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A Study of a Lifting Body as a Space Station Crew Exigency Return Vehicle (CERV)

Ian O. MacConochie
FDC/NYMA, Inc., Hampton, Virginia

October 2000

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Contents

A Study of a Lifting Body As a Space Station Crew Exigency Return Vehicle (CERV)

Ian O. MacConochie

Abstract

A lifting body is described for use as a return vehicle for crews from a space station. Reentry trajectories, subsystem weights and performance, and costs are included. The baseline vehicle is sized for a crew of eight. An alternate configuration is shown in which only four crew are carried with the extra volume reserved for logistics cargo. A water parachute recovery system is shown as an emergency alternative to a runway landing. Primary reaction control thrusters from the Shuttle program are used for orbital maneuvering while the Shuttle verniers are used for all attitude control maneuvers.

Preface

This study was initiated as a result of the increased interest by the agency in the development of permanently manned space stations and the resolve that some type of return vehicle must be provided; one that is docked at the space station and available for use. This document is intended principally to provide descriptions and weights of the various subsystems for a lifting body return vehicle. Vehicle costs are also included.

The paper was prepared in draft form in August of 1992 but was not published at the time. It is being published now inasmuch as the information contained herein still seems to be relevant to current NASA space efforts, especially those centered around space station. The original study was initiated by Delma C. Freeman, Jr., head of the Vehicle Analysis Branch at the time. These smaller (than Shuttle) vehicles proposed (principally for crew transports) have been variously referred to as Assured Crew Return Vehicles (ACRVs) or as Crew Emergency Return Vehicles (CERVs). The vehicle described herein is referred to as a Crew Exigency Return Vehicle.

The writer wishes to acknowledge the assistance of the following for detailed information in the disciplines identified, namely: Christopher I. Cruz, aerodynamics; Richard W. Powell and J. Chris Naftel, trajectories; Kathryn E. Wurster, heating; Charles A. Breiner (Planning Research Corporation), design; Alan H. Taylor, structures; John B. Hall, Jr., Lisa C. Simonsen, and Merle Shuey (Hamilton Standard), environmental control and life support; Dr. Steven L. Schneider (LeRC) and Carl Stechman (Marquardt), propulsion; Clyde J. May and John Giltner/Kurt Brown (Eagle-Picher), power; Howard W. Stone, Jr., and Duane R. Teske (Sunstrand), avionics; Edwin D. Dean, Arlene A. Moore, and Edward H. Bogart (Lockheed Engineering and Sciences Company), costs; Sandy M. Stubbs, landing dynamics; and Patrick A. Troutman, space station interfaces. Unless otherwise identified, the contributor is from NASA Langley Research Center.

Introduction

When the space station becomes operational, some type of return vehicle will be needed. The type of vehicle, size, crew accommodation, and other factors will undoubtable be the subject of considerable study. Even the number of vehicles to remain docked at the space station may be the subject of study, although two such vehicles appear to be a reasonable choice. Major considerations include the cost of the return vehicle versus its ability to return injured crewmen. One of the simplest shapes, and one about which much is known, is the ballistic shape used in the Apollo Program.

The purpose in this study, however, is to identify the characteristics of a lifting-body shape as a space station return vehicle and to make the information available.

Study Guidelines

The study guidelines were as follows:

- Vehicle must be deliverable to orbit in the Shuttle cargo bay
- Passenger and crew accommodations for eight
- Vehicle must have sufficient life support and power for a total of 24 hours including on-orbit loiter and entry time
- 180-day docking period between resupplies
- Runway landings
- N2-02 atmosphere at 14.7 psia with enough gas supply for one re-pressurization
- Ability to enter and land with untrained or disabled crew
- Utilization of current technology where possible to minimize costs

Vehicle Description and Characteristics

The vehicle is 24.6 ft long and is 20.9 ft wide, has accommodation for eight (crewmen and their personal effects), and has a propulsion system sized for return from a space station in a 260 nmi. orbit (Fig. 1).

The vehicle is equipped with a tricycle landing gear with pneumatic tires. An inward opening hatch is provided above and behind the pilots' compartment for docking and access between station and vehicle. Wings (or fins) are equipped with two position locks so that they can be pivoted upward for transport and can be deployed for reentry. The vehicle is equipped with small (25 lbT) thrusters for attitude control and docking, and two (870 lbT) motors for the de-orbit maneuver.

Because of the long periods of relative inactivity of the vehicle (180 days) and quick response required, all storable propellant systems are used. For the same reasons, auto-activable batteries are used for power instead of fuel cells or auxiliary power units.

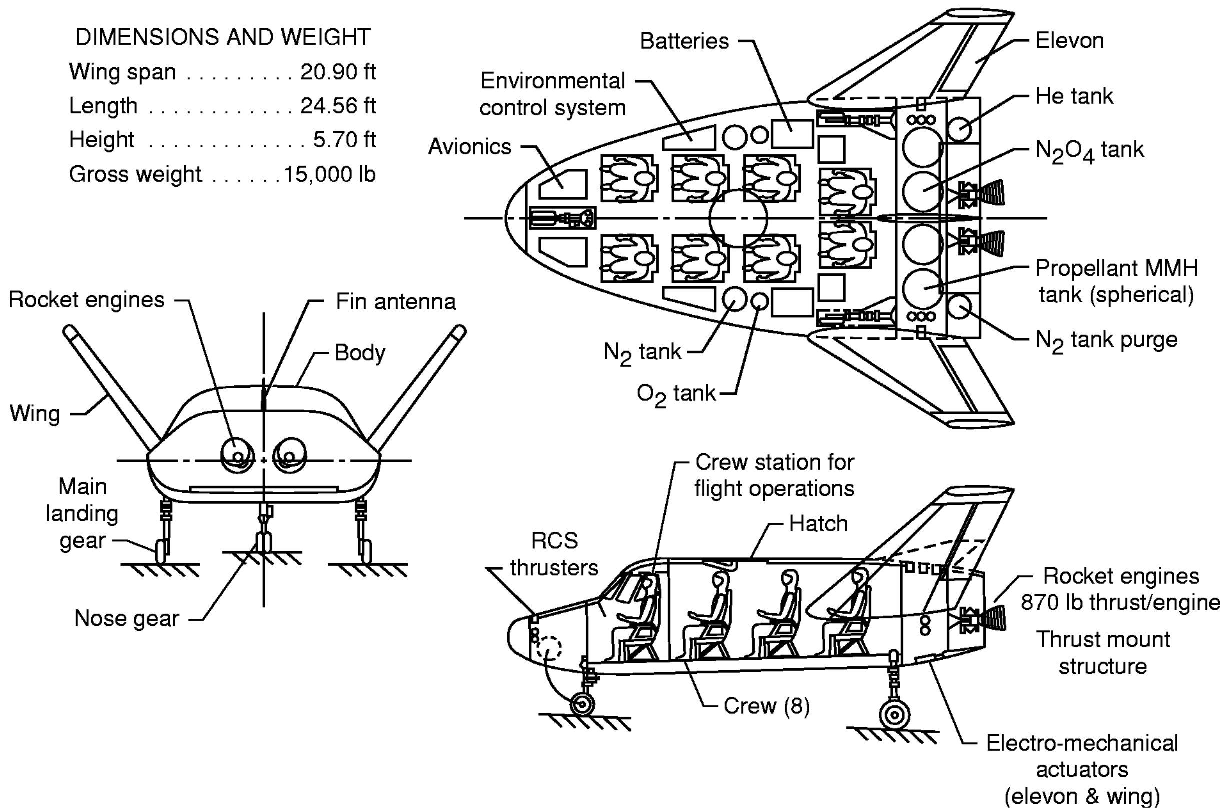


Figure 1. Crew Exigency Return Vehicle inboard profile (subsystems).

The following are some of the characteristics of the CERV as they relate to the space station return application.

Overall Geometry

The lifting body shape is particularly suitable for the requirements set forth in the guidelines since the body volume is large for any given constraints on width and height—in this case the Shuttle cargo bay 15-foot diameter constraint. A (more conventional) vehicle with equivalent body volume and horizontal wings and a vertical tail would be more difficult to store in the circular-cross-section Shuttle cargo bay.

Heating

From the standpoint of entry heating, the CERV (and lifting bodies in general) tend to have lower entry planform loadings than ballistic shapes. Since the lower planform loadings result in somewhat lower peak-heating rates, the design requirements for the heat shield for the lifting body shapes are less stringent than for the ballistic body designs.

When comparing the lifting body to a more conventional winged vehicle, sized for the same body volume, the radii of the nose cap and other body radii are greater. This tends to result in lower temperatures on leading edges and nose cap. The wings are canted at an angle of 50 degrees to the horizontal and are highly swept. This geometry is such that the wings fall within the bow shock during the peak heating period at entry, reducing the requirements for the thermal protection.

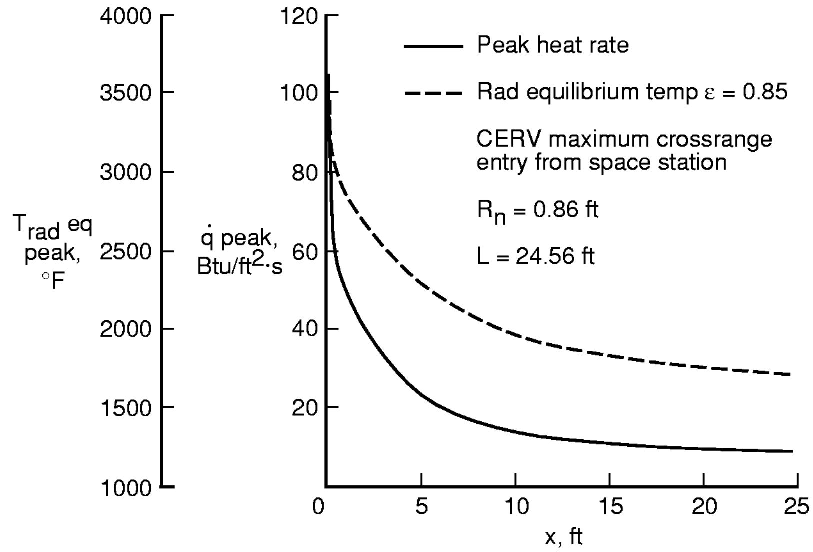


Figure 2. CERV windward centerline peak heating rate.

The windward centerline peak-heating rates and radiation equilibrium temperatures for the CERV are shown in Fig. 2. The peak stagnation temperatures on the nose cap of the CERV are estimated to be several hundred degrees higher than those for the Shuttle, but are estimated to be within the capability of the reinforced carbon carbon (RCC) presently used, particularly since the vehicle will seldom, if ever, be used. The projected low-use scenario drastically reduces the design requirements on the thermal protection material.

Flight and Landing Characteristics

The CERV has a hypersonic L/D of 1.6 to 1.8 and a subsonic L/D of about 3.5. The estimated cross range of the vehicle is 900 nmi. Trajectories have been run for entries from a space station in a 28.5 degree orbit. The assumed entry angle for the CERV is 1.5 degrees.

Five landing sites are required in order to provide for a landing from any space station orbit. These are the runways at the Kennedy Space Center; Edwards Air Force Base; Hickham Air Force Base in Hawaii; Guam; and Dakar, North Africa. These five sites would allow a performance reserve of 250 nmi. and provide for a maximum-sensed acceleration during the entry of 1.5 g's. An example ground track into the runway at the Kennedy Space Center is shown in Fig. 3. Figure 4 shows landing footprint comparisons between the current shuttle and the CERV. Figures 5 and 6 show the time histories of selected state variables for this entry from a 262 nmi. altitude.

Some experimental results give indications as to the accelerations to be expected at touchdown for a lifting-body and a ballistic shape (Refs. 1 and 2). A sample trace of an HL-10 lifting body equipped with skids and a nose wheel, shows a 30 millisecond spike in normal acceleration up to 4.6 g's at main-gear contact. This is followed by a second short duration normal acceleration of about 3 g's at nose-wheel contact. The tests were conducted using a vertical velocity of 10 ft/sec and a horizontal velocity of 205 to 213 ft/sec. Short duration vertical peak accelerations ranged between 4.6 and 5.6 g's for five tests (Ref. 2). Since the current CERV has pneumatic tires on both nose gear and main gear, the peak accelerations should be even less than the HL-10 test article. The seat cushions and support system would have an even greater effect in moderating the peak accelerations and would have to be considered in determining the actual accelerations experienced by the crew.

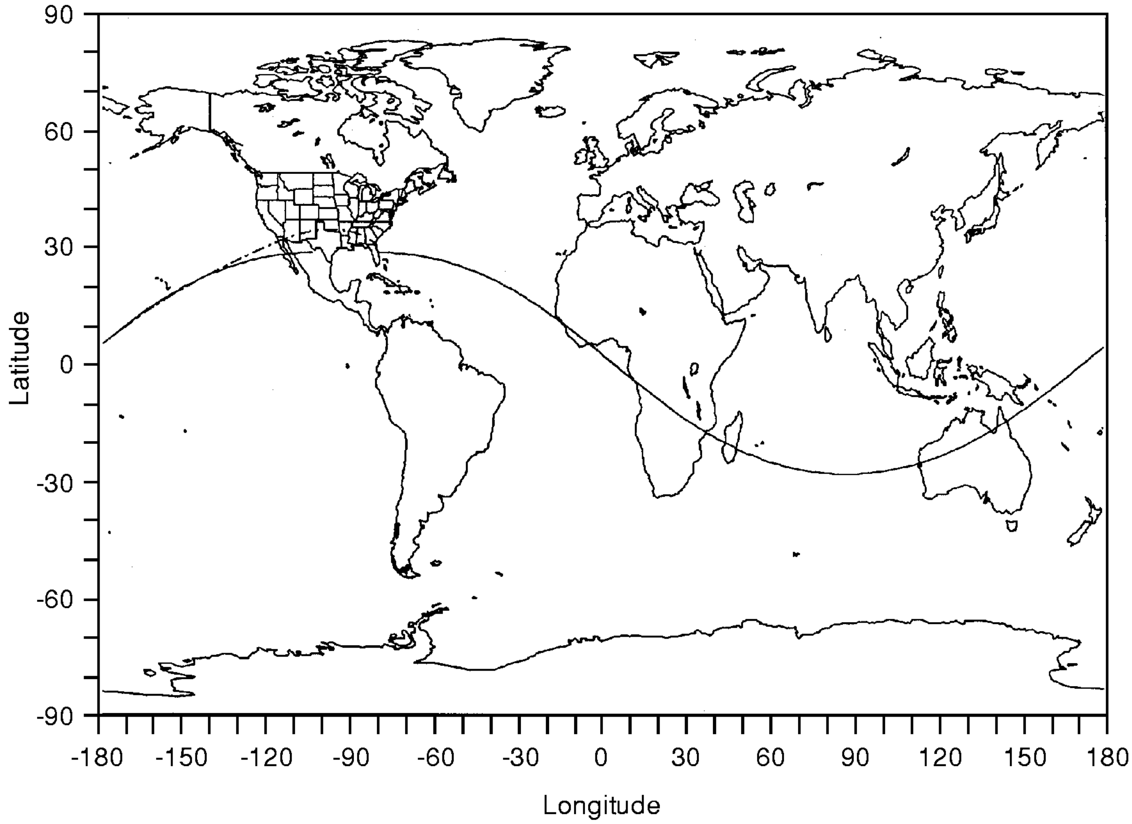


Figure 3. CERV entry into KSC from orbit #1 from 262 nmi. orbit.

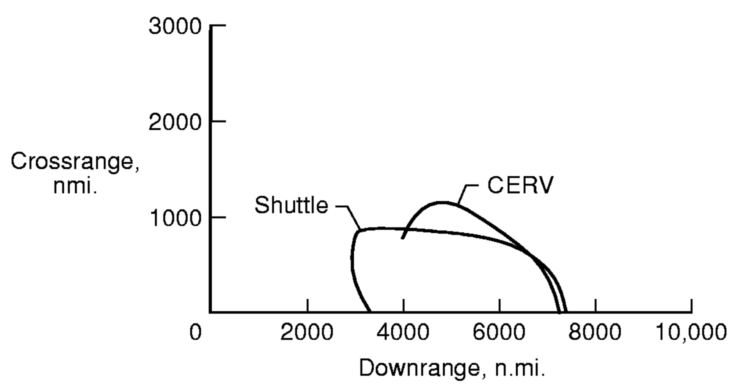


Figure 4. Landing footprint comparison.

Numerous landing tests have been conducted on a scaled model of a ballistic (Apollo) shape. In one series of tests, the maximum normal accelerations are shown for an impact velocity of 23 ft/sec and various water-entry angles (Ref. 1). For a water-entry angle of 30 degrees (a corner entry for the base heat shield), the maximum normal acceleration recorded was about 3 g's. For a zero-degree entry angle, the peak accelerations reached were about 30 g's. The horizontal velocity of the capsule at contact with the water seemed to have little effect on the maximum normal accelerations obtained.

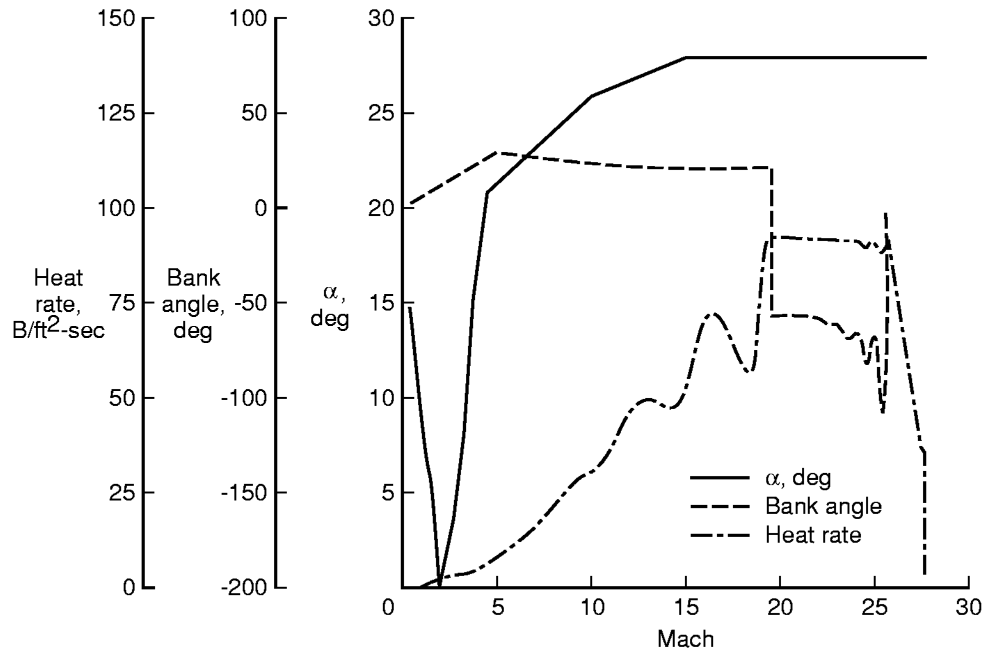


Figure 5. Selected state variables versus Mach number for entry into KSC from a 262 nmi. orbit.

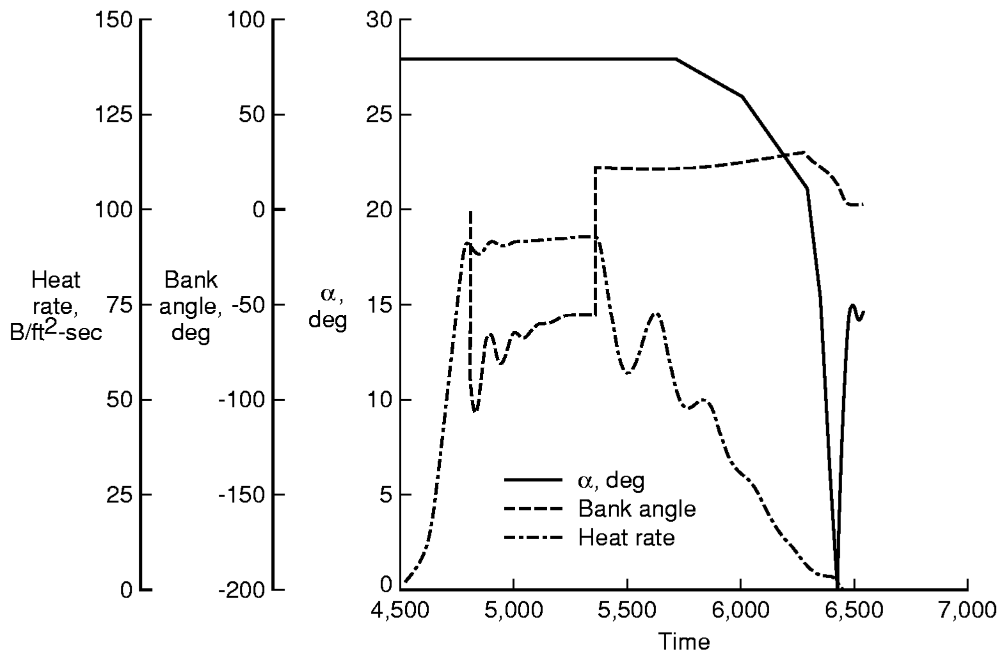


Figure 6. Selected state variables versus time for entry into KSC from a 262 nmi. orbit.

Vehicle Mass Properties

The vehicle mass properties (Table I) are based on the subsystems as described in the following paragraphs. Subsystems details are included in the section entitled Subsystem Characteristics.

Structure and Thermal Protection

The wing, tail, and body constitute the basic structural groups for the vehicle. The body shell, the pressurized crew compartment, the engine thrust structure, and the body flap are included in the body group. The thermal protection system (TPS) includes the nose cap and leading edge pieces, and acreage body TPS used externally. Flexible and bulk insulations are used internally in selected areas, but especially on the leeward side of the vehicle under the titanium body shell not protected externally.

Propulsion

The maneuver and reaction control engines and propellant tanks are listed in weight category 7 in Table I. The pressurization, pneumatics, and purge are listed in category 8. The dry weight of the reaction control system (RCS) and main propulsion system (MPS) combined (including pneumatics) is 466 lb. Propulsion system component weights, from which the overall weights were obtained, are shown in Table II.

Table I. CERV Mass Properties

		Weight, lb
1.0	Wing group	289
2.0	Tail group	0
3.0	Body group	4,480
4.0	TPS	980
5.0	Landing and auxiliary systems	390
6.0	Main propulsion	0
7.0	Propulsion systems (OMS & RCS)	329
8.0	Pressurization, pneumatics & purge	137
9.0	Prime power	1,530
10.0	Electrical conversion and distribution	266
11.0	Surface controls	92
13.0	Avionics	651
14.0	Environmental control	939
15.0	Personnel provisions	546
16.0	Margin	1,063
	Inert weight	11,692
17.0	Personnel (clothes, etc.)	1,925
18.0	Payload accommodation	0
19.0	Payload returned	0
20.0	Residual and reserve fluids	180
	Landed weight	13,797
22.0	RCS propellant	124
23.0	OMS propellant	620
24.0	O_2,N_2, H_2O, He, and NH_3	449
	On-orbit weight	14,990

Table II. Propulsion System Weight Estimates

System/Item	Quantity	Weight/Unit, lb	Weight, lb
Pressurization System			
He pressure level	1	7.2	7.2
Pressure vessel	4	3.9	15.6
Valves	6	3.4	20.4
Fill or vent	3	0.6	1.8
Check valve	8	3.4	27.2
Relief valves	2	4.7	9.4
Plumbing		10.2	10.2
Propulsion System			
MPS thrusters	2	16.0	32.0
RCS	17	3.5	59.5
Propellant tanks	2	21.4	42.8
Valves	88	1.7	149.6
Filters	2	0.8	1.6
Fill or vent	4	2.2	8.8
Plumbing		35.1	35.1

Table III. Propulsion System Parameters

	MPS	RCS
Propellants	MMH/N_2O_4	MMH/N_2O_4
Thrust, lb	1100	32
Feed pressure, psia	320	320
Mixture ratio, (O/F)	1.65	1.65
Area ratio	100	100
Specific impulse, sec	305	292
Minimum impulse, lb-sec	25	0.2
ΔV, ft/sec	426	80
Propellant mass (20% reserve), lb	775	155
MMH mass (20% reserve), lb	293	59
N_2O_4 mass (20% reserve), lb	482	96
MMH volume (20% reserve), ft^3	5.5	1.2
N_2O_4 volume (20% reserve), ft^3	5.5	1.2

The total required propellant mass is 924 lb, which includes a 20-percent residual plus reserve (Items 20, 22, and 23 in Table I). These propellant requirements are derived from the thruster specific impulses, the requirement of 426 ft/sec delta-V for the de-orbit maneuver, and an 80 ft/sec ideal velocity equivalent for attitude control. The parameters used in sizing the MPS and RCS system are given in Table III.

The main propulsion system propellants are stored in 26.3-inch- diameter capillary acquisition tanks weighing an estimated 17.7 lb each. The RCS propellants are stored in 15.3-inch-diameter bladder tanks weighing an estimated 3.7 lb each. Bladder tanks have successfully operated on the Mariner vehicle. The helium pressurization storage vessel is sized by isothermal blowdown from 4500 psia to 320 psia at 530 degrees Rankine. This vessel is estimated to be 9.3 inches in diameter and to weigh 7.2 lb. It contains 0.7 lb of helium.

The nitrogen pneumatic system is estimated to be about one-quarter the size of the helium system. Nitrogen is stored in a 6-inch-diameter tank at 4500 psia. The estimated tank weight is 2.0 lb and contains 1.5 lb of nitrogen.

Prime Power and Surface Controls

The prime power system weight (Category 9) includes the batteries, the bus bar, the vents, the auto-activation system, and the installation hardware. The electrical conversion and distribution system (Category 10) includes all the conversion equipment, control units, electrical wiring, and lighting systems.The surface controls (Category 12) includes the control surface actuators and cockpit controls.The assumptions for the prime power systems weight calculation are listed in Tables IV and V.

Avionics

The avionics includes the guidance, navigation and control, communication and tracking, and the displays and controls systems. The detailed functions (required for a lifting body capable of entry and landing on a runway) are itemized in Table VI. Various options were considered for the various components. Each subsystem selection, if studied in depth, would require consideration of the weight, power, volume, reliability, and cost of each as related to the goals and mission. Information on some of the candidate avionics subsystems (in varying degrees of detail) is given in Tables VII through IX. The total avionics system estimated weight, based primarily on lowest weight, is 651 lb (Table I).

Table IV. CERV Actuator Energy Calculations

- Power requirements – based on Shuttle design Criteria
  - Body flap = 1,329 Watts × 2 = 2,658 Watts
  - Elevon = 414 Watts × 2 = 828 Watts

    3,486 Watts (peak)
- Assumptions
  - Body flap rate = 25°/second
  - Elevon rate = 20°/second
  - Operational time = 10 minute
  - Duty cycle: 10% time at peak load

    90% time at 10% peak load
- Power consumption = 111 Watt-hours

  Safety factor of 3, P = 333 Watt-hours
- Power source options
  - Silver-zinc – auto activate, not rechargeable
    Batt. wt = 36 lb
  - Nickel-cadmium – rechargeable, power switching req'd
    Batt. wt = 52.6 lb/VDC

Notes:

1. If 270VDC actuators are used, 9 NiCd batteries would be required. Wt = 475 lb
2. DC/DC converters could be used to obtain the 270VDC from 28VDC batteries.
3. NiCd batteries (158 lb) and 4 converters (50 lb) = 208 lb

Table V. CERV Prime Power Weights Versus Mission Duration

- Estimated continuous power = 1.0 kW
- Energy source: automatically activated silver-zinc batteries

Misson Length, hr	Total kW-hr	Est. kW-hr Available	Batteries Required	Weight, lb	Volume, ft^3
3	3	4.75	4	160	2
6	6	9.50	8	320	4
9	9	14.25	12	480	6
12	12	19.00	16	640	8
15	15	23.75	20	800	10
18	18	28.50	24	960	12
21	21	33.25	28	1120	14
24	24	38.00	32	1280	16

Table VI. CERV Software Functions Required

- Nav filter
- Nav sensor sops
- Air data processing
- Guidance – entry and landing
- Flight control – entry and landing
- Redundancy management
- Displays and controls
- Actuation
- RCS
- Main engines
- Power management
- Hatch lock

Throughput estimate .... 200 KB/sec
Core estimate .... 200 K Words (16 bit)

Table VII. Telemetry (Voice and Tracking Beacon Only)

	Power, Watts	Weight, lb
• S-band Transceiver (TRW) NASA Std Equipment	37	14.5
• S-band Transceiver (Rockwell estimate)	100	25
• Beacon Transponder (Rockwell estimate)	50	10

Table VIII. CERV Processor Options

	Performance MIPS (DAIS)	Core	Power, Watts	Weight, lb	Cost, $K
• AP 102 (1750A) IBM Fed. Systems	1.0	1024K	114	23.5	
• PACE (1750A) Performance Semiconductor	2.6	variable	2		
• Fairchield 9450 (1750A)	0.7	variable	3		
• Magic V (1750A) Delco	0.7	512K	55	28	
• SPC 1750A RCA	0.5	512K	26	33	
• MDC 281 (1750A) MacDac	0.75	variable	2		
• Sandac IV Sandia	3.5		20	7	75

Table IX. Navigation Instruments

	Power, Watts	Weight, lb	Cost, $\$10^6$
• IMU			
• LINS (Honeywell) - strapdown	85	43	2.0
• IUS (Hamilton Std) - strapdown	88	88	
• Shuttle Replacement (Kearfott) - gimbaled	130	57	
• TAI (Kearfott) - strapdown (missile qualified)	60	24	
• Hemispherical Resonator Gyro, HRG, (Delco)			
• Startracker/Scanner			
• NASA Std (Ball Bros.) Tracker	18	17	2.0
• CT 411 Shuttleorbiter (Ball Bros.) Tracker	20	17	2.5
• DMSP (Honeywell) Scanner	1	6	1.0
• GPS Receiver/Antenna			
• Magnavox	45	45	1.5
• Collins	30	20	.7 (+ 8×10^6 for dev)
• TI	65	40	1.5
• Rockwell (AOA Heritage)	73	26	
• Antenna (Watkin/Johnson or Ball Bros.)	5	2	.1 (for new dev)
• Landing System			
• Bendix ILS (Boeing 767)	80.5	11.3	
• Shuttle MSBLS	213	83.7	
• Radar Altimeter			
• Bendix (Boeing 767)	28	11	
• Shuttle	36	8	
• Air Data System			
• SEADS	30	100	
		600 (including nose cap)	
• Probes (probably deployable)	256 +1000 heaters	150	

Environmental Control, Life Support, and Personnel

Category 14, environmental control and life support system (ECLSS), includes the oxygen and nitrogen supply subsystems, the water and waste management systems, the air temperature/CO2 removal and humidity control, and the heat rejection systems (Table X and XI). The category weights in Table I were obtained from these detailed weights. The personnel provisions include seats for eight passengers and crew with an allowance of 64 lb for equipment stowage, restraints, and fire detection and prevention equipment (i.e., the last three items in Table XII).

Category 17, personnel, includes an allowance of 165 lb for each passenger, 20 lb for clothing, and 605 lb for food, medical supplies, emergency medical equipment, survival, and rescue gear (Tables XIII and XIV). If all crewmen wore suits, as shown, and a crew of eight were returning, the spacecraft would exceed the baseline landed weight of 13,797 shown in Table I. The passengers and crew are shown in full pressure suits to show that this attire is a viable option from the standpoint of seating space for any one of the eight seat locations. Categories 22 and 23 include the usable propellants for maneuver and attitude control for separation from the space station and entry. Category 24 includes the cabin atmosphere supply for nitrogen and oxygen, water for drinking and the heat rejection systems, and ammonia for the heat rejection system for use below 100,000 feet.

Table X. CERV Environmental Control and Life Support Summary

Subsystem	Wet Weight, lb	Dry Weight, lb	Volume, ft^3	Power, kW
LiOH	114	96	4.0	0.22
High pressure O_2	116	82	3.2	0
High pressure N_2	207	148	6.2	0
Air temperature & humidity control	90	90	1.7	0.21
Fresh H_2O supply	119	56	1.7	0
Waste H_2O supply	37	37	1.7	0
Urinal	38	38	0.7	0.16
Solid waste (fecal bags)	12	12	0.7	0.0
2 gas controller & monitoring	60	60	3.5	0.05
Vent & relief	10	10	0.7	0
Ventilation	15	15	6.0	0
Total	818	644	30.0	0.64

Table XI. Heat Rejection System

Item	Weight, lb
Flash evaporator, valves, controls & plumbing	195
Heat transport loops & heat exchangers	100
Dry weight	295
Fluids	
Water	275
Ammonia	10
Total	580

Table XII. Personnel Provisions

Item	Weight, lb Number of Crew		
	2	4	8
Seats	140	250	482
Equipment storage	15	20	30
Restraints	20	20	20
Fire detection and prevention	14	14	14
Total	189	304	546

Table XIII. Personnel

Item	Weight, lb Number of Crew 2	4	8
Crew @ 165 lb each	330	660	1320
Clothes	40	80	160
Food, medical supplies, and emergency medical equipment	75	100	145
Survival and rescue	200	250	300
Totals	645	1090	1925

Table XIV. Emergency Medical*

Item	Weight, lb
Small suction apparatus	15
CPR (firm support table)	25
Emergency supply of drugs (large briefcase)	10
Defibrillator monitor	15
Supplies of IV fluids (500 mm, enough for three persons)	20
Emergency oxygen	10
Total	95

*Note: Above details were used in previous table for medical supplies for 8 man crew. Food weight allowance equalled 50 lb.

Overall Mass Properties and Performance Considerations

The weight for the 24-hour mission vehicle for an eight man crew at separation from the space station is approximately 15,000 lb. The landed weight is 13,797 lb (Table I). The landed weight drops substantially as the specified mission duration is shortened (Fig. 7a). This decrease is primarily due to the reduced battery system weight — a weight that is nearly linear with the required lapsed time of the mission. The estimated landing speed for the baseline with eight crew and batteries for a 24-hour mission is 175 knots at a 20 degree angle of attack (Fig. 7b). Any weight growth will cause an increase in the nominal landing speed and, therefore, an increased demand on the landing gear and control systems. Based on currently available tires, the nominal tire limit from the standpoint of landing speed is about 220 knots.

Except for landing speed, the weight growth would have little adverse effect on the utility of the system. For example, the 15,000 lb gross of the vehicle is well within the Shuttle delivery capability. With regard to the cost of delivery of a heavier vehicle, this would be relatively small for the lifetime of the vehicle since the application is one of stand-by and not one of repeated deliveries to orbit.

The center of gravity of the vehicle is located at approximately 54 percent of length in the empty condition and 55 percent of length with propellant and personnel. Because of the relatively large weight of the crew compared to the vehicle, and because of the potential differences in weight between crew, especially after a long space mission, an accelerometer package is proposed to measure vehicle weight, moments of inertia, and center of gravity location prior to entry (Ref. 3). This information would be of assistance in setting the trim of the vehicle and for offsetting any weight asymmetry in the spanwise direction prior to

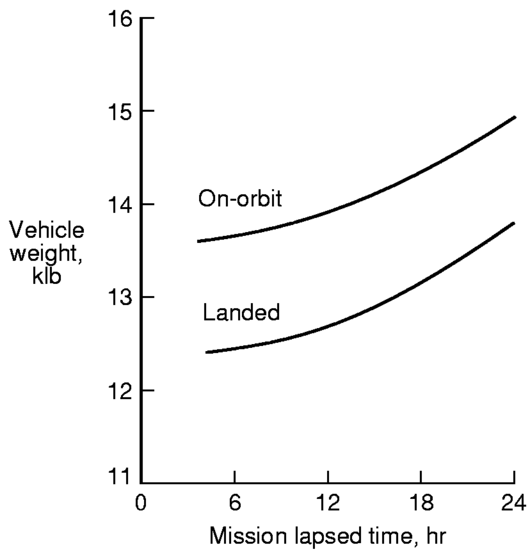


Figure 7a. CERV weight versus mission duration.

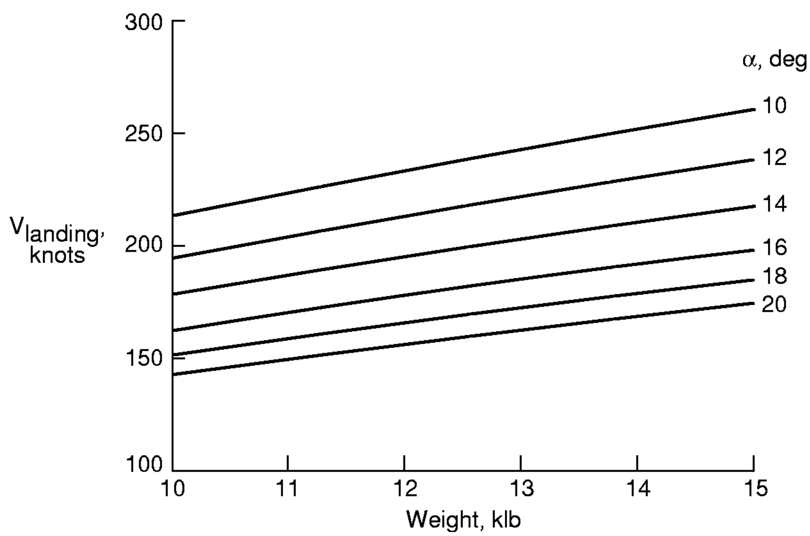


Figure 7b. CERV landing speed estimates.

entry. This weight management, prior to entry, could be accomplished by pumping fluids laterally or axially in the vehicle or by adjusting seats in axial and transverse seat tracks using electrical or mechanical means. Adjustments in vehicle balance would be made using several micro-maneuvers (or functionally required mission maneuvers). The results for total weight and the coordinates for C. G. would be obtained from an average of the values obtained from the maneuvers.

Subsystem Descriptions

Structure and Thermal Protection

The body structure consists of titanium ringframes and skin (Fig. 8). Most of the body structure is minimum gauge due to the very low loading conditions. As a weight-savings feature, the titanium skins on the body are allowed to elastically buckle, a weight-savings design feature made possible because the durable TPS is mechanically fastened (Ref. 4). A viable alternative to durable (fasten-on) TPS and titanium-buckled skin is an aluminum-non-buckled skin structure with Reusable Surface Insulation (RSI). Any final selection for RSI and structure might be driven by considerations of availability and cost. Operational costs should not be a primary driver, however, if the vehicle is used infrequently such as for emergency return.

The engine thrust structure and much of the landing gear structure is boron aluminum metal-matrix-composite material. The pressurized crew compartment is a separate aluminum shell suspended within the titanium body shell. The wings are made of a superalloy honeycomb sandwich with a maximum temperature capability of 1600 degrees Fahrenheit. The body flap and elevons are configured as hot structures, fabricated from an advanced carbon-carbon composite (ACC).

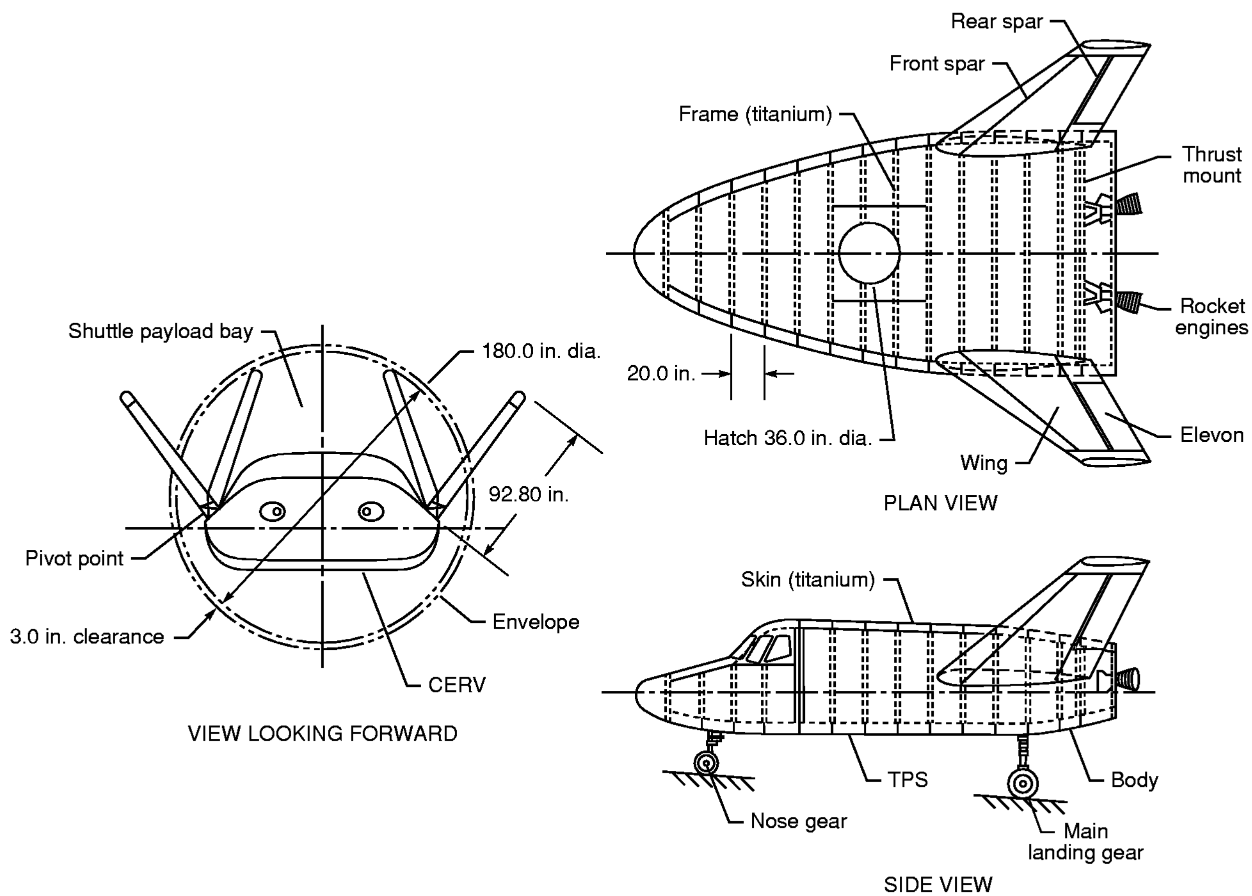


Figure 8. Crew Exigency Return Vehicle structural concept.

Landing and Auxiliary Systems

The landing gear used is derived from high performance fighter plane technology. In the interest of weight and volume savings, single wheels are used on the main gear. Dual wheels could be used, but the necessity for using them for added redundancy is questionable. Any reasonable required level of redundancy could be built into a single tire.

Propulsion

The Main Propulsion System (MPS) and Reaction Control System (RCS) engines on the CERV are derivatives of the primary and vernier thrusters on the current Shuttle. Also, the architectures of the pressurization and feed systems are similar to the Shuttle (Ref. 5). A notable exception, however, is the use of a common pressurization system for both the MPS and the RCS (Fig. 9); commonality being used because of the very small amount of helium pressurant required, particularly for the RCS system. A gaseous nitrogen (GN2) system is used for backup operation of the thruster control valves and for purging the fuel and oxidizer passages downstream of the engine valves.

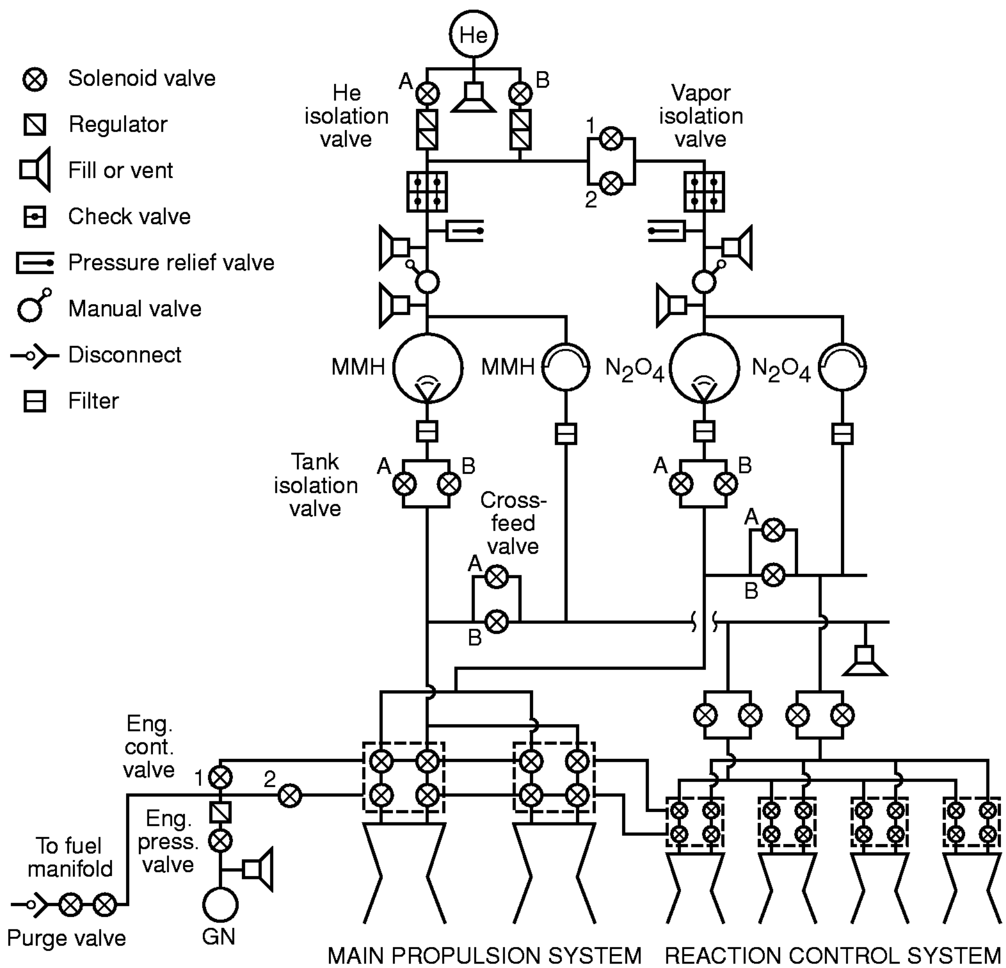
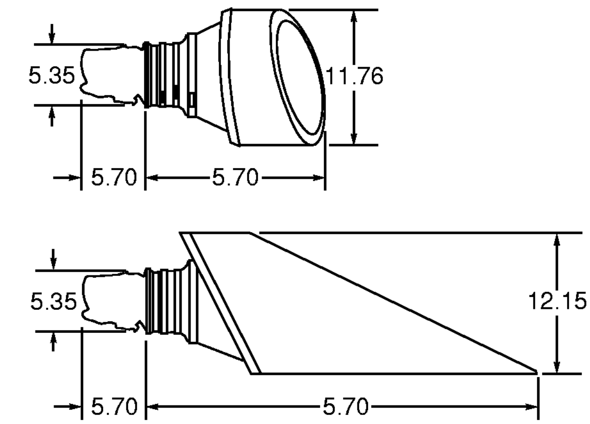


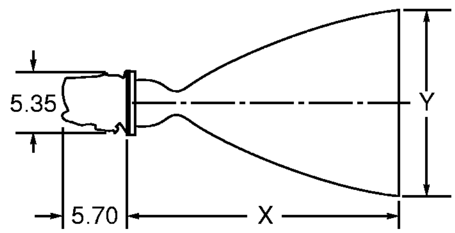
Figure 9. Propulsion system schematic.

R-40A
SHUTTLE ORBITER VERSIONS



	Nominal	Range
Thrust:	900 lb	600-1300 lb
Feed Pressure:	238 psia	150-400 psia
Mixture Ratio:	1.65	
Weight:	16.0 lb (ε = 100)	
Specific Impulse:	309 sec (modified)	} ε = 100
	298 sec (shuttle standard)	

R-40B
ORBIT ADJUST/PERIGEE VERSION

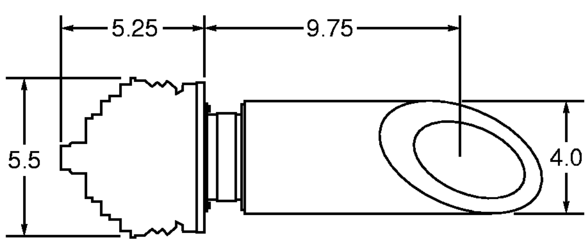


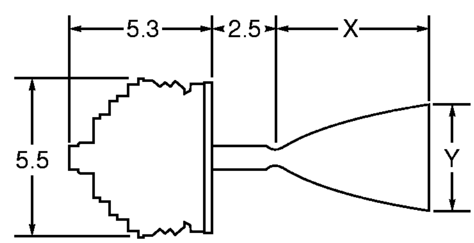
Nozzle dimensions, in.

ε	X	Y	
40	17.7	12.9	
60	22.3	15.8	
100	29.3	20.4	Configuration used in analyses
150	39.7	25.0	

Minimum Impulse Bit:	25 lb sec
Maximum Run Time:	Continuous
Power Consumption:	2.5 amps @ 28VDC
Life:	>23,000 sec
Propellants:	Nitrogen tetroxide/monomethylhydrazine, hydrazine
Usage:	Space Shuttle orbiter, orbit adjust, perigee
Status:	Qualified

Figure 10. Main engine nominal performance and envelope for model R-40 900 lb (4000 N) bipropellant rocket engine.





Nozzle dimensions, in.

ε	X'	Y'	
40	3.27	2.60	
60	4.14	3.18	
100	5.50	4.10	Configuration used in analyses
150	6.88	5.02	

	Nominal	Range
Thrust:	25 lb	15-35 lb
Feed Pressure:	220 psia	150-400 psia
Mixture Ratio:	1.65	1.0-2.7
Voltage:	28 VDC	18-32 VDC
Weight:	3.5 lb	
Specific Impulse:	290 sec (ε = 100)	

Minimum Impulse Bit:	0.2 lb sec
Maximum Run Time:	Continuous
Power Consumption:	1.2 amps @ 28VDC
Life:	82,000 sec (min. demonstrated)
Propellants:	Nitrogen tetroxide/monomethylhydrazine
Usage:	Space Shuttle orbiter, orbit adjust, attitude control
Status:	Qualified

Figure 11. Reaction control thruster nominal performance and envelope for model R-1E 25 lb (410 N) bipropellant rocket engine.

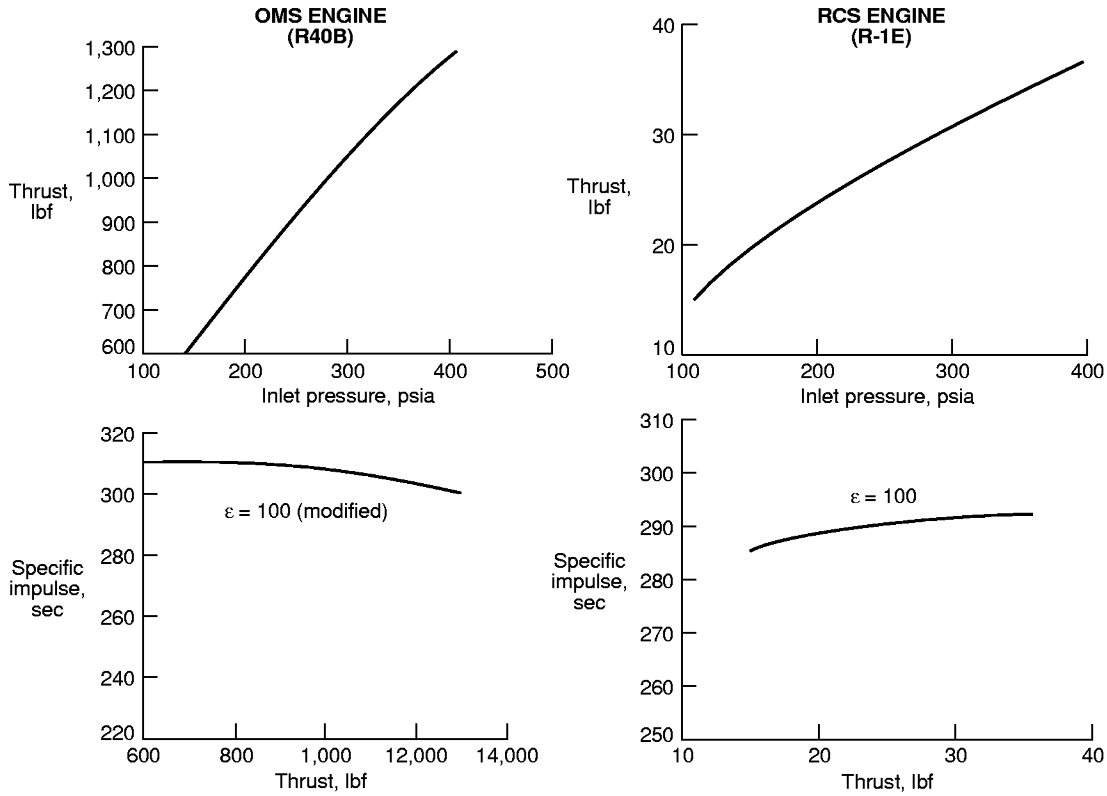


Figure 12. Bipropellant rocket engines. Oxidizer-to-fuel ratio equals 1.65.

Redundancy, typical of a single pod on the Space Shuttle, is used for valves and regulators. Manual valves are provided for ground checkout prior to delivery of the vehicle to a space station. Bladder tanks are used in the RCS system supply tanks to provide positive expulsion in a zero g environment. Propellant management in the MPS supply tanks is accomplished using a system of capillary screens and baffles.

The Marquardt model R-40B engines, with gimbal systems added, are used for the MPS, and Marquardt model R-1E engines are used for the RCS (Figs. 10 and ll). Performance characteristics of the engines from Marquardt data are shown in Fig. 12. For a thrust of 1100 lbT, the inlet pressure required for the engine is 320 psia. The corresponding specific impulse is 305 sec. These performance parameters for a modified Shuttle primary thruster are identified in Reference 6. For the RCS, the specific impulse for the model R-1E engine at the same feed pressure is 292 sec, for a developed thrust of 32 1bT.

Prime Power and Electrical Conversion and Distribution

Primary power for the CERV is supplied from automatically activated silver-zinc batteries. This type of battery was chosen for the CERV application because of its long dry shelf life (usually in years). Batteries were provided for an estimated continuous average power demand of one (1) kilowatt. The battery system is considered to be current technology, having been used frequently in various spacecraft.

The power profile for the ECLS and avionics in the CERV application is shown in Fig. 13. An estimated 1200 watts would be required for checkout of the vehicle and separation from the Space Station. The vehicle could then go into a power-down mode of about 800 watts. About 1300 watts would be required during the de-orbit phase and 1500 watts for entry and landing.

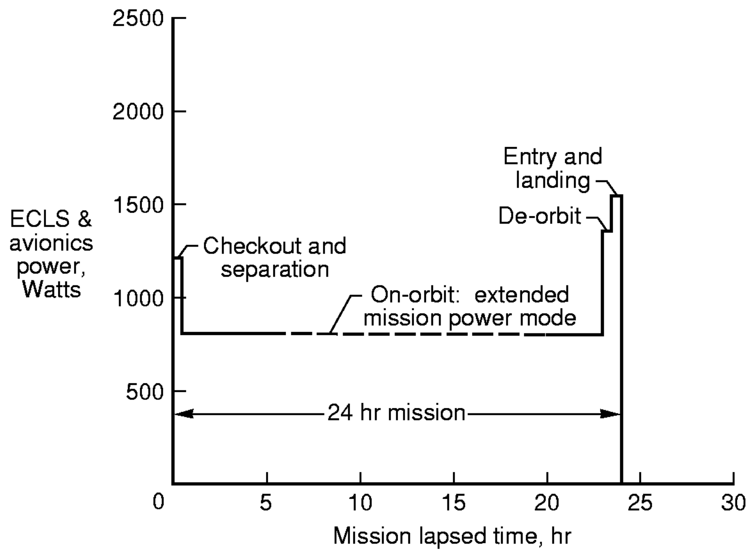


Figure 13. CERV power profile.

The automatically activated silver-zinc batteries are stored in a dry, chemically charged state prior to activation. The electrolyte (potassium hydroxide solution) is transferred from the reservoir to the cells by use of gas pressure and does not depend upon gravity or the position of the cells. This can be accomplished electrically from a remote position. An electrical pulse, supplied from a small carry-on pack of nickel-cadmium batteries, for example, sets off a gas-generating squib within the activation system, and the generated gas forces the liquid into the cells. Activation may take as long as 5 minutes in order to achieve a 24-hour wet stand life. Less activation time is required for a shorter wet stand life.

The batteries have a pressure relief vent. Gassing would be slight, but a venting system would be required. No liquid should be expelled. Each battery is rated at 40 Ampere-hours, with an estimated weight of 40 pounds and volume of one-half cubic foot, and consists of 18 cells at 1.5 volts each for a nominal voltage of 27 volts. Output is at a final voltage of 25 volts with a capacity of 44 Ampere-hours. It is possible that more than one battery could be packaged within the same case, with a common activation system to give a weight and volume savings.

Four batteries, in parallel, will be used to supply power for the ECLSS and Avionics. If a dual-bus system is employed, with loads distributed fairly equally on each bus, two batteries can be connected to each bus. At each 3-to 4-hour interval during the mission, depending upon power usage, another block of four (4) batteries could be activated when the voltage reaches some predetermined value. The 40 Ampere-hour batteries recommended were based on a 4-hour discharge rate with a 1 kilowatt load. Table V lists the number of batteries required for various mission lengths and their resulting parameters.

The actuators presently being considered for the body flap and elevons are electrically operated and require 270 Volts DC. Power requirements are based on the Shuttle design criteria. An operational time of 10 minutes would require about 110 watt-hours of power. Peak power loads would be about 3,500 watts for 10 percent of the time. A battery pack capable of delivering 3 Ampere-hours at 270 volts would be more than adequate to drive the actuators. The unit would weigh about 36 lb. A back-up battery unit should be provided. An alternate method of powering the actuators would be to use DC/DC converters. These units would operate from the vehicle 28VDC bus and provide the 270VDC required by each actuator. The power requirements and duty cycles of the actuator power system are summarized in Table IV.

Surface Controls

All actuators on the CERV are electric. The actuator elements include the controller, the motor, and the mechanical drive (Fig. 14). The motors are of the D.C. permanent-magnet type using rare Earth (samarium cobalt) magnets. This technology offers the high torque-to-inertia ratios needed for high-frequency response applications, such as primary flight controls (Ref. 7). The electromechanical actuator has been demonstrated on the roll control for a C-141 (Ref. 8).

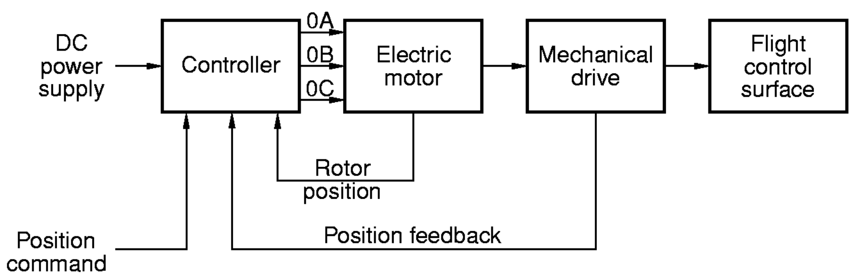


Figure 14. Electromechanical actuator block diagram.

For less-demanding applications, such as those requiring lower rates and response, induction motor systems are possible. Examples of the lower rate-response applications could include the body flap and gear actuation. For the induction motor, the control scheme is much simpler than that required for the brushless D.C. system. Some research is ongoing (at the Sundstrand Corporation) to develop switched reluctance motors for actuation applications. Though still developmental, the technology is attractive because

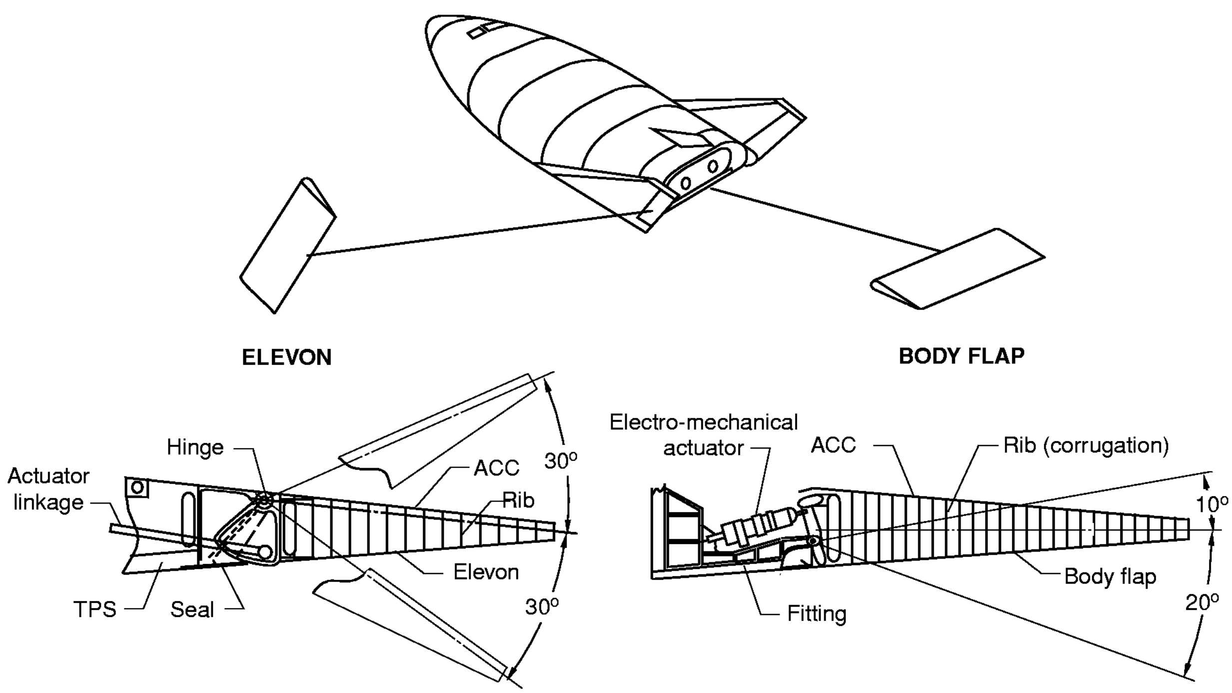


Figure 15. Surface controls.

the motor contains no magnets and is therefore capable of operating at higher temperatures and speeds. Its construction, with individually energized poles, also lends it to fault-tolerant designs. In addition, lighter weight, lower volume actuator motor controllers, are being developed that employ customized hybrid circuits. A trade between cost and weight exists, and a close matching with an already available unit may be desirable.

Both linear and rotary actuators are available. (The Shuttle uses rotary actuators on the rudder/speedbrake system and the body flaps.) For the CERV, linear actuators have been selected (Fig. 15) for all the applications because the surfaces are (thermally) thin, and heating could be a problem if the actuators were located at the control surface hingeline. In order to minimize the heating on the elevon actuators, these motor drives are also located in the body and a torsion linkage used to transmit the torque to the surfaces. High temperature-bearing technologies would be applied to render the control surface pivots operable for the entry conditions (Ref. 9). An in-depth thermal analysis would be required to determine the amount of heat sink material, insulation (or possibly active cooling) needed to carry the actuators through the entry heat pulse and heat soak at landing. The motors are generally limited by the Currie temperature of the magnets, i.e. approximately 390 degrees Fahrenheit. Typical actuators are capable of operating satisfactorily up to 275 degrees Fahrenheit. Currently, research is being conducted on rotary actuators for jet engine thrust reversers suitable for operation up to 700 degrees Fahrenheit. Active cooling for any of the above systems could be complex and expensive and probably should be avoided. Alternatively, some type of phase-change material could be used, located in close proximity to the element for which an over temperature is expected.

Avionics

The CERV avionics architecture is shown in Fig. 16. Three computers are used, each having its own channel, but with cross-strapping to another channel for redundancy. In addition, a bus interface unit is employed giving a fault-containment region so that no one bus can cause failure of the system. Similar systems are described in Ref. 10. Unique features of the system are the precision landing capability and precision air data system, the latter described in Refs. 11 and 12.

The avionics power requirements may exceed the capability of the present battery system, which was estimated at one kilowatt average early in the study. Already, in order to meet the 24-hour mission time, an extensive power-down procedure must be used while the vehicle is on orbit and separated form the space station. Units that would be shutdown to conserve power for extended stays on orbit include the radar altimeter, the inertial measuring unit, S-band transceiver, the star tracker, the landing system, and the air-data system.

A number of alternatives have been investigated for the various avionics components. All the candidate components differ in cost, weight, and power requirements. The selection process would require extensive study. The alternative components (with information available at the time of the study) are listed in Tables VII through IX (pp. 10-11).

Environmental Control

The ECLSS proposed for the CERV is a state-of-the-art open cycle system designed to sustain the crew in a comfortable environment during an exigency return from the Space Station (Figs. 17 and 18). The system is sized to accommodate a crew of eight for a 24-hour mission with a 180-day resupply interval allowed for the CERV while the vehicle is attached to the Space Station. A pressure of 14.7 psia is provided in the cabin. The cabin pressure is made up of N2 at a partial pressure of 11.7 psia, and O2 at a partial pressure of 3.2 psia. Carbon dioxide is removed from the cabin air with LIOH. The system, except for size, is very similar to the present Shuttle orbiter's life support (Ref. 13).

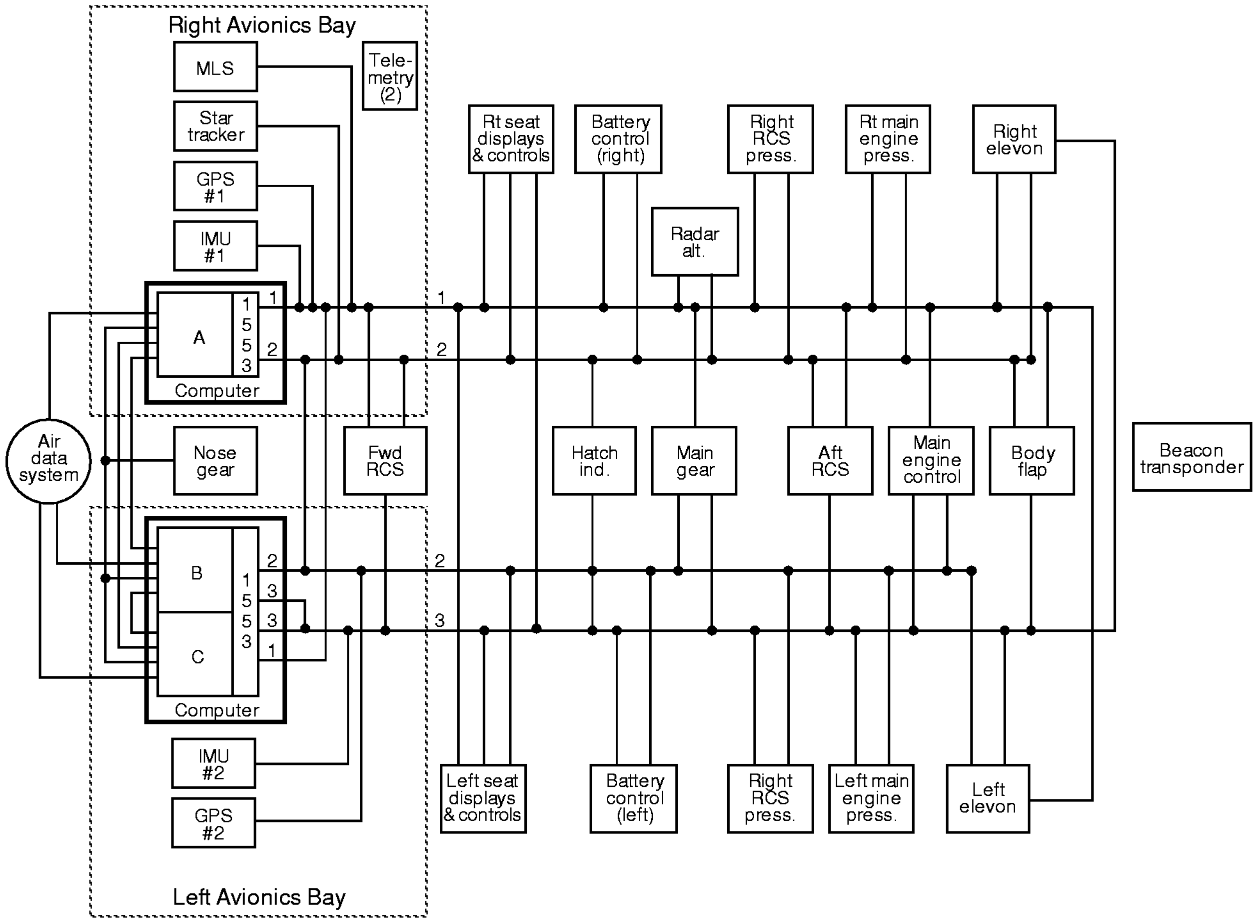
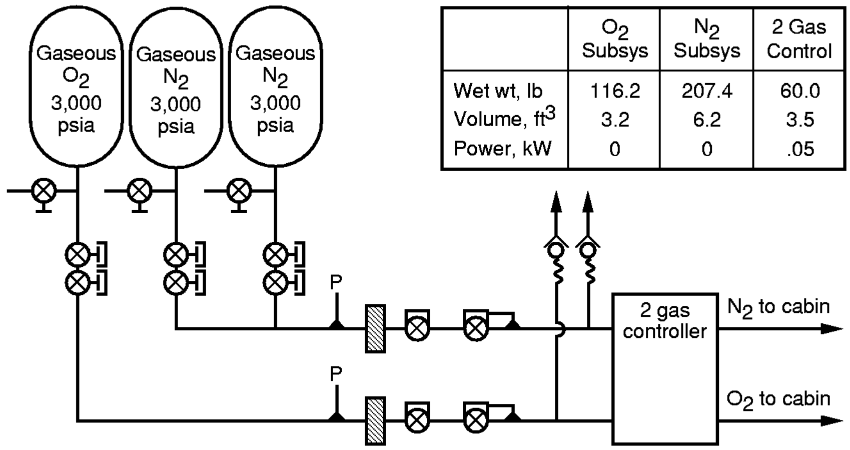


Figure 16. Preliminary CERV avionics configuration.

Contaminants and odor control are maintained with both particulate and charcoal filters. Sufficient oxygen is provided for consumption by a crew of eight for a 24-hour mission. Sufficient O2 and N2 are provided for one cabin re-pressurization and for a leakage rate of 0.25 lb/day for 180 days while the CERV is docked at space station. The O2 and N2 are stored as high pressure (3000 psia) gases and are supplied to the cabin through a two-gas controller. Humidity and sensible heat control are regulated with a single phase 35-degree Fahrenheit water loop and resistance heaters, respectively (Fig. 18).

Drinking and urinal flush water are provided from stored water tanks at approximately 10 psia (Fig. 19). Urine/flush and condensate waste liquids are collected in storage tanks while solid and trash wastes are collected in storage bags. These wastes would be removed after the CERV has touched down at the landing site. Drinking water could be carried on board, or stored in small plastic containers as an alternative to storage in tanks.

A flash evaporator is used for heat rejection. This system is similar in design to the units now used on the Shuttle Orbiter for use from 100,000 ft altitude until the cargo bay doors are opened, exposing the radiator panels to space. Whereas the Shuttle is equipped with a separate ammonia system for use during entry and landing, the CERV fluid evaporator could be configured to operate either on water or ammonia.



	O_2 Subsys	N_2 Subsys	2 Gas Control
Wet wt, lb	116.2	207.4	60.0
Volume, ft^3	3.2	6.2	3.5
Power, kW	0	0	.05

Figure 17. Oxygen and nitrogen supply subsystems.

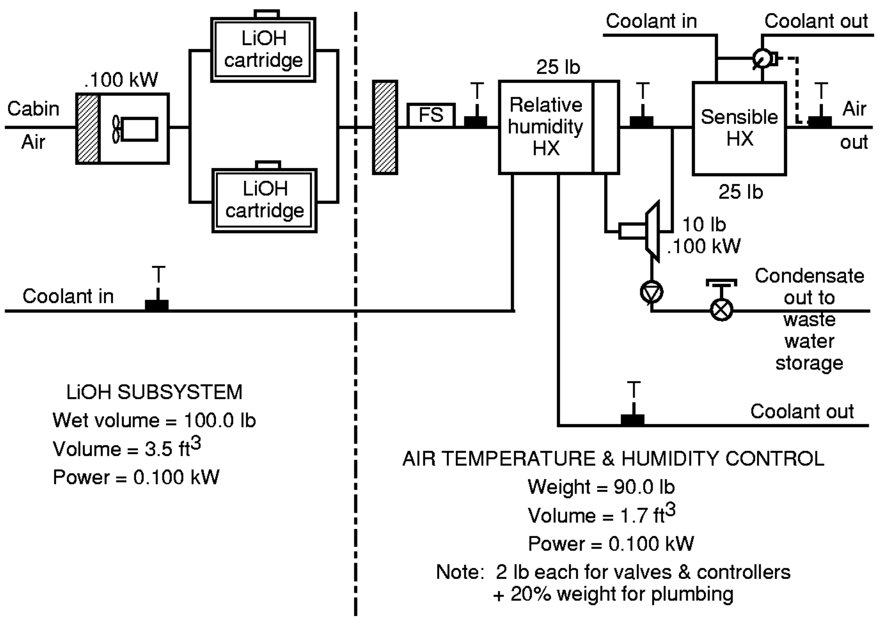
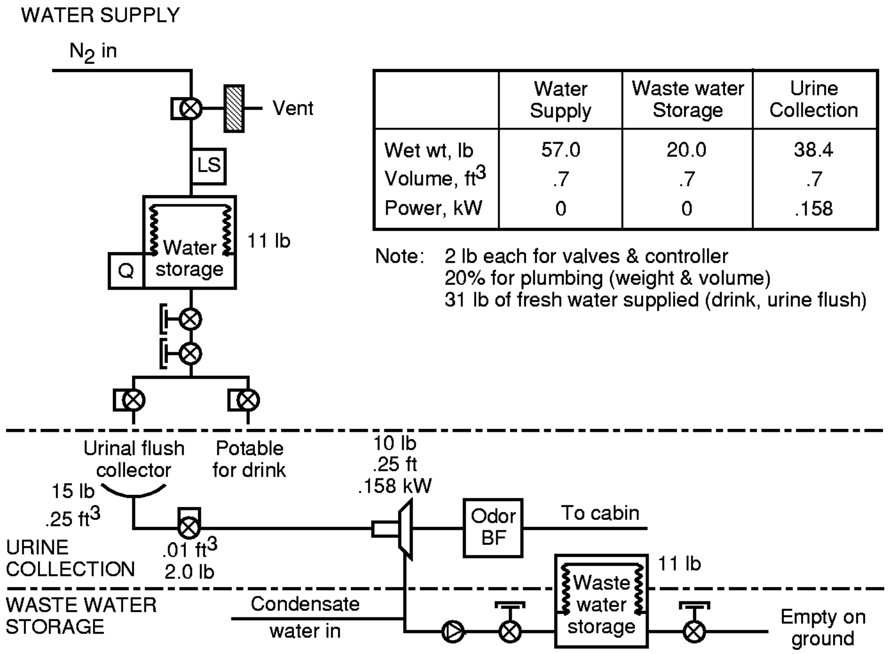


Figure 18. CO_2 removal, and air temperature and humidity control.

Personnel and Personnel Provisions

The seat weight estimates reflect an assumption of an axial load factor of approximately 1 g and are not structured for occupancy during delivery of the CERV in the Shuttle Orbiter. Also, allowance for miscellaneous personnel gear is minimal since the vehicle is configured only for return. Some emergency medical equipment and supplies are included. They include a small suction apparatus, collapsible table for CPR, emergency supplies in a small briefcase, defibrilator-monitor lifepack, supplies of I-V fluids, and emergency oxygen (Table XIV).



	Water Supply	Waste water Storage	Urine Collection
Wet wt, lb	57.0	20.0	38.4
Volume, ft^3	.7	.7	.7
Power, kW	0	0	.158

Note: 2 lb each for valves & controller
20% for plumbing (weight & volume)
31 lb of fresh water supplied (drink, urine flush)

Figure 19. Water supply, waste water storage and urine collection.

Space Station Interfaces

The space station, at the time of the study, consisted of a number of connected elements. These elements included habitat modules, open truss structure, solar panels, and radiators (Fig. 20 and 21). The habitat modules are coupled to other modules using smaller cylindrical sections having multiple docking ports. These sections are referred to as nodes. The problem for the CERV is to find two docking ports to which the vehicles can be docked for long periods of time, safely and without interference with other space station operations.

The space station used in the study (Referred to as the Phase II Configuration) has four resource nodes with open ports potentially available for CERV berthing. Since the CERV hatch diameter is smaller than the space station ports, an adaptor section is necessary. A cone-shaped adaptor, 3 ft long, was chosen providing adequate clearance for most of the docking geometries between CERV wing tips and space station.

The front of node number one is the primary docking port for the Shuttle Orbiter. (The Orbiter is shown at its alternate docking position in Fig. 20). Neither the front, back, top, nor bottom of nodes one and two are suitable for docking the CERV because of the possible presence of the Orbiter. Additionally, the right side of node one is not suitable because of the presence of the remote manipulator system located in the left side of the Orbiter near the top edge of the cargo bay (right side in Fig. 20). For this geometry, the manipulator arm would be restricted in its reach below the transverse boom. Because of the above constraints, the only port available for long-term docking of the CERV is the left side port on node number two.

Node three has the bottom and side ports available. The bottom port is a logistics port but the side port is a good CERV berthing location. Node four has only one free port but it is blocked by the servicing facility. Of the two CERV berthing locations, the one on node two offers the least hazardous escape path (assuming no berthed Orbiter). Some danger exists in a possible collision with the transverse boom during arrival departure for a CERV docked at node three.

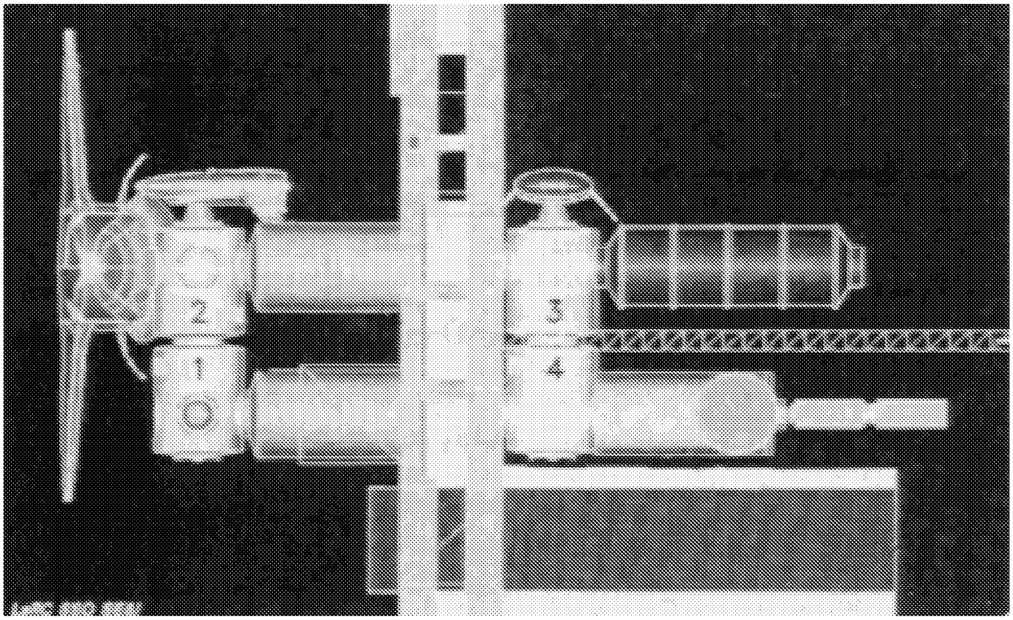


Figure 20. View of the attached CERVs from the upper boom.

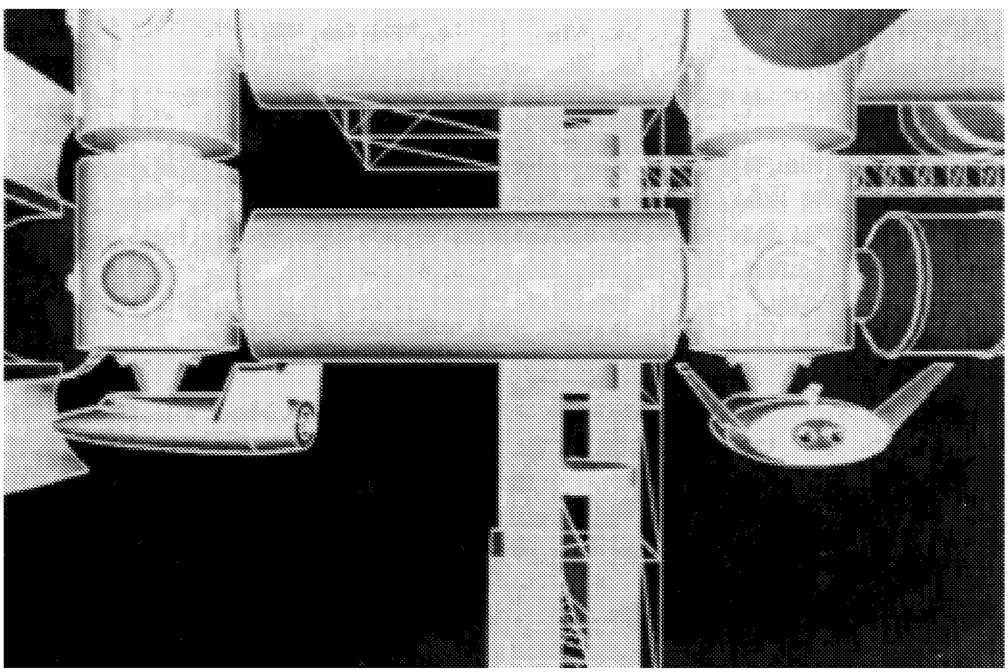


Figure 21. View of the attached CERVs from the lower boom.

Costs

The estimated cost of the CERV represents a rough order of magnitude (ROM) cost based on the vehicle configuration described in this report and on assumptions derived from the technical information generated by the conceptual design study team. Any deviation from the specified configuration and assumptions would have an impact on cost.

Assumptions

The baseline assumptions for the cost estimate are as follows:

(1) The estimate includes two production units, one certification unit, and one flight test unit. Other subsystem and component prototypes are included and vary by subsystem. This estimate represents an “as delivered” cost that includes vehicle hardware software, integration and test, contingencies, General and Administrative (G & A) and fee.

(2) There are no schedule constraints.

(3) Low and perceived estimates are based on only a 12-hour mission. The low estimate assumes that the weight constraint for the landing profile can be met utilizing an “off the shelf” technology base. The perceived estimate assumes that some subsystems might have to use state-of-the art technologies to meet the weight constraints. The high estimate assumes a 24-hour mission; some state-of-the-art and some advanced technology necessary to achieve the weight constraint.

(4) Low and perceived estimates assume that the main engines and the reaction control system (RCS) share common tankage and pressure vessels. For the high estimate, two separate pressurization and tank systems are assumed, as opposed to the common pressurization system baseline.

(5) All costs are given in fixed year 1987 dollars.

Methodology

The estimate was performed using LaRC Cost Estimating Office risk analysis (Ref. 14). In this analysis, a low, perceived, and high cost are obtained from a cost distribution curve (Fig. 22). In the cost-estimating process, a low, perceived, and high cost are estimated for each work breakdown structure (WBS) element (Figs. 23-24). The low cost is based on a set of optimistic assumptions that represents the best possible program scenario and is the lower cost bound. The high cost represents the upper cost bound and is based

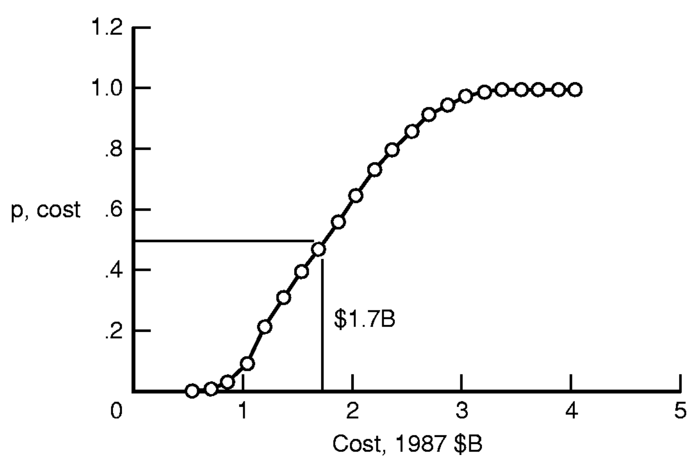


Figure 22. CERV cost distribution.

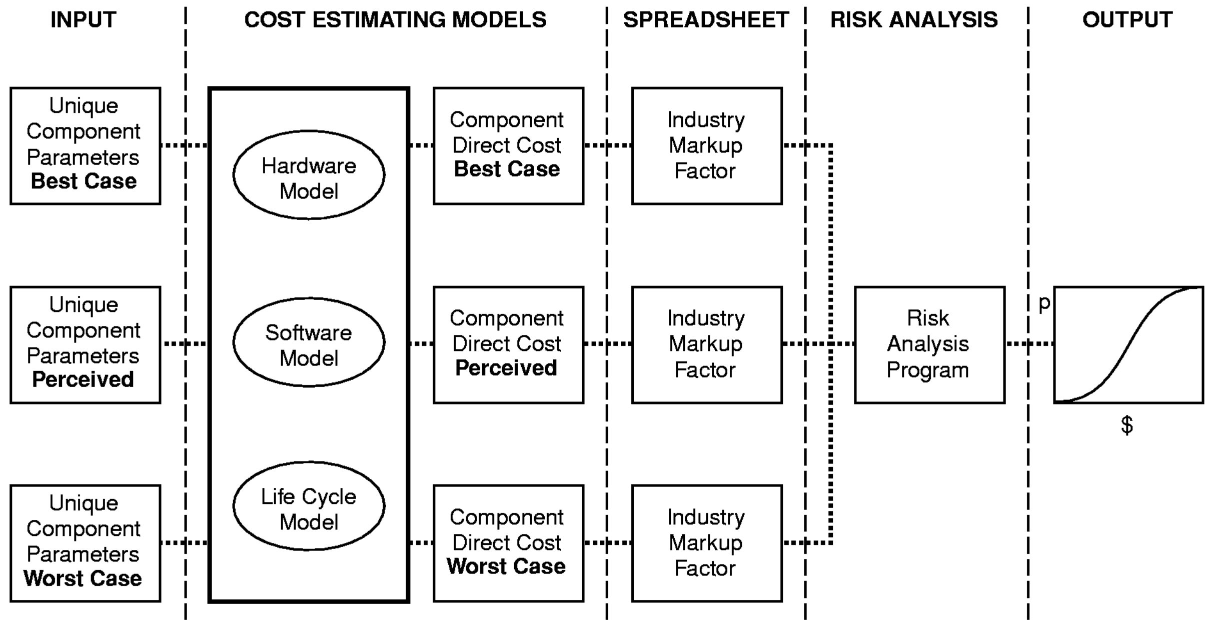


Figure 23. LaRC cost estimating process.

on a set of high-cost assumptions such as high program risks, advanced technologies, and high material cost. The perceived cost estimate corresponds to the engineering assumptions associated with the current conceptual configuration. These three separate cost estimates serve as the basis for generating a cumulative probability distribution using a Monte Carlo simulation risk analysis.

RCA PRICE Systems Price H (hardware) and Price S (Software) cost models were used. Costs were estimated at the lowest level of the WBS at which engineering detail was available (Fig. 24). This level of detail varies between subsystems. Subsystems were calibrated to Shuttle elements whenever appropriate with the advancements in technology from 1972 to present taken into account.

Results

The most probable cost for Design, Development, Test & Evaluation (DDT &E) and 2 production units is $1.7 billion. Risk analysis resulted in the CERV cost distribution curve (Fig. 22). This curve is a cumulative cost distribution in which the scale on the vertical axis indicates the probability of being less than, or equal to, the associated cost on the horizontal axis. The range of costs is $1 to $2 billion with a 0.5 probability of completing the project for less than $1.7 billion.

An analysis was made on the above estimate in order to provide a basis for comparison between the Langley Research Center (LaRC) configuration and the design concepts studied by Johnson Space Center (JSC). To normalize the LaRC estimate to the JSC estimate, programmatic costs were included, and the costs were listed in the same format as the JSC estimate. It should be noted that the LaRC and JSC studies were conducted under different sets of ground rules and assumptions. LaRC studied alternate configurations that potentially would exceed the CERV minimum mission requirements and could be developed as a multi-purpose vehicle to be used as a cargo transport, on-orbit maneuver vehicle, or for alternate access to space. JSC studied several configurations, all of them different from the LaRC vehicle. All could meet the mission objectives. The JSC estimate was extracted from presentation material entitled, “CERV Status Presentation to Langley Research Center, March 26, 1987,” and cites a range of cost from $0.7 to $1.5 billion with an

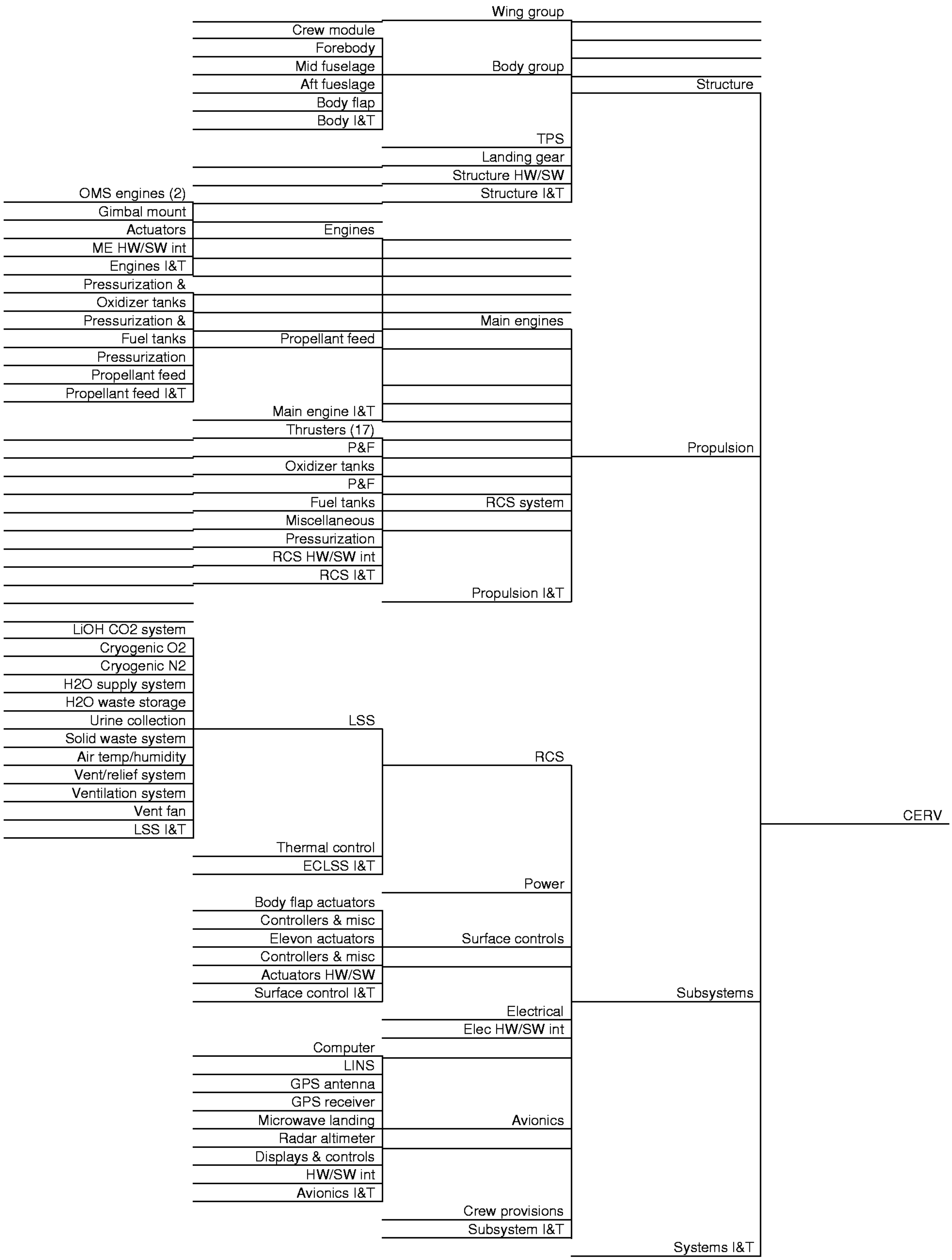


Figure 24. CERV work breakdown structure.

Table XV. CERV JSC/LaRC Cost Estimates (1987 $M).

Cost Category	JSC Estimate Cost Range Low	JSC Estimate Cost Range High	LaRC Estimate Cost Range Low	LaRC Estimate Cost Range Expected	LaRC Estimate Cost Range High
Prime Contractor	494	982	292	1193	2508
DDT&E Cert & 1 flight test unit 2 production units					
GSE/OSE/ASE SE&I/contractor wraps					
Contractor Fee (JSC = 8%, LaRC = 15%)	40	79	44	179	376
NASA Non-Prime (19%)	100	201	55	227	477
MCC updates Crew equipment Simulators/trainers/training Subsystem mgmt					
KSC ops (3 flights) Recovery (1 flight + training)					
APA (JSC = 18%, LaRC = 5,10,32%)	114	227	17	343	923
Totals	748	1,489	408	1,942	4,283
Expected cost	**1,119**			**1,942**	

expected cost of $1.1 billion. This report did not specify the specific configuration used as a basis for their estimate. The Langley Research Center configuration expected cost is $1.9 billion (Table XV).

Alternate Configurations

Some of the alternate applications considered for the basic CERV include modifications for space station resupply, for cargo transport, and for high delta-V orbital maneuvers. They are discussed in the following paragraphs.

Resupply Vehicle

The baseline CERV was reconfigured for use as a resupply vehicle to be delivered to orbit on a Titan III (Figs. 25 and 26). A crew size of 4 and 1000 lb of cargo were selected. The vehicle had to be capable of aborting from the Titan III at a peak acceleration of 8 g's. The necessary changes included increases in the weight allowances for the seats, seat tracks, wings, thrust structure, and other secondary structure in order to withstand the abort. (The basic CERV is delivered without crew in the Space Shuttle at a 3 g maximum axial acceleration). Instead of batteries, the much lighter fuel cells were assumed for prime power; this substitution is regarded as reasonable since the resupply CERV can be serviced on the pad with the LOX and LH2, and the vehicle is not required to remain on orbit unattended for long periods of time as is the case for the exigency return version.

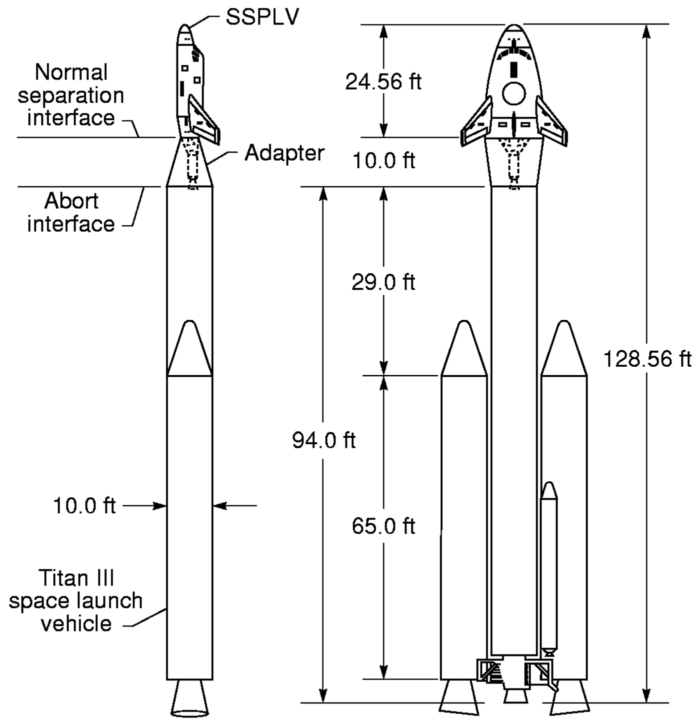


Figure 25. CERV/Titan III launch configuration.

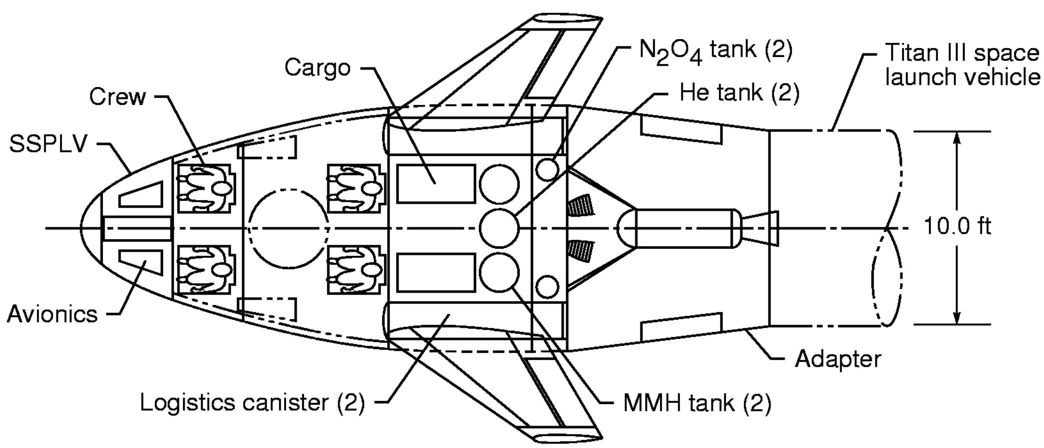


Figure 26. Alternate-access-to-space vehicle.

Because of the 8 g abort requirement, structural weight increased by 820 lb. Personnel provisions changed little because the increase in individual seat weights nearly equalled the savings from the removal of four seats. Personnel decreased principally by the weight of 4 crewman at 185 lb each including their clothing. The landed weight with a crew of 4 and 1000 lb of cargo is 13,808 lb (Table XVI). Approximately 30 percent of the fuel cell reactant was assumed to be on board at landing and is included in the residual and reserve fluids. (Table I, category 20.0).

Table XVI. CERV/STAR Mass Properties (4 man – 48 hr mission)

Lifting Body		Weight,lb
1.0	Wing group	347
2.0	Tail group	0
3.0	Body group	5,300
4.0	TPS	980
5.0	Landing and auxiliary systems	390
6.0	Main propulsion	0
7.0	Propulsion systems (OMS & RCS)	332
8.0	Pressurization, pneumatics & purge	137
9.0	Prime power	1,530
10.0	Electrical conversion and distribution	266
11.0	Surface controls	92
13.0	Avionics	651
14.0	Enviromental control	939
15.0	Personnel provisions	304
16.0	Margin	1,006
	Inert weight	11,070
17.0	Personnel (& clothes, etc.)	1,090
18.0	Payload accommodation	0
19.0	Payload delivered	(1,000)
20.0	Residual and reserve fluids	648
	Landed weight w/o payload	12,808
	Landed weight with payload	(13,808)
22.0	RCS propellant	148
23.0	OMS propellant	1,206
24.0	O2,N2, H2, H2O, He, and NH3	593
	On-orbit weight	15,755
Service Module		
25.0	Abort motor	4,579
26.0	Thrust structure	552
27.0	Adaptor shell	1,670
28.0	Misc. subsystems	750
29.0	Margin	530
30.0	Chute system	741
	Service module	8,822
	Ascent weight (with payload)	24,577

To provide abort capability, a solid rocket motor and parachutes were located in the CERV-to-Titan III adaptor section. The CERV with an alternate (conical) adapter is shown in Fig. 27. In an abort, the vehicle would land on the parachutes either on land or water (Fig. 28). Because of the size of the pressurized cabin, the vehicle would float in the water until the crew is rescued. Parachute risers are located in removable tunnels connecting the three parachutes to the hard points located in the top of the CERV. If the abort system is not used, the adaptor section is separated at the adaptor-to-CERV interface, and the parachute risers at the structural hard points. Depending upon the mission, the adaptor section could be; (a) stowed on orbit at a space station until down cargo space was available in the Shuttle Orbiter; (b) cannibalized for parts at the space station; or (c) separated prior to insertion and allowed to burn up in the atmosphere.

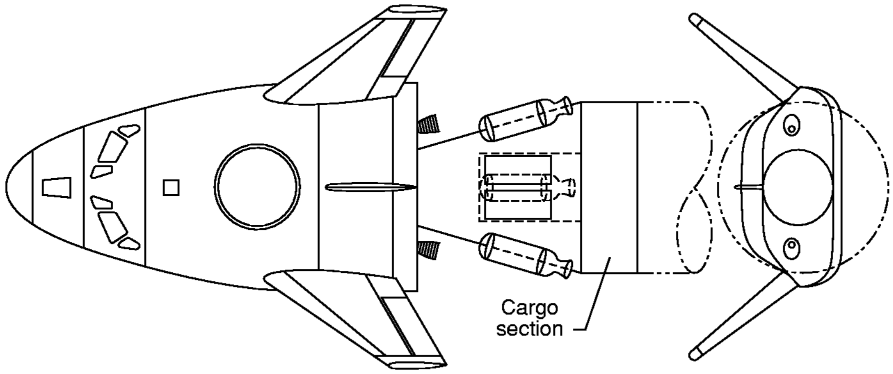


Figure 27. Eight passenger with 8000 lb cargo in adapter section.

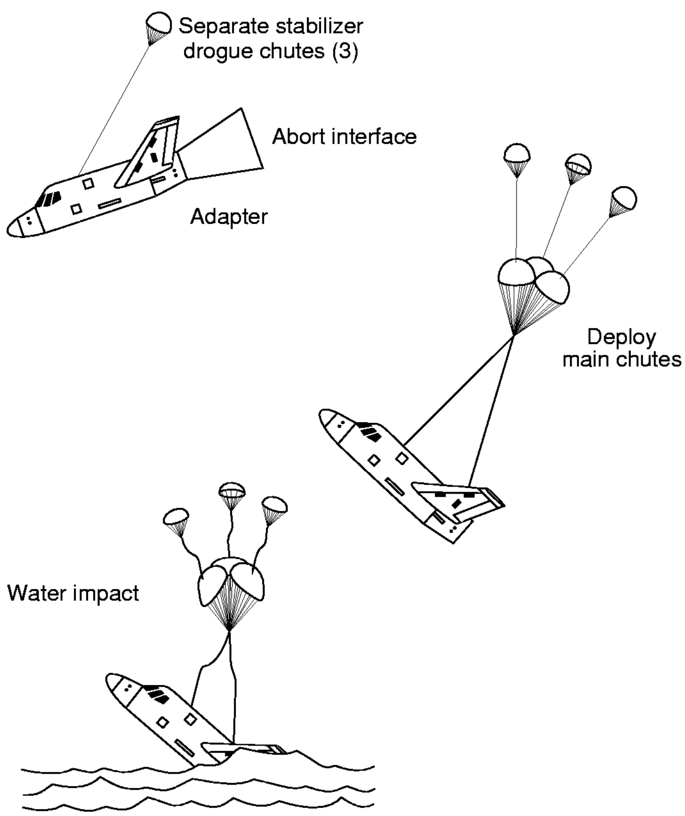


Figure 28. CERV parachute recovery system.

On-Orbit Maneuver Configurations

Two on-orbit maneuver configurations were identified. One in which dual cell tanks were used to contain the N204 and the MMH propellants (Fig. 29). A pressurized compartment ahead of the propellant tanks accommodates two crew. No provisions for stores or cargo is available. If the particular mission does not involve atmospheric flight, stores or cargo could be mounted externally in pods (not shown in Fig. 29). The delta-V capability is approximately 7000 ft/sec for an assumed engine specific impulse of 300 seconds.

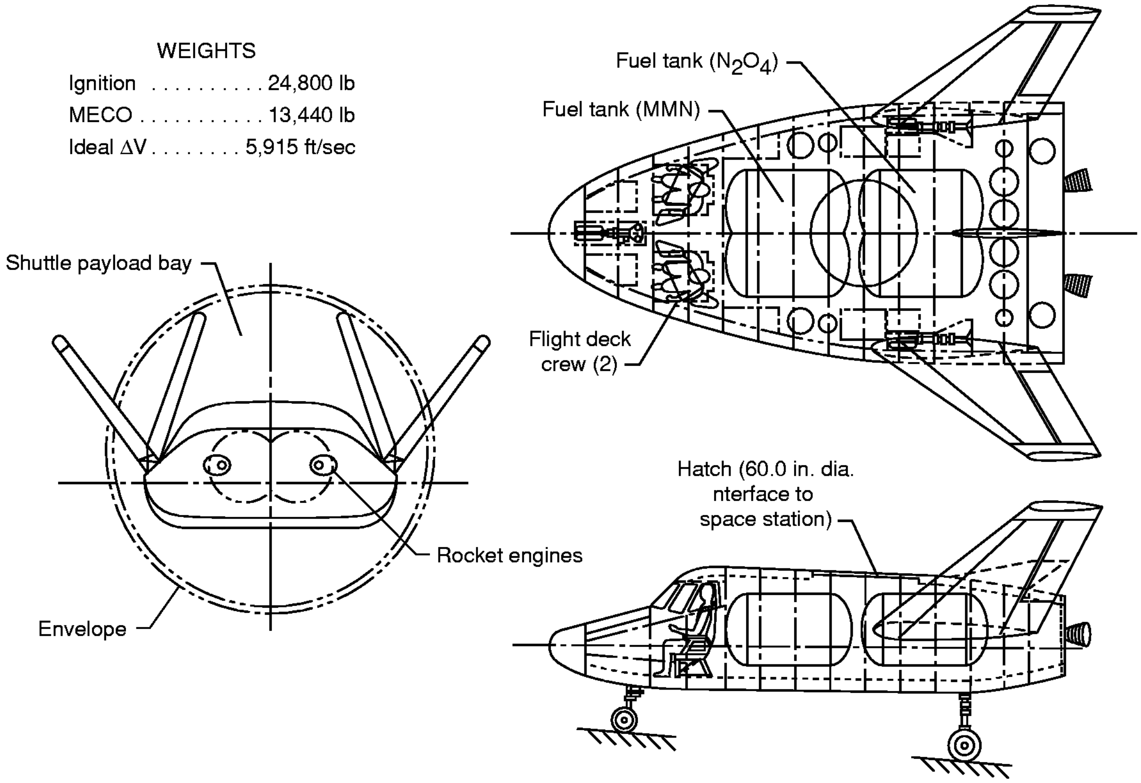


Figure 29. CERV orbit maneuver configuration.

A second maneuver configuration was identified in which the propulsion system was sized for a transfer from a space station in a 28.5 degree orbit to a polar platform in a 98 degree orbit. Extra propellant was placed in drop tanks in order to make the maneuver using the same 24.6 ft long CERV (Fig. 30). The plane change involves a combination of propulsion and aero-assist maneuvers (Table XVII). The first two inclination changes are achieved using the external tanks, followed by atmospheric entry and an aeroassisted turn after the drop tanks have been released. Circularization and de-orbit maneuvers are made on the internal spherical tanks. Two sizes of drop tanks are shown; one set for making the maneuver with a 1500 lb payload, and the larger set for making the maneuver with a 5500 lb payload.

The attachment of the drop tanks is regarded as not a difficult design task, inasmuch as no gravity or aerodynamic forces are present and the thrust-to-weight on orbit is relatively low (about 0.09 at ignition for the smaller drop tanks). Because of the much higher on-orbit weight, a single (6000 lb thrust) OMS engine from the Shuttle Orbiter is needed in order to bring gravity turn losses down to reasonable values when the drop tanks are used in place of the smaller internal tanks.

Cargo Configurations

Two cargo versions of the CERV configuration were identified. One version involves the removal of 6 crewman and all the associated support systems such as seats (Fig. 31). A corresponding reduction in the life support system dry and expendable weights is possible. The weight removed approximately equals the down cargo assumed, namely 2000 lb, thus giving the same landed weight, landing speed, and load on the landing gear and brakes. Cargo packaging density for this configuration is 3 lb/ft^3. (The shuttle cargo packing density is 6 lb/ft^3).

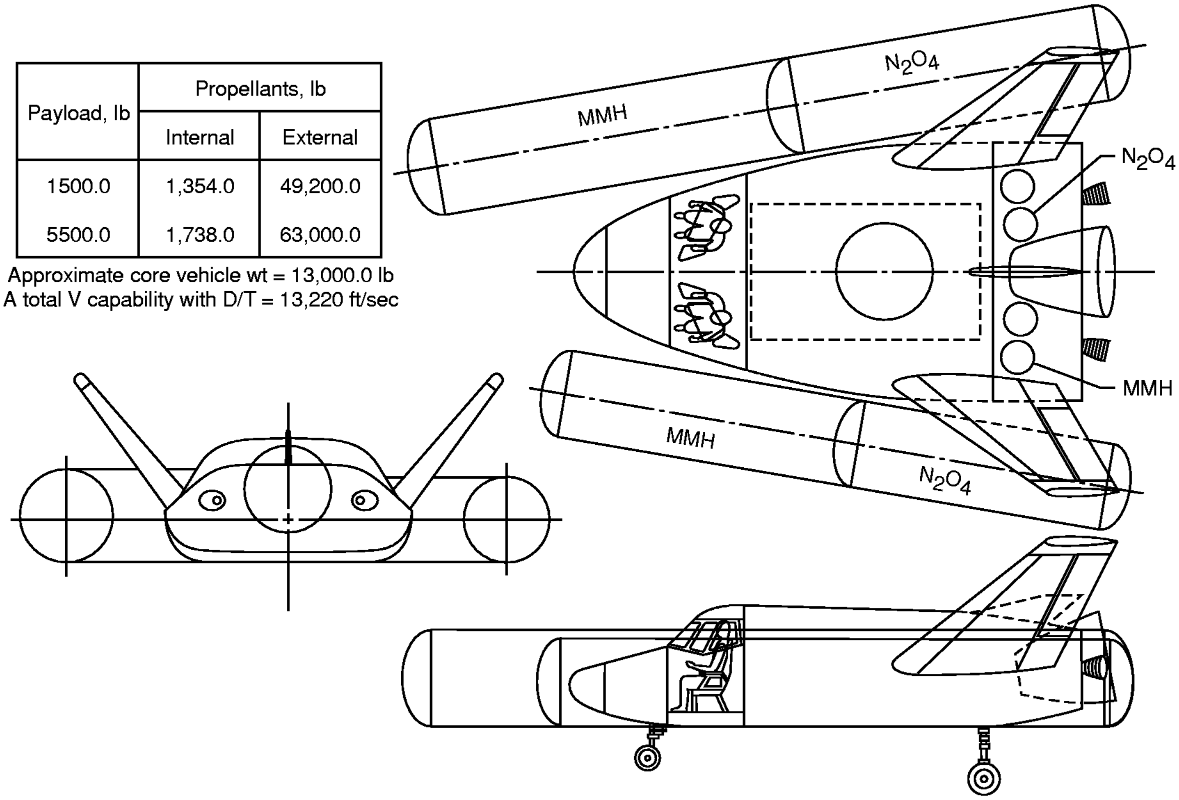


Figure 30. CERV configuration — space station-to-polar platform transfer.

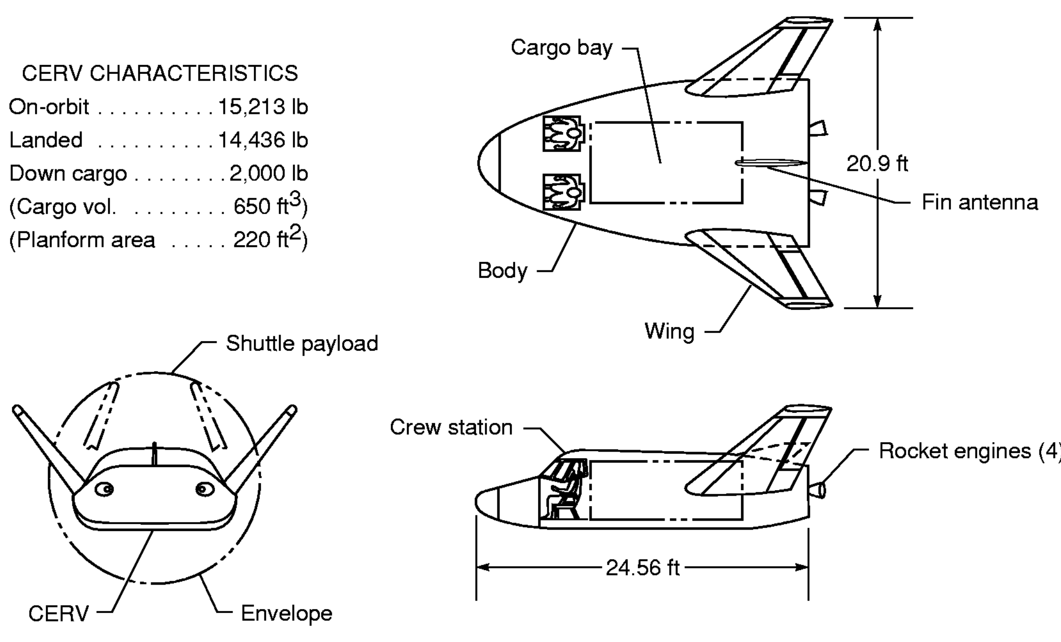


Figure 31. CERV/2000 lb cargo version.

Table XVII. Delta V Budget

Event	Apogee, nmi	Perigee, nmi	Inclination	ΔV Required, fps
Initial orbit	262	262	28.5	0
First burn	19,378	262	30.9	7,885
Second burn	19,378	32.2	80.8	4,430
Atmospheric exit	262	10.6	98.0	0
Circularization	262	262	98.0	443
De-orbit	262	0	98.0	464

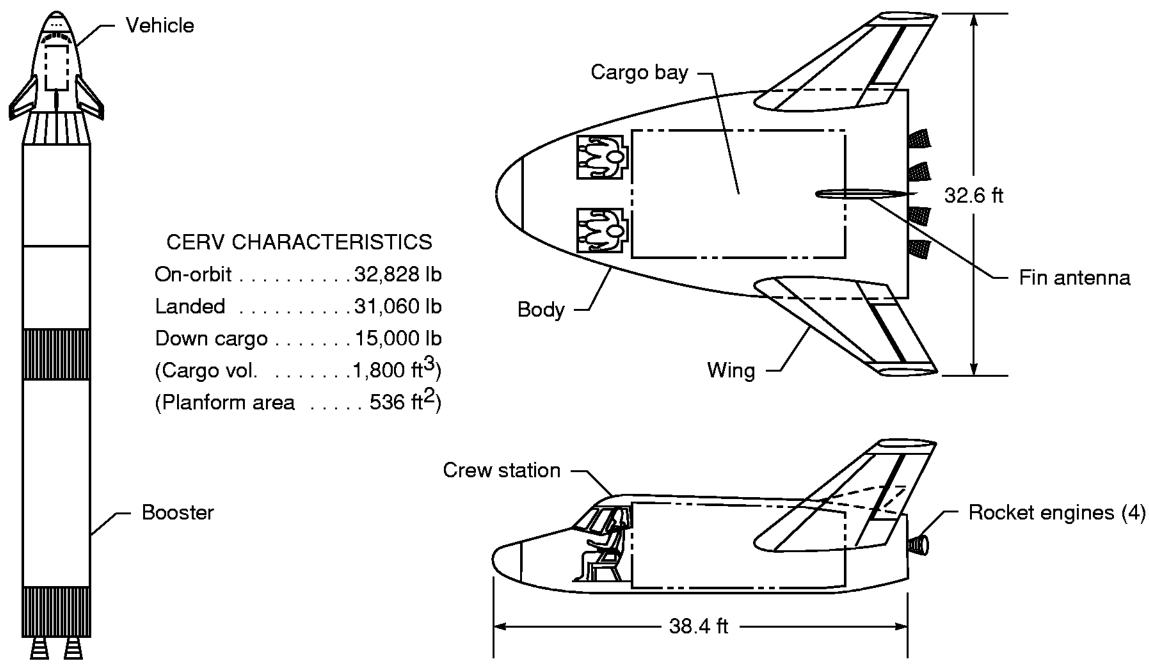


Figure 32. CERV/15,000 lb cargo version.

A second cargo version of the CERV is identified in Fig. 32. In this version the vehicle is increased in size from 24.6 ft to 38.6 ft in order to approximate the same planform loading for a 15,000 lb down payload. Packaging density for this payload is approximately 8 lb/ft^3. Because of the trends in vehicle subsystem weights, it is not possible to maintain both a constant packaging density and planform loading for all possible payloads.

Summary Remarks

A lifting body shape has been studied primarily for use as a means of return from a space station. The vehicle is to be Shuttle deliverable. Subsystem weights have been identified. In addition to the space station rescue function, the vehicle, with minimal modifications, is shown for a number of other space transportation applications. These include use for the return of cargo from space, for missions requiring large maneuvers in space, or for resupply and crew exchange. In the latter application, the vehicle could be delivered to orbit on an expendable booster.

Because of its shape, the lifting body affords one of the highest internal volumes for a vehicle that is constrained to delivery in the current Shuttle cargo bay. Because of its capability for gliding flight (L/D of about 3.5 subsonically), it can land on any orbit at any airport having about a 12,000 ft runway.

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A lifting body is described for use as a return vehicle for crews from a space station. Reentry trajectories, subsystem weights and performance, and costs are included. The baseline vehicle is sized for a crew of eight. An alternate configuration is shown in which only four crew are carried with the extra volume reserved for logistics cargo. A water parachute recovery system is shown as an emergency alternative to a runway landing. Primary reaction control thrusters from the Shuttle program are used for orbital maneuvering while the Shuttle verniers are used for all attitude control maneuvers.

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