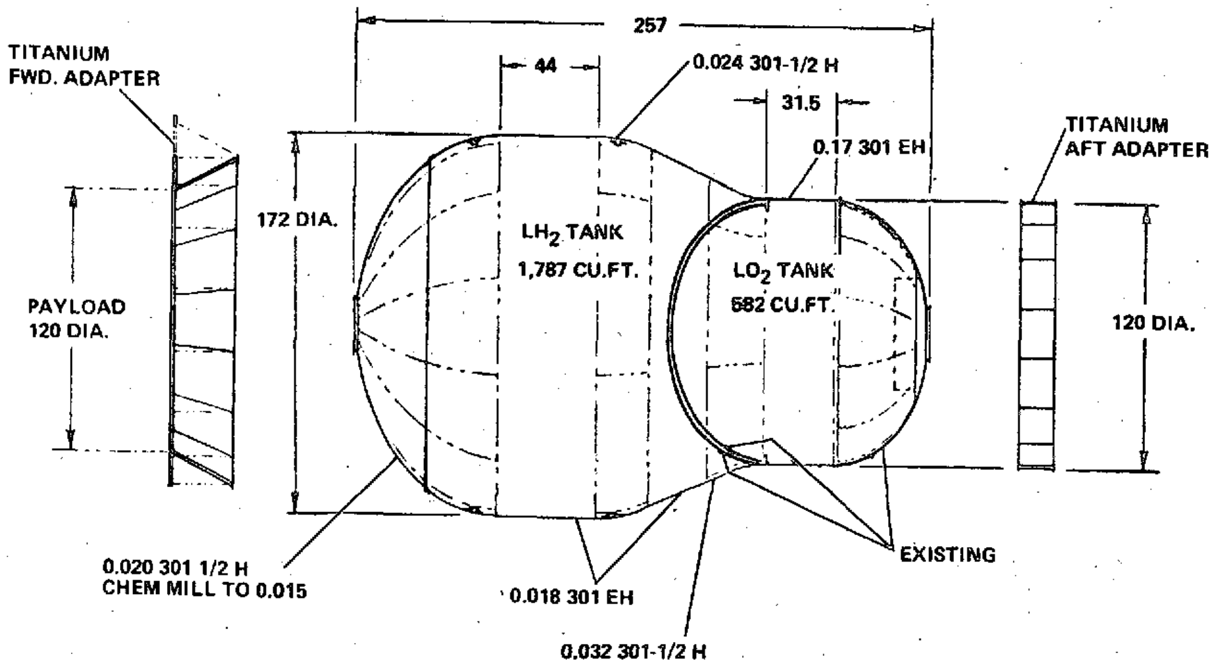


Figure 10-3. Reusable Centaur Tanks and Adapters

unpressurized with stretch loads applied. Axial stretch loads applied to the monocoque tank allow tank handling, maintenance, and transport with tank pressure vented. The LH_2 tank forward bulkhead is assembled from stretch-formed gores chem-milled to final thickness. A 24-inch-diameter (61 cm) forward door allows access to the tank for manufacture and installation of components. A 150-inch-diameter (3.82 m) flange is welded to the bulkhead to provide mounting of the forward adapter. Clips and brackets are resistance welded directly to the tank skins for attachment of tubing, cableways, and components. The intermediate bulkhead is identical with that for D-1T, RLTC, and D-1S(R). Tank life is estimated to be a minimum of 45 flights.

There are two key drivers behind the selection of this tank configuration. It significantly simplifies flight operations and lowers user costs to eliminate kick stages, but this means increased Centaur propellant capacity. Considering the length constraint, the tank diameter has to be increased and the mixture ratio adjusted to 5.8:1. A wealth of detail information already exists on a very similar enlarged tank design from a firm proposal made to NASA LeRC for a Growth Tank (GT) Centaur (Reference 8). Therefore the Reusable Centaur proposed tank configuration is regarded at Convair as a routine design task.

Structural Adapters. The D-1T Centaur has a titanium skin stringer cylindrical stub forward adapter and an aluminum skin stringer conical equipment module. The two units serve to mount the astrionic equipment and payload. Aerodynamic shroud loads are introduced as side loads into the adapter, which distributes the loads into the Centaur hydrogen tank.

The forward adapter for the Reusable Centaur has been configured to mate with the increased diameter tank and to enable distribution of the forward Orbiter attachment load. The Reusable Centaur forward adapter and equipment module are combined into a single unit to take advantage of the space available in the payload bay and to provide an improved load path for both payload mounting and Orbiter load introduction. The adapter is a titanium skin stringer conic section mounting to the LH_2 tank at a diameter of 150 inches (3.82 m) and reducing to the 120-inch (3.05-m) diameter payload interface. Titanium was chosen due to its low thermal conductivity and high strength when compared to aluminum. Graphite/epoxy is an alternate choice with attractive weight savings but is still a growing technology.

An aluminum frame at the forward end of the adapter picks up and distributes the Orbiter attach load into the adapter. Titanium channels both stabilize the frame and, along with fiberglass rings, provide a mounting surface for the astrionics equipment. The fiberglass rings provide further thermal isolation between the hydrogen tank and the astrionics packages.

The Reusable Centaur payload attachment is on the 120-inch (3.05-m) diameter flange of the forward adapter. Either a distributed flange load or concentrated strut loads at 12 points can be accommodated. The skin stringer construction of the forward adapter serves to distribute the concentrated loads into the LH_2 tank forward bulkhead. Electrical and fluid service connections are provided for payload use.

The aft adapter is a titanium skin stringer stub cylinder described in Section 2.2.2. It distributes loads from the 16 deployment latches to the liquid oxygen tank.

10.2.2 MAIN PROPULSION SYSTEM. The heart of all Reusable Centaur versions is the use of the existing twin Pratt & Whitney RL10A-3-3 engines, and with them, virtually the entire propulsion section arrangement on the aft bulkhead of the LO_2 tank. For the long DoD payloads, operating at a mixture ratio of 5.8:1, it is recommended that propellant weight be maximized even though the hydrogen tank is shortened. The slow chilldown technique discussed in Section 2.3 is incorporated. These modifications require adjustment of the controller and cooldown valves, but no new main engine hardware.

Mixture Ratio. Readjusting the engine mixture ratio to burn more oxygen enables more of the denser oxygen and less of the bulky hydrogen to be carried within the

28-foot vehicle length constraint. The resulting increase in total impulse increases the payload capability, even with the slight loss in Isp to 439.8 sec. The effect of mixture ratio change on payload capability within the length constrained Reusable Centaur is shown in Figure 10-4.

The engine mixture ratio shift is accomplished by resetting the engine thrust control. The PU mixture ratio excursion band of ±0.5 remains the same. The RL10A-3-3 has demonstrated the capability of operating at nominal mixture ratios up to 6:1 with no appreciable effect on thrust chamber life. Operation at 6:1 rather than 5:1 reduces the number of chamber thermal cycles from 200 to 190 which is still far in excess of the life requirement of the engine.

Boost Pumps. The existing Centaur steam turbine driven pumps are retained for the Reusable Centaur. The turbine monopropellant peroxide is common with the APS supply. The primary speed controls are adjusted to maintain the same pressure profile for the revised mixture ratio flows. Slow chilldown involves a second peroxide solenoid orificed down to allow approximately 10% flow. The fuel boost pump casting will be slightly modified to reroute the bearing coolant flow from the present exit into the sump cavity to a return line routed higher into the fuel tank. This minor change will eliminate 95% of the energy input to the sump during engine start, which will permit lower LH_2 tank pressurization as discussed in Section 10.2.3.

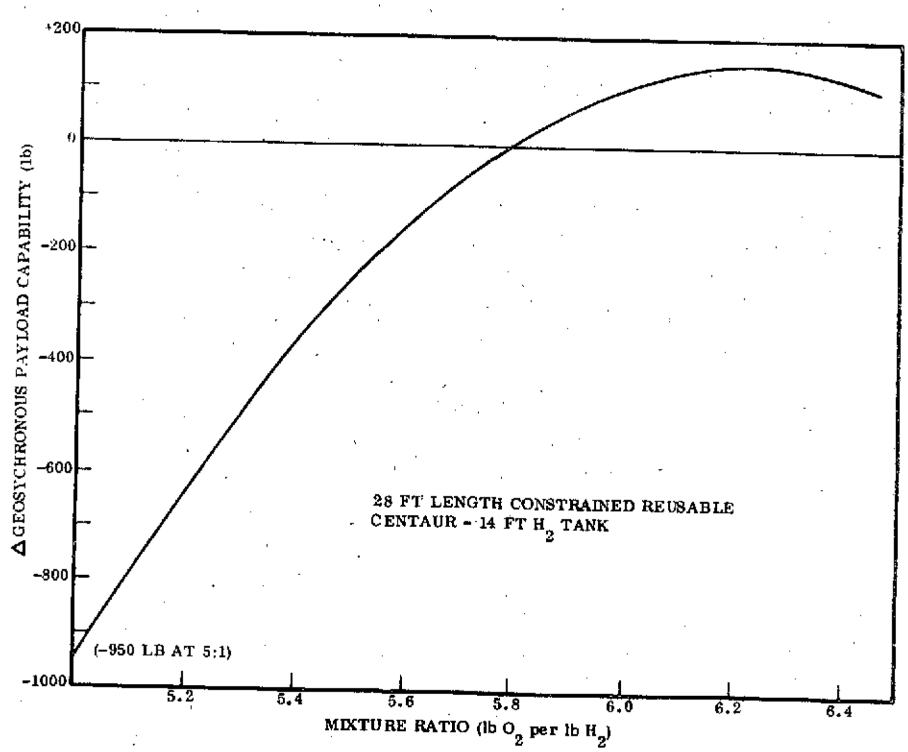


Figure 10-4. Mixture Ratio Effect on Geosynchronous Payload Capability

Thermal Conditioning. The need for the revised thermal conditioning sequence of the engine prior to each start for RC is the same as for RLTC and D-1S(R) (Section 2.3.2). For five engine burns, Table 10-1 shows that the total propellant loss for engine chill-down is 138 pounds (62.6 kg). In addition to the chilldown and start losses shown in Table 10-1, each burn shutdown loss is 12 pounds (5.5 kg), and the mission leakage duration for 27.4 hours results in 52 pounds (23.4 kg) of propellant lost.

10.2.3 PROPULSION SUBSYSTEMS. The propulsion support subsystems are all the same type discussed in Section 2.3 for D-1S(R), with modifications due to the enlarged propellant tanks:

a. Propellant Feed Lines. Same as D-1T plus tank isolation valves and lengthened LH_2 duct leg. Thermal protection by radiation shield, foam not required.

b. Thrust Vector Control. Same hydraulic system as D-1T plus heaters.

c. Pressurization System. Same ambient helium system as D-1T, resized, with more redundancy.

d. Vent System. D-1T tanking boiloff valves (quad redundant), H_2 vent line inside tank plus zero-g thermodynamic vents.

e. Propellant Utilization. Same as D-1T with reshaped hydrogen and oxygen probes.

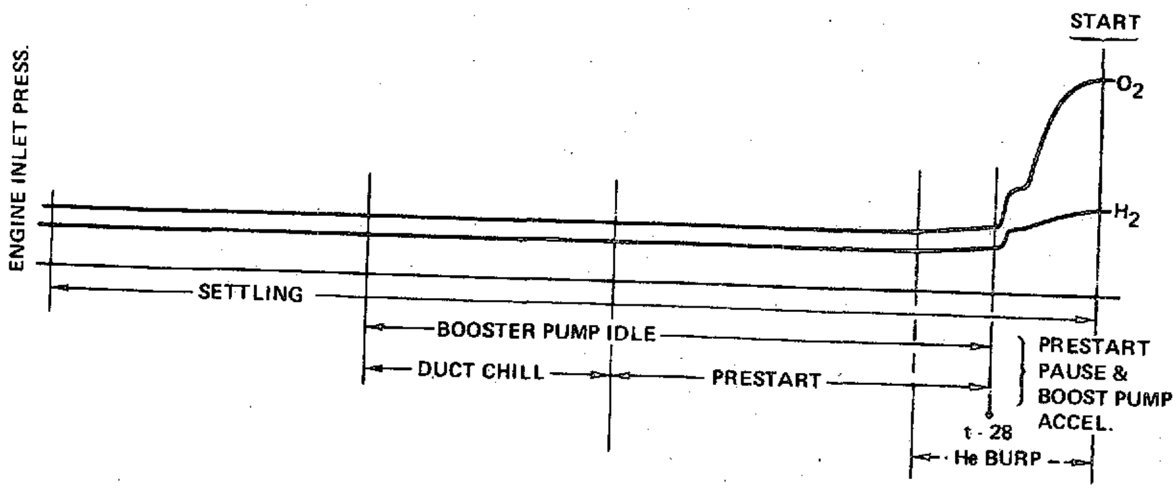
f. Intermediate Bulkhead Vacuum System. Same as D-1T.

g. Fill and Drain System. New valves (quad redundant) and lines to relocated disconnect panels, enlarged for inflight dump.

Pressurization System. The pressurization system for RC is essentially the same for D-1S(R), shown in Section 2.4.

The tank pressurization helium requirements provide a pressure increase of 3 psi (2 N/cm^2) for the LO_2 tank and 1.5 psi (1 N/cm^2) for the LH_2 tank prior to each engine prestart. Current Centaur vehicles use 3 psi (2 N/cm^2) for the LH_2 tank (as well as the LO_2 tank), but 1.5 psi (1 N/cm^2) is adequate for RC. A recent thermodynamic model (verified by simulations matching ground and flight test boost pump operations) concluded that the LH_2 boost pump bearing coolant flow, discharged at the base of the sump, has a predominant effect on the pump NPSP because the flow represents about 95% of the total energy input to the sump region. As a result of rerouting this flow away from the sump, the vapor pressure rise due to sump heating will not exceed 0.1 psi (0.07 N/cm^2). Since the boost pump NPSP requirement is 0.1 psi (0.07 N/cm^2) and the sump heating another 0.1 psi (0.07 N/cm^2), the 1.5 psid (1 N/cm^2) added by the pressurization system prior to each engine start will meet NPSP requirements and provide a 1.3 psid (0.9 N/cm^2) margin. This reduced LH_2 tank pressurization requirement in turn permits smaller helium storage for RC

Table 10-1. Main Propulsion System Thermal Conditioning



Childown (Prestart) Thermal Conditioning Losses

Burn	Coast Time (hr)	H_2 Losses/Engine Time (sec)	(lb)	Overrun (sec)	Overrun (lb)	Total (lb)	O_2 Losses/Engine Time (sec)	(lb)	Overrun (sec)	Overrun (lb)	Total (lb)	Total per Engine (lb)	Total per System (lb)
1	2.54	56	3.5	14	8.7	12.2	70	6	-	-	6	18.2	36.4
2	5.17	48	3.2	-	-	3.2	45	3.7	3	3.1	6.8	10.0	20.0
3	11.43	56	3.5	14	8.7	12.2	70	6	-	-	6.	18.2	36.4
4	5.27	48	3.2	-	-	3.2	46	3.7	2	2.1	5.8	9.	18.0
5	2.78	35	2.9	-	-	2.9	27	1.8	8	9.	10.8	13.7	27.4
					17.4					14.2			$\boxed{138.2}$

Overrunning = 31.6 lb, a possible additional saving by split chill

Total Thermal Conditioning Losses

Burn	Boost Pump Idle Duct Chill (sec)	Prestart (sec)	Prestart (lb_b)	(lb_a)	Prestart Pause (lb_a)	Start Losses (lb_b)	Total Non ΔV Losses lb_a + lb_b = lb Total
1	84	70	36.4	1.36	2.4	10	3.8 + 46.4 = 50.2
2	71	48	20.	1.07	2.4	10	3.5 + 30. = 33.5
3	84	70	36.4	1.36	2.4	10	3.8 + 46.4 = 50.2
4	71	48	18.0	1.07	2.4	10	3.5 + 28. = 31.5
5	62	35	27.4	.87	2.4	10	3.3 + 37.4 = 40.7
							$17.9_a + 188.2_b \sim 206.$

16_a = Monopropellant non ΔV losses
16_b = Main propellant non ΔV losses

than is required for D-1S(R). With the exception of the lower LH_2 tank pressurization level, the technique of pressurization and the hardware required by the system for RC is the same as for D-1S(R).

The pressurization system helium is stored in two 4.26 ft^3 (0.12 m^3) helium supply bottles located on the aft side of the liquid oxygen tank. These helium supply bottles (made of titanium) are currently installed on Centaur vehicles.

An initial charge of 2800 psia (1930 N/cm^2) provides about 15.5 pounds (7 kg) of helium. The final bottle pressure is 500 psia (345 N/cm^2) and a total of about 12 pounds (5.4 kg) helium is available for the five-burn synchronous equatorial mission. Total helium usage consists of the following requirements:

Table 10-2. Reusable Centaur Helium Requirements

LH_2 Tank Pressurization, lb (kg)	5.50	(2.49)
LO_2 Tank Pressurization, lb (kg)	2.29	(1.04)
Engine Start Usage, lb (kg)	0.44	(0.20)
APS Propellant Pressurization, lb (kg)	2.25	(1.02)
Total, lb (kg)	10.48	(4.75)
14% Contingency, lb (kg)	1.43	(0.65)
Total Available, lb (kg)	11.91	(5.40)

The LH_2 tank pressurant requirement is based on post pressurization helium densities obtained from Centaur B-2 test stand (NASA Plumbrook) testing modified for expected average LH_2 tank helium inlet temperature.

The LO_2 tank pressurization is achieved by flowing helium through a bubbler submerged in the LO_2 tank as is the case for Centaur D-1T. Helium usage is based on B-2 tank testing for a tank pressure increase of 3 psi (2 N/cm^2).

Additional helium usage results from engine valve operation and APS propellant pressurization. The propellant tank thermodynamics were estimated from Centaur empirical simulations in order to calculate pressure profiles, venting requirements, and ullage residuals. It was assumed that a zero-g thermodynamic vent and mixer package will maintain thermodynamic equilibrium during coast periods. Zero-g venting controls the pressure to 20 psia (13.2 N/cm^2) in the LH_2 tank and 30.5 psia (21 N/cm^2)

Table 10-3. RC Pressurization System Parameters

Helium Storage Bottle Fill Pressure (Max)	2800 psia (1930 N/cm^2)
Helium Storage Bottle Final Pressure (Min)	500 psia (345 N/cm^2)
Helium Storage Bottle Charge Temp. (Nom)	530°R (294K)
Helium Storage Bottle Volume	8.5 ft^3 (0.24 m^3)
Usable Helium Available	11.9 lb (5.4 kg)
Engine Controls Supply Pressure (Nom)	450 psig (310 N/cm^2)
APS Bottle Supply Pressure (Nom)	290 psig (200 N/cm^2)

in the LO_2 tank. Helium partial pressures of 3 psi (2 N^2/cm^2) and 1-1/2 psi (1 N/cm^2) (LO_2 and LH_2 tank respectively) were assumed to exist at engine start. Table 10-3 lists pressurization system parameters.

Vent System. The function, method of operation and the hardware for the main propellant tank vent system for RC is the same as that described in Section 2.4.3 for RLTC. Table 10-4 gives some zero-g vent system characteristics for RC under deep space conditions. The deep space vent rates listed in Table 10-4 represent the steady state vent rates that would eventually occur during coast in the absence of engine firings. Actual venting will not occur during coast until the tank pressure reaches the vent pressure of the zero-g vent device. Figure 10-5 shows the LO_2 and LH_2 tank ullage pressure histories for RC as controlled by the pressurization and vent systems. The engine firing sequence for RC reduces LO_2 tank pressure so that the LO_2 tank does not actually vent any propellants overboard after the first Centaur engine firing, and the LH_2 tank vents overboard only during a portion of the coast time between burns.

Propellant Tank Fill & Drain System. The main propellant tank fill and drain systems for RC are used to both fill the tanks prior to launch, and to quickly dump tanked propellants during flight if required by an abort condition. Both propellants will be dumped at altitudes greater than 220,000 feet (67 km) by supplying helium pressurant from storage bottles located on the deployment adapter and from vehicle stored helium. Parallel redundant dump valves are installed in each dump line on both the vehicle and the deployment adapter to meet fail-closed/fail-open safety requirements. The lines were sized to permit simultaneous dumping of both full propellant tanks within 250 seconds. The orientation of the propellants during dumping (caused by the Orbiter turnaround maneuver) results in a maximum of 180 pounds (88 kg) of LH_2 and 60 pounds (27 kg) of LO_2 left in the tanks at dump termination. Table 10-5 gives the abort dump characteristics for RC.

Table 10-4. Zero-g Vent System Characteristics

Nominal System Characteristic		Propellant Tank LO_2	Propellant Tank LH_2
Tank Heat Rate (Deep Space)	- Btu/hr. (watts)	583 (171)	2476 (725)
Zero-g Vent Design Flow Rate (P/L Bay Door Open, Max Heat Rate)	- lb/hr. (kg/sec.)	87.3 (0.011)	35.1 (0.004)
No Vent Pressure Rise Rate (P/L Bay Doors Open)	- psi/hr. (Newton/ cm^2 sec)	0.2 (3.8×10^{-6})	0.7 (13.4×10^{-6})
Vent Rate (Deep Space) (Mix During Vent Only)	- lb/hr. (kg/sec.)	6.8 (8.6×10^{-4})	12.5 (15.75×10^{-4})
Vent Rate (Deep Space) (Continuous Mixing During Coast)	- lb/hr. (kg/sec.)	9.6 (12.1×10^{-4})	12.8 (16.1×10^{-4})

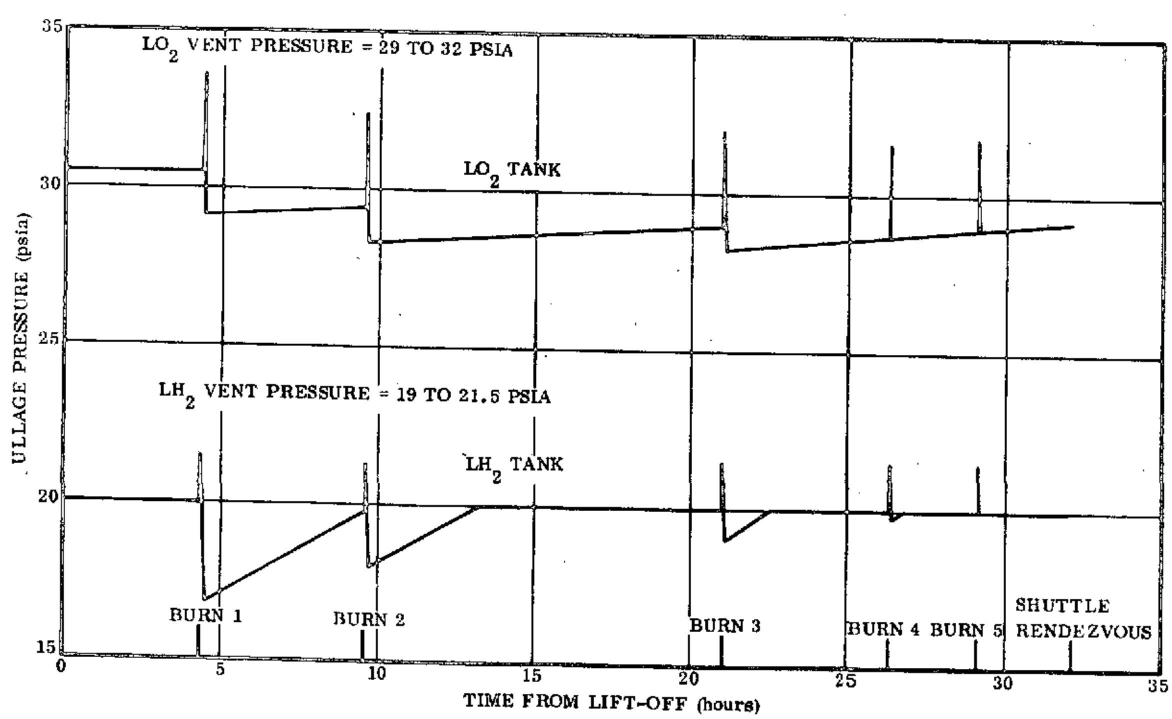


Figure 10-5. Reusable Centaur Propellant Tank Ullage Pressure vs. Time From Shuttle Liftoff

Table 10-5. Abort Dump Characteristics for RC

Abort Dump Characteristics	LO_2 Tank	LH_2 Tank
Dump Pressure	35.0-38.0 psia (24.1-26.2 N/cm^2)	24.5-26.0 psia (16.9-17.9 N/cm^2)
Dump Line Dia.	5.0 in. (12.7 cm)	6.0 in (15.2 cm)
Dump Time	250 sec	250 sec

Figure 10-6 shows the LH_2 fill and drain system, with dual, parallel redundant dump valves on both the vehicle and the deployment adapter. All propellant lines aft of the retractable umbilical panel are vacuum jacketed. The LO_2 system is similar.

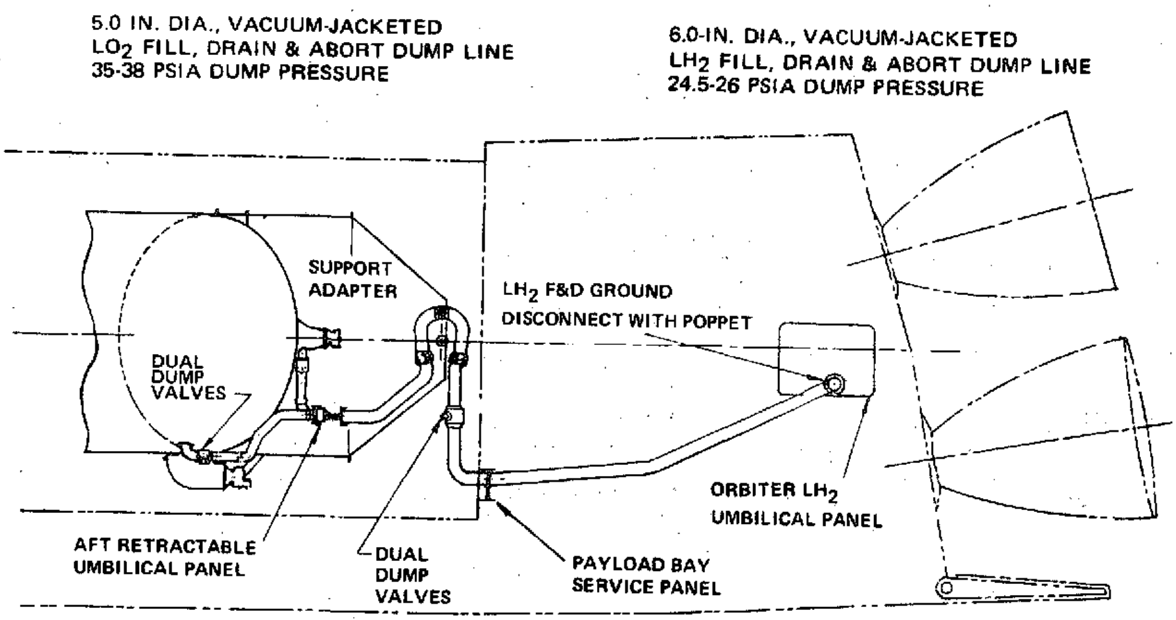


Figure 10-6. LH_2 Fill and Drain System

10.2.4 <u>AUXILIARY PROPULSION SYSTEM</u>. The APS for RC is the same as that for D-1S(R) (Section 2.5.1) with two exceptions. First, the RC system has added shutoff valves to permit isolation of each individual thruster cluster (4 places) in the event a thruster motor fails open. Since the vehicle can function with one cluster out, the system is therefore fail-safe, fail-operational. Secondly, the requirements for H_2O_2 usage are less for RC, so only two storage bottles are required. Table 10-6 lists the H_2O_2 requirements for RC, and Figure 10-7 is a system schematic.

H_2O_2 usage is based on:

a. ΔV requirements (as specified in Section 10.4.3) are based on two midcourse corrections and a payload separation for a total ΔV of about 50 ft/sec. (15 m/sec).

Table 10-6. APS Requirements Summary for Reusable Centaur

Event	Impulse (lb-sec)	H_2O_2 (lb)	(kg)
ΔV	28,200	182	(83)
Settling	16,750	108	(49)
Attitude Control	12,850	107	(49)
Boost Pumps	11,200	74	(33)
Total	69,000	472	(214)
2% Residual		10	(5)
	Total Tanked	482	(219)

Nominal capacity per bottle: 242 lb (110 kg)

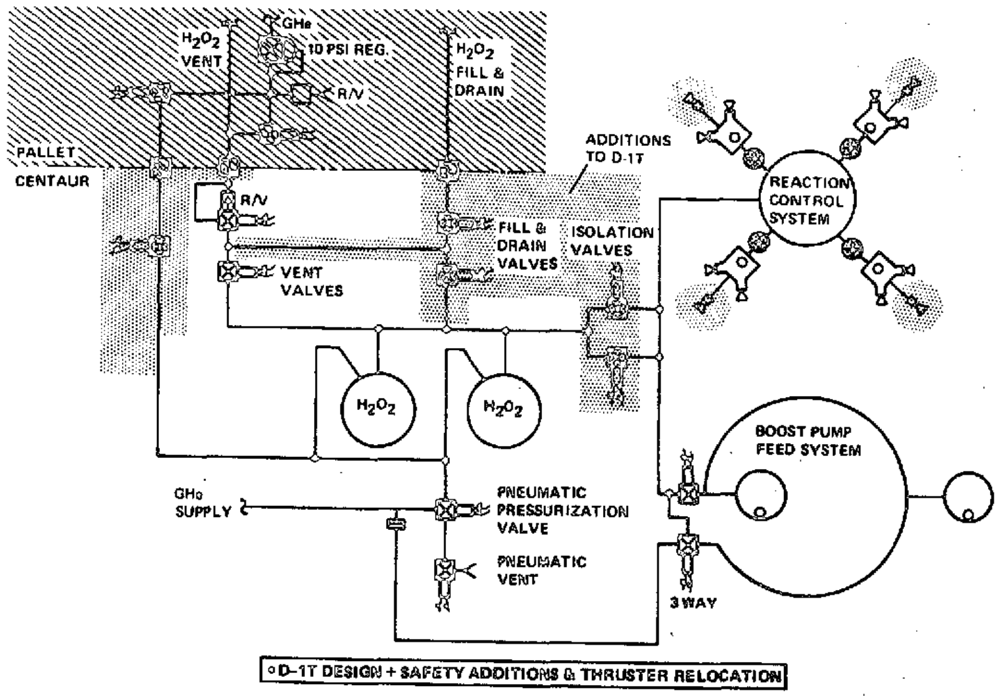


Figure 10-7. Auxiliary Propulsion System

This, of course, is significantly less than the 145 ft/sec (44 m/sec) requirement as originally specified for RLTC. Subsequent deletion of several of the RLTC requirements and reduction of the overly conservative 50 ft/sec to 20 ft/sec (15 to 6 m/sec) for the first midcourse correction have brought the overall ΔV requirements to a more realistic value.

b. Settling requirements are based on five engine starts.

c. Attitude control requirements assume that 10% of each coast is in the fine control mode at 20 lb/hr (9 kg/hr) and 90% of each coast is in the coarse control mode at 2.0 lb/hr (0.9 kg/hr). The resulting average H_2O_2 consumption during coast is about 3.8 lb/hr (1.7 kg/hr). This rate assumed a vehicle thermal maneuver was not periodically required.

d. Boost pump idle and full power modes of 1.27 lb/min (0.6 kg/min) and 5.1 lb/min (2.3 kg/min), respectively, were used prior to and during engine firing.

10.2.5 THERMAL CONTROL. Thermal control systems for RC are similar to those for D-1S(R) and RLTC, and are both active and passive. Electric line heaters and blankets provide active control for the APS system and for the boost pump catalyst beds. An electric heater in the hydraulic system reservoir provides thermal control for the TVC. (The weight breakdown includes the space radiator for the fuel cell.) Passive thermal control is provided by insulation blankets on the main propellant tanks, radiation shielding on TVC components and the main propellant tanks, thermal paints and finishes, and the vacuum environment provided between the propellant tanks by the intermediate bulkhead.

The tank insulation blankets are like those for D-1S(R) and RLTC (Section 2.6). The insulation blankets, in conjunction with the Shuttle payload bay gas conditioning system (and heaters in the deployment adapter, if required), provide a satisfactory payload bay environment prior to launch. Further study is required to optimize the Orbiter and OOS features that contribute to the payload bay environment.

Table 10-7 gives heating rates for the main propellant tanks, and Table 10-4 gives deep space vent rates. Insulation blanket performance data for various flight conditions is given in Section 2.6.

Table 10-7. Propellant Tank Heating Summary (Nominal Values)

Flight Condition	RC Tank BTU/HR (Watts) LH_2	LO_2
Closed Payload Bay, Prelaunch	126,226 (36,972)	53,770 (15,749)
Closed Payload Bay, 0-480 sec	47,575 (13,935)	18,315 (5,364)
Open Payload Bay, 480 sec to Low Altitude Orbit	4,452 (1,304)	3,652 (1,070)
Low Altitude Orbit	2,611 (765)	703 (206)
Deep Space, Continuous	2,476 (725)	583 (171)

10.2.6 ORBITER/CENTAUR INTERFACE. The Centaur with adapters is designed to use the Orbiter four-point support technique. The RC interface consists of a deployment adapter and the astrionics interface equipment. The deployment adapter as shown in Figure 10-8 has the two aft X and Z direction supports which also serve as deployment rotation points and the single Y direction support point.

The forward Z load attaches directly to the RC forward adapter at Shuttle Station 978. All fluid and electrical services for the RC are routed aft on the vehicle and interface with the deployment adapter eliminating any forward umbilical disconnects. Figure 10-9 shows the fluid systems, nine of which interface with the Orbiter.

The fluid interface between the Centaur and the Orbiter are at the aft bulkhead of the payload bay. Flight disconnect of the Centaur lines occurs in the support adapter, independent of Orbiter deflections. About 90 pounds of helium for abort dump of the main propellants is carried on the deployment adapter. The deployment adapter also contains the redundant propellant dump valves and vent valves.

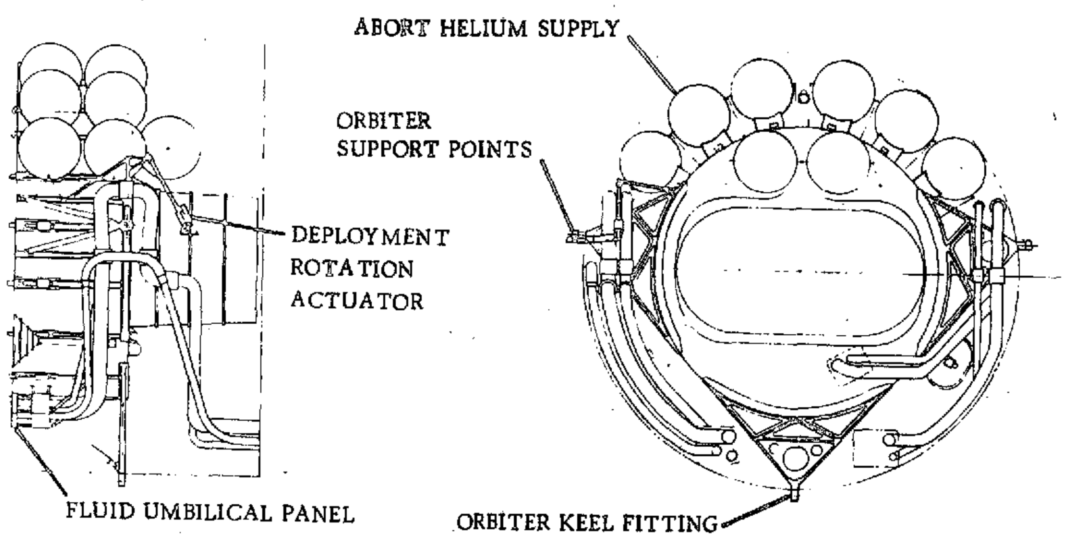


Figure 10-8. Reusable Centaur Deployment Adapter

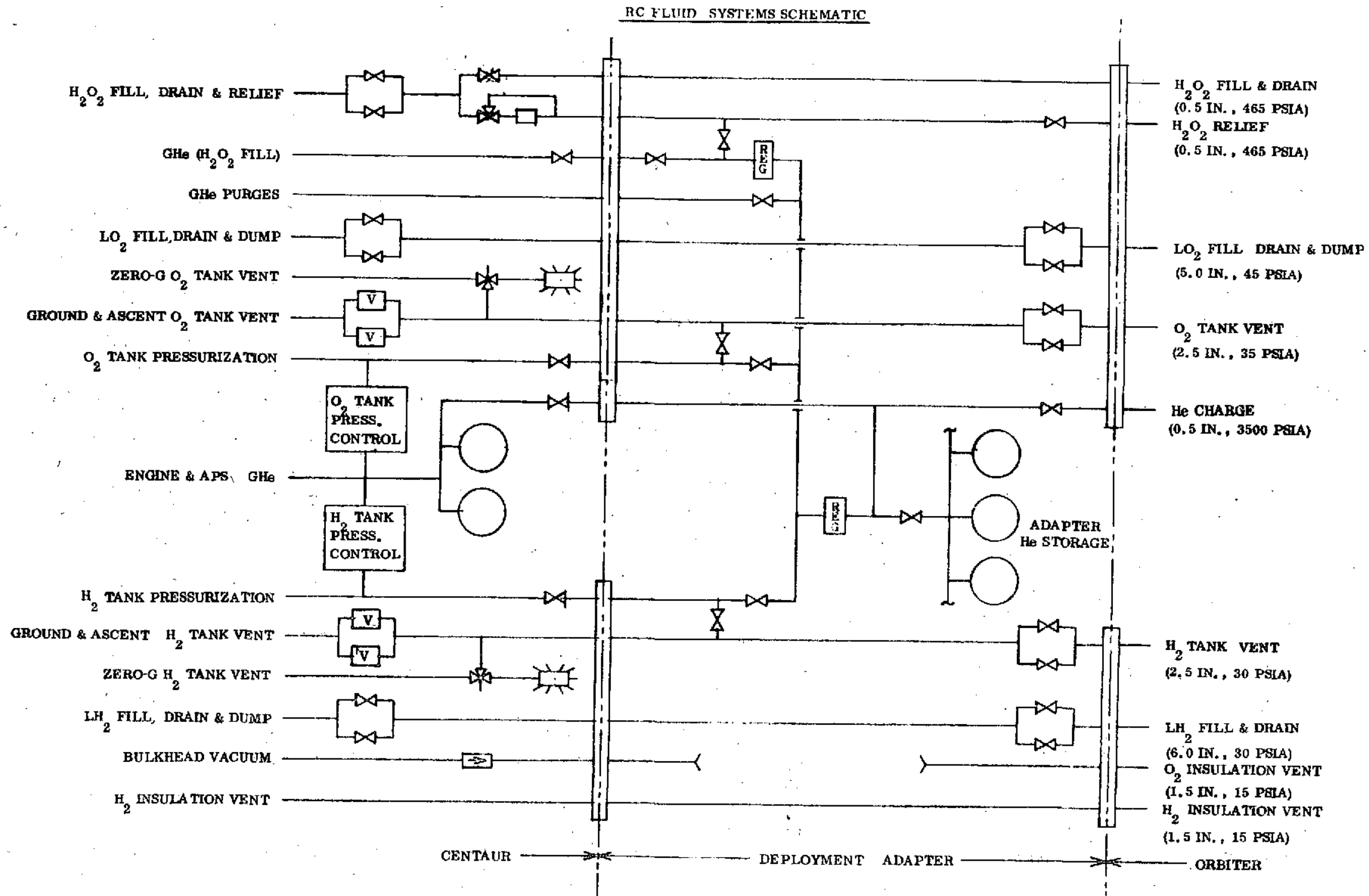


Figure 10-9. Reusable Centaur Fluid Systems Schematic

10.3 ASTRIONICS

The significant recent change on the current D-1A/D-1T Centaur program has been a major improvement of the astrionics system, with substantial associate contracts to Teledyne and Honeywell. Many former hardware functions were integrated into the airborne computer software, resulting in a flexible system readily adaptable for Shuttle operations. The astrionics task for a Reusable Centaur, therefore, is minimum because of these major improvements just completed on Centaur.

The return flight of the Centaur stage to the Shuttle Orbiter and the extended mission times require changes to the expendable D-1T astrionics sufficient for a compatible interface with the Orbiter and for reusability similar to those described for D-1S(R) in Section 3. The following astrionic areas are impacted:

a. Flight software

b. Increased computer main memory

c. Added equipment:

   1. Sun and horizon sensors for attitude update
   2. Dual SGLS communications
   3. Tape recorder
   4. Fuel cell and backup battery
   5. Backup docking attitude control

d. Deletions:

   1. C-Band transponder
   2. S-Band telemetry
   3. Range safety pyrotechnics
   4. Centaur primary batteries

Figure 10-10 is the astrionics system block diagram for the Reusable Centaur and the additions and revisions to the Centaur D-1T are noted.

A single central digital computer is used for data management and control functions and is identical to the Centaur D-1T DCU except added memory and software functions have been added for mission requirements and reusability. Nominal synchronous equatorial mission duration is 36 hours with about 30 hours free flight of the upper stage out of the Orbiter, allowing the payload to walk to its final position. A tape recorder has been added in the data management system for flight history and maintenance data.

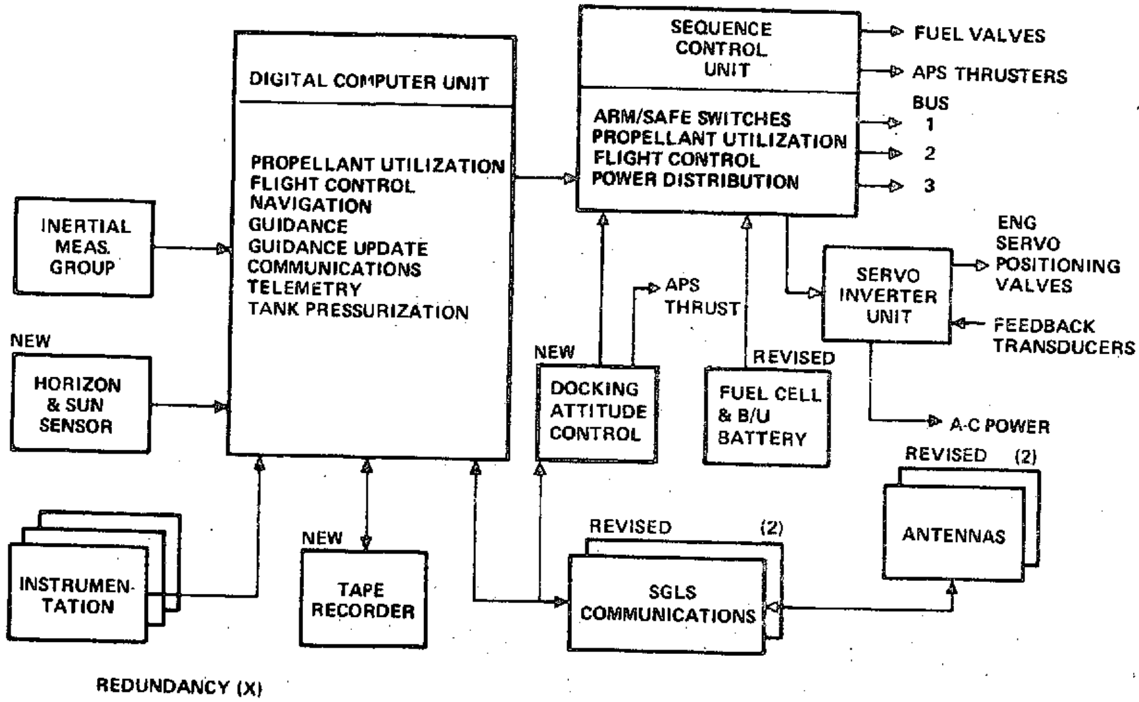


Figure 10-10. Astrionics Functional System

In the GN&C subsystem, the significant change is the inclusion of an attitude update capability to support the ground tracking position and velocity data.

Arm-safe switching has been revised from the normal Centaur to ensure man-rated safety. This requires very minor changes in the sequence control unit. A backup docking attitude control unit provides the capability to preserve stabilization control and safety during the terminal phase of the redocking sequence with the Orbiter.

Communications have been completely revised to provide redundant statusing and override capability. Dual SGLS communications equipment is included which is the same as that defined for the STSS study to satisfy the DoD requirements.

Instrumentation is similar to the standard Centaur D-1T with some minor revisions.

The baseline electrical power supply is a modified Orbiter fuel cell to be developed by NASA for Shuttle. A 36-pound (16.3-kg) backup battery with high discharge rate characteristics is provided for safety and emergency operation up to one hour.

Like Centaur D-1T, simplex electronics is utilized with limited redundancy added for safety in communications, attitude control, and electrical power and instrumentation. Astrionic equipment as described in the following sections is identical in most cases to the D-1T Centaur hardware except for the reusability/safety additions. Most of these electronic components are mounted to the equipment module ring located on the forward end of the Centaur vehicle.

10.3.1 DATA MANAGEMENT SYSTEM. The DMS uses the following equipment:

	Usage	Weight	Power
Digital Computer Unit	D-1T (Modified)	70 lb (31.75 kg)	190 watts
Remote Multiplexer	D-1T (as is)	20 lb (9.07 kg)	25 avg
Tape Recorder	Agena	12 lb (5.44 kg)	10 avg

These units provide the required data processing, reporting, and control functions for the Reusable Centaur vehicle and are designed to operate with man-rated safety. The DMS computer functions include:

a. Navigation

b. Guidance

c. Stabilization control

d. Sequencing

e. Venting/pressurization control

f. Propellant utilization

g. Communications/telemetry formatting and control

h. Instrumentation

i. Power conversion and distribution control

The Centaur digital computer unit is a stored program, random access core machine manufactured by the Teledyne Company. Main memory capacity is 24,576 instruction words of 24 bits each. Memory cycle time is nominally three microseconds. Conservative estimates of the software required show an increase of 5,000 words from D-1T.

The vehicle flight software for the DMS are:

Module	D-1T Ref. (words)	Reusable Centaur (words)
Executive	2388	2500
Attitude Control	1439	1500
Communications	1398	5000

Module	D-1T Ref. (words)	Reusable Centaur (words)
Guidance	2120	2200
Guidance Update	-	1100
Navigation	369	400
Onboard Checkout	-	500
Sequencing	2600	2800
Thrust Vector Control	1205	1200
Total	11,509	17,200

Discussions with Teledyne indicate that increasing the production Centaur DCU TDY-300 from 16 to 24k memory is a straightforward task that can either be done by additions to the basic DCU package or utilizing an additional 8k unit currently being developed for the DIGS computer. The Centaur DCU currently has excess input/output capacity beyond the requirements for Reusable Centaur. The Orbiter interfaces, guidance update, USB/SGLS communications, and the backup docking control do not necessitate the development of any new I/O. It is tentatively recommended to stretch the existing DCU because:

a. Existing TDY-300 machine design permits the addressing of 32k (24 bit) words of memory.

b. Expansion of the main memory to 24k total (16k + 8k) would be best done by stretching the present DCU case by raising the lid by 1 inch.

c. The existing internal power supply is probably adequate for the 8k word expansion.

d. Teledyne's advice is to not use separate units both to avoid external electromagnetic interference problems and for the general desirability of an integral unit.

A flight tape recorder unit is provided to store vehicle status and maintain data to be telemetered. The initial selection is a Lockheed Electronics unit which has been flown on Agena. Total data storage is 4.15×10^8 bits for a maximum tape time of 270 minutes.

Inputs to the DCU from the instrumentation transducers are passed through signal conditioners as required and are tailored by the remote multiplexer to the digital format. The signal conditioners convert the signal and measurement voltages to ranges compatible with the RMU acceptance limits.

10.3.2 GUIDANCE, NAVIGATION & CONTROL. Included in the GN&C subsystem are the subset of guidance & navigation, guidance update and flight control.

Equipment

	Usage	Weight	Power
Inertial Reference Unit (Honeywell)	D-1T (as is)	64 lb (29.03 kg)	-
System Electronic Unit	D-1T (as is)	25 lb (11.34 kg)	185 watts
Horizon Sensor (Barnes 13-161)	X-15	10 lb (4.54 kg)	5 avg
Sun Sensor (Adcole 1402)	NASA dev.	4 lb. (1.81 kg)	2 avg
Sequence Control Unit	D-1T (modify)	75 lb (34.02 kg)	17 avg
Servo Inverter Unit	D-1T (as is)	45 lb (20.41 kg)	40 avg
Docking Attitude Control	New	40 lb (18.14 kg)	5 avg

The Centaur inertial measurement group (IMG) consists of the inertial reference unit and the system electronic unit and is identical to that described for D-1S(R) in Section 3.

Continued use of the Honeywell platform is a satisfactory, low cost approach. The use of dual DIGS skewed axis Hamilton Standard platforms would be a desirable extra cost feature with superior system reliability. Both types of platforms have adequate accuracy for Reusable Centaur missions.

Alignment of the Centaur gimbaled platform may include the following:

a. Thrust acceleration alignment - during ascent

b. In-flight transfer alignment - prior to deployment

c. Update from the ground (state vector) and from on board attitude sensors during free flight.

The Orbiter prior to deployment provides state vector and attitude data from the Orbiter reference system. This input is compared in the DCU with the outputs of the Centaur inertial reference unit. The Centaur IRU is then precision torqued to the desired 3 axis matched output condition for initialization. Ground optical alignment is not a requirement. Honeywell has been studying the IRU gimbal resolver chain accuracies and torquer requirements for Convair during the past month. No mechanical change to the IRU is needed to achieve adequate gimbal angle transfer. A small amount of DCU software, about 150 words, is needed to adjust values in a direction cosine matrix to correspond to the pre-calibration of the existing resolvers. For precision torquing, some minor circuit redesign is needed to provide a constant current precision torquing supply. The D-1T Centaur has a lesser precision requirement.

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 axis, and outputs the components via the a-c D/A converters in the same way detailed for D-1S(R) in Section 3.

Guidance Update. Increased mission times for the Reusable Centaur (36 hour nominal) with the added engine burns and maneuvers over the D-1T will result in larger error dispersions for the guidance function. New equipment has been added in the form of a horizon sensor and a sun sensor to assist ground tracking inputs by 3 axis attitude update, the same as for D-1S(R), Section 3. This extra capability results in a Level III type autonomy preserving a moderate onboard computer capacity and requiring position and velocity update to be RF uplinked from the ground.

Flight Control. The DCU receives analog attitude outputs from the inertial measurements group, converts them to digital form, operates on them in accordance with the software instructions, generates output commands, and converts them to analog signals for powered flight control and digital commands to the coast phase control system. The backup attitude system used during the retrieval and docking with the Orbiter in the event of primary failure is identical to D-1S(R), Section 3.

Docking Control Backup System. Safe retrieval of the Centaur upper stage into the Orbiter bay requires the satisfaction of two principal conditions: (1) the Centaur must be statused via the RF communication link, and (2) no hazardous last minute failures should take place prior to full deactivation of the upper stage subsystems. Stabilization of the stage must be maintained until full control is achieved with the lock-on of the Orbiter manipulator arms. This seems to necessitate continued operation of the attitude control system and the APS thrusters to the point of actual contact.

Dual redundancy in the form of an emergency backup stabilization control has been designed into the flight control subsystem. This backup electronic package has been called the docking attitude control unit and consists of an attitude sensor package containing three inertial quality gyros of moderate precision. These gyros would have electronic caging to provide rate signals and would be activated before the return approach to the Orbiter. The backup attitude gyro signals would be compared to the main IRU and if the errors exceeded a certain level, with command concurrence from the Orbiter the backup would take over from the primary system and directly control the vehicle. Control logic electronics is included to bypass the primary system and permit manual override from the Orbiter as needed with visual monitoring by the mission specialist.

Intended operation is to stay on the primary flight control with the full complement of inertial reference unit, systems electronics unit, and digital computer unit until a failure occurs.

In the event of failure during the terminal situation of retrieval of the upper stage, the mission specialist on the Orbiter would be alerted and take action. He would monitor the following:

a. Visual closure between Orbiter and upper stage

b. Output differences in attitude rate (telemetered via the dual Communications) between the primary inertial platform and the control backup attitude gyros.

The switching to the backup system would be at his command. Upon switching, automatic stabilization will be possible, with manual control override if needed to provide safety and attitude adjustment.

Communications. All elements of the communications subsystem are revised from the D-1 Centaur, with the exception of the DCU involvement, so as to be compatible with the Space Shuttle upper stage requirements. The previous C-Band tracking RF system and the S-Band telemetry transmitter have been deleted and replaced by dual S-Band transponders which are DoD SGLS/SCF compatible. RF transmission/reception elements are reconfigured to operate in the assigned frequency ranges:

DoD:	Downlink	2200 to 2300 MHz
	Uplink	1750 to 1850 MHz
NASA	Downlink	2200 to 2300 MHz
	Uplink	2025 to 2120 MHz

The selected units are the same as previously presented for D-1S(R) communications in Section 3.

10.3.3 ELECTRICAL POWER/POWER CONVERSION AND DISTRIBUTION

Electrical Power. Baseline recommendation for the Centaur upper stage electrical power source is a modified Orbiter type fuel cell system which for the nominal 36-hour design mission allows significant dry weight savings and overall improvement in vehicle performance. The selected system is not dependent on one type of fuel cell, and could alternatively be implemented with a light weight version of the fuel cell being developed for USAF and NASA.

A comparative analyses with battery system alternates has been done and is reviewed in the following paragraphs.

Primary electrical power for Centaur has traditionally been furnished by on-board silver zinc batteries for relatively short missions of a few hours or less duration. The standard Centaur batteries are a high discharge design with a capacity of between 40 to 55 watt hours per pound of weight. For longer missions, this results in the carrying of excessive vehicle dry weight.

Computed electrical load requirements for the new Reusable Centaur, based on a 36-hour nominal DoD mission with 31.5 hours (maximum) in free flight outside the Orbiter, can be summarized as follows:

	Average Power (W)	Peak Coast Power (W)	Peak Burn Power (W)
(Duration)	(31.5 hr)	(variable)	(0.18 hr)
Subsystem			
Data Management	225	255	255
GN&C	254	388	367
Communications	48	107	107
Instrumentation	70	80	80
Heaters	135	486	-
Propulsion Support			
Main Engine	-	-	504
Other Support	130	196	433
	862	1512	1746

The average power drain, considering the various duty cycles, is 862 watts, or 30.8 amperes at 28 Vdc. For standard Centaur 150 A-hr, 81 pound (36.7 kg) batteries, this average drain would require in 31.5 hours a total of 970 A-hr supplied from a minimum of seven batteries at a weight of 567 pounds (257.2 kg).

With lighter weight, low discharge rate battery designs like the Eagle-Picher, type 30 used for Ascent Agena with 400 A-hr at 134 pounds (60.78 kg) each, the calculated loads could be handled by three of these type 30 batteries plus a 100 A-hr high discharge rate Centaur 66 pound (29.9 kg) battery, for a total weight of 468 pounds (212.3 kg).

Comparisons with fuel cell designs for the 36-hour mission - either modified Space Shuttle cells or the Pratt & Whitney lightweight cells under development test - show significant advantage to the use of fuel cells for the Centaur cryogenic upper stage with its hydrogen/oxygen fuel system. Fuel cell system weights for a modified Orbiter fuel cell and conventional space radiator are: (1 kilowatt design)

Fuel cell unit	63 lb (28.57 kg)
Heat exchanger	12 lb (5.44 kg)
Associated plumbing	10 lb (4.54 kg)
Space radiator	32 lb (14.51 kg)
Backup battery (23 A-hr)	36 hr (16.32 kg)
Total	153 lb (69.40 kg)

This fuel cell system weight is 315 pounds (142.9 kg) lighter than the above lightweight battery combination, and 414 pounds (187.8 kg) less than the standard Centaur battery pack. While there is a significant development cost involved, the savings in dry weight and performance improvement offset this.

Fuel consumption for any of the fuel cell candidates is estimated at about 1 pound per hour of free flight.

Fuel Cell Selection. A modified Orbiter fuel cell (either P&W or GE) has been selected for the Reusable Centaur, primarily to take advantage of the committed Orbiter fuel cell development funding and schedule. Although the accessory equipment for the 21kW Orbiter unit would have to be resized for the 1kW Reusable Centaur, the technologies involved with materials, manufacturing and quality control reduce the risk associated with a modified Orbiter unit. The P&W lightweight unit recommended in Section 3 for RLTC is a viable option, being technically closer matched to the Reusable Centaur, but lacks committed funding.

The Reusable Centaur reactant supply differs from the Orbiter's, creating different cell stack considerations. Where the Orbiter system is built around dedicated supercritical reactant storage, the Centaur unit can simply draw low pressure propellants from the main cryogenic propellant tanks, or even use boiloff gases. Either way, using propellant grade as reactants requires more venting to purge contaminants through the cell stacks. In addition, the Reusable Centaur may have up to 5% helium in the main propellant tanks. To ensure an adequate purging, an additional 10% propellant usage has been included for purging. These additional contaminates in propellant grade reactants do not affect the Reusable Centaur unit to the same extent as it would the Orbiter. Although they both have an allowable voltage variation of 2.5 Vdc, the Reusable Centaurs current demand varies only 2:1, where the Orbiter must maintain the voltage control over a 7:1 power fluctuation.

The Reusable Centaurs low pressure (16 psia) versus the Orbiter 60 psia requires more cell area to maintain the same cell current density. This, in addition to the contamination compensation, increases the number of cells provided for a 1 kW rating. The Reusable Centaurs design life of 800 hours (for about 20 flights) simplifies the fuel cell stack requirements over the Orbiter's 30,000 hour requirement. Table 10-8 is a summary of these differences.

The accessory equipment between the alternate (GE and P&W) fuel cell concepts is significantly different. This is the mechanical equipment associated with reactant feed, waste heat rejection, product water removal and contamination purging. All this equipment currently sized for the 21 kW Orbiter unit should be resized for a 1 kW

Table 10-8. Primary Requirement Differences Between Orbiter and Reusable Centaur Fuel Cell

	Orbiter	Reusable Centaur
Operating Pressure, psia	60	16
Reactant Purity	Reactant grade	Propellant grade & up to 5% He
Voltage Level, Vdc	30 ± 2.5	28 ± 2.5
Power Level	3 to 21 (7:1)	0.8 to 1.7 (2:1)
Water Rejection, Heat Rejection	Used for life support	Throw away
Life, hr	30,000	800
Accessory Sizing, kW	21	1

unit. Where the Orbiter fuel cell is interrelated to the life support system to use these byproducts, the Reusable Centaur treats them as waste, rejecting them to space, which simplifies the design of the accessory equipment.

Power Conversion & Distribution. The electrical power distribution system uses the fuel cell and backup battery to supply DC power to three separate busses so as to isolate equipments that tend to be generators of electromagnetic interference from equipment which may be sensitive. Bus 1 supplies power to the digital computer unit, sequence control unit, multiplexers, and signal conditioners. Bus 2 provides power to the communications subsystem, propellant utilization system, servo inverter unit, and instrumentation. The loads on Bus 1 and Bus 2 are primarily loads that tend to be sensitive to electromagnetic interference. Bus 3 carries loads of a switching nature such as solenoids, relays, and motors that tend to be generators of electromagnetic interference.

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

Vehicle power is provided from the Orbiter prior to deployment. The power changeover switch is activated prior to deployment and connects the internal power source to the power distribution system. The power changeover switch features a make-before-break contact arrangement to ensure uninterrupted power to the loads during the switching. The electrical 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.

The changeover switch is a revised Centaur D-1 design capable of carrying 65 amperes per pole continuously at 28 Vdc, with a voltage drop of less than 100 millivolts on each of its multiple poles. The switch also has single pole, single throw, break-before-make contacts capable of carrying 7 amperes continuous for each set with less than a 77-millivolt drop. These latter contacts are primarily for signal type circuits. Total maximum transfer time for the switch is 170 milliseconds. The switch is hermetically sealed and pressurized to one atmosphere with 95% dry nitrogen and 5% helium.

Electrical System Harnessing. Generally the harnessing consists of H-film insulated wires and is physically separated into three basic classifications: (1) wires that may be sensitive to electromagnetic interference, (2) wires that connect equipment that tends to generate electromagnetic interference, and (3) wires that do not fall into any of the above categories. The third group is routed between the other two groups. This routing provides some isolation and minimizes interaction between the various electronic systems.

Estimated weight for the electrical harnessing and connectors for the vehicle is 160 pounds (72.58 kg).

10.4 MISSION PERFORMANCE

This section summarizes the mission performance of the Short Reusable Centaur (RC) Space Tug. This configuration has been defined to accommodate payloads up to 32 feet (9.75 m) in length. The reusable Centaur is 28 feet (8.53 m) in length (installed in the Orbiter). A shorter RC, 25 feet (7.62 m) can be configured by removing a 3-foot (0.91 m) section from the LH_2 tank to accommodate longer payloads.

Since much of Section 4.0 applies to the RC, only additional material specific to the RC is presented in this section.

10.4.1 MISSION REQUIREMENTS. The driving mission requirements for the short RC is to accommodate payloads of 32 feet (9.75 m) in the 60-foot (18.29 m) long Orbiter cargo bay. NASA missions are defined in the NASA-MSFC "NASA/NON-NASA Mission Model" of September 1973 and DoD missions are defined in the "DoD Space Mission Model", of 16 August 1973, Revision 1. Missions for the four year 1980-1983 period are to be accomplished. Figure 10-11 summarizes payload characteristics from these models.

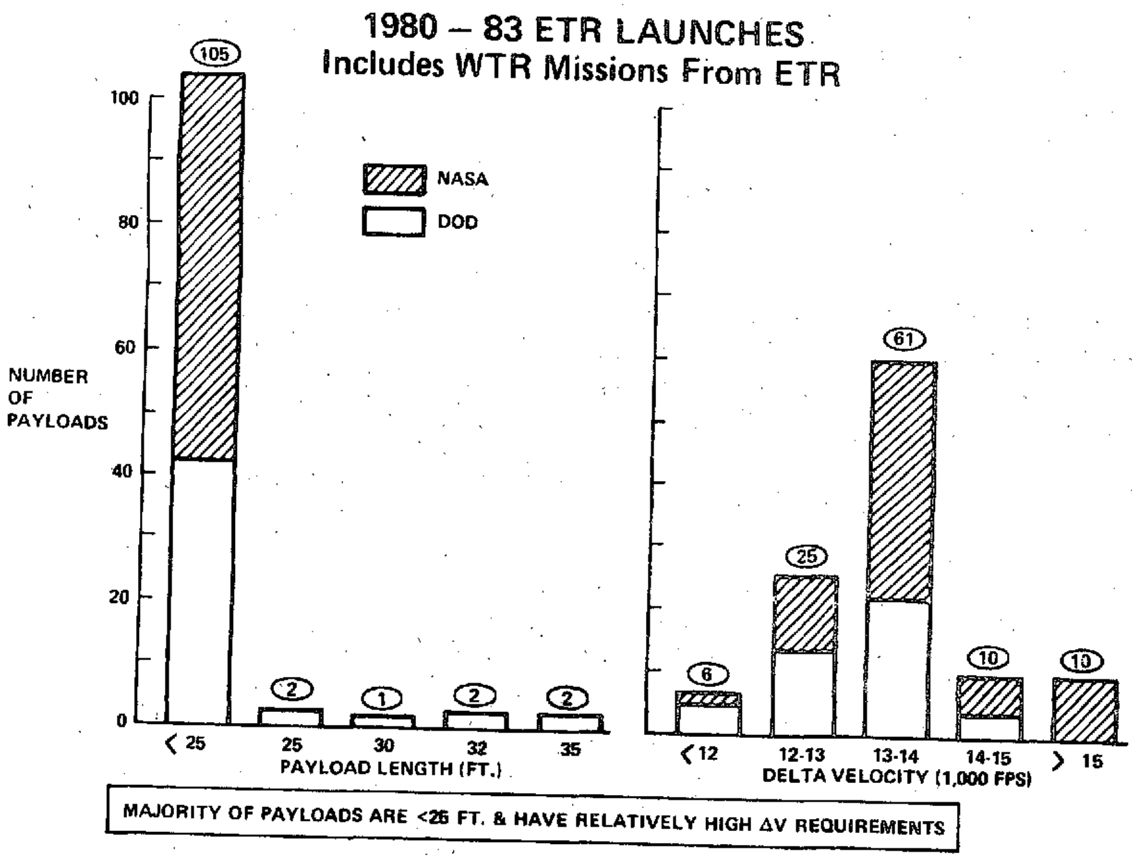


Figure 10-11. Initial Upper Stage Payload Characteristics

The 28 foot (8.53 m) RC is to be capable of accomplishing all NASA missions with the Centaur reusable except for a few extremely high energy planetary missions where the RC may be expended with the payload. The 28 foot (8.53 m) RC will also accomplish all DoD missions in the reusable mode, except for those payloads requiring the shorter configuration. A 22 or 25 foot (6.7 to 7.62 m) RC is reusable when delivering these payloads. To simplify the study, General Dynamics assumed that WTR is not activated for RC launches during the 1980-83 time period. The RC is capable of deploying polar and sun synchronous payloads when launched from ETR.

Payloads have "walking" capability which provides the payloads with the ability to place themselves within an orbit; the RC is only required to deliver payloads to the appropriate orbit. The Reusable Centaur is capable of multiple payload delivery with one, two, or three payloads deployed during a single flight. Only single payloads are deployed on planetary missions.

10.4.2 WEIGHT AND MASS PROPERTIES. Reusable Centaur vehicle weight summary is shown in Table 10-9. The weight summary is based on the following vehicle configuration conditions:

Structure. The astrionics module/forward adapter and the aft adapter have been redesigned using titanium skin-stringer design.

The fuel tank and the LO_2 tank were sized for 46,000 pounds of propellants at 5.8:1 engine mixture ratio. The intermediate bulkhead, and the aft bulkhead and thrust structure are existing manufacture for the Centaur D-1T configuration.

The vehicle length installed in the payload bay is 28 feet.

Propulsion. The main engine system including engine actuation and support is basic existing Centaur D-1T design.

The ACPS system is a modified existing H_2O_2 system sized to mission requirements. Two H_2O_2 tanks are included, as well as the four thruster modules with a forward thruster added. The pressurization is largely D-1T Centaur components, including two large ambient storage helium bottles, with revisions made to the system for plumbing and control. D-1T H_2O_2 boost pump weights are used in the feed system weights. Other components are modified D-1T design.

The fill and drain system includes two LH_2 dump valves and two LO_2 dump valves.

The vent system includes the zero-g vent valves and associated plumbing and hardware.

Table 10-9. Weight Statement

CONCEPT:	MISSION:	CONFIG. NO.
Reusable Centaur 28'	Payload Delivery to Synchronous Orbit	RC-1

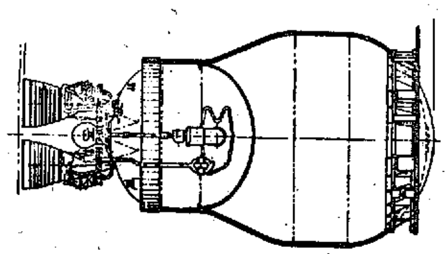
	Lbs	Kg
Structures	(1723)	(781.6)
Body Structure	395	179.2
Fuel Tank & Supports	773	350.6
Oxidizer Tank & Supports	387	175.5
Thrust Structure	37	16.8
Equipment Mounting Structures	61	27.7
Meteoroid Structure	—	—
Interface for P/L & Shuttle	50	22.7
Umbilicals	20	9.1
Propulsion	(1717)	(778.8)
Main Engine (ME) Assembly	580	263.1
ME Actuation & Support	154	69.9
ACPS & Support	195	88.4
Pressurization	265	120.2
Fill, Vent & Drain (Incl. Feed)	459	208.2
Purge System	—	—
Propellant Management	64	29.0
Thermal Control	(488)	(221.3)
Fuel Tank Insulation	182	82.6
Oxidizer Tank Insulation	158	71.7
Purge System	110	49.9
Radiators (Inc. Fluid Loop, Heaters, Coatings)	38	17.2
Avionics	(787)	(357.0)
Rendezvous & Docking	—	—
Data Management	122	55.3
Flight Controls	134	60.8
Guidance & Navigation	89	40.4
Guidance Update	14	6.3
Power	121	54.9
Power Conversion & Distribution	160	72.6
Instrumentation	90	40.8
Communication	57	25.9
STAGE DRY WEIGHT	(4715)	(2138.7)
Contingency	280	127.0
TOTAL DRY WEIGHT	(4995)	(2265.7)
NON-USABLE FLUIDS	(665)	(301.7)
Trapped Propellants - Main	122	55.4
Trapped Propellants - ACPS	10	4.5
Trapped Gases	498	225.9
Trapped Helium	10	4.5
Propellant Leakage	—	—
Propellant Reserves - Main	**	**
Propellant Reserves - ACPS	—	—
Propellant Utilization	24	10.9
Trapped Water	1	.5
BURNOUT WEIGHT	(5660)	(2567.4)

	Lbs	Kg
NON-IMPULSIVE EXPENDABLES	(807)	(366.1)
ME Childown	188	85.3
ME Leakage	52	23.6
ME Shutdown	60	27.2
Pressurants - Main	5	2.3
Pressurants - ACPS	—	—
Fuel Cell Reactants	34	15.4
Fuel Line Chilldown	—	—
Gas Generator	—	—
Thermal Control Fluids	—	—
Boiloff Vented	468	212.3
PROPELLANTS	(46112)	(20916.4)
*Main Engine	(45640)	(20702.3)
Main Propellant - Fuel	6712	3044.6
Main Propellant - Oxidizer	38928	17657.7
Attitude Control Propellants	472	214.1
*FIRST IGNITION WEIGHT	(52579)	(23849.9)
SHUTTLE INTERFACE ACCOMM	(2571)	(1166.2)
Adapter Structure	(816)	(370.1)
Supports	533	241.7
Attachment Fittings	104	47.2
Deployment Mechanism	179	81.2
Propellant Lines Umbilicals, Tankage	(1395)	(632.8)
Fill, Drain, Dump	207	93.9
Dump Pressurization System	1188	538.9
Avionics Interface	(360)	(163.3)
Shuttle/Payload in Bay	270	122.5
Mission Specialist Station	90	40.8
Bay Purge	—	—
Gases	—	—
Bottles	—	—
Payload Auxiliary Support	—	—
STAGE PAYLOAD	(4581)	(2077.9)
Payload	4581	2077.9
Payload Adapter	—	—
AUXILIARY STAGE	—	—
TOTAL WEIGHT IN SHUTTLE	(59731)	(27094.0)

$$\text{*Mass Fraction} = \frac{\text{Main Engine Propellants}}{\text{First Ignition Weight}} = .868$$

** 135# included in main propellants

CONFIGURATION SKETCH:



REMARKS:
1) Mission - 5 burn 30 hours
2) Engines - RL10A-3-3
3) ACPS - H_2O_2 (2)
4) Boost Pumps - H_2O_2
5) Propellant Tanks - 46000# Cap. Stainless Steel
6) Forward adapter / Aft Adapter } TI Skin Stringer
7) Insulation - Goldized Kapton Dimplar
8) Electric power - Fuel cell
9) Helium - Ambient Supply
10) Avionics - Modified D-1S Centaur
11) Contingency - 10% New Design, 2% Existing

Propellant management includes the propellant utilization system and the propellant loading system as presently used on the D-1T Centaur.

Thermal Control System. Insulation of the propellant tanks is accomplished by using two goldized Kapton dimplar blankets each 3/4 inch thick and three radiation shields, the inner one of which serves as a purge bag. The weights include the insulation, purge bag, purge system, and all supports and seals required. A space radiator for the fuel cell is also included.

Astrionics. Additional computer capability has been included and a tape recorder has been added for data management.

An IRU unit for vehicle guidance and a solid inverter for flight control has been included. Sensor equipment has been added for guidance update navigation and control.

A revised antenna and component parts have been added as part of a standard communication system, including DoD encryption devices.

Electric power is supplied by a modified Orbiter type fuel cell including heat exchanger and plumbing, plus a backup battery.

A typical electrical distribution system and an instrumentation system have been included.

Contingency. The contingency weight for this vehicle has been defined as 10% of the vehicle dry weight for new items and 2% for existing items. Contingency is itemized in Table 10-10.

Fluids. All propellant weights are based on 5 burn, 30 hour solo mission requirements. Residuals and non-impulsive propellants weights are calculated values.

Shuttle Interface. The weights in this section included all required adapters, attachment fittings, deployment mechanisms required to release and return the RC vehicle to the Shuttle. Also included is a dump pressurization system and its associated plumbing and controls.

All vent and dumping interface requirements are included.

A weight allowance of 360 pounds (163.3 kg) has been included for the astrionics interface.

A comparison to the existing D-1T Centaur vehicle is presented in Table 10-11. The necessary configuration changes are listed for all the subsystems. Vehicle center of gravity data is presented on Figure 10-12. All Shuttle landing condition points fall well within the stated tolerance. (Inflight dump provisions preclude the possibility of reentry and landing fully loaded.)

Table 10-10. Reusable Centaur Weight Contingency
(28 ft Version, 4715 lb Stage Dry Wt.)

Subsystem	Exist	New or Mod	Weight Allowance (lb)	
Structure				127
Body (Fwd-Aft Adapters)		X	35	
Fuel Tank		X	74	
Oxidizer Tank	X		10	
Thrust Structure	X		0	
Equipment (Astrionics) Mounting	X		1	
Interface for Payload & Orbiter		X	5	
Umbilicals		X	2	
Propulsion				77
Main Engines	X		12	
M. E. Support (TVC, etc)	X		3	
Fill, Vent, Drain, Dump (although Boost Pumps same)		X	46	
ACPS	X		5	
Pressurization	X		5	
Propellant Management		X	6	
Thermal Control				30
Fuel Tank Insulation		X	12	
Oxid. Tank Insul incl. blkhds	X		3	
Purge System		X	11	
Radiators, incl. Fuel Cell Rad.		X	4	
Astrionics				46
Data Management	X		2	
with Increased Memory		X	3	
Flight Controls	X		4	
with Backup Package		X	4	
Guidance, Navigation + Update		X	10	
Power Supply (Fuel Cell with H. X.)		X	12	
Power Distribution	X		3	
Instrumentation	X		2	
Communications		X	6	
Total Contingency				280

Table 10-11. Reusable Centaur Weight Change Breakdown

System	D-1T Centaur (lb)	Reason for Change	Δlb	Reusable Centaur (lb)
Structure	1413		(+310)	1723
		Redesign forward adapter/ equipment module	- 98	
		Remove P/L truss adapter	- 74	
		Add new aft adapter	+ 94	
		Remove I/S pyro separation system	- 22	
		Redesign fuel tank	+310	
		Redesign LO_2 tank	+ 70	
		Redesign interface & umbilical system	+ 30	
Propulsion	1500		(+217)	1717
		Add forward thrusters	+ 10	
		Add helium bottle	+ 81	
		Add isolation valves	+ 30	
		Revise fill & drain	+ 29	
		Add zero-g vents	+ 57	
		Add redundant LO_2 vent valve	+ 10	
Thermal Control	354		(+134)	488
		Add fuel tank insulation blankets	+ 87	
		Add LO_2 tank insulation blankets	+ 13	
		Add fuel cell radiator	+ 38	
		Misc insulation changes	- 4	
Astrionics	761		(+ 26)	787
		Add 8k computer memory	+ 25	
		Add tape recorder	+ 12	
		Remove RSS system	- 53	
		Revise multiplexer	+ 6	
		Add backup docking attitude control	+ 40	
		Remove S-Band	- 31	
		Remove C-Band	- 15	
		Revise RF uplink	- 4	
		Add guid. update sensors	+ 14	
		Add fuel cell system	+121	

Table 10-11. Reusable Centaur Weight Change Breakdown (Continued)

System	D-1T Centaur (lb)	Reason for Change	Δlb	Reusable Centaur (lb)
Astrionics (Continued)		Remove battery	- 81	
		Simplify electrical wiring	- 40	
		Add SGLS communications	+ 31	
		Add cryptor equipment	+ 22	
		Misc. simplifications	- 21	

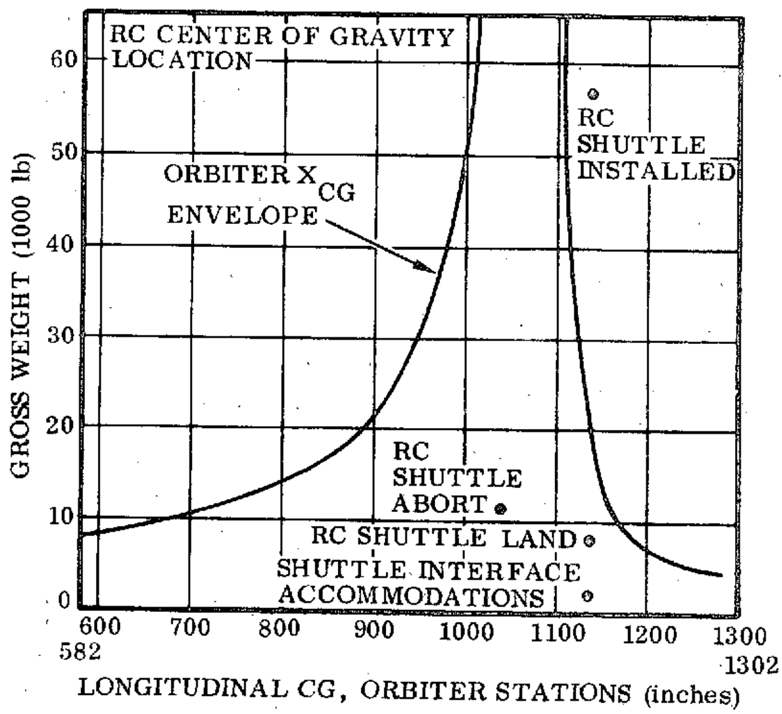


Figure 10-12. Reusable Centaur Center of Gravity Location

10.4.3 PERFORMANCE ANALYSIS. Performance calculations were based on the propulsion characteristics given in Table 10-12.

A kick stage is required for some high energy NASA planetary missions. The kick stages used with the RC are based on the Thiokol TE-M-364-4 solid rocket motor. Kick stage characteristics are described in Section 4.3.1.

Table 10-12. Reusable Centaur Propulsion Characteristics

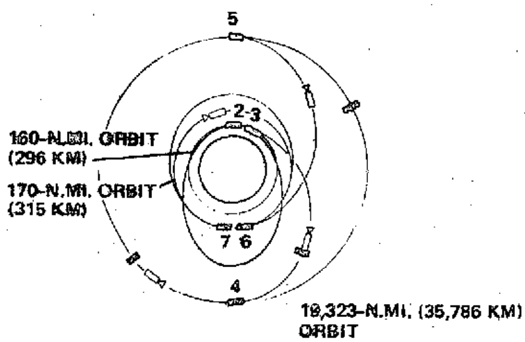
Main Engines (2)	RL10A-3-3
Thrust (each engine)	15,000 lb (66,723 N)
Specific Impulse	439.8 sec (4313 Nsec/kg)
Mixture Ratio (O_2/H_2)	5.8:1
Main Engine Propellant (usable)	45,640 lb (20,702 kg)
ACPS Propellant	H_2O_2
ACPS Specific Impulse (steady)	155 sec (1520 Nsec/kg)
Flight Performance Reserve	1% ea ΔV
Stage Length (installed)	28 ft (8.53 m)

The baseline mission defining system design requirements is a synchronous equatorial deployment mission by a Reusable Centaur without the use of velocity packages (kick stages) or other auxiliary hardware. The mission profile is shown in Figure 10-13 along with the sequence of major events, time durations, and the velocity budget.

This mission requires five main engine burns after the Shuttle delivers the RC to a 160 n. mi. (296 km) circular orbit inclined at 28.5 degrees. The first burn occurs at 4.17 hours after liftoff and inserts the RC on the transfer trajectory to synchronous altitude. A plane change of 2.2 degrees is combined with this burn with the remainder of the necessary 28.5 degrees of plane change combined with the second, mission orbit insertion, burn which occurs following a coast period of 5.17 hours. A period of 11.43 hours is spent in the synchronous equatorial orbit deploying the payload and coasting to the next opportunity to return. On the return trip to a 170 n. mi. (315 km) orbit a phasing orbit is used to bring the RC into the proper alignment with the Shuttle. The fifth, and final burn, brings the RC to the 170 n. mi. (315 km) circular orbit for rendezvous. Approximately 30 hours has elapsed from RC deployment to recovery by the Shuttle and 46.6 hours from Shuttle liftoff to landing.

The velocity budgets for both the main and the attitude control engines are shown in Figure 10-13. The attitude control engine velocity budget contains 20 fps (6.10 m/s) for midcourse correction on the outboard trip, 10 fps (3.05 m/s) for separation from the deployed payload, and a 20 fps (6.10 m/s) midcourse correction on the return trip. On the Space Tug System Studies a midcourse correction delta-V of 50 fps (15.29 m/s) has been used; the 20 fps (6.10 m/s) value is consistent with the ground rules established for the DoD interim Tug studies.

SEQ. NO.	EVENT	TIME COAST (Hr.)	BURN (MIN.)	TOTAL (Hr.)
	LIFTOFF			0
1	ORBITER BURNOUT			
2	CIRCULARIZE (160 N.MI.)			1.63
2-3	COAST & DEPLOY TUG	2.54		
3	TRANSFER ORBIT INSERT.		5.7	4.27
3-4	COAST	5.17		
4	MISSION ORBIT INSERT.		2.4	9.48
4-5	DEPLOY P/L & COAST	11.43		
5	TRANSFER ORBIT INSERT.		1.3	20.93
5-6	COAST	5.27		
6	PHASING ORBIT INSERT		0.5	26.21
6-7	COAST	2.70		
7	CIRCULARIZE (170 N.MI.)		0.5	29.60
7-8	RENDEZVOUS & COAST	16.90		
8	DEORBIT			45.90
9	TOUCHDOWN			46.60



VELOCITY BUDGET

EVENT NO.	ΔV MAIN-FPS (m/s)		ACS-FPS (m/s)	
3	8080	(2463)	–	–
3-4	–	–	20	(6.10)
4	5849	(1783)	–	
4-5	–	–	10	(3.05)
5	5844	(1781)	–	–
5-6	–	–	20	(6.10)
6	3870	(1180)	–	–
7	4159	(1268)	–	–

Figure 10-13. Baseline Reusable Centaur Mission

Payload deployment capability to synchronous equatorial orbits is given in Table 10-13 for the 28 ft (8.53 m) RC. The 22 ft (6.70 m) RC is configured to deliver long payloads to orbits less energetic than synchronous equatorial; it is capable of delivering an 18,000 lb. (8165 kg) payload to a 12 hour orbit (the driving mission) in the reusable mode.

Table 10-13. 28 ft (8.53 m) RC Synchronous Equatorial Payload Capability

Flight Mode	Payload Capability (lb)	(kg)
Reusable RC	4,581	(2078)
Reusable RC - Expended Kick Stage	9,400	(4264)
Expended RC	17,000	(7711)
Reusable RC - Sortie	1,240	(562)

Payload deployment capability versus delta velocity above a 160 n.mi. (296 km) circular orbit is shown in Figure 10-14 for the flight modes applicable to the 28-foot Reusable Centaur. These curves are representative of a five-burn profile for the recovered RC missions, and a three-burn profile for the expended RC missions. Gravity losses are included in the ΔV. The Shuttle Orbiter parking orbit altitude is 160 n.mi. (296 km) at RC deployment, and 170 n.mi. (315 km) at RC recovery. The Shuttle is launched from ETR with an inclination of up to 57 degrees. Synchronous equatorial missions are launched due east from ETR.

Performance sensitivities (payload partials) for the reference synchronous equatorial mission are given in Table 10-14.

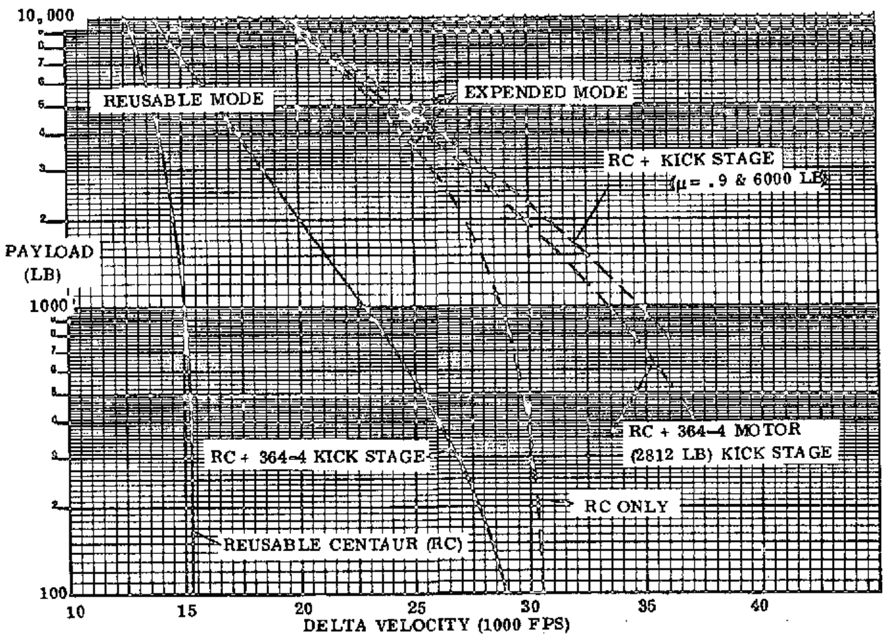


Figure 10-14. Payload Weight vs. Delta Velocity

Table 10-14. Reusable Centaur Synchronous Equatorial Performance Sensitivities

Variable	Payload	
	Partial	Value
Burnout Weight	$\partial W_{PL}/\partial W_{BO}$	-3.7 lb/lb (-3.7 kg/kg)
Main Engine Propellant	$\partial W_{PL}/\partial W_{P}$	+0.6 lb/lb (+0.6 kg/kg)
Trapped Main Engine Propellant	$\partial W_{PL}/\partial W_{(trapped)}$	-4.3 lb/lb -4.3 kg/kg
I_{SP}	$\partial W_{PL}/\partial W_{I_{sp}}$	+132 lb/sec (+6.106 kg/Nsec/kg)

10.4.4 MISSION CAPTURE. Reusable Centaur mission capture over the 1980-83 period is presented in Figure 10-15 for a combined NASA/DoD mission model. This model has been synthesized from payload descriptions and traffic described in the NASA-MSFC "NASA/Non-NASA Mission Model" of September 1973 (current design payloads) and the "DoD Space Mission Model" of 16 August 1973, Revision 1. Polar payloads normally launched by a Tug from WTR have been launched from ETR. No Shuttle flight buildup constraint has been superimposed on the mission capture results.

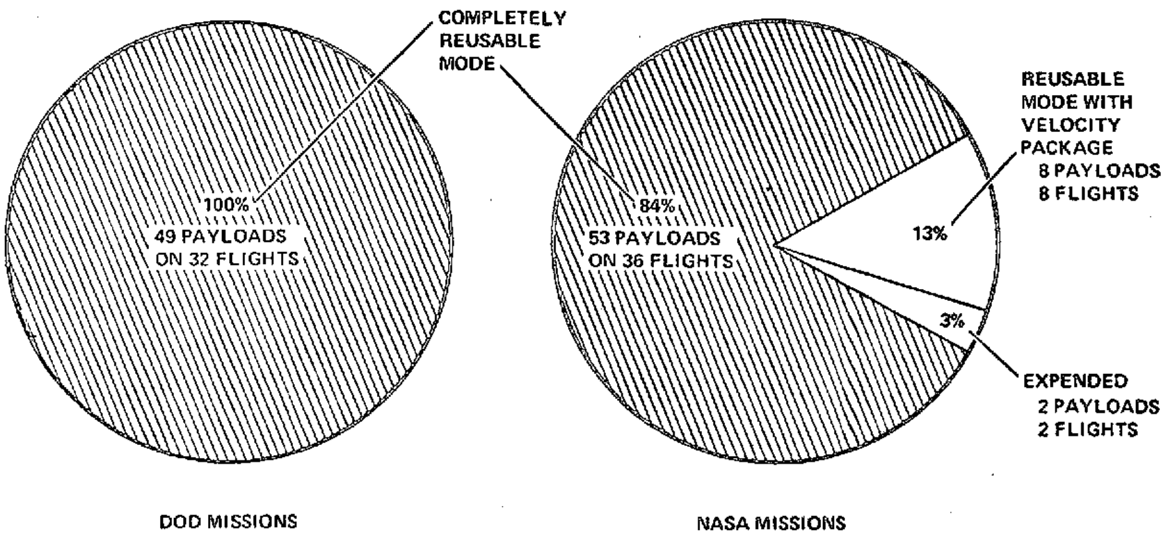
Guidelines governing permissible payload combinations are an extrapolation of those provided by the Government for the Space Tug Systems study. Two of the most significant are: only specific, and like, DoD payloads may be combined; and no more than three payloads may be deployed on a single RC flight. Table 10-15 shows that a total of 78 RC flights are required to deploy 112 payloads over the four-year period from 1980 through 1983. On 41 flights a single payload is deployed, two payloads are deployed on 20 flights, and three payloads are deployed on seven RC flights. In addition, eight velocity packages and 2 RCs are expended. An average of 1.44 payloads is deployed per RC flight.

Table 10-15. RC Mission Model Flight Summary

Flight Mode	Year 80	81	82	83	Total
Deployment					
1 Payload with RC Return	9	11	10	11	41
2 Payloads with RC Return	3	7	6	4	20
3 Payloads with RC Return	2	2	2	1	7
RC Return & Expended Velocity Package	1	6	1	0	8
Expended RC	1	1	0	0	2
Total	—	—	—	—	—
RC Flights	16	27	19	16	78
Payloads Deployed	23	38	29	22	112

The Reusable Centaur captures 100% of the OOS missions defined for the 1980-84 period. Ninety-one percent of these payloads can be placed in orbit with the RC in a completely reusable mode (no kick stages, drop tanks or other auxiliary hardware items are required). Polar and sun synchronous payloads normally launched from WTR can be deployed from ETR by the RC. Since WTR will probably not be activated for Shuttle/Tug operations until after 1984, the RC's excellent performance capability is used to launch these payloads from ETR.

Figure 10-15 shows that all DoD missions are accomplished with the RC completely reusable. Shuttle/RC flights (and hence operations costs) are reduced by using the



91% OF PAYLOADS CAN BE DELIVERED TO THEIR RESPECTIVE ORBITS IN THE COMPLETELY REUSABLE MODE WITHOUT EXPENDING KICK STAGE OR DROP TANK

Figure 10-15. Reusable Centaur Mission Capture

multiple payload delivery capability of the RC to deploy a total of 49 payloads with 32 Shuttle/RC flights.

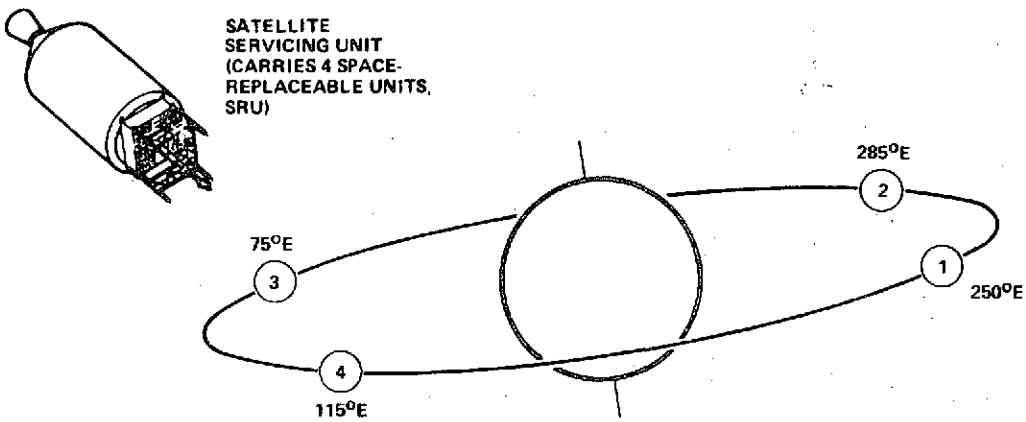
Eighty-four percent or 53 of the 63 NASA payloads are delivered with the RC completely reusable. By using the multiple payload capability of RC only 36 Shuttle/RC flights are required. Eight near-planet and Sun-synchronous payloads are delivered by a reusable RC with a kick stage, using an existing Thiokol 364-4 motor, expended. Two RCs are expended to deploy two far-plant payloads requiring extremely high energy levels.

10.4.5 SERVICE MISSION. The Reusable Centaur can service two spacecraft in the reusable flight mode based on the service mission requirements defined in Figure 10-16. Necessary design changes include:

a. Increased ACS propellant to provide 50 fps (15.2 m/s) delta velocity per satellite serviced

b. Addition of rendezvous and docking equipment.

With these changes and the service mission ground rules given in Figure 10-16, the Reusable Centaur can carry a 500 pound (227 kg) service unit to orbit along with 900 pounds (408 kg) of space-replaceable units, 450 lb (204 kg) per satellite, and return to the Orbiter with the service unit.



- REQUIREMENTS
  - SERVICE 4 SATELLITES/MISSION
  - SERVICE TIME: 2 HR/SATELLITE
  - SERVICE UNIT WT = 500 LB; NOT INCLUDING SRUs
  - RENDEZVOUS AND DOCK WITH SATELLITE
- TWO SERVICE MODES
  - I EXPEND ALL REPLACED SPACE REPLACEABLE UNIT (SRU)
  - II RETURN ALL RETRIEVED (SRU); EQUAL PAYLOAD UP AND DOWN

Figure 10-16. Service Mission

10.5 FLIGHT OPERATIONS

The discussion in Section 5 is applicable to a Reusable Centaur including the subjects of autonomy, communications, and inflight abort. The Reusable Centaur program is considered a national program with provisions for control centers at both Sunnyvale and JSC or MSFC, ground software for both, and development of both DoD and NASA airborne communications hardware.

10.6 SAFETY AND RELIABILITY

All the safety features for D-1S(R) discussed in Section 6 are incorporated in the Reusable Centaur weights including:

a. Rapid on-pad fill and drain.

b. Propellant leak containment with purge bags vented overboard, isolation valves, etc.

c. Inflight propellant dump system with oversized lines, redundant valves and high capacity helium supply for outflow pressurization.

d. Tank pressure control with redundant pressurization, sensing and venting.

e. Twin intermediate bulkheads with vacuum cavity between.

f. Auxiliary propulsion isolation valves, redundant set of thrusters, and overboard peroxide dump provisions.

g. Dual communication system.

h. Battery to backup fuel cell.

i. Backup docking control system.

j. Dedicated computer integrated with Caution & Warning panel in Orbiter.

The reliability of the Reusable Centaur was determined for a baseline geosynchronous payload placement mission. The reliability measurement starts with deployment from the payload bay and ends with Tug recovery by the Orbiter. The failure rate data used in the analysis was taken from Centaur historical data, MSFC data (Saturn Component Failure Rates and Failure Rate Modifiers), and Pratt & Whitney RL10 engine data. Failure rate modifiers were used to adjust the failure rates to reflect the influence of the environment during various mission phases. The failure rate modifiers are:

Mission Phase	Failure Rate Times
Centaur Engine Burn	20
Centaur Coast	1

The resulting Centaur reliability, including the reliabilities of each of the major subsystems, is summarized in Table 10-16. As indicated in this table, the Centaur reliability for a geosynchronous mission is approximately 0.970. The principal candidate for reliability improvement lies in the astrionics subsystem. This is essentially a single string system and, should the need arise to increase the Centaur reliability for longer duration missions, the reliability improvements of this subsystem can be achieved with relative ease. The addition of a back-up guidance package, for example, would increase the vehicle reliability to 0.978.

A significant reliability plus for the Centaur emanates from its performance capability. This performance is sufficient to allow payload placement for any of the presently anticipated DoD missions without having to resort to a kick stage. Consequently, the reliability problems associated with kick stage separation, stabilization, ignition, guidance etc. are absent.

10.7 GROUND OPERATIONS

For a Shuttle reusable upper stage vehicle, ground operations covers the time span from initial delivery of the new vehicle at the launch range through test and checkout, spacecraft mating, Orbiter mating and checkout, pre-launch operations, launch, landing, safing, removal from the Orbiter, maintenance and refurbishment, and storage or preparation for the next mission. In planning ground operations for the Reusable Centaur vehicles, STS ground rules were followed to assure upper stage compatibility with Shuttle operations and turnaround timelines. Planning includes transportation, maintenance, refurbishment, checkout, GSE, and facility requirements, utilization of existing resources, and logistics, integrated to assure smooth operational flow and low cost.

Table 10-16. Reusable Centaur Reliability

Subsystem	Reliability	Redundancy Level	Design Mission Time (Hrs)	Remarks
Structure	0.9999	---	28.5	Based on structural factors of safety design into structure
Propulsion	0.9880			
- Main Engine System	0.9900	Component	0.18	Redundant valves
- Thrust Vector Control	0.9996		0.18	
- Press., Vent, Fill, Drain	0.9988	Component	28.5	Redundant valves
- PU, Purge	0.9996	Component	0.18	
- ACPS	0.9999	Component	28.5	Redundant engines, valves
Astrionics	0.9819			
- Data Management	0.9923		28.5	
- Guidance & Navigation	0.9919		28.5	
- Flight Control	0.9989		28.5	
- Guidance Update	0.9998		28.5	
- Communications	0.9999	Component	28.5	Redundant transceiver, antenna
- Electrical Power & Distribution	0.9989		28.5	
Interface Systems	0.9999		5	Dual and triple He purge seals on all fluid lines, dual and triple electrical disconnects Note: Reliability for entire mission; lift-off to landing ≈ .963
Vehicle Reliability	0.9701			Reliability for Centaur while out of payload bay

10.7.1 TEST & CHECKOUT CONCEPT. Ground operations plans require that all payload preparations, checkout, and testing be complete prior to installation in the Orbiter, to the maximum extent possible. Only mandatory Orbiter/payload composite testing and interface validation can be allowed in the installation and checkout periods at the MCF and launch pad. To satisfy these objectives, test and checkout philosophy for Centaur is planned on the same basis as that for the Tug, i.e.:

a. Test and checkout for first flight of each vehicle based on complete test and checkout to attain maximum reliability, maximum probability of mission success, and assurance of vehicle return - the "Test & Retest" philosophy that has achieved proven results on existing fly-right-the-first-time launch vehicles such as Centaur D-1.

b. Test and checkout for remaining flights of each vehicle based on "Condition Monitored Maintenance with Pre-Flight Testing" (CMM_{PF}), or "Preceding flight data is the best test and checkout for the next flight."

When first delivered to the Cape, each Centaur vehicle will undergo complete mechanical, fluid, and electrical systems checkout at ETR Complex 36 prior to first flight in an Orbiter. The vehicle, in its deployment adapter, will be installed in a simulated Orbiter payload bay at complex 36 to verify Orbiter interfaces and disconnect functions, then tested. The deployment adapter includes all Centaur in-flight support systems and equipment, deployment and separation mechanisms, and separation disconnects. Mated to Centaur, the deployment adapter allows complete testing of these in-flight functions on the ground, before the unit is installed in the Orbiter. The deployment adapter remains in the Orbiter payload bay after Centaur deployment, is removed after Orbiter landing, and is recycled to the Centaur handling facility (Hangar J) at KSC for refurbishment and preparation for another flight.

Upon completion of test and checkout at complex 36, Centaur will be transferred to a designated spacecraft preparation and mating facility for spacecraft mating. All interface connections and combined Centaur/spacecraft systems will be checked out here, prior to moving to the Orbiter mating area, to assure that the Shuttle payload is in all respects ready for installation in the Orbiter, with minimal possibility of interference with the Shuttle turnaround schedule.

10.7.2 GROUND OPERATIONS

10.7.2.1 Transportation. For ground handling, Centaur will be moved on its own transport pallet only, supported by simulated Orbiter attach points at the forward adapter and by either a deployment adapter or shipping adapter aft, mating with simulated Orbiter attach points on the transport pallet. The transport pallet can be moved locally on its own detachable wheels, or on a transport trailer.

For air transport, the reusable Centaur will be air shipped in the NASA Super Guppy or equivalent, on its transport pallet.

10.7.2.2 Receiving and Inspection. All production vehicles will be shipped to the Cape for pre-first-flight testing before assignment to KSC or WTR for Shuttle operations. After landing and unloading at the CKAFS skid strip at Cape Kennedy (Figure 7-2), Centaur will be transported to Hangar J for receiving-inspection. Centaur and its adapters will be given a brief electrical inspection, and a visual check for transportation damage, corrosion and overall completeness, then transferred to Complex 36 for complete systems checkout.

10.7.2.3 Pre-First Flight Checkout and Testing. Prior to installation, tanking, and launch in an Orbiter for the first time, each Centaur upper stage vehicle will undergo pre-first flight test and checkout at existing Centaur Launch Complex 36A at the Cape.

Complex 36A will be modified to provide a dual capability: launch of Atlas/Centaur vehicles, and cryo tanking test and checkout of the Reusable Centaur for Shuttle. A payload bay simulator will be provided at Complex 36A, mounting directly on the existing launcher, with service lines tapped off existing service lines at the base of the umbilical mast to permit propellant loading of Centaur within the payload bay simulator, Figure 10-17.

The simulator will duplicate the Orbiter cargo bay with respect to payload attach points, service line connections, and thermal characteristics of the payload bay wall. It will be fabricated in two sections: the lower half which can be used alone for Centaur tanking tests to check out the vehicle itself, and the upper or spacecraft half which can be mated to the lower half to run complete thermal tests of the Centaur/Spacecraft payload when required.

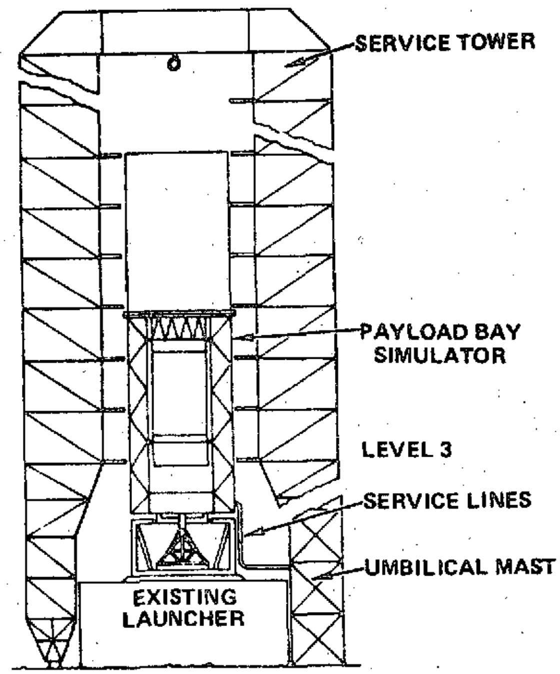


Figure 10-17. Centaur Checkout and Cryogenic Test Facility, KSC Complex 36

After installation in the simulator, Centaur and its adapter will undergo initial fit and function checks of all systems. Functional testing will include reaction control system engine firing, and Centaur cryo tanking. This will be the first and only time that the propulsion systems of each new vehicle will be exposed to propellants and proofed as complete systems, before final loading and launch at Shuttle Complex 39. Individual subsystems and components (main engines, vent valves, etc.,) will have been exposed to cryogenic temperatures and functional tests, however, during individual acceptance tests (IAT), prior to vehicle final assembly at San Diego.

Centaur vehicle testing in the payload bay simulator at Complex 36 will be done on the first flight article with two objectives: vehicle checkout, and payload bay thermal testing. Follow-on production vehicles will be tested with respect to Centaur flight readiness only, unless mission-peculiar spacecraft tests are also required. Centaur testing in the payload bay simulator is summarized for the first flight article, and for follow-on production vehicles, as follows:

First Flight Article. Centaur will be installed in the simulator with a dummy spacecraft. Temperature sensors will be mounted on the simulator walls, payload, and critical points on Centaur. A normal tanking test will be run, followed by ground hold thermal tests with payload bay GN_2 conditioning introduced, duplicating the Orbiter environmental conditioning system with respect to temperature, humidity, flow rates and distribution. Objectives of the test will be:

a. Vehicle Checkout.

1. Compatibility checkout of Centaur/Payload Bay interfaces.
2. Combined electrical readiness test (CERT).
3. Terminal countdown demonstration (TCD) of all Centaur system functions under tanked conditions.
4. Flight events demonstration (FED).
5. Determination of Centaur ground hold propellant boiloff rates.
6. Emergency propellant dump tests.

b. Thermal Environment Test (with no spacecraft heat load).

1. Temperature patterns within the payload bay.
2. Payload bay wall temperatures.
3. Centaur temperatures (equipment module, thrust section and deployment adapter, electronics, peroxide, hydraulic systems).
4. Payload temperatures.
5. Need for modification of planned environmental (payload bay) conditioning parameters.

As a result of the b.5 test objective, further thermal testing can be run with conditioning gas flow rates and temperatures as variables, to obtain optimum payload bay conditioning, or configuration changes may be indicated within the payload bay to obtain better flow distribution and temperature patterns.

If justified by payload peculiar requirements or characteristics, operational spacecraft for Centaur deployment may be scheduled for testing in the full-length simulator with follow-on Centaur production vehicles.

Production Vehicles. After initial delivery and R&I at the Cape, each production Centaur will be installed at Complex 36A for test and checkout before its first flight. Test and checkout will duplicate that for the first flight article, plus vehicle temperatures (6.3 above). To minimize operating and handling costs, only the lower half of the simulator will be used, unless special requirements must be met, such as verification of spacecraft temperatures when mated to Centaur and installed in the Orbiter payload bay.

10.7.2.4 Centaur/Spacecraft Mating. Following checkout and tanking tests in the payload bay simulator at Complex 36A, the mated Centaur/deployment adapter unit will be transported to a NASA-designated spacecraft mating facility for mating with a spacecraft. NASA maintains three spacecraft facilities at KSC, west of the Banana River in the NASA Industrial Complex on Merritt Island: Spacecraft Assembly and Encapsulation Buildings 1 and 2 (SAEB 1, SAEB 2), and the Spacecraft Operations & Checkout Building, M7-355, referred to also as the Manned Spacecraft Operations Building (MSOB). All three facilities are air conditioned to Class 100,000 clean room rating, and have floor area, door clearance, crane capacity and hook height adequate to handle any payload for Centaur.

In the clean room of the spacecraft mating facility, Centaur protective covers will be removed and the forward interface prepared for spacecraft mating. After mating, all interface connections and combined systems will be checked out, and electrical and electronic systems run through an abbreviated countdown culminating in transfer to internal power, arming, and initiation of a simulated flight sequence. The test will be similar to the complex 36 electrical checkout except for the abbreviated sequence and omission of Centaur pyrotechnic monitoring.

10.7.2.5 Payload/Orbiter Mating. "Payload" here refers to the complete Centaur/Spacecraft unit, as a single Orbiter payload.

Shuttle Preparations. After completion of Orbiter inspection, maintenance and refurbishment, mission-peculiar equipment is installed in preparation for payload mating. In addition to LH_2 and LO_2 fill and drain lines, propellant vent and other service lines, the major item of mission-peculiar equipment to be installed in Orbiter in preparation for payload mating will be the payload specialist station system, in support of the Centaur monitor and control system (CMACS). Once the CMACS capability has been installed in Orbiter and checked out, together with any non-permanent TBD mission-peculiar service lines, the Orbiter will be ready for payload installation.

Centaur Preparations. On arrival at the Orbiter MCF 9 work days (11 to 13 calendar days) before Shuttle launch, Centaur final preps for Orbiter mating will be performed, including removal of protective covers and desiccant plugs, and installation of flight batteries and pyrotechnic squibs.

Payload/Orbiter Mating & Checkout. After Centaur/Orbiter mating, test and checkout at the MCF will consist of Orbiter/Payload interface checkout and verification, and an overall avionics operations test (AOT). The AOT, while undefined as yet, presumably will provide electrical and electronic verification of all Orbiter/Centaur/Spacecraft related electronic systems, before Orbiter mating to the Shuttle external tank (ET) and the mobile launcher.

After AOT data review and discrepancy correction, Orbiter will be mated to the external tank on the mobile launcher, Orbiter/ET interfaces connected, fluid and electrical systems checked out, integrated electrical interference tests run, ordnance installed, and the Shuttle/ML prepared for transfer to launch complex 39.

10.7.2.6 Launch Pad Operations. Two work days or 37 hours before launch, the mobile launcher and Shuttle are transferred to Complex 39 on the transport crawler, arriving 31 working hours before launch. During the first 9 hours at the launch pad, all ML/Pad interfaces are connected and verified, Centaur guidance and control system (CMSCS) activated, Shuttle powered on, and a final overall combined AOT run. The next 20 hours, until 2 hours before launch, are taken up in data review, cabin closeout and payload servicing, and countdown preparations. Details of the combined Orbiter/Centaur propellant loading sequence and final two hours of countdown for the D-1S(R) and RLTC vehicles are discussed in Section 7. They are applicable to the Reusable Centaur vehicle with minor differences in flow rates and sequencing details.

10.7.2.7 Countdown and Launch. Launch operations for the RC vehicle are identical to those for the D-1S(R) and RLTC, covered in Section 7.2.7.

10.7.2.8 Post-Landing Operations. Post-landing operations for the D-1S(R), RLTC and RC vehicles are identical, and are covered in Section 7.2.8.

10.7.2.9 Payload Changeout. See Section 7.2.9.

10.7.2.10 Maintenance and Refurbishment. Restoration of a Centaur vehicle, regardless of configuration details, is based on results of the refurbishment and cost analyses performed for the Space Tug Launch Site Service Interface Study. Details of maintenance and refurbishment philosophy, and Hangar J Refurbishment Facility at CKAFS, ETR, are discussed in Section 7.2.10.

10.7.2.11 Reusability. See Section 7.2.11.

10.7.2.12 Storage. Centaur vehicles not actively engaged in turnaround activities or in preparation for operational activation will be held in storage at Hangar H, CKAFS, in close proximity to the Centaur MCF (Hangar J), and other spacecraft and Centaur facilities at the Cape. Details of the Centaur storage facility and procedures are discussed in Section 7.2.10.

10.7.3 CENTAUR TURNAROUND TIME. Centaur turnaround time, within a few hours, will be the same for D-1S(R), RLTC, and the RC vehicle: approximately 275 to 285 working hours. This turnaround time provides a capability of flying 13 to 14 missions per year on a 5-day week, 2-shift per day basis, and since the fleet utilization schedule (see Section 10.8.6) indicates a maximum utilization of only nine flights per year under the worst attrition conditions, turnaround time is not pacing and is of importance only in regard to task and scheduling functions. Five or ten hours difference in maintenance time between the three configurations is therefore inconsequential for planning. Details of turnaround time are covered in Section 7.4.

10.7.4 GROUND OPERATIONS FACILITIES. Facility requirements to support Centaur ground operations at KSC are based on utilization of existing facilities, providing an extremely low cost investment for the program. Usage of and modifications to existing facilities are discussed in Section 7.5, covering Hangar H (Administration & Storage), Hangar J (Centaur Maintenance & Refurbishment Facility), Hangar K (GSE Maintenance), and Complex 36 (Centaur Cryo Test Facility.

10.7.5 GROUND SUPPORT EQUIPMENT. The inventory of GSE required to support any launch vehicle in use today covers a wide variety of equipment, from major items such as transport vehicles, shipping pallets, handling adapters, electronic checkout systems, propellant loading valve skids, and engine service units, down to the normal list of minor items such as hand tools, tie-downs, dollies, fixtures and covers.

For the Reusable Centaur vehicles, much of this inventory is already in existence in support of Centaur D-1A and D-1T operations, particularly in the transport and handling categories, and in the category of minor items. Most of the existing equipment is in sufficient quantity to partially if not wholly support concurrent D-1A, D-1T and Centaur for Shuttle operations. Some items of existing equipment will require minor modification, and some new equipment will be required.

New, modified, and major related items of GSE required to support Reusable Centaur operations are listed in Table 10-17. The list includes those ground operations items also used in manufacturing, to give a total GSE requirement summary with allocation and numbers required. Most of the test and checkout items of support equipment are self-explanatory and need no discussion. Other items are fully described in Reference 1, Volume 6. The tables provide inventory requirements for the 4-year program, existing hardware identification, and net purchase requirements for the program.

10.8 PROGRAMMATICS

Programmatics encompasses the scheduling and resource planning associated with development, procurement, and deployment of the Reusable Centaur Shuttle upper stage. Trade studies and risk analyses support test plans, manufacturing plans, and operations (ground and flight) plans which are developed and synthesized to formulate a viable program for the Reusable Centaur.

Key elements of the program which impact resource requirements are the development test plan and fleet operations plan. Development testing must be carefully considered since it can easily account for 25 to 30% of the development phase cost, sufficient to minimize risk and ensure reliable hardware. Fleet size and degree of reusability directly impact the procurement and deployment phase costs. Careful attention to all

Table 10-17. Reusable DoD Centaur - GSE

Item No.	Name	Design Status	Program Requirement SD	Program Requirement KSC	Program Requirement TOT	Existing Hardware	Purchase
1401	Centaur Transport Pallet	New	3	2	5	-	5
1402	Aft Support Adapter	New	3	2	5	-	5
1404	Transport Trailer	Mod	1	2	3	-	3
1405	Transport Cover, Centaur	New	3	2	5	-	5
1407	Lift Sling, Centaur Trans Pallet	New	2	2	4	-	4
1408	Deployment Adapter Trans Pallet	New	1	2	3	-	3
1409	Dep. Adapter Pallet Lift Sling	New	1	2	3	-	3
1413	Payload Bay Simulator	New		1	1	-	1
1414	Payload Bay Simulator Workstands	New		1	1	-	1
1415	Hydrogen Peroxide Transfer Kit	Exis		1	1	1	0
1416	H_2O_2 Vacuum Drying Unit	Exis		1	1	1	0
1417	H_2O_2 Hot Firing Kit	New		1	1	-	1
1418	H_2O_2 Transfer Unit (Cx 39)	New		1	1	-	1
1419	LH_2 Fill & Drain Skid	New		2	2	-	2
1420	LO_2 Fill & Drain Skid	New		2	2	-	2
1421	CCLS (Computer Cont Launch Set)	Exis	1	2	3	3	0
1422	FAP (Flight Acceleration Prof.)	Exis	1	0	1	1	0
1423	GDIE (Grd Dig Interface Elect)	New	1	1	2	-	2
1424	Launch Control GSE	Mod		1	1	1	1, Mod
1425	Factory Checkout Trailer	Mod	1	0	1	1	1, Mod
1426	TLM Ground Station	Mod	1	1	2	2	2, Mod
1427	TLM Data Station	Mod	1	0	1	1	1, Mod
1428	CMACS Simulator	New	1	2	3	0	3
1429	Subsystem Support Equipment	Mod	1	0	1	1	1, Mod
	a. DCU/RMU SE						
	b. IMG SE						
	c. CCTE						
	d. SCU, SIU, SC Test Sets						
1430	Maintenance, Assembly & Test St.	New		2	2	-	2
1431	Deployment Adapter Dolly	New		1	1	-	1
1432	Pneumatic Test Panel	Exis		1	1	1	0
1433	Ultrasonic Scan Unit	New		1	1	-	1
1434	IMU Rate Table	New		1	1	-	1
1435	Acoustic Leak Detection Unit	New		1	1	-	1
1436	Radiography Unit	New		1	1	-	1
1437	Mass Spec Leak Detection Unit	New		1	1	-	1
1438	MLI Purge Metering Unit	New		1	1	-	1
1439	Borescope & Fiber Optics	Exis		1	1	-	1

Table 10-17. Reusable DoD Centaur - GSE (Continued)

Item No.	Name	Design Status	Program Requirement			Existing Hardware	Purchase
			SD	KSC	TOT		
1440	Main Engine Test Kit	Exis		1	1	1	0
1441	Engine Handling Dolly	Exis		2	2	2	0
1442	Spacecraft Docking Simulator	New		1	1	-	1
1443	GH_2 Vent Line	New		1	1	-	1
1444	GO_2 Vent Line	New		1	1	-	1
1445	Purge Lines	New		1	1	-	1
1446	Star Simulator	New		1	1	-	1
1447	Horizon Scanner Test Target	New		1	1	-	1
1452	ACPS Engine Test Kit	Exis		1	1	1	0
1453	Propellant Utilization Kit	Exis		1	1	1	0
1454	Pallet Tractor	Exis	1	1	2	2	GFE
1455	Tow Tractor	Exis	1	1	2	2	GFE
1456	Escort Vehicles	Exis	2	2	4	4	GFE
1457	C-5A Cargo Aircraft	Exis			1	Yes	GFE
1458	Super Guppy Aircraft	Exis			1	2	GFE
1459	Cargo Lift Trailer	Exis	1	1	2	2	GFE
1460	Fuel Cell Test & Checkout Unit	New		1	1	-	1

programmatic issues is vital to assure that the Centaur program is successfully developed, procured, and deployed on schedule and within budget.

10.8.1 PROGRAM SCHEDULE. Development of the Reusable Centaur follows an approach aimed at low development cost but high confidence in an early mission capability. State of the art applied in the design of the vehicle is such that minimal development testing is required to attain this confidence. A large portion of the hardware and components selected for the Reusable Centaur are either existing Centaur D-1T hardware, off-the-shelf-type hardware or well within current state-of-the-art technology. The main engines (RL10A-3-3) are the same as used on D-1T; minor adjustments made to the cooldown valves to minimize chilldown propellant losses do not require significant development or increased procurement spans. The boost pumps are the same as for the current D-1T but with an idle mode added to conserve chilldown propellants. The propellant tanks are D-1T tankage, except for an enlarged diameter LH_2 tank, which changes the internal PU hardware and software. The tank design, however, is based on Centaur and Atlas technology, and on previous design experience for the large tank GT Centaur effort. Cryogenic insulation and ground vent systems are either D-1T hardware or existing technology. A new zero-g thermodynamic vent mixer device is based on Convair's tested prototype hardware, and the titanium structural adapters are based on D-1T experience at Convair with R&D and production efforts. The astrionics system design and development also lean heavily on D-1T experience; except for some minor revisions, this system is essentially the same as D-1T. Reusability, revised mission reliability requirements and man-safety requirements have increased the number of astrionic functions and necessitates revalidation and qualification for some equipment. A horizon scanner is added but the star sensor is no longer needed as the guidance system will be updated from ground. A modified Shuttle fuel cell is considered for additional power and will support the proposed development schedule.

The above considerations and the fact that the early Tug will be used for deployment - only missions permits a minimal development effort to attain the desired confidence at IOC. Figure 10-18 shows the major development milestones leading from the conceptual phase through the proposed four-year operational phase. Figure 10-19 is the summary development schedule for the Reusable Centaur. It is predicated on a one-shift engineering and production basis and reflects the minimum subsystem development and vehicle integration activities considered necessary to adequately support an IOC by July 1980. This schedule reflects a 45 month development program to IOC. Current Centaur and D-1T flight data along with a dedicated RC flight test will provide ample verification and demonstration of this Reusable Centaur as a Shuttle third stage. After the DT&E phase, when additional production vehicles become available, the flight test vehicle will be refurbished as necessary and returned to fleet use.

Only one test article is provided for structural loads and life-cycle testing. Other major ground tests include the RC deployment mechanism for functional and dynamic testing as well as the astrionics equipment module for systems and software integration testing. Component and subsystem hardware items from the ground test program which have not been subjected to high-stress conditions will be utilized on the dedicated flight test vehicle. Such items include the astrionics module and equipment, payload adapter, main engines, some propellant feed system hardware, and the Orbiter interface structure.

The flight test vehicle is delivered to the operational site some 7 months prior to the initial flight. During this period the Reusable Centaur will be integrated and checked out with the site to verify compatibility, procedures and operations (including Pad 36 and Hangar "J" ringout) required in support of pre-flight activities. The remaining production vehicles will also be delivered to Hangar J for checkout, followed by initial tanking tests at Centaur Pad 36 prior to payload mating and delivery to the Shuttle VAB for Orbiter integration.

During site integration at Pad 36, ground hold insulation and propellant dump verification tests will be performed with the flight test vehicle prior to its initial preflight preparations. The initial flight is planned as a technical evaluation and demonstration of Shuttle compatibility and safety, vehicle deployment and performance, vehicle retrieval, refurbishment and reusability. The initial operational flights will further demonstrate general reusability and maintenance and refurbishment techniques. Initial OT&E is also assigned to these flights, to demonstrate general utility and operational effectiveness and suitability.

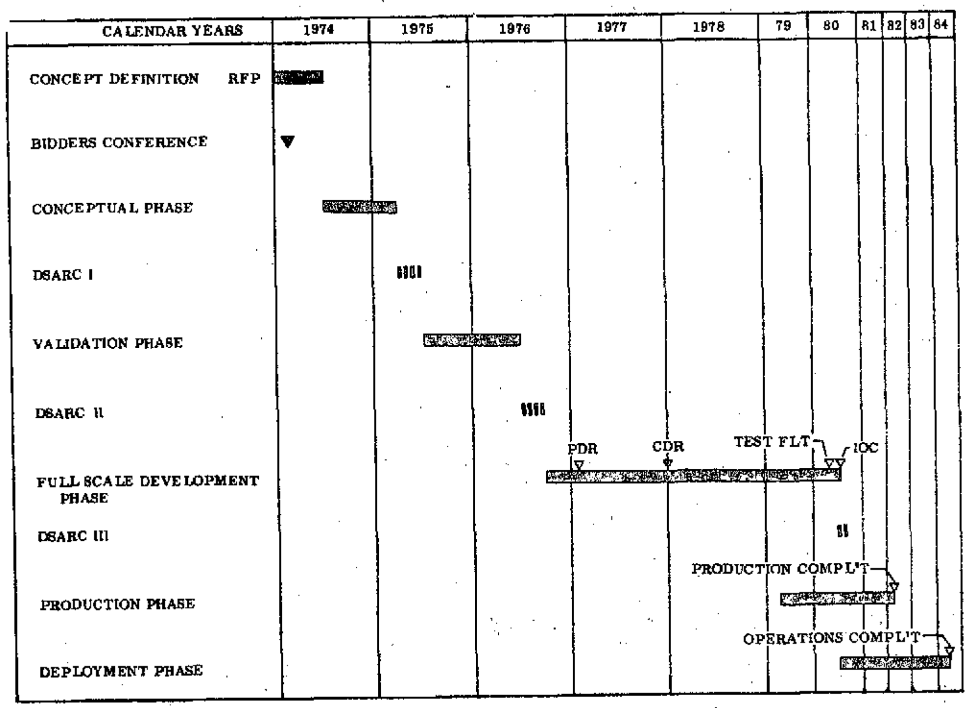


Figure 10-18. Reusable Centaur Program Development Milestones

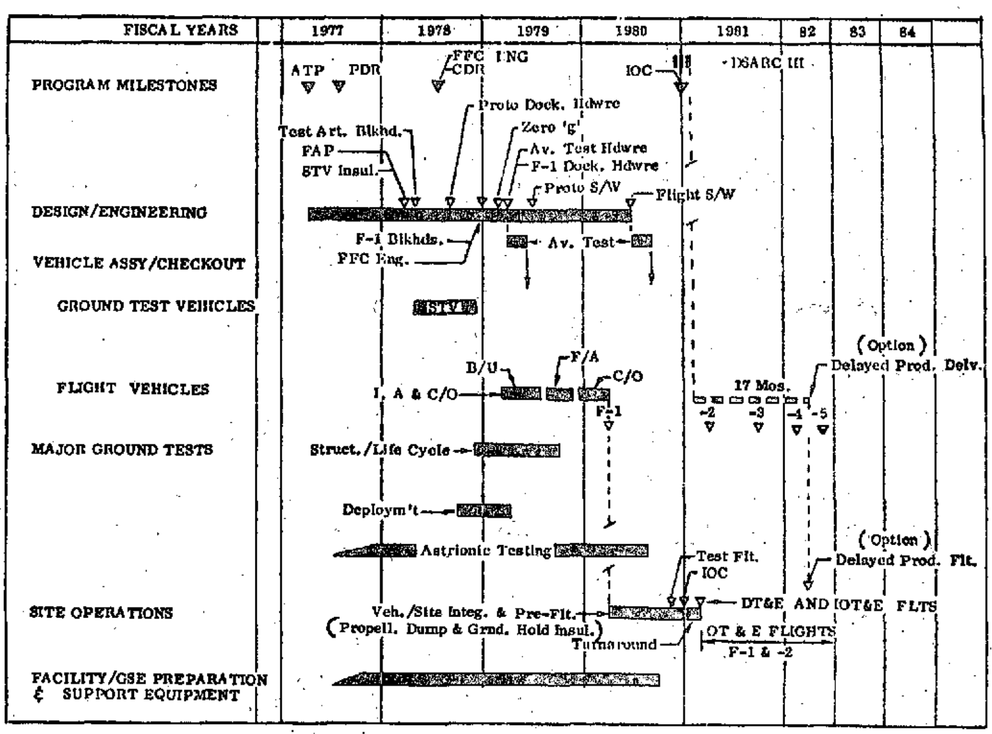


Figure 10-19. Reusable Centaur Development Program, Summary Schedule

DSARC-III is initiated during the test flight turnaround phase and completes after the IOC flight turnaround phase. OT&E flights are continued beyond DSARC into FY'82 as necessary to satisfy the specific test requirements.

If further production were constrained by completion of DSARC-III, the second flight vehicle would not be available until mid-FY'82. This would leave only one vehicle to satisfy the 26 flights in FY 1981 and '82 (15 DoD and 11 NASA), which would be impractical. One vehicle may satisfy the 15 DoD flights in these years assuming no unplanned or unscheduled contingencies occur. Of course, the event of an early attrition flight in these years would stop flight activities until mid-FY'82. Since only 4 years are considered in the operational program and only 5 Reusable Centaurs are required to satisfy the complete Tug flight model, it is suggested that minor exception to AFSCP 800-3 be taken to permit initiation of follow-on production (4-vehicles) prior to DSARC-III with delivery to KSC delayed until completion of DSARC-III as is indicated in the summary schedule. Start of fabrication of these vehicles is sufficiently beyond the close of ground qualification testing to minimize or eliminate any potential problems or redesign delays.

The 45-month development program for the Reusable Centaur is considered nominal in view of the existing Centaur vehicle status and the level of design and testing detail to be accomplished during the validation phase. The single long-lead development item is flight software, which will be qualified and integrated in the flight test vehicle prior to initial flight. A prototype software program is used for the contractor's factory checkout and later flight-site integration.

10.8.2 <u>DEVELOPMENT TESTING</u>. In accordance with the intent of DoD Directives 5000.1 and 5000.3 and AFSCP 800-3, development test and evaluation of the Reusable Centaur shall commence early in the program and continue through completion of full-scale development as necessary to support progressive reduction of acquisition risks and to determine user worth. Test and Evaluation consists of two major phases, DT&E and OT&E. The DT&E phase demonstrates that the system will meet specifications and that development is complete. The OT&E phase assesses the system's operational effectiveness and suitability as well as user utility.

Table 10-18 presents an overall summary of the test programs required to support development of the Reusable Centaur as the Shuttle third stage. There is a dedicated flight test prior to IOC, and this test vehicle is instrumented to obtain performance and environmental data, as well as provide verification data on reusability, refurbishment and initial operational utility and suitability.

10.8.2.1 <u>Ground Test Program</u>. The ground test program is based on test requirements necessary to support development of the Reusable Centaur vehicle and ancillary equipment concepts in support of Shuttle missions. Current Centaur test philosophy and criteria along with Centaur D-1A and D-1T testing and experience were used as a basis for determination of unique Reusable Centaur test requirements. Emphasis was placed

Table 10-18. Reusable Centaur Test Program Summary

Test Activity	Test Site	Test Hardware	Test Duration	Remarks
Component/Subsystem Qualification	Convair & Vendor Labs	Preproduction and/or production components & hardware	15 Months	All new hardware requires qualification & some Centaur hardware tested to different environments.
Propulsion Tests	P&W (E5)	Engines & feed systems hardware	Part of Engine Mod. Validation	Engines used on flight test vehicle.
Vehicle Structural Tests	Contractor or Government Facilities	Propellant tanks & feed system hardware, insulation and adapters	10 Months	Loads tests on non-Centaur structure & functional tests on all hardware, incl. life cycle data.
Ground Hold Insulation & Propellant Dump Tests	Pad 36	Uses Flight Test Article	3-4 Months	Insulation verification and propellant dump evaluation.
Deployment System Tests	Convair	Orbiter interface structure & deployment hardware, & Centaur aft structure simulation	6 + Months	Functional capability demonstration.
Astrionics Systems	Convair, JSC (SIL) & KSC	All astrionic systems and supplemental structure and equipment	22 Months	Prototype & production hardware, software and GSE (system integration)
Flight	Simulated Operational Mission	Flight Test Vehicle	36 Hours	On-orbit testing performed, including simulated payload deployment

on identifying major testing necessary to support design activities, providing qualification status for hardware, and establishing the high degree of confidence required for performance of the Space Shuttle systems. Basic test requirements for the Reusable Centaur configurations under consideration are within current state-of-the-art techniques. Primary considerations during program formulation were low cost and high confidence.

Figure 10-20 is a summary schedule of the overall integrated ground test program, showing the major test elements, their specific time phasing and intra-program relationships.

Most of the existing design components to be used on Reusable Centaur have been qualified to environments at least as severe as those to be encountered in the Reusable Centaur mission mode. New components will be subjected to complete qualification testing and existing hardware will, as required, be subjected to additional testing to qualify under re-use criteria.

The RL10A-3-3 engines proposed for Reusable Centaur are essentially identical to the engines in current use on Centaur. Changes in the technique of thermal conditioning of the propulsion system require some verification testing of the total propulsion feed system. Minor engine adjustment and subsequent integrated testing with support subsystem hardware can be economically accomplished by the engine contractor at his facilities. Data on engine and propellant feed system reusability and refurbishment requirements will also be obtained.

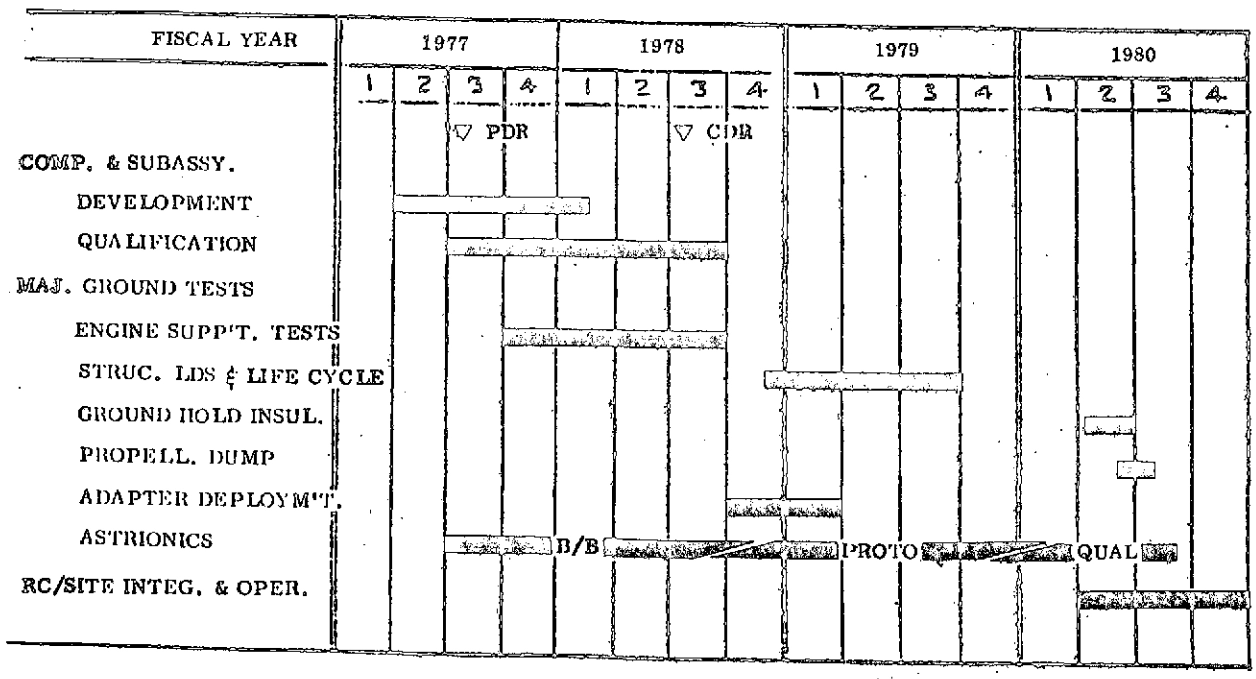


Figure 10-20. Reusable Centaur Development Test Program Summary

Structural testing of Centaur derivative vehicles for Shuttle is necessary to qualify design changes, verify function, and obtain reusability data. One dedicated structural test vehicle (STV) as shown in Figure 10-21 is sufficient to support the program. The test article will consist of the major tank structure with its attached forward and aft adapters, and the deployment adapter which stays in the Orbiter and mates with the Centaur aft adapter or latch skirt. The STV will include tank insulation, and supporting propellant load and vent systems. Engines and astrionics equipment will not be included.

Structural testing of the STV will include static load tests for strength and deflection to qualify the redesigned fuel/oxidizer tank structure, combined cryo-pressure cycles to obtain reusability data, and dynamic testing for vibrational loads if desired. The most suitable and economic facility for conduct of these tests exists at MSFC at the S2 Structural Test Position. The stand is equipped with load rings, is compatible with Centaur configuration and size, is connected to a 336,000 gallon liquid hydrogen supply system through an 8 inch vacuum jacketed line, has an adequate hard stand area to support liquid nitrogen tankers for cryo-conditioning of the Centaur LO_2 tank, and has 200 data channels to the MSFC 4670 West Blockhouse and 4619 Data Acquisition Facility. Testing can be phased in at the end of the Shuttle external tank structural test program at the adjacent S-1C static test stand, with the same MSFC test crew.

Thermal vacuum insulation performance and reusability characteristics can be tested at MSFC Test Position 300, if desired. Much of the necessary space environment

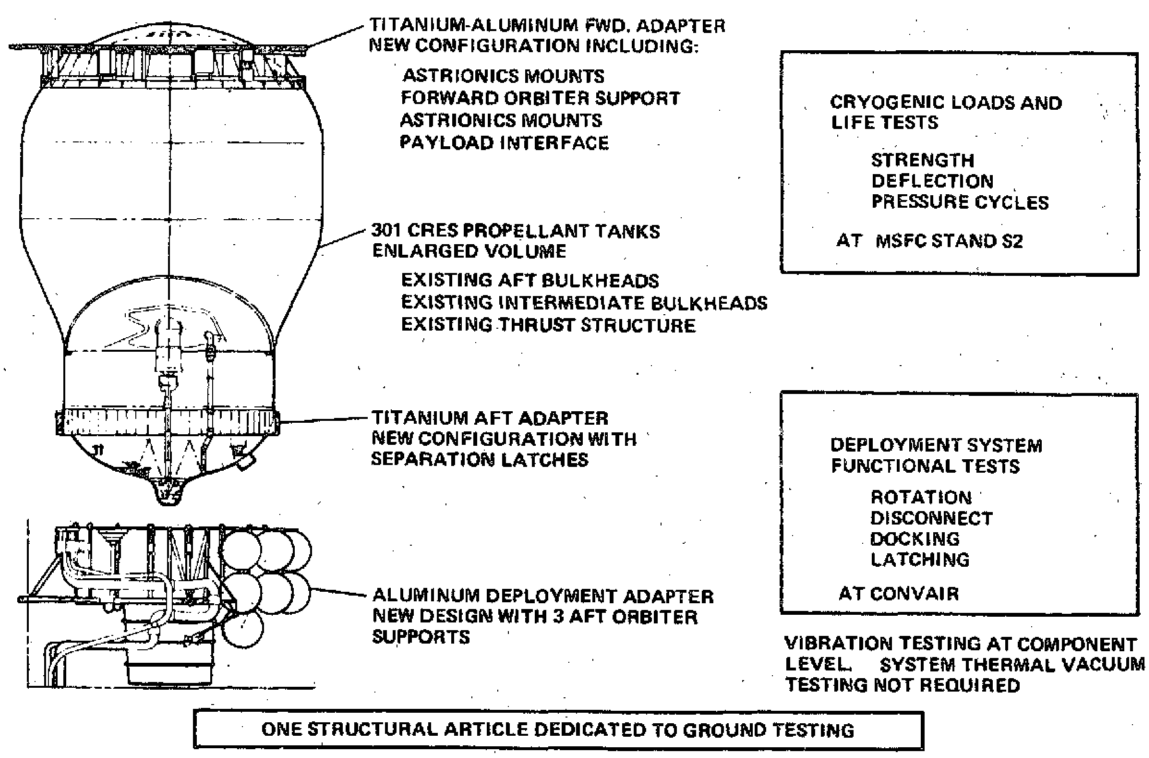


Figure 10-21. Structural Testing

thermal data has already been accrued from previous Centaur development test and operational flights, however, and additional data will continue to be compiled on D-1T missions. While thermal system vacuum testing is therefore not required, TP 300 testing would be advantageous from a standpoint of gathering reusability data under conditions of simulated Orbiter payload bay pressurization and ascent venting.

Deployment system functional tests will be run in San Diego at GD/CA's Harbor Drive or Kearny Mesa facility under non-cryo conditions. Tests will include deployment rotation, disconnect, separation, docking, and latching. Non-cryo static load tests will also be run on the deployment adapter and vehicle adapters at these facilities.

Astrionic systems testing starts early, around the preliminary design review (PDR), with breadboard type testing, and leads through prototype to qualification test hardware. The majority of applicable Centaur components are already qualified to environments at least as severe as those to be encountered in the Reusable Centaur mission mode. Before the subsystem level tests, qualification testing of remaining components will be completed and all parts, components and subassemblies will receive unit-level acceptance tests at the contractor's plant before installation on the module. Astrionics units will be installed on the Astrionics Module in a flight configuration and hardware interfaces, procedures, and computer control software planned for testing of electronic systems on the Centaur will be finalized and demonstrated during subsystem-level tests.

The Centaur astrionics module with associated GSE will be tested for compatability with the Shuttle astrionics systems and associated GSE in the Shuttle Vehicle SIL at JSC and integrated with the operational flight site at KSC, as part of the flight test vehicle.

10.8.2.2 Flight Test Program. A flight test is planned to demonstrate the validity of ground test environments, loads and test results. It will also impose effects of zero-g and other space environmental conditions which cannot be totally duplicated in ground testing. Purpose of this flight test is to verify and demonstrate compatibility and safety of the Reusable Centaur as a Shuttle third-stage vehicle, vehicle deployment and retrieval capability, and on-orbit performance. Post-flight evaluation will assess reusability and refurbishment capability of the vehicle and subsystem elements.

The Reusable Centaur's inherent capability for relatively long orbital stay times will permit an extensive amount of testing to be performed during a single flight mission. A typical flight will be divided into numerous phases for testing different Reusable Centaur operating modes. These phases will be structured so that operations involving progressively more stringent conditions will be separated by periods of data evaluation. The Reusable Centaur's operational flexibility will also complement this approach by permitting operational parameters such as chilldown times, burn times, etc. to be changed and reprogrammed from the ground (if capability exists) and/or selected from options stored in the airborne computer prior to launch.

The flight test vehicle is delivered to the test site some seven months prior to initial flight. During this period the vehicle will serve to verify complete compatibility with all operational facilities, support equipment and procedures (including Centaur Pad 36 and Hangar "J" where all production RC vehicles will be checked out prior to Shuttle integration). During the vehicle/site integration, the test article will be used at Pad 36 for a thermal evaluation of the vehicle ground-hold insulation and for demonstration of the propellant rapid-dump system capabilities. The initial flight will precede IOC by two months, allowing this same time for the test flight turnaround activities.

A single flight test along with ample related flight-test data from the current Centaur and D-1T programs will adequately demonstrate capability of the Reusable Centaur to satisfy IOC requirements with high confidence. The early operational flights with this test vehicle will further strengthen maintainability, refurbishment and reusability data acquired throughout the RC development program, and will verify operational suitability and user utility.

10.8.3 VEHICLE MANUFACTURING. The manufacturing plan required to produce the Reusable Centaur is almost identical to that for the RLTC, using existing resources used in the production of current D-1A and D-1T vehicles. Wherever possible, existing tools, facilities and operations have been incorporated in the manufacturing plan with a minimum of modification, to assure low costs and minimum scheduling risk. Details of the manufacturing plan are discussed in Section 8.3 of this report.

10.8.4 FACILITIES AND GSE. Use of existing manufacturing facilities at San Diego for manufacture of the Reusable Centaur vehicle follows the same philosophy and planning as that for the D-1S(R) and RLTC vehicles. Details of manufacturing and test facilities and GSE requirements for the RC vehicle are almost identical to those for the RLTC, discussed in Section 8.4 of this report.

10.8.5 LOGISTICS. Spares, training, handling, and transportation logistics for the Reusable Centaur vehicle are identical to the D-1S(R) and RLTC, and are discussed in Section 8.5 of this report.

10.8.6 FLEET SIZE. The number of reusable third stage vehicles needed to satisfy the STS mission model, annually and for the entire program, has a significant influence on the DDT&E, investment, and operational costs, as well as on scheduling and planning. Determination of optimum value for the active (annual) fleet size and for the total program fleet size is based on consideration of a number of programmatic factors:

a. Payload mission model

b. Vehicle capability for multiple payloads

c. Number of expendable vehicles required

d. Ground turnaround time

e. Attrition estimate

f. Launch Center scheduling

g. Production rate efficiency

h. Vehicle Reusability in terms of flights or total mission duty cycles (TMDC).

For the Reusable Centaur program, consideration of the above factors shows a recommended total program buy of seven vehicles and one spare tank: six "standard" 28-foot vehicles, one "short" 25-foot vehicle, and a short tank spare for build up in event of loss of the short tank vehicle. The six standard length vehicles include the flight test article, and the number of vehicles provides for a normal attrition loss of one vehicle. In the following analysis, each of the programmatic factors is discussed in the order listed above, with regard to its influence in determining the recommended number of vehicles for the fleet.

The Reusable Centaur has a performance capability that permits 100% capture of the 112 payloads in the four-year mission model with 78 flights, using multiple payload delivery on many missions, velocity packages on some, and expending the Centaur vehicle on two missions. The short vehicle delivers single payloads only, two of which are long DoD payloads which must be flown on the short vehicle. The traffic schedule for the program is as shown in Table 10-19.

Ground turnaround time for the Reusable Centaur is approximately 280 working hours, permitting a maximum of 14 flights per year for a Centaur vehicle flying 47-hour missions, if launches occur at even intervals. A minimum of two vehicles will therefore

Table 10-19. Reusable Centaur Minimum Active Fleet Requirements, Four Year Program

	Annual Traffic				
	1980	1981	1982	1983	Total
Flight Requirements					
DoD Flights	5	10	11	6	32
NASA Flights	11	17	8	10	46
Total Vehicle Flights	16	27	19	16	78
Vehicle Requirements					
For Turnaround Time	2	2 (3)	2	2	
Expendable Vehicles	1	1	0	0	
Attrition Vehicle	1	1	1	1	
Minimum Active Fleet Requirement	4	4 (5)	3	3	

always be required as an active fleet to satisfy turnaround requirements during the four-year program, when annual traffic is 16, 27, 19, and 16 flights per year.

In addition to vehicle requirements to satisfy turnaround time, an allowance must be made for attrition. An attrition rate of 1% has been set for the Tug programs, based on a 97% probability of mission success and recovery of the vehicle in two out of three cases. For the Reusable Centaur programs, this 1% loss is applied conservatively in determining the number of attrition vehicles lost; i.e., it is assumed that vehicles will be lost on the 1st, 101st, 201st, etc., flights. For the 78 flight four-year program, therefore, it is assumed that one of the six standard 28-foot vehicles will be lost through attrition.

Minimum active fleet requirements as summarized in Table 10-19 indicate a minimum active fleet need of four vehicles during the first two years of the program and three during the last two years. With a maximum of 27 flights in 1981, however, two vehicles (28-flight turnaround capability) are barely adequate to assure turnaround in that year, assuming even launch centers. To provide for launch schedule flexibility, contingency, and uneven launch centers, three vehicles are recommended in 1981 to satisfy turnaround requirements, or a minimum active fleet requirement of five vehicles rather than four.

The effects of reusability on fleet size are shown in Figure 10-22, plotting sensitivity of vehicle total mission duty cycles (TMDC) against fleet size as a controlled variable. The curves are based on the assumptions that expended vehicles are full-term, that the attrition vehicle is lost early in the program, and that non-attrition vehicles equally share the attrition vehicle's uncompleted missions. TMDC curves are plotted for three conditions of attrition:

a. Early attrition: loss of the attrition vehicle on its first flight.

b. Normal attrition: loss of the vehicle on its mid-life flight.

c. Late attrition: loss of the vehicle on its last scheduled flight.

As shown, if a minimum program buy of five vehicles is assumed, the maximum number of flights that any one vehicle would fly would be 18, under normal conditions of attrition. With early attrition, i.e., with one of the five vehicles lost on its first flight, the rest of the fleet would be required to fly 19 to 20 missions each, including the two vehicles scheduled for expendable missions. Such TMDC values of 19 and 20 are acceptable for Reusable Centaur, and a fleet size of 5 can therefore be established as a minimum requirement for the program.

The factor of reusability becomes significant if maximum TMDC becomes large, i.e., if vehicles are required to fly 50 to 100 missions in their expected life spans. For the four-year Reusable Centaur program, the significance of reusability becomes relatively minor with a maximum expected TMDC of 20 flights. With few exceptions, Centaur

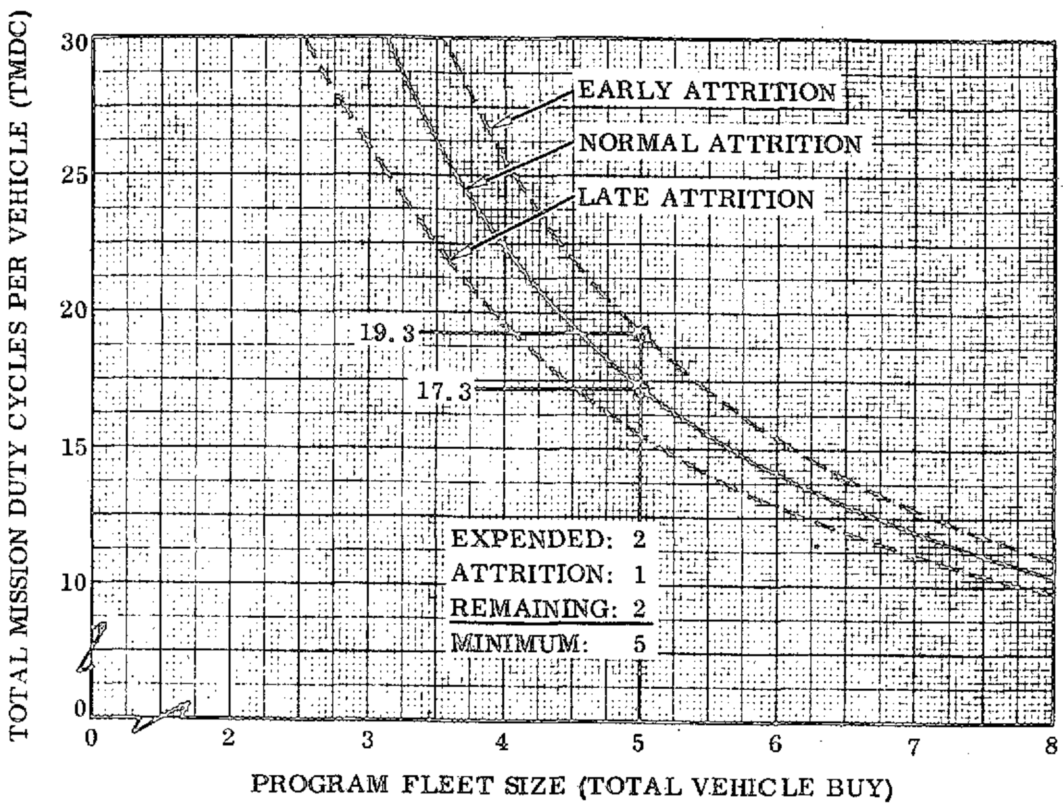


Figure 10-22. Reusable Centaur Fleet Size vs. Mission Duty Cycles

components and systems have an inherent TMDC capability of 20 or greater. The exceptions will be requalified or modified to provide the required reusability capability.

While five vehicles are the indicated minimum fleet size requirement, the actual requirement must be increased to provide for uneven scheduling and for TMDC constraints on individual vehicles. TMDC values will be low for those vehicles flying early expendable missions, and for payload constrained configurations such as the DoD short vehicle. As previously noted, seven vehicles are recommended for the Reusable Centaur program to assure accomplishment of the complete mission model.

Effects of early, normal, and late attrition on scheduling and TMDC are shown in the Fleet Requirement and Utilization Schedules, Tables 10-20 and 10-21. The schedules are illustrative only. Mission assignments are flexible and can be reassigned at will to accommodate attrition losses, schedule changes, and other contingencies. The schedules show the total fleet buy in terms of delivery and need dates, assignments, and vehicle disposition for each condition of attrition. In each case, vehicles left in inventory at the end of the program have completed full life spans, except for the payload-limited short vehicle, providing efficient usage of investment.

As indicated in the tables, a 7-vehicle fleet provides a maximum TMDC of 17 flights per vehicle in the case of early attrition, 16 flights for a normal attrition, and 14 if no attrition occurs. No vehicle will be required to fly more than 10 missions per year if early attrition is considered, or 7 per year for normal attrition. Average flights per

Table 10-20. Reusable Centaur Fleet Requirement and Utilization Schedule

Active Fleet (7 Vehicles)				Annual Traffic				Total Flights
Vehicle No.	Delivery Date	Need Date	Disposition	1980	1981	1982	1983	
				With No Attrition				
NASA-1	Apr 79	Nov 79	Expended	3Δ				3
NASA-2	Jan 80	Jan 80	Expended	7	4Δ			11
NASA-3	Sep 80	Jan 81	Inventory		5	4	5	14
NASA-4	Jan 81	Jan 81	Inventory		5	4	5	14
DoD-1	Sep 79	Jan 80	Inventory	(4)	(4)	(5)		(13)
DoD-2S	May 80	May 80	Inventory	1+(1)	3+(2)	(1)	(2)	4+(6)
DoD-3	May 81	May 81	Inventory		(4)	(5)	(4)	(13)
NASA Flights				11	17	8	10	46
DoD Flights				(5)	(10)	(11)	(6)	(32)
Total Flights				16	27	19	16	78
Active Fleet				4	6	5	5	
				With Early Attrition				
NASA-1	Apr 79	Nov 79	Expended	3Δ				3
NASA-2	Jan 80	Jan 80	Attrition	1*				1
NASA-3	Sep 80	Jan 81	Inventory		7	5	5	17
NASA-4	Jan 81	Jan 81	Expended		7Δ	3+(2)	5	15+(2)
DoD-1	Sep 79	Jan 80	Inventory	6+(4)	(4)			6+(8)
DoD-2S	May 80	May 80	Inventory	1+(1)	3+(2)	(1)	(2)	4+(6)
DoD-3	May 81	May 81	Inventory		(4)	(8)	(4)	(16)
NASA Flights				11	17	8	10	46
DoD Flights				(5)	(10)	(11)	(6)	(32)
Total Flights				16	27	19	16	78
Active Fleet, Including Attrition				4	5	4	4	

( ) DoD Flight
* Attrition Loss
Δ Expended Vehicle

year for all vehicles is 4 to 5. In most cases, vehicles are Agency-dedicated, although some sharing of missions is required to average TMDC values or to accommodate schedule changes required by attrition losses.

Regardless of attrition, the minimum active fleet requirements established in Table 10-19 are satisfied by the Utilization Schedules shown, assuring completion of the mission model unless abnormal attrition losses were to occur.

Table 10-21. Reusable Centaur Fleet Utilization Schedule with Normal Attrition

Active Fleet (7 Vehicles)				Annual Traffic				Total Flights
Vehicle No.	Delivery Date	Need Date	Disposition	1980	1981	1982	1983	
NASA-1F	Apr 79	Nov 79	Expended	3Δ				3
NASA-2	Jan 80	Jan 80	Expended	7	4Δ			11
NASA-3	Sep 80	Jan 81	Attrition		5	3*		8
NASA-4	Jan 81	Jan 81	Inventory		5	4	7	16
DoD-1	Sep 79	Jan 80	Inventory	(4)	(5)	(5)	(1)	(15)
DoD-2S	May 80	May 80	Inventory	1+(1)	3+(2)	(1)	(2)	4+(6)
DoD-3	May 81	May 81	Inventory		(3)	1+(5)	3+(3)	4+(11)
NASA Flights				11	17	8	10	46
DoD Flights				(5)	(10)	(11)	(6)	(32)
Total Flights				16	27	19	16	78
Active Fleet, Including Attrition				4	6	5	4	

( ) DoD Flight
* Attrition Loss Assumed
Δ Expended Vehicle

S = Short Stage
F = Flight Test Article

10.8.7 MANPOWER REQUIREMENTS. Centaur ground turnaround crew requirements at KSC were developed by determining manpower requirements on a task basis, and plotting these requirements against detailed turnaround timelines on an hourly basis. Hourly variations were then averaged by shift, and again across the total turnaround cycle to arrive at total direct support manpower quantity and type requirements as shown in Table 10-22. Project management, systems engineering support and Level II shop support manpower requirements were then added based on projected workload, numbers of direct support personnel, and anticipated Level II support workloads derived in conjunction with failure rate data in the spares analysis tasks. This basic crew can accomplish and support ground turnaround operations on a single vehicle within the previously determined 280-hour turnaround time span.

Crew size and composition is such that, when the number of launches increases in accordance with the traffic model and mission capture analysis, the basic turnaround crew is capable of accommodating concurrent operations on two vehicles. These manning levels also result in a number of direct support personnel who are not working on direct turnaround tasks 100% of the time during a Centaur turnaround cycle. This is particularly true during the long Orbiter host time following installation in the payload bay, when only a two man monitor crew is required. During these slack times, direct support personnel will be used to assist in Level II shop support tasks and accomplish GSE and facility maintenance.

Table 10-22. KSC Ground Turnaround Crew Size

Manpower Skills	KSC Shift 1	KSC Shift 2	Total
Direct Support			
Astrionics	10	8	
Propulsion - Fluids	9	7	
Structures - Handling	7	6	
Quality Control	2	2	
Safety	1	1	
Engineer	4	2	
Other	4	2	
Subtotal	37	28	65
Engineering & Admin. Support			
Logistics & Administration	13	6	
Design-System Engineering	6	3	
Scheduling	2	1	
Procedures-Test Reqt.	4	1	
Data Analysis	4	1	
Programming	2	1	
Subtotal	31	13	44
Level II Shops			
Metal	2	1	
Paint	2	1	
Plastics-MLI	1	1	
Pneumatics	2	1	
Avionics	2	1	
Standards-Inspections	1	1	
Subtotal	10	6	16
Total	79	46	125

10.9 REUSABLE CENTAUR PROGRAM COSTS

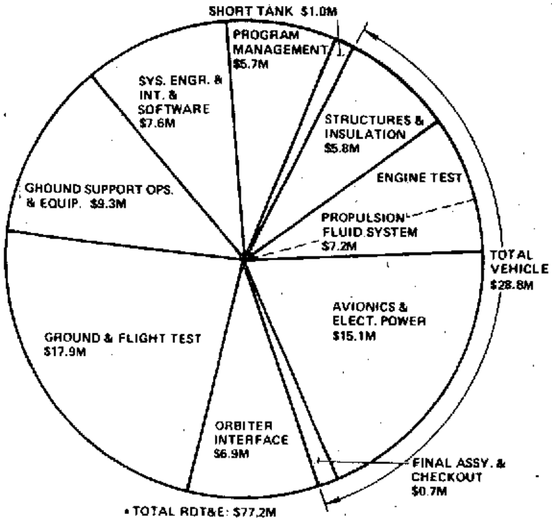
The Reusable Centaur incorporates many configuration and technology features from the current operational Centaur and at the same time offers the economies inherent in a reusable vehicle. The RC vehicle configuration resembles the D-1S(R) configuration with the single exception that the main propellant tanks are larger diameter like the RLTC tanks; hence, the development costs of RC correspond more closely with D-1S(R) than RLTC. For example, the RC does not incorporate the triple redundant guidance computers and other astrionics associated with Level I autonomy used on RLTC. Additionally the RLTC incorporates a new hydrazine ACPS and composite structures while the RC retains the existing hydrogen peroxide ACPS and conventional aluminum and titanium structures.

The basic methodology employed for estimating Reusable Centaur program costs incorporates carefully screened cost data to build up point estimates at or below the subsystem or task level identified in the WBS.

Program task costs were developed largely from recent Convair Aerospace experience dealing with the development of Centaur D-1, the development and production of D-1T and D-1A Centaur configurations, and the launch services at ETR. Additionally the Titan/Centaur integration tasks performed by Convair Aerospace provided an excellent analog from which to extrapolate Shuttle/Centaur integration costs. Centaur D-1 development was initiated in mid 1969 and the Titan/Centaur proof flight, incorporating four main engine burns, is scheduled for February 1974; therefore, the extensive cost data base is current and provides Reusable Centaur program cost estimates that are more valid than would normally be the case in a Phase A study.

Convair Aerospace experiences with launching Atlas/Centaur D provided a sound reference point from which to estimate Centaur/Shuttle launch service and operational costs. Where direct analogs were not available, parametric cost estimating relationships were formulated from the above Centaur experience, and judgement was then applied to modify and tailor the data to specific Centaur/Shuttle program tasks. In some cases the judgement process included an abbreviated "grass roots" estimating procedure or comparison with tasks and costs not related to Centaur. In many instances, the individuals who would be responsible for performing Centaur/Shuttle program tasks provided manpower estimates for their respective tasks.

The nonrecurring costs are summarized in Figure 10-23. Total stage DDT&E cost is $28.8M and includes development and component qualification of all new hardware. Included in the propulsion and fluid system development is engine testing required to demonstrate the changes associated with engine childown conditioning and mixture ratio. Astrionics and electrical power changes include the addition of sun and horizon sensors, DoD and NASA communications systems and modified Shuttle fuel cell unit. The Orbiter interface equipment includes the deployment adapter and associated mechanical and fluid systems as well as the interfacing astrionics identified as CMACS (Centaur Monitor



• TOTAL RDT&E: $77.2M
• ESTIMATES EXCLUDE PRIME CONTRACTOR FEE
• ESTIMATES INCLUDE 10% CONTINGENCY

Figure 10-23. Preliminary Reusable Centaur Nonrecurring Costs

and Control System). Unlike the D-1S(R) and RLTC programs, the RC test program incorporates a dedicated flight test. Although the ground test program for RC resembles the D-1S(R) and RLTC ground test programs, the total RC test cost including flight, is not appreciably greater because much of the ground test hardware is used in the flight test article. Additionally the ground testing of RC is reduced in some areas, such as thermal evaluations, where the data would be redundant with flight test data.

The nonrecurring cost item identified as Short Tank is the added cost to develop a 22- or 25-foot-long version of the RC. A cylindrical section of the LH_2 tank is removed necessitating minor changes to the propellant utilization system, tank insulation, wiring harnesses, and pneumatic tubing. The vehicle changes, together with a handling adapter to allow the shorter version to interface with GSE sized for the longer baseline configuration, can be incorporated for less than one million dollars.

Total program costs for RC are summarized in Figure 10-24. All NASA and DoD payloads for the initial 4-year operational period are flown from ETR. Costs are expressed in constant 1973 dollars without prime contractor fee.

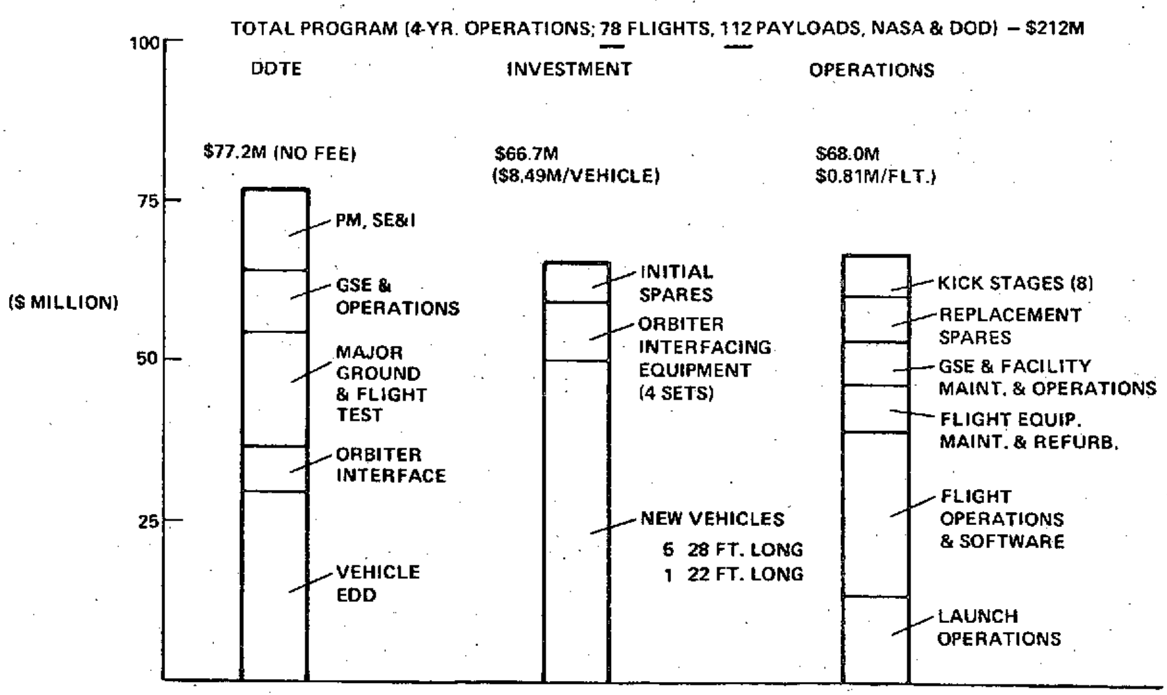


Figure 10-24. Reusable Centaur Program Costs

SECTION 11

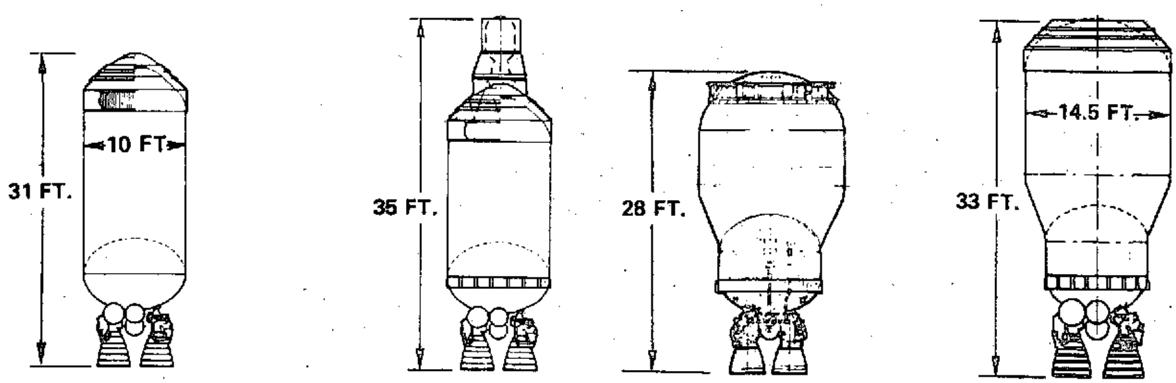
CONCLUSIONS AND RECOMMENDATIONS

This study used the current D-1T Centaur as an actual basis to develop a range of concepts for an initial upper stage for Shuttle. The basic drivers were: (1) low development cost, (2) safety and reliability, (3) reusability and low user cost, and (4) high performance.

An optimum solution would depend on the actual funding available and a realistic mission model taking into account the buildup in Shuttle flights and the transition from expendable launch vehicles. Of course, specific ground rules and directions from the development agency finally selected will influence the final configuration.

Figure 11-1 compares the three principal configurations resulting from this study. Although the D-1S(R) is very similar to the D-1T and therefore has the lowest development costs, it usually requires a kick stage.

The RC was conceived most recently to satisfy new DoD requirements for very long payloads. (The shortest, 22-foot version is omitted for clarity.) The RLTC is offered as a more sophisticated, autonomous version with greater performance but at higher development cost.



D-1T	D-1S(R)	RC	RLTC
30,000 LB. PROPELLANT	29,000 AT 5.8:1	47,000 AT 5.8:1	52,000 LB. AT 5.0:1
7 HR., 4-BURN	24 HR., 5-BURN	36 HR., 5-BURN	47 HR., 6-BURN
4,028 LB. DRY WEIGHT	5,018 LB.	4,995 LB.	5,510 LB.
FLIGHT-PROVEN	85% EXISTING COMP.	62.5% EXISTING	CURRENT TECHNOLOGY
SIMPLEX ASTRIONICS	ADDED CAPABILITY	DUAL SAFETY ITEMS	REDUNDANT/AUTONOMOUS
EXPENDABLE	REUSABLE >10 FLTS.	SAME	SAME
SYNC. EQ. P/L	4,096 W. SRM	4,581 W.O. SRM	4,501 W.O. SRM
DEVEL. COST ≃ $70M	$63.3M	$77.2M	$122M

VERSIONS SATISFY DIFFERENT REQUIREMENTS

Figure 11-1. Comparison of Centaur Versions

All of these Reusable Centaur versions employ the existing twin RL10A-3-3 main engines, LO_2 tank aft bulkhead with thrust structure, and twin-member intermediate bulkhead. The computer-controlled astrionics system just recently developed for D-1T is kept essentially intact. Although tank geometry varies from one version to the next, which is a very visable difference, only changes to thin stainless steel sheet metal are involved.

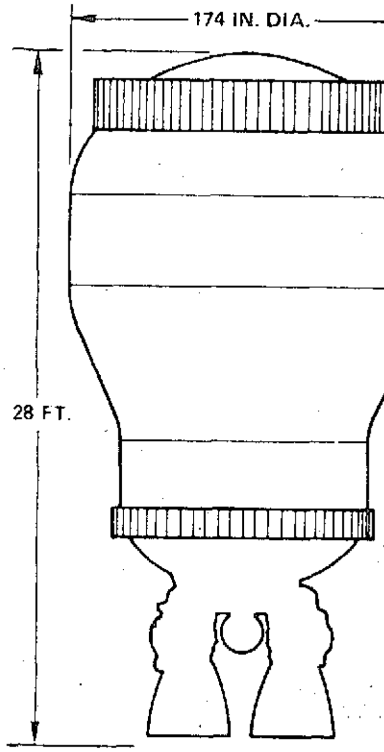
The final solution may be an evolution from one version to another, perhaps from expendable to reusable then to improved capability.

11.1 REUSABLE CENTAUR HIGHLIGHTS

We believe that currently the best OOS candidate is a Reusable Centaur, the main features of which are shown in Figure 11-2. The configuration is 176 inches in diameter and 28 feet long. It retains the basic construction technique of the Atlas and Centaur. A shorter version (down to 22 feet long) is envisioned (and included in the development cost) for accommodating the few long payloads which may be required. The configuration was derived from the existing Centaur and driven by the requirement to keep the development cost significantly below \$100M. It utilizes the high performance inherent in a LO_2/LH_2 cryogenic stage and will place 4600 pounds of payload to synchronous equatorial orbit and return the stage to the Shuttle for return to earth and reuse without the need for an expensive expendable solid upper stage. It is a low risk development since it needs no new technology. Sixty-three percent of the components are flying today on Centaur, 25% are modifications to existing hardware and only 12% are new to Centaur (mostly adding DoD communications equipment). It has inherent reusability due to its propellant, oxygen and hydrogen. The Reusable Centaur is a safe configuration and is compatible with manned operations in the Shuttle (in fact the Orbiter is attached to 1.6M pounds of LO_2/LH_2 and uses it for ascent propulsion). These facts allow the routine reuse of the Centaur to realize a truly low recurring cost of \$800,000 per launch which give it very low life cycle cost and make it a very cost effective OOS candidate, with good growth capability to payload retrieval capability later, should that be desired.

11.2 SUPPORTING RESEARCH AND TECHNOLOGY REQUIREMENTS

While the Reusable Centaur programs do not depend on any technology breakthroughs, there are a few areas requiring further development. There are backout positions for each item, and with the inherent Centaur performance margin, alternates can be used. The D-1S(R) intentionally uses nearly all existing hardware, while the RLTC uses more advanced technology. These technology and development differences should be resolved during the next major program phase (Phase B or Validation) to further reduce uncertainties and risk. The identified items are not really research or technology as much as they are advances or specializations of the current state of the art. Many of them are the subject of current technology contracts which need only to include Shuttle requirements. Some of them could be postponed until the



- DESIGN TO COST

  $77.2M RDT&E

- HIGH PERFORMANCE WITHOUT KICK STAGE

  4,580 LB. TO SYNC. EQ.
  >4,000 LB. TO 12-HR. ORBIT

- LOW RISK, NO NEW TECHNOLOGY

  62.5% EXISTING HARDWARE
  25% MODIFIED HARDWARE
  12.5% NEW HARDWARE

- REUSABLE > 10 MISSIONS

- COMPATIBLE WITH MANNED OPERATIONS

- VERY LOW RECURRING COST = $0.8M/FLIGHT

REUSABLE CENTAUR IS MOST COST-EFFECTIVE OOS CANDIDATE

Figure 11-2. Reusable Centaur Highlights

beginning of a hardware development phase, but it would be preferable to develop the data with which to make decisions concerning alternate implementations. The costs and schedules associated with each area of technology are estimates of the cost and calendar time required to develop the data necessary to make a decision about which of alternate techniques should be chosen. Sufficient data would be developed to allow reasonable requirements to be specified and to guide the choice of appropriate design allowables. Thus, the tasks and their associated costs and schedules do not reflect development into an actual product, but rather the development to the point where the data to determine applicability and risk are sufficient.

Table 11-1 summarizes the major technology efforts to support Reusable Centaur. Since the optimized, slow burn SRM required for D-1S(R) may be a major task, the RC and RLTC have been sized to avoid the use of velocity packages except on occasional use of the existing TE-M-364-4 size motor. A separate data sheet is included for each of these supporting research and technology items.

Table 11-1. Supporting Research and Technology

Technology or Development Area	Centaur Benefit	Priority	Cost ($)	Duration (yr)
Zero-G Thermodynamic Vents	Performance	1	1.5M	2
Fuel-Cell (Advanced or Orbiter)	Performance	1	1M	2
Advanced Composite Materials	Performance	2	750k	2
Postflight Checkout Techniques	Lower user costs	2	2M	3
Guidance Update Accuracy	Autonomy	2	500k	3
Slow Burn Kick Motor	Payload stress	3	1M	2

11.2.1 HEAT EXCHANGER TYPE ZERO-G VENT

Status: Prototype testing of bulk exchanges and mixers has been accomplished by Convair and others. Development of flight weight units leading to simulated or actual orbital tests is the next step in their development. A distributed exchanger in lieu of the mixer should continue to be investigated.

Justification: Tank lockup, screen containment, and thrust settling are the alternates to this system, and each would present weight penalties and/or safety hazards to the OOS. Development of a bulk exchanger zero-g vent system is required for long-term space residency.

Objectives: To verify the weights and powers of units developed for both LH_2 and LO_2 tanks. To resolve some detail design questions concerning electric drive system design. To develop flight weight prototypes that could be tested in an orbital flight on a planned Centaur launch after payload release. To investigate the distributed exchanger concept as an alternate to the powered mixers.

Resources Required: Flight type tanks and insulation system for each fluid would be required. Acutal flight test including qualification testing would cost $1.5 M and require about 2 years.

11.2.2 LIGHTWEIGHT FUEL CELL DEVELOPMENT

Status: Previous manned space programs Gemini and Apollo demonstrated the practicality of electrical power generation by fuel cell. The R&D for Shuttle-sized fuel cells currently being developed will provide the technology but not in the power range required for the Tug.

Justification: Fuel cells offer considerable mission flexibility for longer duration mission (greater than 48 hours). The Shuttle fuel cells could be utilized for OOS but they would be inefficient and complex. Advanced lightweight fuel cells, which offer significant advantages are being developed by P&W under USAF and NASA sponsorship.

Objectives: To develop a rugged, reliable, inherently stable, low weight fuel cell capable of reliable, high performance for numerous missions requiring a wide range of power with good voltage regulation.

Resources Required: The requirement is to adapt and repackage a Shuttle-type fuel cell which would require 2 years and $750k technology funding prior to contract to ensure feasibility.

11.2.3 ADVANCED COMPOSITE MATERIAL STRUCTURE

Status: Filamentary composite materials such as graphite/epoxy, boron/epoxy, and boron/aluminum have been fabricated into structural assemblies and significant weight savings demonstrated. Development of material properties data for thin gage face sheet material is insufficient for design implementation. Effects of meteoroid impact, acoustics, vibration and exposure to LH_2 temperatures have not been determined.

Justification: OOS performance can be improved using lightweight structure. Composites offer the potential to minimize weight but this potential must be demonstrated. The cost associated with the use of composites must be reduced.

Objectives: To determine realistic weights and costs of OOS structure utilizing composite materials. To develop material properties and design allowables for thin gage sandwich composites applicable to OOS adapter design. To evaluate the effects of the OOS environment on the composite materials (including LH_2 temperatures).

Resources Required: The use of the NASA hypervelocity impact facility and acoustic facility would be required. This would require $500k expenditure over a two-year time period.

11.2.4 POSTFLIGHT CHECKOUT TECHNIQUES

Status: Current preflight checkouts of expendable launch vehicles consist mainly of test and retest techniques which consume large amounts of time and money and often require temporary changes to the flight hardware. An OOS just returned from a successful mission should only be checked for inflight or postflight problems. Currently available field techniques are inadequate in areas such as:

a. Propellant tank leaks/cracks detection including meteoroid impact

b. Metalized insulation, deterioration measurement

c. Self checking maintenance support instrumentation

d. Checkout isolation of redundant hardware

Justification: Reuse of vehicles that have flown in some ways simplifies checkout, but also introduces new problems. Efficient, rapid, low cost postflight checkout techinques must be developed to reduce OOS turnaround costs. X-ray, cryogenic tanking, vacuum chamber checks, etc., are too expensive. Over only a four year span, turnaround costs will exceed initial development costs, and therefore must be minimized.

Objectives: To develop nondestructive methods and techniques for condition assessment monitoring of tankage to detect and characterize flaw growth, to detect leakage, and to evaluate and assess damage. To assess the methods of acoustic emission monitoring, remotely directed ultrasonics, thermal detection, leak tests, etc., and their applicability to the OOS requirements. It will also be necessary to establish acceptance standards for tanks, composite structure, and multilayer insulation which has been impacted by a meteoroid. To develop an accurate, inexpensive, nondestructive condition assessment technique to evaluate multilayer metalized insulation which may have deteriorated from solar radiation, moisture, etc., and which is largely hidden from visual inspection. To develop nondestructive test techniques to the point where lightweight, reliable sensors may be permanently attached to the OOS (if feasible) to exploit the condition monitored maintenance (CMM) concept. To develop checkout techniques applicable to a complex system with many redundant components reconfigurable under software control.

Resources Required: A wide range of specialized test equipment and representative test hardware would be required. The several investigations could require a total of 2 million dollars over a 3 year span.

11.2.5 GUIDANCE UPDATE ACCURACY

Status: Landmark tracking has limitations at high altitude. Horizon sensors are marginal both in accuracy (horizon variance) and dynamic range (100 n.mi. to 60,000 n.mi. altitudes). Non-optical alignment needs to be demonstrated.

Justification: Improved sensor accuracy would improve OOS performance both in ΔV correction required and placement accuracies as well as reduce the Shuttle rendezvous requirements. Improved accuracy is required for more autonomous operations.

Objectives: To develop more accurate horizon model and to develop horizon sensor with sufficient dynamic range and accuracy. To investigate unknown landmark tracking as a supplement to horizon and star sensors. To develop the digital computer programs necessary to implement the guidance update (recursive filtering, altitude damping, etc.) and to evaluate overall accuracy. To develop and demonstrate non-optical alignment for prelaunch and update from the Orbiter.

Resources Required: This would be a program that would require approximately three years of effort and the expending of about $500,000.

11.2.6 SLOW BURN KICK MOTOR

Status: All OOS and Tug designs require a kick motor for outer planet probe missions. The D-1S(R) uses kick motors even for synchronous equatorial missions. Current motors, like the Burner II subject payloads to accelerations greater than the 3.0 ± 20% g's allowed in Shuttle/payload specifications. Slow burn kick motors are on the edge of current technology. Thiokol has performed some slow burn feasibility demonstrations for JPL.

Justification: A slow burn grain is required to satisfy current spacecraft desires for environment more benign than current; i.e., accelerations below 3.6 g. D-1S(R) performance could be doubled using an optimum size kick motor.

Objectives: To demonstrate a slow burn motor the size of the TE-M-364-4 but at 1/3 thrust (3 times burn time). To demonstrate a slow burn motor about twice this size (approximately 1.1 million lb-sec total impulse).

Resources Required: It is estimated these tasks would require at least 1 million dollars and two years.

11.3 SUGGESTED ADDITIONAL EFFORT

This study has provided a family of Reusable Centaur versions which makes attractive candidates for the Initial Upper Stage or OOS. The optimum solution depends on the real program requirements and funding constraints. Based on these new ground rules and requirements, the study should be continued, emphasizing the following specific areas:

a. Flight and Ground Operations

   1. Payload requirements from new mission model, considering buildup and transition.

   2. ACPS requirements considering payload requirements, guidance accuracy, rendezvous envelope, etc.

   3. Multipayload combinations considering operations and hardware.

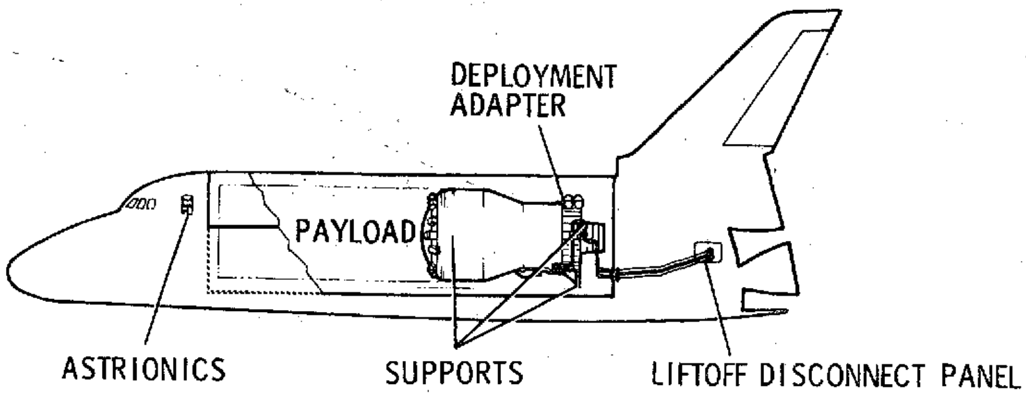
4. Optimize performance of open loop PU and bias versus mixture ratio.
5. Inflight abort propellant dump operations. (Centaur provisions for complete inflight dump may be unnecessarily conservative.)
6. Post flight safing requirements, routine and post abort.
7. Trade autonomy level against hardware, software, communications, and operations costs and ground operations complexity, including fault isolation and system testing.
8. Maintenance operations considering Centaur component life expectancy, available test techniques, and learning curve on CMM.

b. Orbiter and Payload Interfaces (Figure 11-3)

1. Structural loads and deflections in the Centaur and its payload.
2. Environment including prelaunch chilling after tanking.
3. Single and multiple payload support and release concepts.
4. Fluid and mechanical interfaces, including a hydrogen vent to Orbiter fin tip.
5. Caution and warning functions and Mission Specialist operations.
6. Inflight abort propellant dump.
7. Software Development Plan covering onboard software for Centaur, Orbiter support equipment software, and ground (flight operations) support software. NASA Shuttle and DoD compatibility. Recommend format and language and the possible maximum utilization of the existing Centaur program library.

c. Continued Reusable Centaur Studies

1. Define evolution capability from expendable to reusable to greater performance.
2. Optimize performance vs. development cost.
3. Trade reliability redundancy vs. cost and weight, also defining redundancy management.
4. Compare hydrazine vs. hydrogen peroxide ACPS cost and weight.
5. Compare aluminum/titanium to epoxy graphite structures.
6. Minimize chilldown and engine start losses.
7. Compare fuel cells vs. batteries; include Orbiter type and advanced fuel cell.
8. Evaluate cryogenic helium storage using fuel cell heat exchanger.
9. Optimize ground insulation and radiation shields.



- FOUR-POINT STRUCTURAL SUPPORT ATTACHMENT
- CENTAUR DEPLOYMENT LATCHES
- DOCKING GUIDE/SHOCK ARRESTMENT
- PROPELLANT, FLUID & ELECTRICAL SERVICE INTERFACES (CENTAUR & PAYLOAD)
- ABORT PRESSURIZATION & INSULATION PURGE HELIUM SUPPLY
- PROPULSION SYSTEM LEAKAGE CONTAINMENT
- ASTRIONICS MONITOR & CONTROL INCLUDING DEDICATED COMPUTER IN ORBITER
- DEPLOYMENT ROTATION MECHANISM

MOST EQUIPMENT FUNCTIONS ARE COMMON TO ANY TYPE STAGE

Figure 11-3. Orbiter Interfaces

10. Refine development and qualification test requirements and costs including flight test.
11. Refine software tasks and costs, considering language compatibility.
12. Refine operating costs for payload and mission peculiar support, ground support during flight, maintenance and spares.

11.4 CONCLUSIONS

1. Cryogenic stages have inherent high performance:

   Nominal 444 seconds with existing RL10A-3-3 engine

2. High performance results in many benefits:

   Allows multiple payloads, reducing number of launches
   Minimizes need for velocity packages and drop tanks
   Relieves necessity to use advanced technology
   Provides program insurance in case of Orbiter under-performance

3. Reusable Centaur is a low risk program:
   Mostly existing flight-proven components.
   Fuel cell is natural on a hydrogen-oxygen stage.
   Alternate solutions available.

4. Safety and reliability have been designed in:
   Its unique structural concept was man-rated on the Mercury program.

5. Based on current data, Centaur is inherently reusable for about 10 flights.

6. Realistic development cost data is derived from the current, similar, D-1T program.

7. Reusable Centaur has the lowest user costs and is therefore the most cost-effective approach to initial upper stage capability.

SECTION 12

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