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Study of Integral Launch and Reentry Vehicle System

FINAL REPORT

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FINAL REPORT

STUDY OF
INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEM

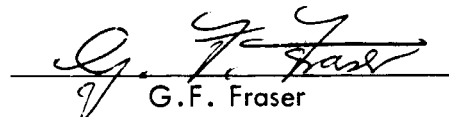
(Contract NAS9-9205)

VOLUME I

SUMMARY REPORT - SECOND PHASE

DECEMBER 1969

APPROVED:


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Space Division
North American Rockwell

FOREWORD

This report is submitted to the NASA Manned Spacecraft Center, Houston, Texas, in accordance with contract NAS9-9205. The report is submitted by North American Rockwell Corporation (NR), through its Space Division (SD), and presents results of the Study of Integral Launch and Reentry Vehicle System.

Acknowledgement is given to NASA MSC for the aerodynamic and aerothermal test data provided in support of the study. The NASA MSC contracting officer's representative directing the study was R.H. Bradley.

This document is Volume I of a five-volume report. It presents a summary of design, performance, and resource studies of a two-stage reusable logistic shuttle vehicle concept investigated during the second phase of the program.

The volumes of the complete final report are as follows:

STUDY OF INTEGRAL LAUNCH AND REENTRY VEHICLE

Volume	Title	SD Report No.
I	Summary Report	SD 69-573-1
II	Technology Development Requirements	SD 69-573-2
III	Technical Report - Second Phase Environment and Performance	SD 69-573-3
IV	Technical Report - Second Phase Design and Subsystems Analysis	SD 69-573-4
V	Technical Report - Second Phase Operations and Resources	SD 69-573-5

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1.0 INTRODUCTION

Future space exploration will be constrained within a limited budget and will benefit from the development of an economical logistic system. NASA is therefore directing investigations of reusable logistic vehicles which offer potential for major operational cost reduction compared to current systems. A number of contracted design studies have been directed by NASA during 1969. This document is presented to the NASA Manned Spacecraft Center, Houston, Texas, in accordance with contract NAS9-9205. The document is submitted by North American Rockwell Corporation (NR), through its Space Division (SD), and contains results of design, performance, and resource studies performed during the second phase of the program.

The contract was negotiated on 31 January 1969, and activity started on 3 March 1969. The study schedule reflects an eight-month technical activity plus reporting period. The major objectives of the study were to define logistic systems which achieve an order-of-magnitude reduction in recurring costs and to advance inherent safety. The program was divided into two phases: the first phase was two months in duration and encompassed comparison of logistic vehicle concepts using expendable boosters. The second phase encompassed investigations of a two-stage reusable vehicle conceived by NASA/MSC. This vehicle has similar shaped booster and orbiter elements with fixed low sweep wings and the main propulsion system uses LO₂/LH₂ propellants. In the study, vehicles were configured to deliver 10,000-, 25,000- and 50,000-pound payloads to a 270-nautical-mile altitude, 55-degree-inclination orbit.

A conceptual design definition was prepared for a selected vehicle, and the technology required to support development of the system was identified.

2.0 STUDY OBJECTIVES

This design study investigates low cost space transportation systems. The objectives of the program are as follows:

1. Derive conceptual designs, plus resources analysis, identifying the development, manufacturing, procurement, testing, RDT&E, and operations requirements for logistics space vehicle systems to support earth orbital programs in the post-1974 period. The major design approaches attempt to:
 - a. Achieve order of magnitude reduction in recurring costs of the total logistic support operational system
 - b. Achieve significant advances in the inherent safety of the space vehicle system
2. Establish requirements for research and technology development, and discuss potential benefits in terms of systems sensitivities, development risk, etc.

3.0 RELATIONSHIP TO OTHER NASA EFFORTS

Four design studies of Integral Launch and Reentry Vehicles (ILRV) were directed by NASA during 1968 and 1969. This program is one of the four studies performed to develop conceptual designs of advanced logistic systems. The studies have been made in accordance with similar ground rules with vehicles configured to satisfy identical mission requirements. Results will enable NASA to compare various advanced space shuttle vehicle concepts.

NASA is currently performing studies of an earth orbital space station and results of the ILRV investigations will support definition of the space station logistic system.

The space shuttle vehicle studies also serve to identify technology development programs required by NASA in support of future space programs.

4.0 METHODS OF APPROACH AND PRINCIPAL ASSUMPTIONS

It is assumed that the advanced logistic vehicle studied will become operational in the post-1974 period. The vehicle is designed with major emphasis on the space station logistics mission which imposes the nominal requirements defined in Table 1. It is also assumed that a high chamber pressure engine using LO₂/LH₂ propellant will be available, and the same basic engine design will be used on all stages of the space shuttle vehicle.

Table 1. Nominal Mission Requirements

• Orbit • Launch Site • Crew • Payloads - Passengers - Cargo weight (including passengers and cargo-bay size) • Life/duration • Launch rate • Ranging	270-nautical-mile, 55-degree inclination Eastern Test Range (ETR) 2 (booster may be unmanned) 10 10,000-pound 25,000-pound, 12-ft diameter x 48-ft, 15-ft diameter x 30-ft and 15-ft diameter x 60-ft 50,000-pound, 15-ft diameter x 60-ft and 22-ft diameter x 60-ft Payload for 3 million and 3.5 million pounds gross weight 7 days active, 30 days in orbit 24 to 50 per year Requirement for 24-hour recall; investigate lateral ranging capabilities
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The vehicle studied during the second phase of the program was conceived by NASA MSC. The system (Figure 1) has two reusable stages with similar aerodynamic shape and fixed, low sweep wings. The main propulsion systems of both booster and orbiter use LO_2/LH_2 propellants with high chamber pressure bell or aerospike engines. The vehicle is configured for vertical takeoff with the orbiter mounted forward on the booster. This orbiter location is to provide a forward center of gravity during the ascent phase of the mission and thus achieve an aerodynamically stable configuration. This results in minimizing thrust vector control, angle of attack, and aerodynamic loads. A major requirement of the logistic vehicle is to minimize operational costs. It therefore becomes necessary to provide a reusable thermal protection system which makes it desirable to limit temperatures experienced on the vehicle during reentry. The approach adopted is to design vehicles with large planform area and flat base and to enter at a high angle of attack to minimize the heating rate.

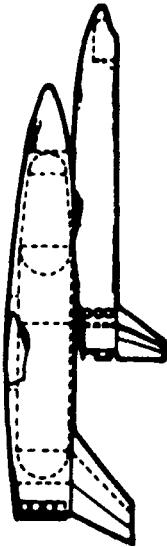


Figure 1. Baseline
Vehicle

With the specified vehicle, the booster element must cruise back to a land landing site after separation of the orbiter. For this purpose, turbojets are used for the mission cruise phase. Turbojets are also used on the orbiter for landing. A high subsonic lift-to-drag ratio (L/D) is provided with booster and orbiter for efficient cruise capability and low landing speeds.

The second phase of the study encompassed:

1. Parametric vehicle sizing analysis
2. Design and resource studies of a selected vehicle

To facilitate definition of vehicle sizes for the previously described payloads, weight scaling data were generated from previous aircraft and space vehicle statistics. These were used in conjunction with baseline vehicle characteristics to obtain preliminary configuration definitions of vehicles with payload delivery capabilities up to 50,000 pounds.

NASA selected one vehicle for detail design and resource studies. The vehicle's aerodynamic and control characteristics were established and used in conjunction with trajectory analyses to define vehicle environments. Environmental data were used in a structural analysis to obtain accurate weight estimates. Trade studies were also performed to select subsystem concepts, and a conceptual design definition was generated for use in resource investigations. Preliminary resource plans were prepared for development, manufacture, and operation of the space shuttle vehicle and program costs were estimated.

5.0 BASIC DATA GENERATED AND SIGNIFICANT RESULTS

This section summarizes results of environment, performance, design, and resource studies performed during the second phase of the study.

FLIGHT PROFILE AND VEHICLE SIZING—Flight analyses established the optimum propellant distribution between orbiter and booster vehicles to minimize the system gross weight. Based upon

the use of two main engines on the orbiter and the same basic engine design on the two vehicle stages. 11 engines were selected for the booster to provide an optimum vehicle design.

The flight profile for the selected vehicle is illustrated in Figure 2. The two-stage vehicle is launched vertically, and staging occurs at an altitude of 230,000 feet with a velocity of 10,800 feet per second. Separation is achieved by rockets installed on the orbiter to displace the vehicle laterally while the attitude of both stages is held constant by the reaction control system. After staging, the booster element rolls with an angle of attack of 52.5 degrees to minimize travel down range and, thus, minimize propellant required to cruise back to the landing site. The orbiter vehicle, after staging, is injected into a 50- by 100-nm elliptical orbit, and coasts to a 100-nm altitude where the orbit is circularized. A Hohmann transfer mode is then used to transfer the orbiter vehicle to a 270-nm, 55-degree-inclination rendezvous orbit. The orbiter vehicle may remain with the space station from 5 to 30 days. After separation from the space station, a phasing operation is performed to provide a reentry orbit compatible with location of the landing site. It is noted that the minimum orbital maneuvering velocity required to accomplish the mission is less than the specified design requirement. The orbiter vehicle reenters at a 60-degree angle of attack and maintains this attitude until the vehicle achieves subsonic velocities. After reentry, the orbiter vehicle is pitched over to a glide attitude of approximately 5 degrees to return to the landing site.

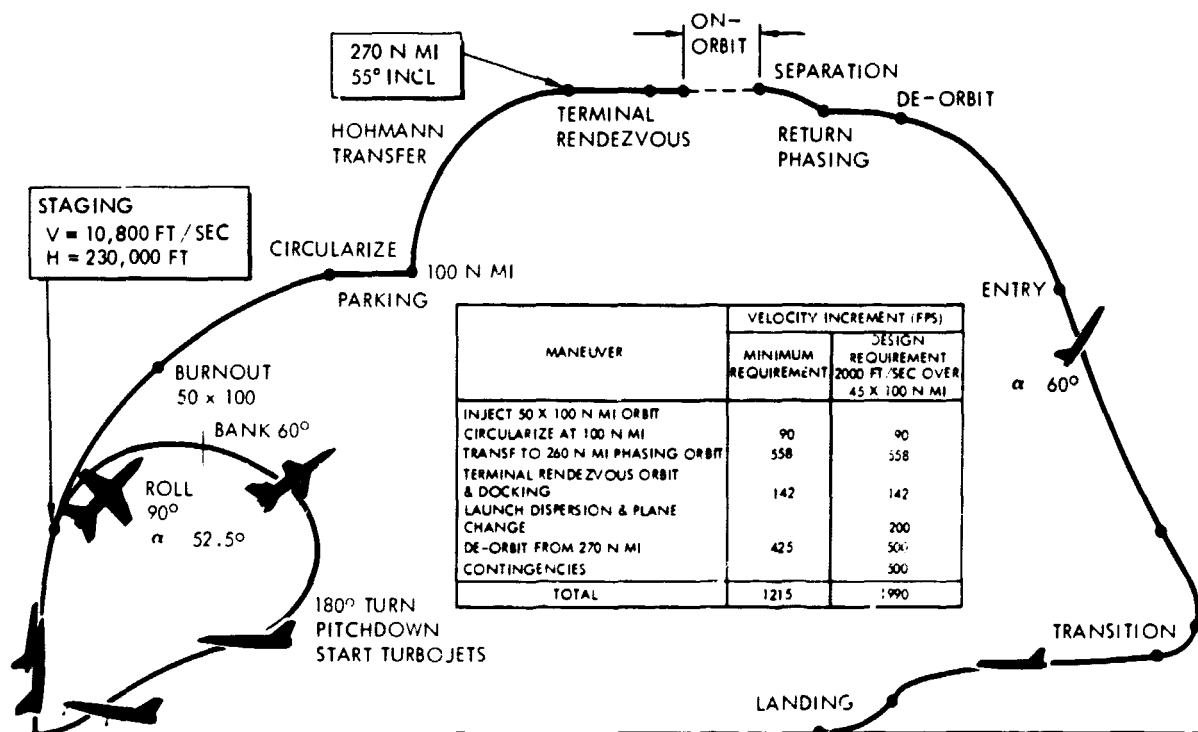


Figure 2. Flight Profile

AERODYNAMIC CHARACTERISTICS—A key factor associated with selecting an advanced logistics system is the aerodynamic characteristics of the configuration. The tandem launch configuration is statically stable during boost. The entry flight mode selected for the shuttle vehicle studied is to reenter at high angle of attack to maximize the lift coefficient and then the vehicle makes a transition to low angle of attack for cruise and landing.

Based upon wind tunnel tests and aerodynamic analysis, the characteristics of the shuttle vehicle have been established. Table 2 summarizes aerodynamic characteristics of the orbiter and booster at hypersonic speeds and for the transition and glide phases of the mission.

Table 2. Aerodynamic Characteristics Summary

	Booster	Orbiter
Hypersonic		
Trim α (entry)	52.5 degrees	60 degrees
Lift coefficient (C_L)	1.87	2.10
Elevator deflection	-7 degrees	-24 degrees
Lift-to-drag ratio (L/D)	0.69	0.53
Subsonic		
Trim α (entry)	52.5 degrees	60 degrees
C_L	2.2	2.20
Elevator deflection	0 degrees	-28 degrees
L/D	0.3	0.3
Trim α (cruise)	4 degrees	2 degrees
C_L	0.5	0.5
Elevator deflection	-5 degrees	-2 degrees
L/D	8.4	7.3

The horizontal tail of the vehicle is sized to obtain a 7-percent static stability margin and to provide the aerodynamic moment to pitch the vehicle over from high to low angle of attack at subsonic speeds. For this transition maneuver, the horizontal is deflected down, and during the cruise phase of the mission, the orbiter and booster achieve maximum subsonic L/D ratios of 7.3 and 8.4, respectively. At hypersonic speeds, the maximum lift coefficient C_L occurs at a 50- to 55-degree angle of attack. To trim the vehicle at high angle of attack at hypersonic speeds, the horizontal must be deflected up to decrease the effective area. The hypersonic L/D ratio is 0.53. It is possible that at subsonic and transonic speeds, unsteady aerodynamic forces may be applied to the vehicle in the yaw direction and slight reshaping of the vehicle may be desirable to assure either continuous flow separation around the body or assure flow attachment. The study has shown that the vehicle will trim at high and low angle of attack and will be statically stable about pitch and roll axes and dynamically stable about the yaw axis.

CONTROL—Vehicle control is achieved through main-engine thrust-vector control (TVC) during the ascent phase of the mission, through use of the reaction control system (RCS) for orbital maneuvers and reentry, and by aerodynamic surfaces during subsonic flight.

To establish the vehicle loading environment, it is necessary to establish the vehicle control characteristics. In the study, the vehicle was analyzed for the ascent phase of the mission with wind and gust profile based upon a 99-percent probability of nonoccurrence. With a programmed flight-path-angle control mode, and a 7-degree main-engine gimbal limit, the vehicle is subjected to maximum q of 2970 psf degrees.

A six-degree-of-freedom control simulation was performed for vehicle reentry and showed that a control system authority of 0.5 deg/sec² acceleration is adequate. A simulation of the transition

to cruise maneuver confirmed that pitchover was smooth and well damped using an elevator command incorporating gain scheduling.

ABORT An analysis of the impact of potential system failures which was performed indicates that failure of a single engine on either the booster or the orbiter will not prevent completion of the primary mission. A system failure requiring premature staging will necessitate action to limit reentry loads on the orbiter and propellant dumping. For mission termination at low altitude, the orbiter with two engines operating will abort to low altitude. With one or two engines operating on the orbiter, the engines may be used to shape the trajectory and prevent excessive reentry loads. Failure of both orbiter engines, however, will cause excessive reentry loads unless propellant dumping is provided, and possible water landing will result.

THERMAL ENVIRONMENT—To define thermal-protection-system requirements, it is necessary to identify a thermal environment. During ascent, the vehicle is subjected to areas of interference heating on the upper surface of the body. The temperature of 500 degrees experienced on the upper surface of the orbiter is the critical design temperature condition and is in excess of that experienced during entry.

The most severe thermal environment condition for the under surface of the vehicle occurs during reentry. Figure 3 illustrates the radiation-equilibrium temperature for the orbiter and booster. For reentry, the temperature is illustrated for two locations on the body and also for a number of locations on the wing. The temperature is defined as a function of body station or percentage distance across the wing chord. The temperatures are based on vehicle reentry with a lift loading (W/CLS) of 50 psf. As shown in the figure, the orbiter vehicle experiences temperatures

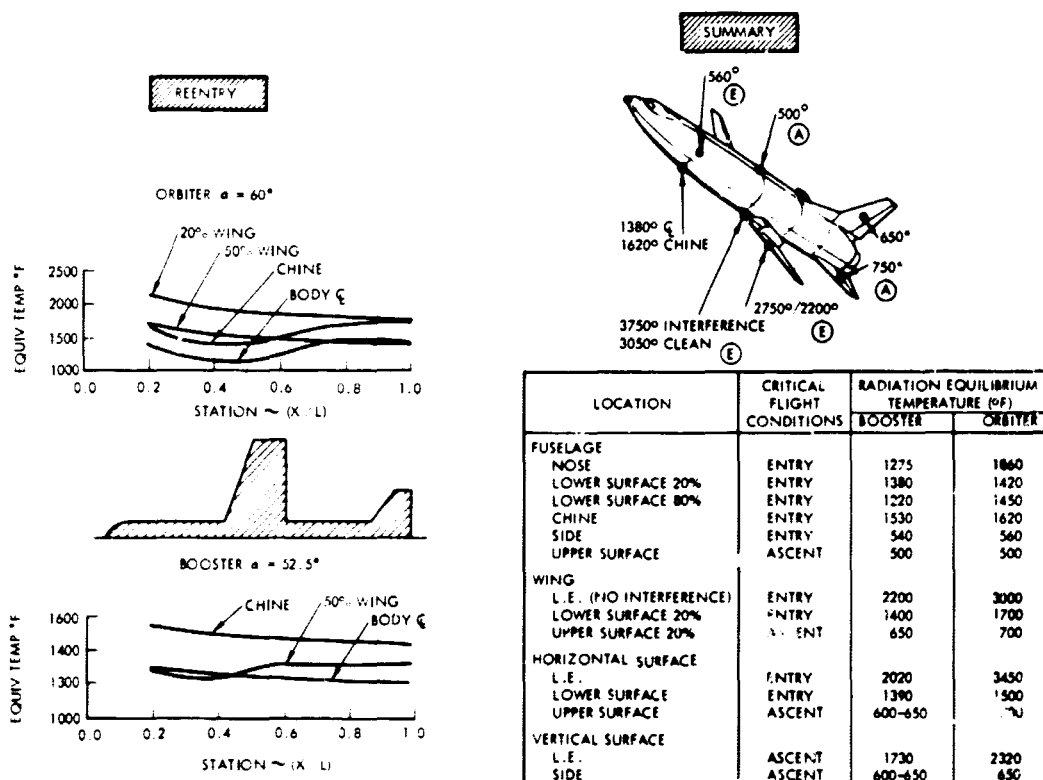


Figure 3. Thermal Environment

below 1500 F on 90 percent of the body. This thermal environment is based upon reentry at a 60-degree angle of attack. It should be noted, however, that the equilibrium temperatures on wing leading edges during reentry are significantly greater than temperatures experienced generally on the body. Radiation-equilibrium temperatures on the wing leading edges are 2800 F to 3200 F without interference effects. Flow interference effects at 20-percent wing span could significantly increase leading-edge temperatures. Skin temperatures, however, will be less than radiation temperatures because of conductance.

As illustrated in the figure, temperatures experienced by the booster during reentry are significantly lower than those experienced by the orbiter and are less than 1400 F on 90 percent of the vehicle.

LATERAL RANGING EFFECTS AND POTENTIAL—In view of the desire to establish vehicle potential for increased lateral range, investigations were performed to establish the impact of reentering at lower angles of attack. Figure 4 illustrates orbiter body and wing temperatures and heat load experienced during entry at various angles of attack. As illustrated, the vehicle can achieve approximately 1240-nm lateral ranging capability at a 20-degree angle of attack without exceeding the temperature limitations of the proposed densified quartz material, except for wing leading edges. The increased heat load, however, with the greater lateral ranging, will require an increase of approximately 22,000 pounds in thermal protection system weight.

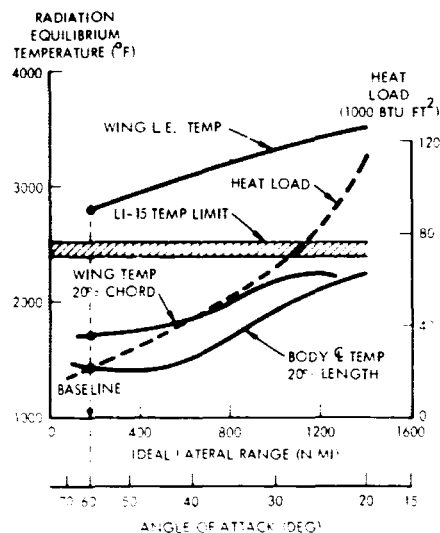


Figure 4. Orbiter Lateral Range Potential

The lateral range capability of the vehicle can also be increased to 1500 nm through cruising with the turbojet engines, off-loading payload, and adding turbojet fuel.

APPROACH AND LANDING—To limit logistic vehicle weight, it becomes extremely important to optimize booster cruise and orbiter landing system weights. Figure 5 illustrates the mission flight mode and results of cruise and landing studies. The booster vehicle is required to cruise approximately 300 miles back to base after staging. A number of turbojet systems were considered for this cruise-back operation. As illustrated, the use of four JT9D-15

turbojets will result in the lowest combined system and propellant weight. With this system, the optimum booster cruise condition is at an altitude 20,000 feet and Mach number of 0.38. This requires a cruise propellant of approximately 10 percent of the vehicle landing weight, and the total system weight, including propellant, is 91,600 pounds. The landing speed for the booster is 113 knots on a standard day. A significant reduction in cruise propellant weight may be achieved through the use of hydrogen as fuel for the turbojets. The orbiter vehicle, in accordance with study requirements, is configured to provide a go-around capability. With the selected system using four JT3D-7 turbojets, the combined go-around system and propellant weight is 20,240 pounds. Elimination of go-around and climb-out requirements will allow removal of two turbojets. This would reduce system weight by 10,120 pounds. The landing speed for the orbiter is 128 knots. It is noted that significant weight reduction would be achieved in the orbiter go-around system weight by the use of high thrust-to-weight turbojet systems currently being developed.

PERFORMANCE—In the study, vehicle gross weights were established for various payloads. Figure 6 illustrates vehicle gross weights for various payload delivery capabilities to 270 nm, 55-degree inclination orbit. The weights illustrated include a 10-percent contingency for subsystem weights. To provide a 50,000-pound payload delivery capability with orbiter designed for a 15-foot-diameter by 60-foot-length cargo bay, the vehicle gross weight with sequential firing of booster and orbiter is 4,490,849 pounds. For a vehicle configured to provide a 12-foot-diameter by 48-foot-length or 15-foot-diameter by 60-foot-length cargo bay and deliver a 25,000-pound payload, the gross weight is 3,310,000 pounds.

If a 10-percent increase in orbiter dry weight is incurred in excess of the 10-percent weight contingency, the vehicle gross weight will increase by more than 0.5 million pounds to maintain a 50,000-pound-payload delivery capability. The use of a simultaneous booster and orbiter firing at liftoff would result in significant payload degradation. As previously discussed, feasibility of the system is contingent upon limitation of system weights. The baseline vehicle is configured to provide a 50,000-pound-payload delivery capability allowing 2000 feet per second for on-orbit maneuvers. Without contingencies, the maneuvering ΔV allowance could be reduced to 1215 feet per second and facilitate a reduction of approximately 200,000 pounds in vehicle gross weight. A number of options exist to improve vehicle performance. Vehicle gross weights for the specified payload delivery requirement may be reduced by approximately 0.75 million pounds.

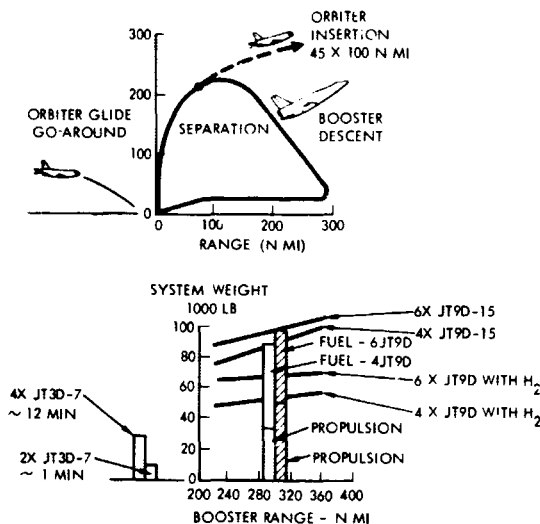


Figure 5. Approach and Landing

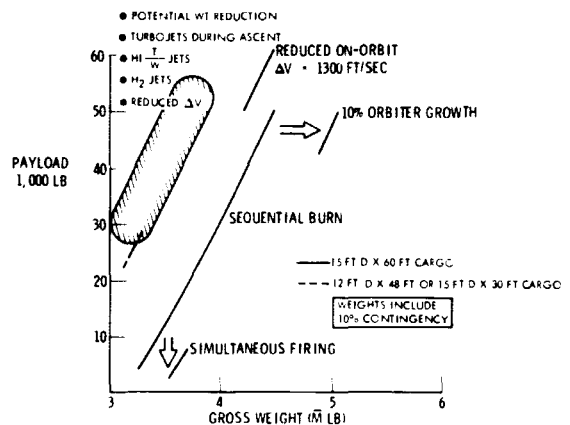


Figure 6. Performance Summary

Payload delivery capabilities of the vehicle for various inclined orbits were established and are:

28.5-degree inclination	60,000 pounds
55-degree inclination	50,000 pounds
90-degree inclination	32,700 pounds

CONFIGURATION DEFINITION—The vehicle sizing and environmental analyses were used to develop definition of the two-stage vehicle illustrated in Figure 7, which is designed to deliver a 50,000-pound payload. As previously indicated, the logistics system uses two reusable stages of similar aerodynamic shape and fixed, low sweep wings. A weight summary for this vehicle is presented in Table 3

The booster uses eleven 510,000-pound sea-level-thrust bell engines with an expansion ratio of 35:1. The LO₂ and LH₂ tanks have a common bulkhead and provide the capability to carry prelaunch, launch, and reentry aerodynamic loads. The booster is 280 feet long, has a wing span of 244 feet, and a lift off weight of 3.62 million pounds.

The orbiter vehicle uses two engines with the same basic design as used on the booster. These engines provide 590,000-pound vacuum thrust capability and have a two-position nozzle with maximum expansion ratio of 120:1. The orbiter element is configured with the 15-foot-diameter by 60-foot-length cargo bay located at the vehicle c.g., to provide a vehicle which can be trimmed on reentry and during cruise with or without payload. This cargo bay location requires separation of propellant tanks. Small cargo modules can be transferred through a pressurized hatch in the main cargo doors. These doors can also be opened for removal of large cargo container. LO₂ and LH₂

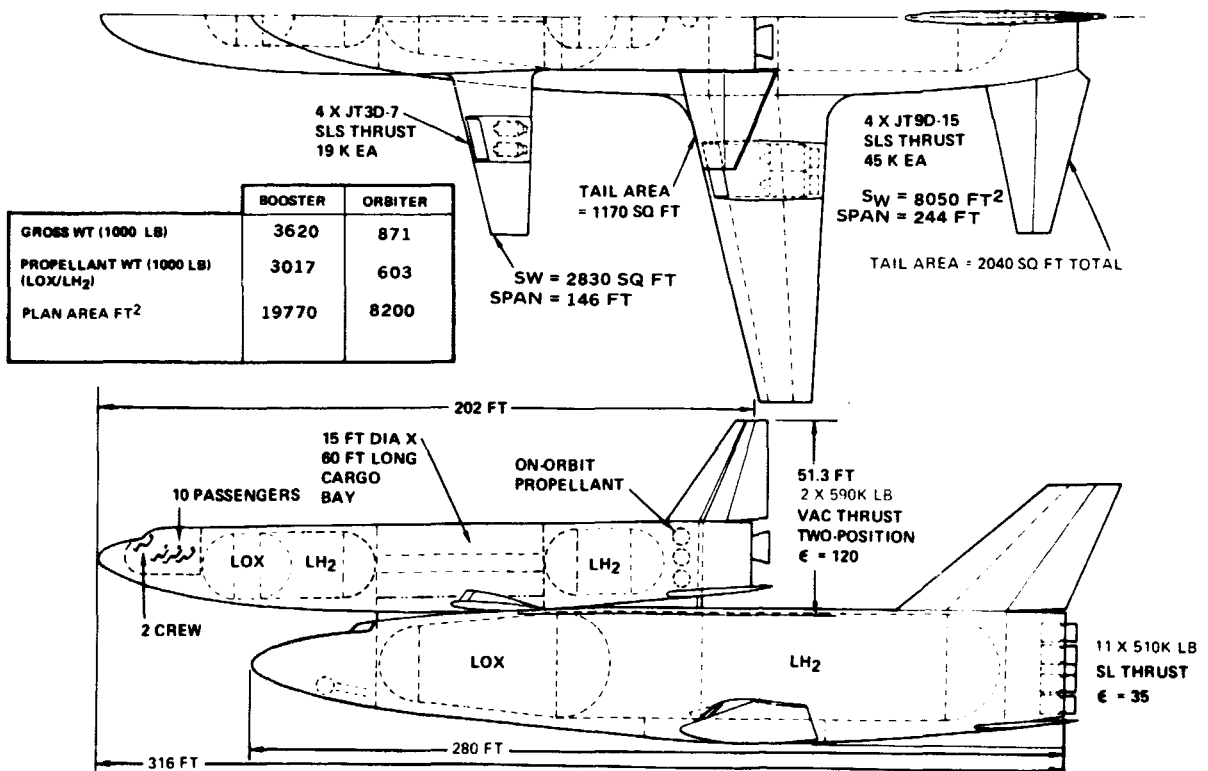


Figure 7. Configuration Definition

**Table 3. Final Weight Statement Summary, 50,000-Pound Payload
 (15-Foot-Diameter by 60-Foot-Length Cargo Bay)**

Item	Weight (pounds)	
	Booster	Orbiter
Aerosurfaces	95,400	26,177
Body structure	229,642	61,658
Induced environmental control	25,572	29,438
Systems	157,078	80,334
Cargo	0	50,000
Residual and service items	24,310	7,200
Landing weight	532,002	254,807
Expended fluids	55,550	3,836
Reentry weight	587,552	258,643
Expended fluids	15,111	48,297
Boost burnout weight	602,663	306,940
Separation rockets and fluids	209	4,037
Ascent propellant	3,017,000	560,000
Liftoff weight	3,619,872	870,977
Total vehicle	4,490,849	

tanks are located forward with a common bulkhead separating the propellants. A second hydrogen tank is located aft of the cargo bay. The tanks are suspended and are not subjected to vehicle aerodynamic loads. Propellants required for on-orbit maneuvers are stored in separate tanks with superinsulation to provide the on-orbit stay-time capability. The orbiter is 202 feet in length, has a wing span of 146 feet, and weighs 871,000 pounds at lift-off.

STRUCTURE AND THERMAL PROTECTION—The most significant weight element in the logistic vehicle is the structure and thermal protection system. Trade studies were therefore performed to establish the lowest-weight structure and thermal protection design for concepts using integral load-carrying tanks and for systems with suspended tanks and external load-carrying structure.

Figure 8 illustrates the orbiter structure and thermal protection design which is considered the lowest weight system. Propellant tanks, which are aluminum, are suspended and therefore not subjected to vehicle aerodynamic loads. A polyurethane insulation is used on the inside of the LH₂ tank. Separate tanks with superinsulation are used to contain propellant for orbital maneuvers. The primary load-carrying structure is a skin-stiffened titanium shell. To withstand the entry thermal environment, the titanium structure is protected on the bottom and sides of the fuselage by a densified quartz, external insulation. Bare titanium is satisfactory on the top of the orbiter because of the mild thermal environment. A fiberglass insulation is used inside the titanium structure to limit the temperature of the tankage and insulation. The orbiter uses fixed, low sweep wings fabricated from titanium.

The booster propellant tanks provide the primary load-carrying structure. The tankage is fabricated from 2219Al with external stiffened structure. The LO₂ tank is located forward of the

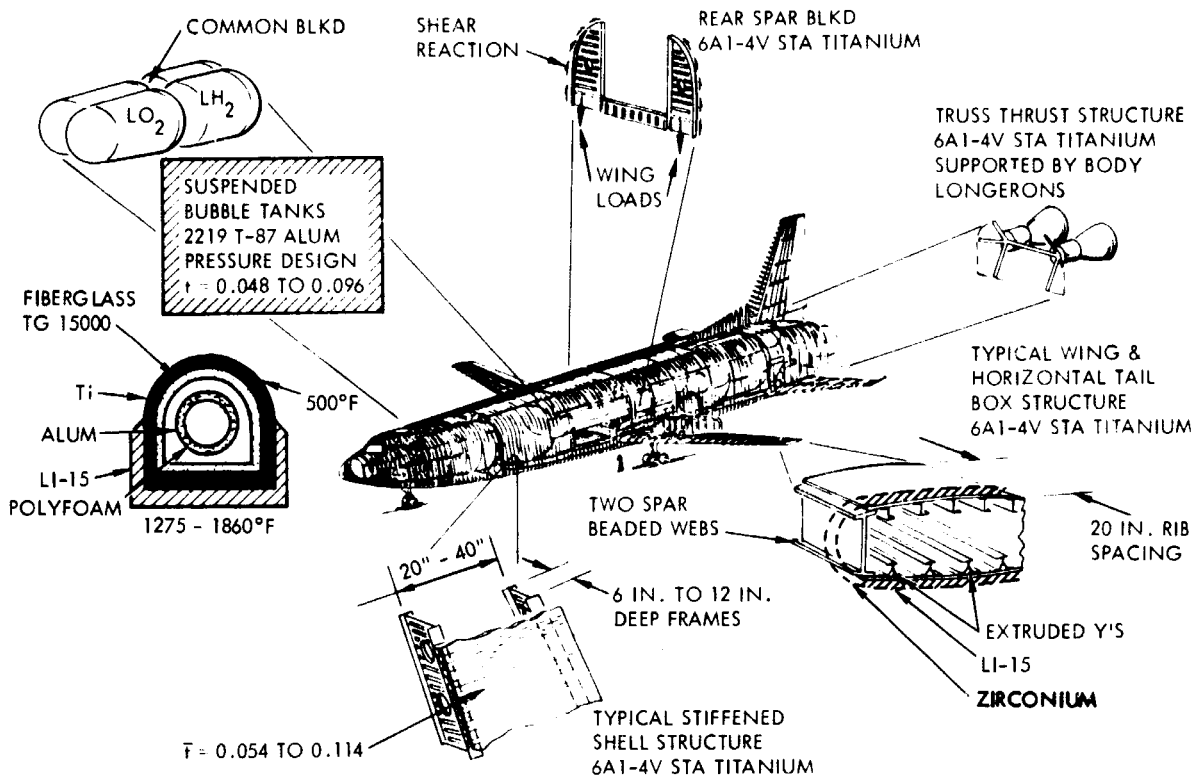


Figure 8. Orbiter Structure Thermal Protection Definition

LH₂ tank with a common bulkhead separating the propellants. LH₂ tank thermal control requirements are satisfied by polyurethane foam installed internally in the LH₂ tank. An aluminum truss and reinforced phenolic and polyimide honeycomb-sandwich substructure is provided to maintain the desired flat base. This structure is also used to bond densified quartz external insulation to the booster bottom and side. The mild thermal environment on the top of the booster is within the capability of honeycomb-sandwich material. The booster uses fixed, low sweep wings fabricated from titanium.

PROPULSION—Figure 9 schematically illustrates the orbiter main propulsion system. The system uses two high-pressure bell engines with a vacuum thrust of 590,000 pounds. The engine has a two-position nozzle with expansion ratios of 58 and 120:1, and provides 7-degree pitch and yaw gimbal capability. Engines are throttled to 10-percent maximum thrust for orbiter maneuvers. The LO₂ tank is located forward to provide vehicle stability and is pre-pressurized at launch from 30 to 32 psi. Helium is used as the pressurant before launch. Pressurant during ascent is obtained from the engine, and tank pressure is regulated to provide 20 and 30 psia at end of boost in the LO₂ and LH₂ tanks. Pressure is required in the tank to prevent collapse on reentry.

Investigations of the orbiter go-around system were performed to identify the desirable turbofan engines. The system selected to provide a go-around capability uses four JT3D-7 engines. These engines, which use JP fuel, have a rated sea level static thrust of 19,000 pounds and have a thrust-to-weight ratio of 4.5:1. It will be necessary to qualify these engines for a space environment. A significant reduction in go-around system weight could be achieved through the use of high thrust-to-weight engines. Technology for these engines, currently under development, could result in an increase of approximately 10,000 pounds in vehicle payload delivery capability.

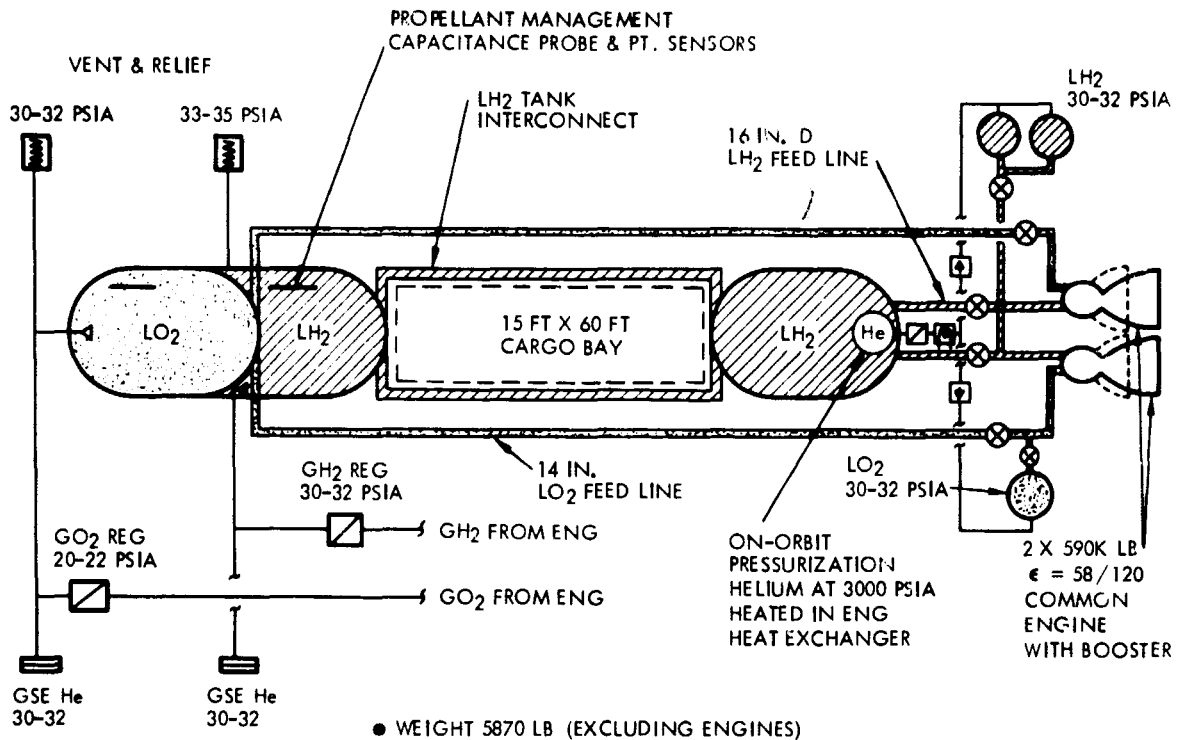


Figure 9. Orbiter Main Propulsion System

A significant system development for future logistic vehicles is the reaction control system. A gaseous oxygen and hydrogen system was selected to provide a clean system, subject to minimal maintenance requirements. The installation is based upon the use of four thruster modules, and full vehicle control capability is possible with a single thruster failure. A number of design options exist. The system illustrated is based upon use of propellant stored in the on-orbit maneuvering tanks. The propellant is pumped in liquid state through heat exchangers and transferred as a gas to accumulators for use in the RCS thrusters. Twenty-eight thrusters with 300-psi chamber pressure, each exerting 800-pound force, are used. It is possible that residual propellant gases contained in the main propellant tanks could be used for the RCS. This approach would require relatively high horsepower compressors to transfer the gases to RCS accumulators. A performance and weight trade has not yet been performed to establish the most desirable RCS design approach. In the subject study, RCS propellant is obtained from the maneuvering tanks.

The booster main propulsion is provided by 11 high-pressure bell-nozzle engines, each delivering 510,000-pound sea level thrust. The engine has a fixed nozzle with 35:1 expansion ratio and provides 7-degree pitch and yaw gimbal capability. The same basic engine is used in the orbiter and booster with altitude and sea level nozzles attached at an expansion ratio of approximately 5:1. Studies indicate that the desirable cruise system for the booster will use JT9D-15 turbojets. These engines have a rated sea level static thrust of 45,000 pounds. The JP propellant weight required to provide capability for cruising 310 miles back to a landing site and go-around is 57,800 pounds. Use of hydrogen fuel for these engines in place of JP would facilitate a significant reduction in propellant requirements. The booster RCS uses gaseous oxygen and hydrogen and has 24 thrusters, each delivering 1800-pound thrust.

GUIDANCE AND NAVIGATION—In establishing the guidance and navigation system for future logistic vehicles, functional and equipment requirements were established. The future G&N system will require the use of a computer integrated with a total electronics system to provide checkout capability and also dual redundancy. Two other major new system requirements are development of an automatic docking and an automatic landing system. It is projected that the docking system will use a three-beam radar system with reflectors installed upon the space station. Approach and landing is likely to use ground-based radar and a computer, and the ground base would transmit signals to the logistic vehicle autopilot. It is considered that repackaging and qualification of the guidance and navigation equipment would be necessary for operation in a space environment.

INTEGRATED ELECTRONICS—To provide a minimum weight and flexible electronic system, integration of the various subsystems is desirable. Integration of the various electronics subsystems using common equipment will provide increased flexibility. A key element in the integrated electronics system is a real-time computer using three processors to provide capability for safe mission termination with two system failures. With the logistic vehicle, a large amount of data must be displayed to the crew. Therefore, it becomes desirable to use an electrooptical system for display of mission information. This display system presents mission data as required. A manual override may be used to request display of data not preprogrammed. The system will also automatically display information associated with system or subsystem performance which is outside tolerance. A multiplexing system will be used in place of hard wiring to transmit signals and thus minimize weight and complexity. Built-in test equipment will be used to the subsystem and component level to provide an onboard checkout capability. Automatic checkout procedures of non-electronic equipment will require development of techniques such as sonic analysis, ultrasonic testing, fiber optics, etc.

POWER—Electrical and hydraulic power equipment requirements for the booster and orbiter are summarized in Table 4. The dc power supply is obtained from fuel cells and two 80-ampere power batteries. These batteries are recharged from ac and dc supply systems. The ac power is obtained from an alternator which is powered by four auxiliary power units using cryogenic propellants. These auxiliary power units are also utilized to drive the hydraulic pumps which deliver pressure to the aerodynamic surface actuators and landing gear.

Table 4. Power System Requirements

Booster	Orbiter
Electrical 4 x 250-hp APU, O ₂ /H ₂ 2 x 80-ah batteries 4 x 12-kva alternators	Electrical 3 x 5-kw fuel cells, O ₂ /H ₂ 4 x 150-hp APU O ₂ /H ₂ 2 x 80-ah batteries 4 x 12-kva alternators
Hydraulic 4 independent systems, 4000 psi 2 x 40 gpm pumps and associated equipment	Hydraulic 4 independent systems, 4000 psi 40 gpm pump, reservoir N ₂ pressurization, filters, relief valve, actuators

ENVIRONMENTAL CONTROL AND LIFE SUPPORT—The environmental control and life support system (EC LSS) provides thermal and contaminant control of the crew and passenger compartment and life support systems. Thermal control is also provided for the vehicle equipment. Thermal control is supplied by dual-loop water-glycol system and water evaporators for backup and peak loads.

MANUFACTURE—An analysis of the baseline shuttle vehicle and development of a manufacturing breakdown indicates that manufacture of development hardware will extend over approximately a 4-1/2-year period. The total manufacturing program encompassing fabrication of six flight vehicles at a rate of two per year will extend over approximately 7-1/2 years. Significant manufacturing progress will be necessary for installation of the vehicle thermal protection system and vehicle handling. Existing facilities can be adapted with minor modifications for production of the orbiter vehicle, but new facilities or existing facilities with major modifications will be required for the booster. However, the consideration of facility collocation effects of cost reduction led to the selection of a new final assembly facility located at Palmdale, California.

DEVELOPMENT—Qualification of the shuttle vehicle will be based upon an aircraft-development approach where possible, using nondestructive testing techniques and increasing severity of test conditions. It is planned that taxi, cruise, glide (low angle of attack), and landing tests will be performed using turbojets. Main-rocket propulsion is not recommended for horizontal flight tests. Separate tests with vertical takeoff of the orbiter and booster are possible. Integrated-system dynamic and flight tests will be necessary.

OPERATIONS—The study requires a vehicle capable of rendezvous with the space station within 24 hours of lift-off. The orbiter vehicle must provide capability for five days active life of systems and may remain with the space station for 30 days, using the power source from the space station or from the payload. The orbiter vehicle must also provide the capability for return to a selected landing site within 24 hours of recall. The system goal will be to provide the capability for maintenance, checkout, and turnaround within a two-week period. To achieve this, use of onboard checkout and nondestructive testing techniques such as fiber optics, sonic analysis, etc., will be necessary. Figure 10 illustrates a normal operation cycle for booster and orbiter elements of the shuttle system. For the orbiter vehicle, the flight cycle ranges from 15 to 49 days, and for the booster, 8 to 15 days will be required.

In considering the period required for turnaround, an analysis was performed to establish the time necessary for maintenance, vehicle assembly, cargo loading, and checkout. Figure 10 summarizes the vehicle cycle for the ILRV, based on the goal of vehicle turnaround within two weeks. Based upon the philosophy that a large percentage of checkout operations will be automatic and performed during the normal mission, and that airline maintenance type operations will be performed, the minimum estimated period required for turnaround is 66 hours. Nonscheduled maintenance would, however, require an increase in this turnaround period. It is considered that in the early phases of the space shuttle program, the turnaround period of two weeks would be exceeded, and additional information and experience are still required to confirm the feasibility of a two-week turnaround cycle.

Operations analysis indicates that approximately 200 direct personnel will be required for maintenance and launch operations to sustain a rate of 50 launches per year.

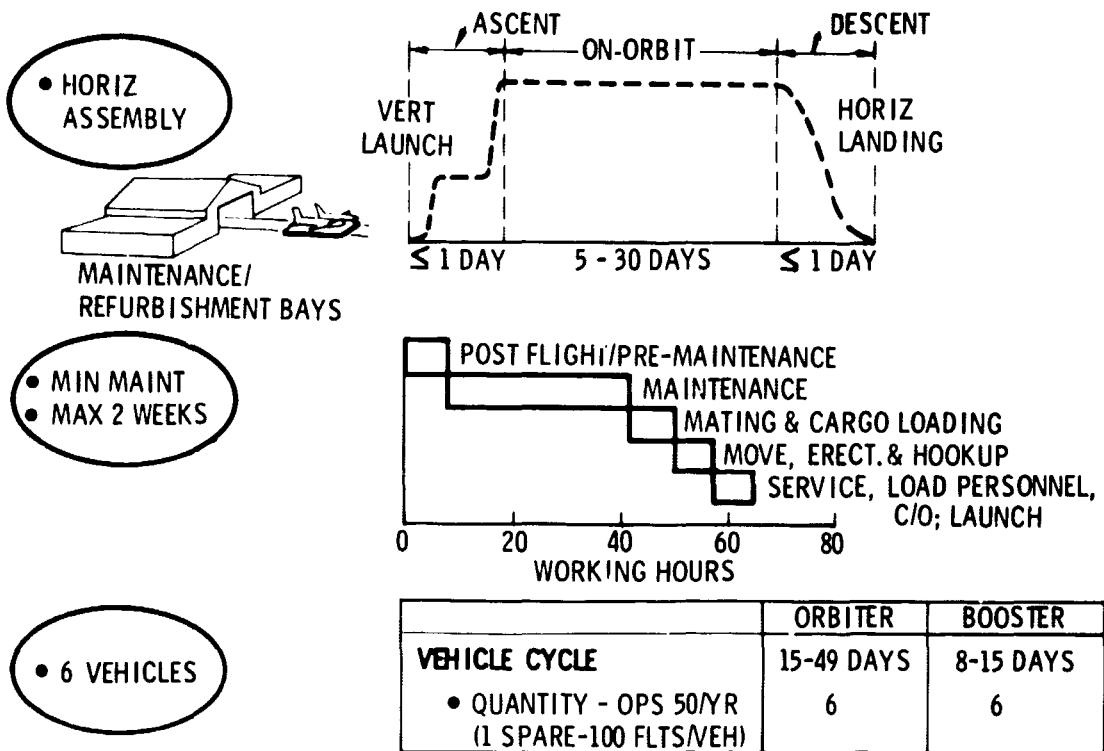


Figure 10. Operations Summary

The maintenance operations and associated turnaround times were used in analyses to establish the number of logistic vehicles required. For a 14-day turnaround cycle and 50 launches per year, three orbiters would be required if the vehicles were capable of more than 170 reuses per vehicle. In the subject program, however, a life constraint of 100 flights per vehicle is used. This identifies a requirement for a minimum of five orbiter vehicles in a 10-year program for 50 flights per year. One space vehicle was added to the quantity required to satisfy the life constraint.

ASSEMBLY AND LAUNCH—A number of approaches for assembly and launch of the two-stage logistic vehicle were considered. The various assembly and launch options are illustrated in Figure 11, and analyses, indicate that off-pad horizontal assembly is desirable. Horizontal assembly of the vehicle would not be possible within the existing Vertical Assembly Building (VAB) at Complex 39. A major extension to the VAB or provision of a new assembly building will therefore be necessary. The booster and orbiter elements would be checked out in a horizontal position and the orbiter hoisted above the booster for mating. The integrated system would then be transported to the launch site in the horizontal position and erected on the launch platform.

The study required launch and landing of the ILRV at the Eastern Test Range. Investigations were performed to establish the landing strip that should be used. The conclusion resulting from the studies is that either the existing skid strip at KSC or a new landing strip should be used. A new strip would result in greater initial expenditure but would provide a more efficient facility. This landing strip would be built adjacent to Complex 39.

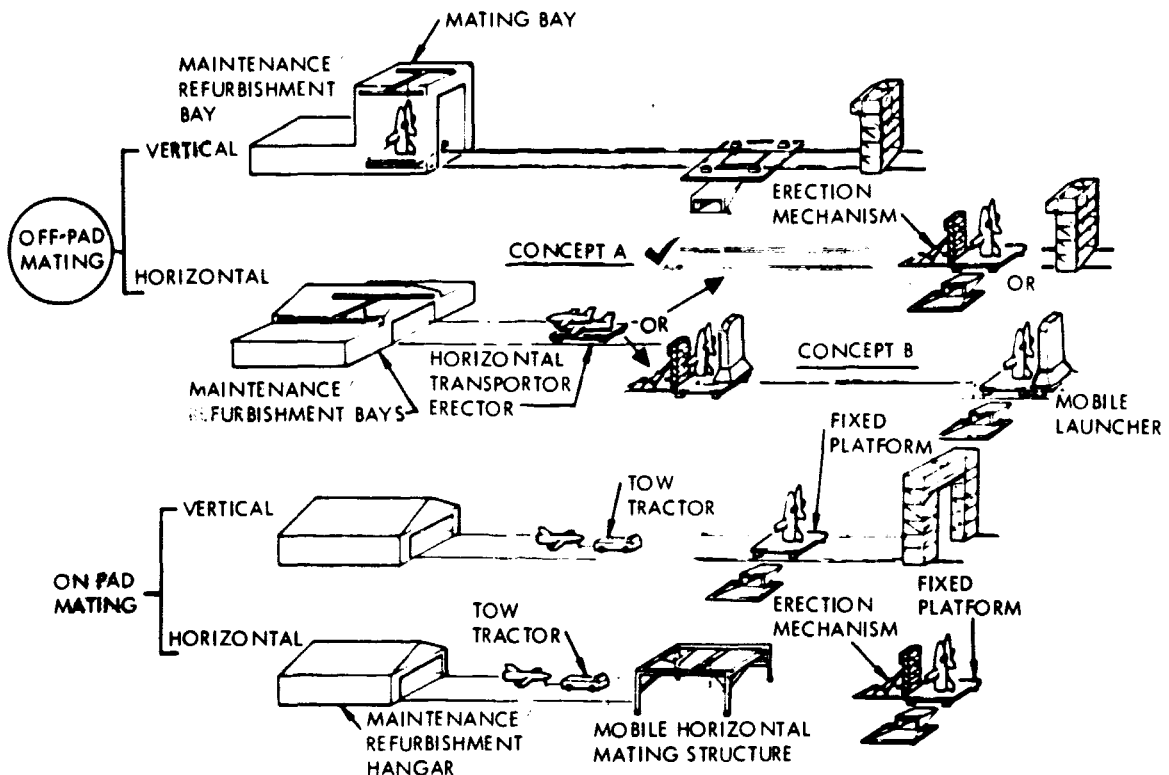


Figure 11. New Assembly and Launch Facility Options

SCHEDULE AND COST ANALYSIS—Based upon a schedule analysis which assumes a successful program and technology available to support the space shuttle, it is considered that program initial operational capability could be achieved 7-1/2 years after initiation of Phase B.

Preliminary cost estimates have been developed, also based upon a successful program and availability of technology. These cost estimates are illustrated in Figure 12 for various launch rates and show a nonrecurring cost of approximately \$7 billion and program cost of \$10 billion for 50 launches per year.

CONCLUSIONS—Based upon the investigations, the following may be concluded:

- The subject vehicle will trim at high and low angles of attack; therefore, the concept of entering at a high angle of attack to minimize temperature and subsequently pitching down to a low angle of attack for glide is a feasible operating mode.
- Reentry of the vehicle at a high angle of attack in a stalled condition will minimize the lift loading and reduce temperatures experienced upon reentry. It is projected that temperatures on the orbiter vehicle will generally be less than 1500 F. Temperatures on leading-edge and flow-interference areas, however, will be significantly greater than 1500 F. The capability exists for the orbiter to return to the selected landing site within 24 hours of recall without reducing payload. At a low angle of attack, the vehicle lateral range capability may be increased to 1500 nm, but this would require an increase of approximately 22,000 pounds in insulation weight. A portion of the ranging must also be obtained through vehicle cruise.

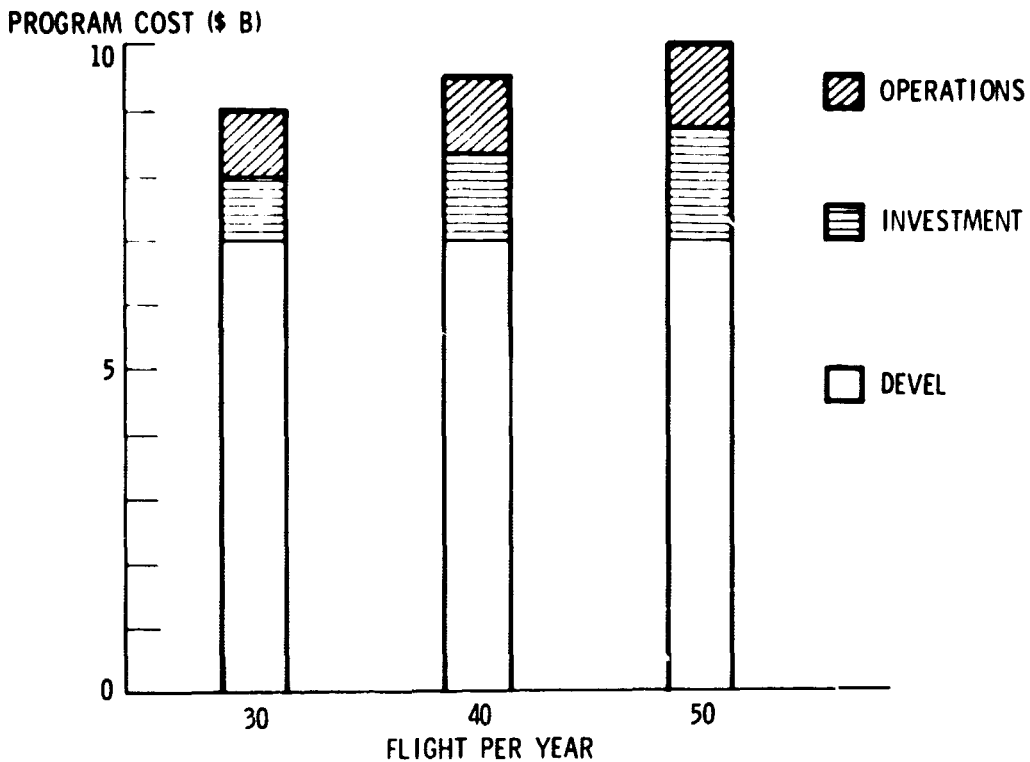


Figure 12. Cost Summary (10-Year Program)

- The projected landing speed of both orbiter and booster elements will be less than 130 knots which is comparable to existing transport aircraft.
- The studies show that sequential firing of booster and orbiter maximizes performance, and with the two-stage system configured to deliver 50,000 pounds of payload, the gross weight is approximately 4-1/2 million pounds.
- Analyses of the structure and thermal protection system indicate that suspended tanks and external load-carrying structure are desirable in the orbiter to minimize weight and increase design flexibility. The use of densified quartz as the external thermal protection also minimizes system weight and provides a safe system. The safety aspect associated with the use of densified quartz is attributable to the capability of the thermal protection to ablate under more severe reentry thermal environments than projected, thus preventing catastrophic vehicle failure. To assure availability of the vehicle in a timely manner, early operations could be performed using an ablative material for external thermal protection. It is projected that for the booster element, integral load-carrying tanks should be used to minimize system weight.
- For the vehicle configured to deliver a 50,000-pound payload, eleven 510,000-pound sea-level-thrust engines are desirable on the booster. By requalification for space environment, existing turbojets may be used on orbiter and booster elements for cruise and go-around. Benefits could also be obtained through high thrust-to-weight turbojets and engine using hydrogen as fuel.

- To provide low weight, flexibility, and a system with automatic checkout capability for rapid turnaround an integrated electrical system is desirable.

6.0 STUDY LIMITATIONS

The study has indicated the feasibility of a two-stage, reusable, fixed wing concept. Further design studies and development testing of the system will, however, be necessary.

The aerodynamic technology for the ILRV is well defined for the ascent phase of the mission and for cruise and landing. Aerodynamic technology for flight at high angle of attack from hypersonic to subsonic velocities, post-stall flight, and transition from high to low angle of attack requires advancement. Establishment of the aerodynamic characteristics of the ILRV in this study has been based upon analysis and a limited amount of testing up to Mach number 2. The effects of Reynolds number and Mach number on vehicle aerodynamic characteristics have not been clearly defined. The potential flutter and buffeting which may be experienced with the vehicle have also not been analyzed. Testing to date has been at Reynolds numbers considerably less than would be experienced during flight. The horizontal tail surface could be too large when sized on the basis of low subsonic Mach number test data measured at low Reynolds numbers. Tendencies toward directional instability at high angles of attack indicated in testing at low Reynolds numbers would be expected to be nonexistent at flight Reynolds numbers. Trade studies to establish the aerodynamic shape of the system have been limited.

In this study, the abort analysis has been directed toward establishment of trajectories and environmental conditions which would occur if premature staging becomes necessary and mission termination is required. A failure-mode analysis has not been performed to identify system failures which would require premature staging and mission termination. Extensive analysis will be necessary to establish whether the system should be designed to provide for premature staging and to define the flight mode which will be used for abort at any phase of the mission.

The thermal environment for the vehicle has been based on analysis and a limited amount of test data. Tests indicate potential flow interference areas and resulting increases in heat rates. Further testing will be necessary to obtain more information on the flow interference effects and methods of preventing excessive temperatures. A limited amount of effort has been expended in establishing desirable trajectories to minimize heating. The analyses of vehicle reentry also at low angle of attack have encompassed only the heating effects on the base and leading edges of the system.

Establishment of the loading environment for the system has been based upon a static loads analysis with factors applied to reflect magnification associated with a dynamic response. The effects of acoustic environment on secondary structure also have not been considered.

To obtain complete confidence in the weights projected for the system, a further comprehensive analysis of all subsystems will be required.

A key factor in the capability of the system to operate at low cost is the availability of a reusable thermal protection system. In the study, definition of the thermal protection has been based upon an extremely limited amount of data relative to densified quartz. Flight analysis and

definition of the main propulsion system have not considered the impact of using main engines for orbital maneuvers. The preconditioning requirements for the main engines to perform orbital maneuvers have not been defined; therefore, the impact upon vehicle performance has not been established. While preliminary definitions have been developed for the vehicle subsystems, a detailed definition will be required for development of the vehicle. It will also be necessary to establish the degree of integration required for the electronics system.

Definition of resource plans for the ILRV has been based upon a conceptual design definition and a limited amount of information relative to materials. More definitive data relative to each subsystem will be necessary to refine the program plans and to facilitate more accurate estimation of system costs.

7.0 IMPLICATIONS FOR RESEARCH

The previous section has identified limitations of this conceptual design study. It has also served to identify areas requiring technology advancement. A critical area requiring development of technology for the shuttle vehicle is the aerodynamic characteristics of the system. Further analytical studies and extensive wind-tunnel testing will be necessary to establish the following:

1. Vortex development and shedding and boundary layer separation associated with stall and post-stall flight
2. Flutter transonically and in stalled flight
3. Reynolds-number and Mach-number effects at high angle of attack
4. The effects of configuration geometry (body shaping, wing and tail geometry, etc.)

Interference heating and boundary-layer transition probably represent the most difficult problems to solve because of the many variables involved, lack of analytical tools, and unknown scaling laws.

Analyses and test data indicate two major areas of concern: shock-wave interactions on orbiter and booster wings and horizontal tail, and shock-wave and support-structure interference during mated booster and orbiter ascent. Extreme temperatures may occur without proper design. Risk can be minimized through extensive test program and subsequent design modifications.

It is generally recognized that critical areas for development are the external structure and external and internal thermal-protection systems. It will be necessary to establish a low-weight reusable system. In the study, the selected external thermal-protection concept is based upon the use of densified quartz. Presently, this material has not been fully developed and has certain disadvantages. The material is subject to water absorption and possibly erosion under operating conditions, and it will be necessary to provide an external coating. This material is a critical development item for the ILRV. It is also recognized that current insulation used for the cryogenic propellant tankage does not have extensive reuse capabilities. It will therefore be necessary to ensure that the polyurethane insulation material or the replacement will be capable of reuse with minimum maintenance without causing hazards to system operation. Current programs encompass

the development of a high-chamber-pressure bell engine. The program to develop the main propulsion system must be continued to facilitate development of the logistic vehicle. Another major area for development is the reaction-control system using gaseous oxygen and hydrogen. Previous programs also have not developed an auxiliary power unit of the size and type selected for the ILRV, and this also will require further development. Another major area for technology advancement is the demonstration of the reusability of the logistic vehicle with minimum maintenance.

8.0 SUGGESTED ADDITIONAL EFFORT

Prior to development of the logistic vehicle, additional effort will be required in further study activities. These studies include investigations of various modifications of the basic ILRV shape which offer potential for improving vehicle stability, reducing severity of the thermal environments, and decreasing system weight. Specific additional activities should include potential improvement concepts which have been identified in the study and these items are listed in Table 5. Future programs must encompass extensive testing to develop the materials required for satisfactory performance of the shuttle system.

In general, an in-depth analysis will be necessary to develop comprehensive definitions of all subsystems, to develop confidence in the design projected weights, and to facilitate a more comprehensive resource analysis.

Table 5. Recommended Design Improvement Studies

Aerodynamics	Wind fillet and sweep to reduce interference Wing location and body shape to minimize tail size and weight Body-corner radius to reduce tail size and minimize buffeting Wing size and shape to improve flyback range and reduce weight
Cross-range	Low alpha effects Configuration changes to improve cross-ranging capability if desired
Heating	Fly constant-heat-rate reentry Increased radius on orbiter wing leading edge to reduce temperature
Abort	Failure-mode analysis to identify abort modes for projected failures Water-impact effects
Performance	Turbojet use on liftoff
Design/reduced gross weight	Reduce • On-orbit ΔV • Cargo weight/size Modify body shape, reduce wing if increased landing speed acceptable High thrust-to-weight turbojets and H ₂ fuel Canard configuration for booster to improve L/D and reduce weight Integrated structure and thermal protection refinement Use of composites for structure Reduce prelaunch loads by inclination or facility support or windshield Optimize maximum-ascent acceleration Modify wing incidence to reduce ascent loads

APPENDIX: VEHICLE DESCRIPTION

This table presents a descriptive summary of the integral launch and reentry vehicle configured to deliver a 50,000-pound payload within a 15-foot-diameter by 60-foot-length cargo bay.

Table A-1. Description of Final 50,000-Pound-Payload Vehicle

	Booster	Orbiter
Vehicle Weights		
Gross weight (lb)	3,619,872	870,977
Entry weight (lb)	587,522	256,515
Landing weight (lb)	532,002	254,807
Wing Geometry		
Span (ft)	244	146
Sweep (deg)	14	14
Taper ratio	0.319	0.333
Exposed area (sq ft)	6300	2010
MAC (ft)	37	21.4
Leading edge radius (in.)	7.1	4.05
Vehicle Dimensions		
Length (ft)	280	202
Planform area (ft ²)	19,770	8,200
Body wetted area (ft ²)	36,000	17,111
Wing wetted area (ft ²)	12,600	4,020
Horizontal-tail wetted area (ft ²)	4,080	2,340
Vertical-tail wetted area (ft ²)	3,180	1,170
Body mold-line volume (ft ³)	320,000	97,690
Volumetric efficiency ($V^{2/3}/A$)	0.12	0.124
Main Propulsion		
Propellant type	LO ₂ /LH ₂	LO ₂ /LH ₂
Mixture ratio	6:1	6:1
Engine thrust (1,000 lb)	510 (sl)	590 (vac)
Expansion ratio	35:1 (sl)	58:1 (sl)/120:1 (vac)
Number of engines	11	2
Firing sequence	1	2

Table A-1. Description of Final 50,000-Pound-Payload Vehicle (Cont)

	Booster	Orbiter
Turbojet Propulsion		
Engine type	JT9D-15	JT3D-7
Number of engines	4	4
Engine-out capability	Yes	Yes
Engine thrust, SLS (1,000 lb)	45.5	19
SFC (lb/hr/lb)	0.364	0.55
Launch Data		
T/W (initial)	1.25	1.36
Engine-out capability	Yes	Yes
Maximum acceleration (g)	3	3
Burnout time, from lift-off (sec)	220	440
Burnout velocity (fps)	10,866	25,855
Burnout altitude (ft)	245,000	303,806
Burnout range (nm)	98	680
Burnout flight path angle (deg)	6.9	0
On-Orbit Data		
Inclination (deg)		55
Injection conditions (nm)		50 x 100
Station altitude (nm)		270
Maneuver ΔV (total) (fps)		2,000
Entry Data		
Reentry angle (deg)	6.5 (Staging)	-1.5
Reentry velocity (fps)		25,134
Lateral range, (nm) $\alpha = 60$	48	190
Longitudinal range (nm)	310 (Flyback)	2,545
Lateral range, TMAX (nm)		1,240
Maximum acceleration (g)	4.0	2.08

Table A-1. Description of Final 50,000-Pound-Payload Vehicle (Cont)

	Booster	Orbiter
Recovery Data		
Flyback range (nm)	310	0
Landing speed (knots)	113	128
Touch-down angle of attack (deg)	8	8
Aerodynamic Parameters		
Hypersonic (L/D) maximum (inviscid)	2.05	1.53
Hypersonic (L/D) design	0.69	0.53
Subsonic (L/D) maximum	8.40	7.30
Entry planform loading (psf)	30	31
Landing planform loading (psf)	27	31
Landing wing loading (psf)	67	88
Reserves		
RCS (lb)	460	950
ECS (lb)	0	50
Power (lb)	200	880
Turbojets, go-arounds (lb)	10,600	3,900
Orbit man system 500 fps (lb)	---	9,600
Primary propulsion, FPR (lb)	0	4,700
Safety Factors		
Crew compartment	2	2
Primary structure	1.5	1.5
Secondary structure	1.5	1.5
Pressure tanks (propellant)	1.5	1.5
(pneumatic)	2.0	2.0
Payload Definition		
Weight (lb)		50,000
Size, diameter x length (ft x ft)		15 x 60

Table A-1. Description of Final 50,000-Pound-Payload Vehicle (Cont)

	Booster	Orbiter
Weight Factors		
Design stress allowable:		
Aluminum (2219-T87)	58.5 at 175 F	58.5 at 175 F
F_{tu} (ksi)	78.5 at -323 F	78.5 at -323 F
Titanium 6Al-4V STA,	129.5 at 400 F	129.5 at 400 F
F_{tu} (ksi)		
Safety Factors		
Crew compartment	2.0	2.0
Primary structure	1.5	1.5
Secondary structure	1.5	1.5
Pressure tanks (Propellant)	1.5	1.5
(Pneumatic)	2.0	2.0
Weight Margin		
Percent inert weight	10 percent and statistical data	10 percent and statistical data
How applied	Added at subsystem level	Added at subsystem level
Tank Sizing		
Ullage, percent LOX/H ₂	1.1 x Usable W_p	1.1 x Usable W_p
Tolerances, percent LOX/H ₂		
Residuals and Losses		
RCS, (lb)	100	200
ECS, (lb)	0	0
Power, (lb)	35	80
Turbojets, (lb)	200	100
One-man system, (lb)	-	330
Primary propulsion, (lb)	26,490	4,550