

Technology Influence On the Space Shuttle Development

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FOREWORD

NASA and the Air Force are engaged in studies to define transportation systems that will characterize the post-1995 space programs of the United States. A primary objective is the identification of technology efforts that should be initiated now so that the nation has alternatives to the technologies that determined the current Space Shuttle systems.

In the early 1970's, the systems for the present Space Shuttle were formulated by drawing upon the technology base established during progressive research and development programs of the 1960's. As part of the Space Shuttle program, three major technologies were developed and advanced significantly:

- the high-pressure, high-impulse oxygen/hydrogen main engine
- the primary avionics software systems that reside in the four primary general purpose computers
- the reusable surface insulation thermal protection system at one-tenth the mass/density of previous ablative systems

Other Space Shuttle systems made use of technologies that, although state of the art at the time, were initial configurations.

This report reviews the history of Space Shuttle development and operations, defines areas in which available technology limited program options, and identifies technical issues that can benefit future space transportation systems.

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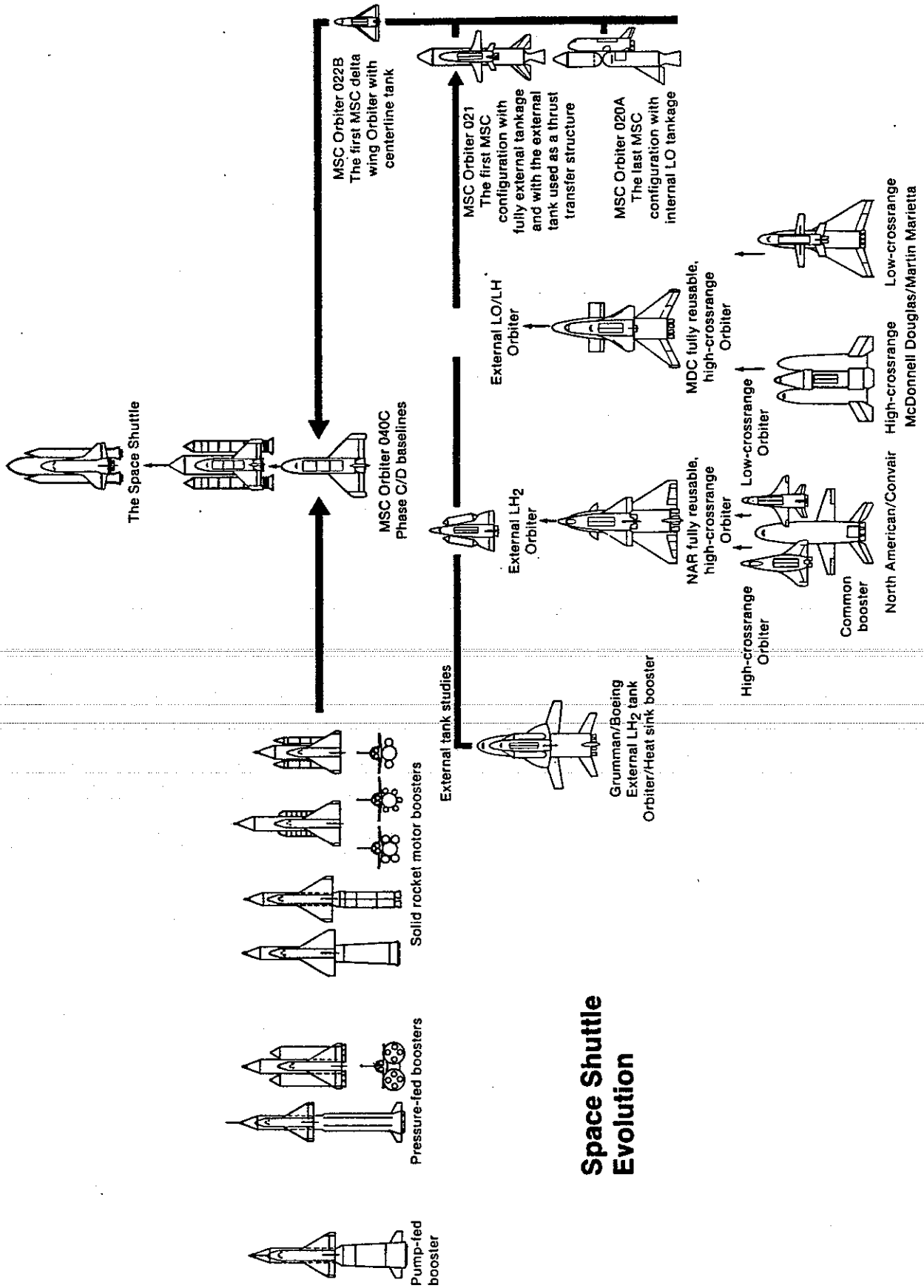
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Space Shuttle Evolution



The Space Shuttle

MSC Orbiter 040C
Phase C/D baselines

External tank studies

External LH₂
Orbiter

External LO/LH
Orbiter

MSC Orbiter 022B
The first MSC delta
wing Orbiter with
centerline tank

MSC Orbiter 021
The first MSC
configuration with
fully external tankage
and with the external
tank used as a thrust
transfer structure

MSC Orbiter 020A
The last MSC
configuration with
internal LO tankage

MDC fully reusable,
high-crossrange
Orbiter

NAR fully reusable,
high-crossrange
Orbiter

High-crossrange
Orbiter

Low-crossrange
Orbiter

Common
booster

North American/Convair

High-crossrange

Low-crossrange

McDonnell Douglas/Martin Marietta

SECTION 1
EXECUTIVE SUMMARY

SECTION 1
EXECUTIVE SUMMARY

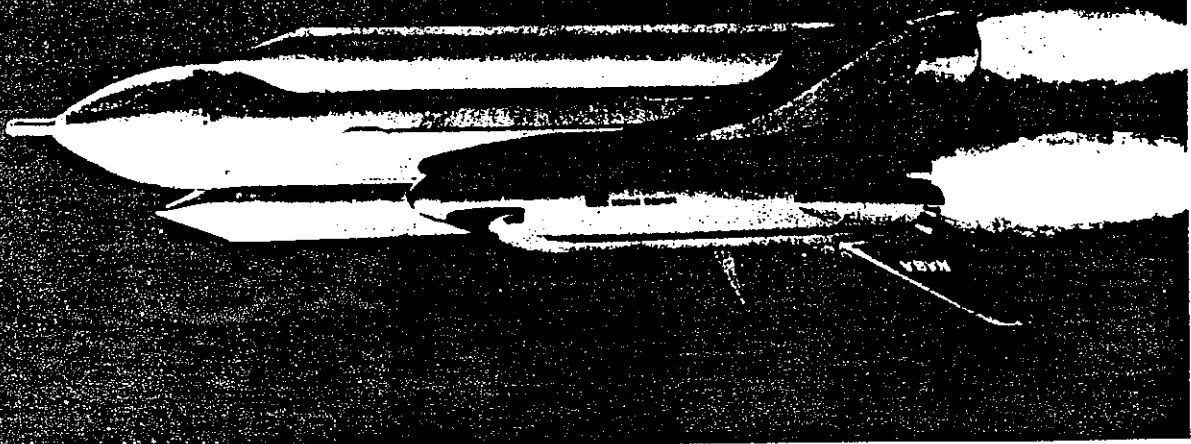
1.1 GOALS

The objective of the Space Shuttle Program was at its inception in the late 1960's, and still is today, "to provide a low-cost, economical space transportation system." To meet this objective, NASA has concentrated on a balance between a low development cost and the lowest practical operations cost. Factors that have had a significant influence on the operational cost of the system are flight rate, refurbishment cost, and expendable element cost.

The key technologies necessary to accomplish the program objective as identified during the Phase A and Phase B studies were as follows.

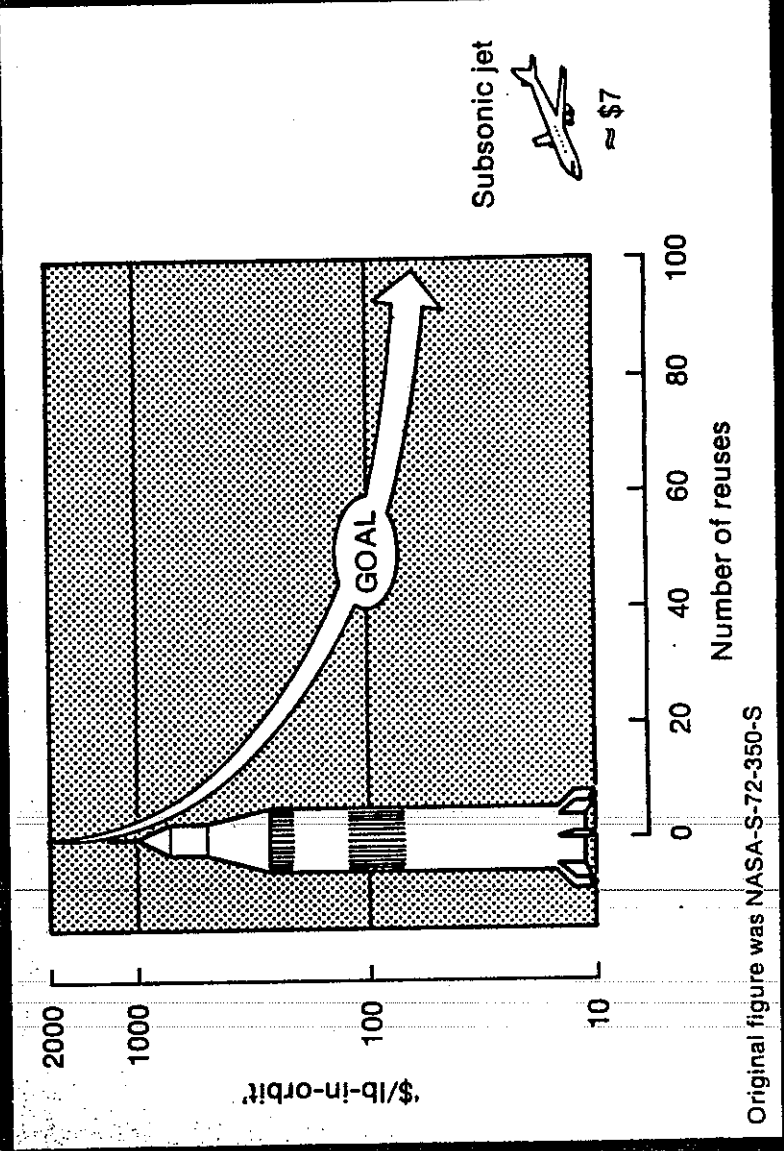
- A reusable, high-performance LO₂/LH₂ propulsion system, in the 500,000-pound thrust class, that would have a 50-mission life without overhaul. This objective was considered mandatory.
- A reusable Orbiter thermal protection system that could be refurbished within 40 hours.
- Element designs that would provide for 100 Orbiter and 20 solid rocket motor reuses.
- Aerodynamic and thermodynamic characteristics compatible with ascent, entry, and landing phases of flight.
- Avionic systems that would provide autonomous operation during ascent, entry, and return. Onboard checkout and malfunction detection.
- Subsystems that would require minimal consumables and which would be reusable for many cycles.

Original figure was NASA-S-73-254-S



SPACE SHUTTLE PROGRAM OBJECTIVE

ACHIEVEMENT OF AN ECONOMICAL SPACE TRANSPORTATION SYSTEM



Original figure was NASA-S-72-350-S

1.2 ACHIEVEMENTS: DEVELOPMENT COST

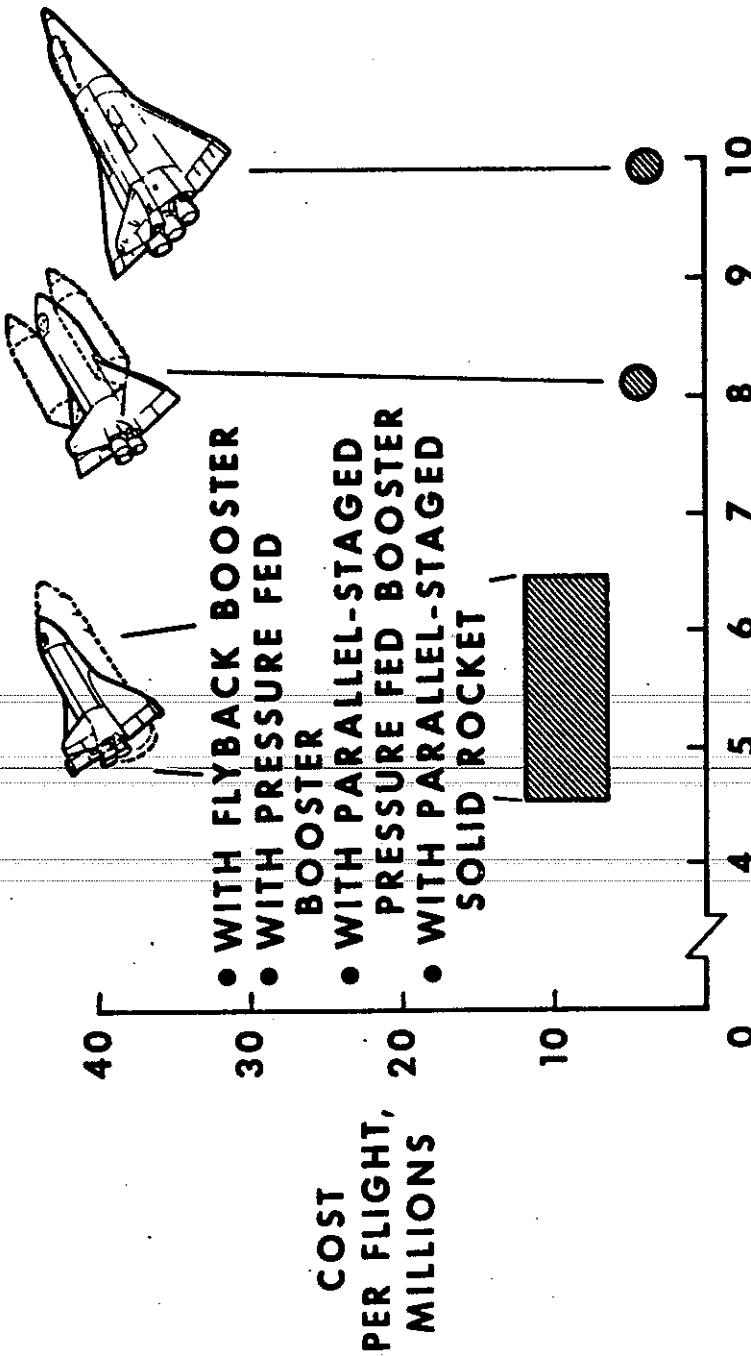
The program development cost was within the budget guidelines of \$5.15 billion after the 1970 inflation factors are subtracted from the total. To stay within the development cost guidelines, the number of new technologies, maintainability provisions, and logistics spares were minimized, which has caused an escalation of operations cost.

✓ In effect, only those technologies needed for performance goals were developed and those impacting cost of ownership were postponed. In many cases, the technology needed was simply neither at hand nor readily conceived. In either case, proven technology was selected to reduce development risk, even at recognized increased maintenance cost and turnaround time.

SPACE SHUTTLE COST COMPARISON

65,000 LB P/L - DUE EAST LAUNCH
15 FT DIA x 60 FT PAYLOAD BAY

| | EXTERNAL LO ₂ /LH ₂ TANKS | EXTERNAL LH ₂ TANKS | INTERNAL TANKS |
|--|---|-----------------------------------|-------------------|
|--|---|-----------------------------------|-------------------|



1.3 ACHIEVEMENTS: OPERATIONAL COST

The lack of sufficient technology advancements to date has prevented the Space Shuttle from achieving the desired low refurbishment and reuse rate. A higher-than-desired refurbishment rate for the Orbiter and Space Shuttle main engines (SSME's) and low flight rate have contributed to a higher-than-anticipated operating cost. The Space Shuttle Program System Specification (JSC 07700, volume X) requires the following as related to maintainability, refurbishment, and reuse.

- The Space Shuttle flight systems shall be designed such that a 160-hour (10 working days) turnaround can be achieved (ref. p. 3-132, par. 3.5.2.1).
- The Orbiter vehicle shall be capable of performing 100 orbital missions. The thermal protection system (TPS) shall be capable of performing 100 orbital missions and the time to accomplish maintenance/refurbishment shall not exceed 40 hours (ref. p. 3-133, par. 3.5.3.1.1).
- The SSME's shall be capable of a service life, between overhaul, of 55 starts or 7 1/2 hours (27,000 seconds) total time including not more than 14,000 seconds at full power level (FPL) (ref. p. 3-134, par. 3.5.3.1.5).

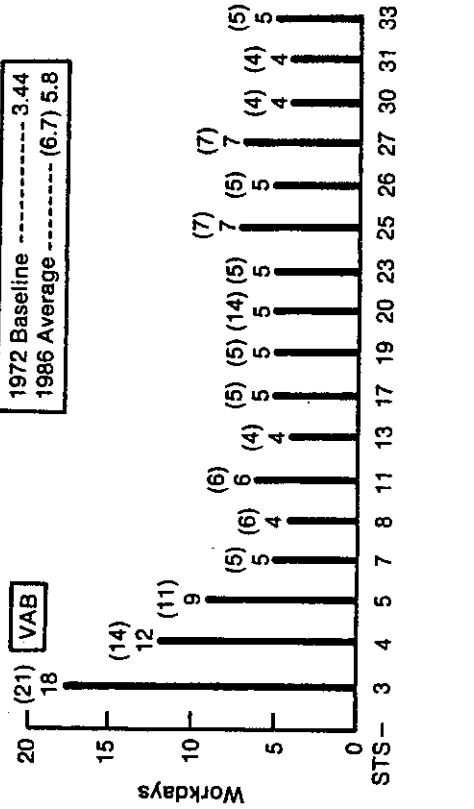
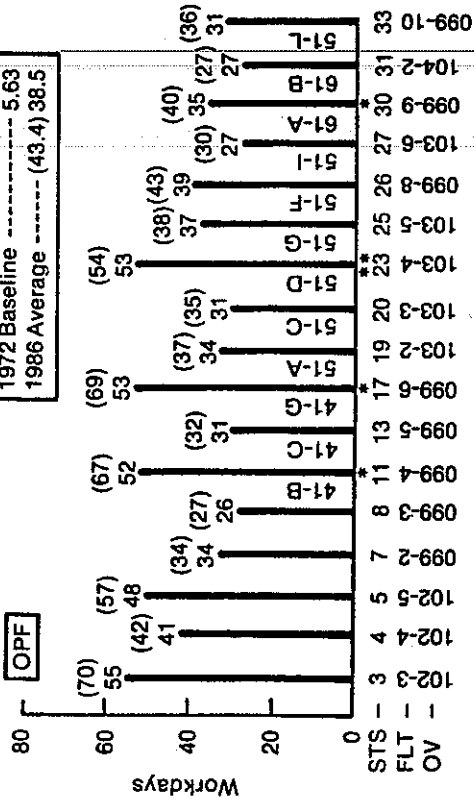
A comparison of the baseline specification turnaround time allocations versus the present actuals experienced to date is depicted on page 1-7. These actuals, which exceed 1972 baseline predictions, have contributed to the Space Shuttle Program not currently meeting operating-cost goals and indicate the need for greater technology emphasis on maintainability, refurbishment, and reuse.

The current significant factors influencing turnaround are listed as follows:

- SSME reverification is in the critical path of Orbiter Processing Facility (OPF) flow
 - oxidizer and fuel turbopumps
 - leak and functional test
 - frequency response test
 - heat shield removal and replacement
 - hydraulics
- Payload bay reconfiguration
- TPS repair and refurbishment
- Operational maintenance instruction (OMI) tasking

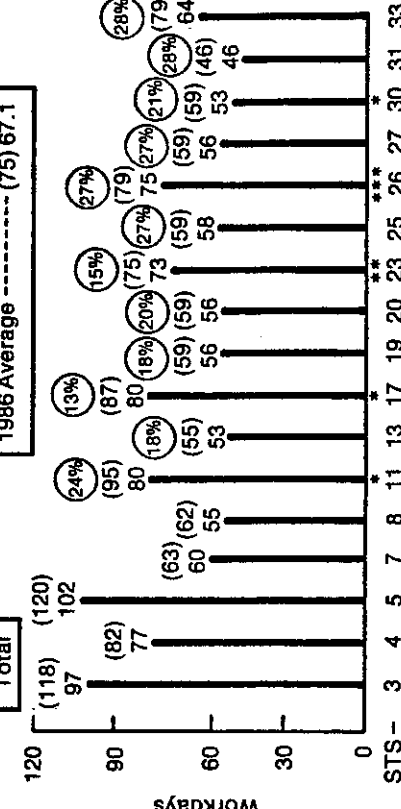
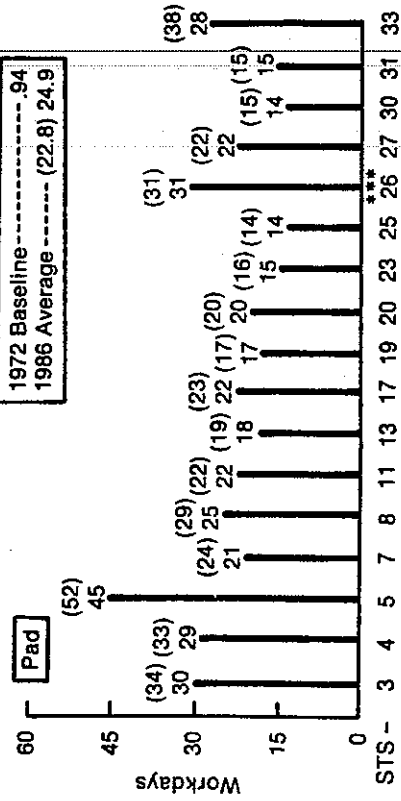
Space Shuttle Processing Workday "As Run" Summary

SO-MPO
March 1986



1972 Baseline ----- 5.63
1986 Average ----- (43.4) 38.5

1972 Baseline ----- 3.44
1986 Average ----- (6.7) 5.8



1972 Baseline ----- 94
1986 Average ----- (22.8) 24.9

1972 Baseline ----- (14) 10
1986 Average ----- (75) 67.1

Note: Excluded - 1st flows each Orbiter (STS-1, 6, 14, 28) and unique flows (STS-2, 9, 24, 32)
* Powerdown mod/Centaur mod times in OPF not included
** Cargo manifested during flow plus payload bay door (PLBD) incident
*** On-pad abort

Legend: (O) (Cal. days)
Work days

Figure 1-3.- Turnaround comparison - 1972 baseline vs. 1986 average.

- Brake repair and refurbishment
- Crew module leak test
- Environmental control system (ECS) circulation pumps, fans, and filters
- Hydraulic horizontal and vertical test - gas in system

SECTION 2
INTRODUCTION

SECTION 2 INTRODUCTION

This study reviews Space Shuttle design decisions involving key technology issues. These decisions are discussed in sections 3 through 6 in chronological order, Pre-Phase A through Phase C/D. Some of these technology issues have now been resolved; others are unresolved and will be significant in the design of future vehicles. These technology issues are listed in section 7.

No decision in a large, complex program is the consequence of a single factor. In such programs, there is a significant interaction between budget availability and technology choice; most technical issues can be resolved if adequate resources can be brought to bear and development time can be allowed. Because technology uncertainty or development maturity can be assessed in terms of cost, there is often a sense that cost constraints determine program content. It is equally valid to observe that technology maturity determines cost and schedule. While there were significant budget limitations on the Space Shuttle development due to the cost of the Vietnam war and other budgetary pressures, it is equally true that the technology base at hand in 1970 had an equally constraining effect on the configuration options for a new space transportation system.

Coincident with the initiation of the Apollo Program, public and private planners began to contemplate space activities for the post-Apollo period. Prior to this time, a significant technology data base was evolving in the area of advanced structural/thermal materials, digital computer based controls, cryogenic propellant rocket engines, lifting-body entry vehicles, and man-rating concepts for highly reliable systems.

NASA personnel had gained considerable experience from laboratory research, rocket and lifting-body flight programs, and manned space flight programs. From the knowledge gained from the Mercury, Gemini, Apollo, and Skylab programs with highly focused research and development (R&D) objectives, NASA personnel were ready to apply their experience to the design of space logistics vehicles to be used in an operational mode. The focus of the Space Shuttle Program was on a fully reusable system which would evolve and improve technologically and economically during its operation. The ultimate goals under consideration in the early 1960's included interplanetary travel (primarily to Mars), establishment of a lunar base, and activation of a space station. All of these goals required various forms of logistics vehicles as part of each mission.

By 1962, NASA had established a Space Station Office at the Manned Spacecraft Center (MSC) in Houston, Texas, to begin planning for a program which recognized the requirement for a logistic vehicle. Two questions prevailed in the consideration of the logistic vehicle at this time: (1) Should a totally reusable booster and orbiter be built? and (2) Should off-the-shelf Apollo hardware be used? Subsequent studies turned the focus on the reusable booster/orbiter concept.

In the late 1968 to early 1969 timeframe, NASA planners viewed the space shuttle as only one of many projects of the nation's overall space effort. A 1969 "Summary of Current Office of Manned Space Flight (OMSF) Integrated Plans for the 1970's and 1980's" depicted the space shuttle as the "Earth to Orbit Space Shuttle," or simply one of a number of shuttle craft to be built for Earth orbit and lunar operations. Over the next 15 years, however, the Space Shuttle became, in fact, the cornerstone of spacecraft development for the nation.

The significant conceptual goal of the shuttle-planning effort in 1968 was to produce a space vehicle operation that would achieve an order of magnitude recurring cost reduction over previous space launch vehicle systems with greatly improved safety. The NASA request for proposal (RFP) (Phase A) for study of an integrated launch and reentry vehicle (ILRV) was released for bid in November 1968. This RFP described the program objectives as the development of "safe and economical two-way transportation of men and cargo between Earth and orbit." The approach emphasized the derivation of a new integrated system utilizing advanced logistic spacecraft and expendable launch vehicles designed for operational economy and safety. Advanced technologies and techniques such as medium-to-high lift-to-drag and 1-1/2-stage-to-orbit concepts, as typified by an integrated launch vehicle/spacraft and expendable tanks, were to be considered. Potential cost reductions were to be investigated, including recovery and reuse of expensive components; simplification of test and checkout operations; low-cost fabrication techniques; simplification of specification and documentation requirements; and emphasis on economy and safety rather than payload performance. The investigations were to include the use of large safety factors or other suitable vehicle design features that could preclude catastrophic failure modes.

As the various study and advisory groups active in the 1968-70 period refined their thinking and the results of the Phase A investigations became available, the recommendations of the various groups and the NASA Space Shuttle System Program Definition Phase B Statement of Work (SOW) confirmed the goal of establishing an airline-type space operation for passenger and cargo transport. The Phase B RFP, released in June 1970, clearly enunciated the requirement for a reusable system with a high-launch-rate capability, short turnaround and reaction times, a flexible capability to support a variety of payloads and missions, and an operational mode that would reduce costs an order of magnitude below those prevailing at the time.

Phased program development planning, the procedure in use for new start program definition in the 1960's, called for four phases of activity. The first phase, phase A, is an assessment of future requirements and preliminary definition of systems that might satisfy requirements. The objective of phase B is to refine the definition of the best choice among the alternative systems identified in phase A. Customarily, the phase B study starts with the "best fit" to the requirements of the phase A studies and relaxes requirements as the design definition process identifies inadequacies of available technology and fits the development plan to the expected budgetary resources and schedules. Phase C/D effort provides for the development and test of the system defined in phase B and again accommodation must be made to the technology shortfalls and achievements experienced in the evolving development process. Finally, there is the operational experience of the system in which the technologies selected may not prove as robust as anticipated or hoped, and budgetary constraints may impose unanticipated operating constraints because the capitalization of the operating system (spares, facility upgrades, etc.) may not be as robust

as anticipated in definition and development. Most, but not all, systems achieve significant capability, mature in operation, and satisfy most if not all of the goals established for them. The disparity in goals defined and achievements attained is as often an overstatement of the goals as it is a shortfall of the accomplishment.

The themes of substantial cost reduction in space operations through low-cost refurbishment and maintenance, and flexibility in supporting a wide range of scientific, defense, and commercial uses continued to mature as the more specific requirements of the proposal for design, development, test, and evaluation (DDT&E) were released for bid on March 17, 1972. Requirements for the system described an orbiter capable of 500 reuses over a minimum period of 10 years. The requested system included an orbiter vehicle with installed main engines and reusable solid rocket motors which burned in parallel with the LO₂/LH₂ main engines. Other important features included an expendable main propellant tank; orbiter self-ferry; and the capability of crossrange maneuvers during entry, aerodynamic flight, and horizontal landing. Recognizing the inherent limitations of flight frequency, an evaluation of the impact of lowering the reuse target to 100 was requested.

By 1973, the Systems Requirements Review baseline configuration of the Space Shuttle flight and ground system had been established to satisfy the requirements of the Space Shuttle Flight and Ground System Specification, JSC 07700, volume X. This specification required the useful life of the Space Shuttle system to be a minimum of 10 years, and the orbiter vehicle was to be capable of performing 100 missions, including ground turnaround operations, with scheduled subsystem maintenance and/or refurbishment, or replacement as determined to be most cost effective. The TPS for the orbiter vehicle was to be capable of performing 100 orbital missions, with unscheduled subsystem maintenance and/or refurbishment, or replacement, as determined to be most cost effective. Turnaround time of the TPS was not to exceed 40 hours and was to be consistent with normal scheduled maintenance activities. The TPS thermal/structural design was tied to a 100-mission life as was the orbiter structural design. However, as a goal, the orbiter vehicle structure was to be capable of 500 reuses.

Other major requirements included an expendable external hydrogen/oxygen tank and two reusable solid rocket boosters capable of 20 vertical launches with recovery, retrieval, and refurbishment after each mission. The SSME's were to have a service life, between overhaul, of 55 starts or 7 1/2 hours (27,000 seconds) total time including no more than 14,000 seconds at full power level. All of these volume X requirements remain in the current edition of the specification.

The actual reuse capability of the orbiter as specified by the volume X specification was to be determined as the flight program progressed through analysis of flight results and/or additional structural tests. Refurbishment and/or modifications to achieve the specified reuse capability were permissible. The subject of reuse was a critical issue as it pertained to the characterization of the Space Shuttle operations era.

This study provides a look at the key technology decisions within the context of this Space Shuttle Program evolution and the supporting rationale. It does so in the light of subsequent events; it also identifies areas where future technological development effort can be directed to meet the needs of advanced space transportation systems.

SECTION 3
PRE-PHASE A ACTIVITIES

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PRE-PHASE A ACTIVITIES

The following chronology, taken entirely from the Rice University Space Shuttle History Project Chronology (ref. 3-1), sets the stage for Space Shuttle development. The need for and technology required to develop a reusable Space Shuttle are stressed.

A. JULY 16, 1962

Edward H. Olling, of the MSC Space Station Program Office, submitted a proposal for a Space Station Program which incorporated consideration of a number of logistic spacecraft for support of the Space Station. These were

- Apollo
- Diana (Apollo modified - reusable - six man)
- Gemini
- Twelve-man transport - designed to Space Station requirements

Olling envisioned launch rates of up to 70 per year if the Gemini spacecraft were used for logistic support of the Space Station and if a Space Station crew of 25 with a 1-month rotation cycle were used. His cost comparisons of the various spacecraft options showed a significant advantage in total program costs for a reusable 12-man transport (ref. 3-2).

B. SEPTEMBER 14, 1966

The Ad Hoc Subpanel on Reusable Launch Vehicle Technology . . . issued its final report. Running to some 138 pages of text plus introduction and appendixes, the report primarily addressed technical issues, while it largely avoided making quantitative or economic analyses. About its inquiry into reusable spacecraft, the subpanel gave this general assessment:

"It is important to note that no single, most desirable vehicle concept could be identified by the Subpanel for satisfying future [Department of Defense] DOD and NASA objectives. Consequently, a number of reusable launch vehicle configurations were selected by the Subpanel and operating modes of greatest potential interest to the DOD and NASA were defined to provide a realistic means for the identification and assessment of the critical supporting technologies. The selected vehicle concepts included both fully recoverable and partially recoverable vehicles. These advanced concepts were specifically chosen to be typical and

representative of future development possibilities, and to reflect a time-phased evolutionary pattern of growth capability consistent with potential needs beyond the early 1970's.

* * * * *

"The Subpanel notes that a partially reusable launch vehicle involving recovery and reuse of a stage or certain major components would cost less to develop and might be amortized in a shorter period with fewer flights. For this reason the Subpanel has emphasized that partially reusable concepts could be competitive with uprated existing systems and advanced expendable vehicles in the 1975 period."

The subpanel identified the lack of wind tunnel data on "realistic vehicle configurations" as "the most serious deficiency in the aerodynamics of reusable launch vehicles." It found aerodynamics technology to be "well advanced" for reusable ballistic spacecraft, except for land landing systems, to be "advancing" at a reasonable rate for moderate lift-to-drag (L/D) lifting-body configurations, and "less developed" for "higher L/D spacecraft and ... integral upper-stage/spacecraft combinations." Problems in structures and materials were described as "pressing," with problems associated with thermal protection systems and cryogenic tankage identified as particularly critical. The promise of LO/LH rocket engines capable of "up to 50 reuses before major overhaul" was specifically underlined, as was the importance of competition between solid and liquid-fueled "expendable first-stage applications." The report identified the hydrogen-fueled turboramjet as having promise for horizontal-landing booster application by the mid-1970's (ref. 3-3).

C. FEBRUARY 1967

The President's Science Advisory Committee (PSAC) issued a report, The Space Program in the Post-Apollo Period, that helped set in motion the process of planning for future space activity by NASA and the nation that would ultimately lead to the Space Transportation System. The PSAC was concerned with the conceptual underpinnings of planning for post-Apollo space flight rather than the details of such activity, and, as a result, it did not attempt to suggest new technology, or even predict it, for future space activity. Thus, the idea of a space shuttle, as it later came to be developed, was never mentioned. But the types of goals, and the problems associated with those goals discussed by the PSAC, were the medium out of which the idea of a space shuttle first emerged and then finally grew into the Space Transportation System.

The PSAC maintained that "an attractive long-term objective for space exploration is a program directed ultimately at the exploration of the planets by man." While clear on the ultimate goal, the PSAC was less certain about the best means for achieving that goal: "The specific steps involved in moving towards planetary exploration are less clear, especially regarding the proper relationship between the manned and unmanned programs."

The PSAC had "serious reservations" about pushing too soon for manned interplanetary space flight. But it saw a space station as a valuable research and development tool for preparing for that more distant goal: "The primary justification for a [space] station with an indefinite lifetime is that NASA must deal with the space environment in almost all of its planning for manned missions, and needs to make long-time studies in this environment on a systematic basis."

The PSAC recognized that a space station, or at least frequent human entry into space, required some means for lowering the cost of putting people into space: "Because of the continuing requirements for manned and manned-attended systems we visualize that an important problem will be posed for a long time by the cost associated with taking men to and from orbit." The PSAC specified some solutions that involved existing expendable spacecraft. But the PSAC also recommended: "For the longer range, studies should be made of more economical ferrying systems, presumably involving partial or total recovery and reuse." (Ref. 3-4.)

D. FEBRUARY 27, 1968

George E. Mueller, Associate Administrator for Manned Space Flight, gave a speech to the Economic Club of Detroit, Michigan. In this speech, Mueller affirmed NASA's commitment to manned space flight, and postulated that fully reusable spacecraft were to be "the backbone of our space activities of the future." Projecting that costs could be reduced to "\$50 per pound or even \$5 per pound for getting into and out of orbit," Mueller predicted that "low-cost space transportation . . . in time, could be competitive on a cost-per-pound/mile basis with other forms of transport. Such a transport system could be developed, using reusable launch vehicles and spacecraft which would take off and land at 'spaceports' much like our present-day airports." Mueller went on to say: "I feel that this matter of low-cost transportation is going to be an area of maximum payoff in the years ahead in the space program. It may well turn out to forecast the whole future of our economy in a transportation sense." (Ref. 3-5.)

E. JULY 1, 1968

The OMSF issued a summary of selected NASA advanced vehicle planning studies entitled "MSF Vehicle Studies." OMSF prepared this summary to "parallel" the advanced missions planning that was also being conducted by OMSF as part of the general post-Apollo planning effort that would soon result in the decision to build the Space Transportation System. The OMSF summary consisted both of a brief overview and assessment of selected vehicle studies and of charts listing the studies evaluated with brief outlines of their salient points. The OMSF summary focused on four different areas of vehicle studies concerned with the "support of future manned and unmanned flight programs in the 1970 decade," but only one of these, "Space Logistics System," had direct relevance to the development of the space shuttle.

About the logistics studies, the OMSF summary explained: "Studies of space logistics systems were conducted in two major groupings: (1) those directed toward the evaluation of a near-term operational system, and (2) those directed toward an advanced, fully reusable transportation system." Although the OMSF summary noted that the studies of a near-term operational system evaluated "logistics spacecraft having hypersonic L/D ratios varying from 0.25 to 3.0," the summary observed:

"From these studies it became apparent that the ballistic spacecraft performs a nominal logistics mission at lowest cost and with adequate operational performance for 1970-1975 operations. In addition, the attractiveness of crossrange capability provided by the high L/D study configurations was reduced by crew survivability and spacecraft reuse considerations for all nominal and contingency landing situations."

The type of logistics studies that had the most relevance to the development of the space shuttle were those concerned with "an advanced, fully reusable transportation system." According to the OMSF summary: "The second major grouping of FY 63, 64, and 65 studies were directed toward the conceptual design of a fully reusable 'air transport type' logistics support system for round-trip Earth-orbital transportation." Based on the studies of a fully reusable transportation system, the OMSF summary concluded:

"Such a system has an apparent high-development risk and cost and requires very high annual launch rates over an extended period (15 to 20 years) to amortize its development costs . . . For post-1976 operations, a new partially reusable transportation system is feasible and appears attractive from operations economics and versatility. Salient system characteristics are noted: reusable, increased crew size spacecraft with lifting plan form [sic] (hypersonic L/D on order of 1.5 to 2.0), and integral propulsion system; onboard checkout system; integral cargo-carrying capability. Launch vehicle stages are primarily expendable; payload performance is on order of 60,000 to 80,000 lb. Fully reusable "aircraft type" launch systems appear at this time to be all in the 1985 and beyond period. (Ref. 3-6.)

F. AUGUST 10, 1968

George E. Mueller, Associate Administrator for Manned Space Flight, gave a speech to the British Interplanetary Society (BIS) of London, England, outlining NASA's aspirations for future space programs in a way that stressed the importance or reducing the cost of space operations. In his speech, Mueller made the first public reference to the Space Shuttle as a name for a specific vehicle concept; before this, space shuttle had only been used as a generic term for a reusable Earth-orbit spacecraft. Stressing economy in space operations, Mueller gave this explanation of why he saw the Space Shuttle as being crucial to NASA's future operations:

"Our studies show that using today's hardware, the resupply cost for a single three-man orbital space station for a year equals the original cost of the space station. This type of cost analysis had led us to carefully evaluate concepts for more efficient resupply systems."

Manufacturing in space, fuel and supply storage for deep space operations, and life support for crews onboard space stations require not tons, but thousands of tons of material, to be shuttled in and out of space.

Therefore, there is a real requirement for an efficient Earth-to-orbit transportation system -- an economical space shuttle. This need has been under study by long-range aerospace planners for over a decade. The objective of these investigations is to find a design that will yield an order of magnitude reduction in operating costs. The elements to which we must look for cost reductions are aircraft manufacturing techniques, aircraft development test procedures, maximum flexibility for multiple use and volume production, long life components for repetitive reuse, and airline maintenance and handling procedures for economy of operation.

In a chart that he displayed to his BIS audience, Mueller listed the following features that he considered necessary for an effective Space Shuttle.

Operational Characteristics of the Space Shuttle

1. Take off and land from air bases or major airports
2. Refuel with ordinary cryogenic fuels
3. Self-contained onboard checkout
4. Continental and/or intercontinental traffic control
5. Standardized ground support equipment
6. All weather operation
7. Passenger safety and comfort comparable to large transport aircraft
8. Standardized equipment and processes for refurbishment
9. Progressive maintenance
10. Will carry 25,000 to 50,000 pounds payload at a cost approaching \$5 per pound

As envisioned by Mueller in his speech, the Space Shuttle was to be a vertically launched vehicle with "crews similar in size to those required for intercontinental jet(s)." The Space Shuttle was to "glide to a runway landing, with practically no noise," and "the landing would be completely automated with prime dependence upon the spacecraft guidance system but with ground control backup." (Ref. 3-7.)

G. OCTOBER 30, 1968

MSC and MSFC jointly issued an RFP for "Study of Integrated Launch and Reentry Vehicle System (ILRV)." The ILRV study eventually came to be known as Phase A of the Space Transportation System (STS) Program. The RFP set forth the basic criteria for the ILRV study. This was an 8-month study "to examine several configurational concepts and operational approaches to a versatile round-trip transportation system, with a goal of achieving order of magnitude recurring cost reductions and greatly improved safety." The RFP specifically emphasized that "the systems discussed herein do not represent approved programs, and this study will not necessarily lead to hardware projects." The following "Guidelines and Assumptions" were set forth in the RFP (ref. 3-8).

1. The mission model included such basic assumptions as the ILRV providing logistic support for a space station with the nominal orbital inclinations "varying between 28° and 90° with 50° as the preferred inclination, and altitudes varying between 100 and 300 nautical miles with 260 nautical miles as the preferred altitude." Also the mission model stipulated: "the duration of the primary mission is 24-hour ascent time, up to 180 days quiescent orbital storage, and 24-hour descent time . . . The spacecraft will have the capability to land at a preselected site(s) located in the continental U.S. at least one/24-hour period for the worst case mission."
2. The ILRV's payload was to be in the 5000- to 50,000-pound range.
3. The system will accommodate up to 12 people plus significant return cargo inside a reusable vehicle. Cargo and passenger provisions will be interchangeable. There may be external provisions to accommodate one-way cargo in disposable containers.
4. A new integrated system utilizing advanced logistic spacecraft and expendable launch vehicles designed with emphasis on operations economy and safety. This effort should include consideration of solid and liquid stages of simplified design.
5. A new logistic space vehicle system which would utilize more advanced technologies and techniques such as possibly medium to high L/D and 1-1/2-stage-to-orbit concepts as typified by an integral launch/spacecraft and expendable tanks. Expendable "zero" boost stages may be considered in the system sensitivity and configuration analysis for purposes of additional mission versatility, performance, etc.

6. Areas to be investigated for potential cost reductions should include: recovery and reuse of expensive components; simplification of test and checkout operations; low-cost fabrication techniques; simplification of specification and documentation requirements; and emphasis on economy and safety rather than optimized payload performance. As an example, these considerations should include the use of large safety factors or other suitable vehicle design features which could preclude catastrophic failure modes.
7. The vehicle will be operational in the post-1974 time period.

SECTION 4
PHASE A (1969)

SECTION 4
PHASE A (1969)

The Phase A activity, designated as integrated launch and recovery vehicle (ILRV), was devoted primarily to a search for a low-cost-per-flight transportation system. This process began in March 1969 with the selection of four study contractors. The four subsequent contracts, plus one unfunded study, were devoted to various combinations of expendable and totally reusable modules. In all cases, the orbiting element was designated as reusable.

The conclusions of the Phase A studies, in conjunction with the MSC in-house studies, resulted in the recommendation to proceed into Phase B with the totally reusable booster and orbiter approach as being the best cost-per-flight solution, considering the mission model, which indicated a continuing and significant growth in traffic. The technical risks were perceived to be with the engines, thermal protection system, configuration, and avionics.

Most of section 4 was taken, only slightly modified, from the NASA Space Shuttle Task Group Report (ref. 4-1) produced at the end of Phase A. As such it represents directly the Phase A results, costs, and decisions existing at that time. The object of this study was not to judge or assess in hindsight those decisions, but to make explicit for future studies the implicit and explicit assumptions that governed or became embedded in those decisions.

All dollars referred to in section 4 are 1969 dollars.

4.1 VEHICLE CONFIGURATIONS

Three major classes of vehicles were studied in Phase A:

- Concept I - An expendable booster with a reusable flyback spacecraft
- Concept II - A partially reusable vehicle
- Concept III - A fully reusable system

Each concept considered was assessed in terms of its reuse and expendable value. Several of the systems proposed were advertised to have a high degree of reusability.

4.1.1 Expendable Booster Plus Reusable Spacecraft (Concept I)

The configuration that represents this category consists of a large solid first stage, a low-cost, pump-fed second stage, and a reusable, lifting-entry spacecraft, with an integrated payload compartment of 10,000 ft³. Several body shapes and decoupled (or semidecoupled) landing modes were considered. A parabolic-shaped spacecraft is shown in figure 4-1 as a representative concept in this class. Integrated electronics equipment was an integral part of the reusable spacecraft, providing substantial savings on these expensive elements. Semiballistic entry vehicles were examined for lower cargo weights, but became impractical for return payloads in the 50,000-pound class.

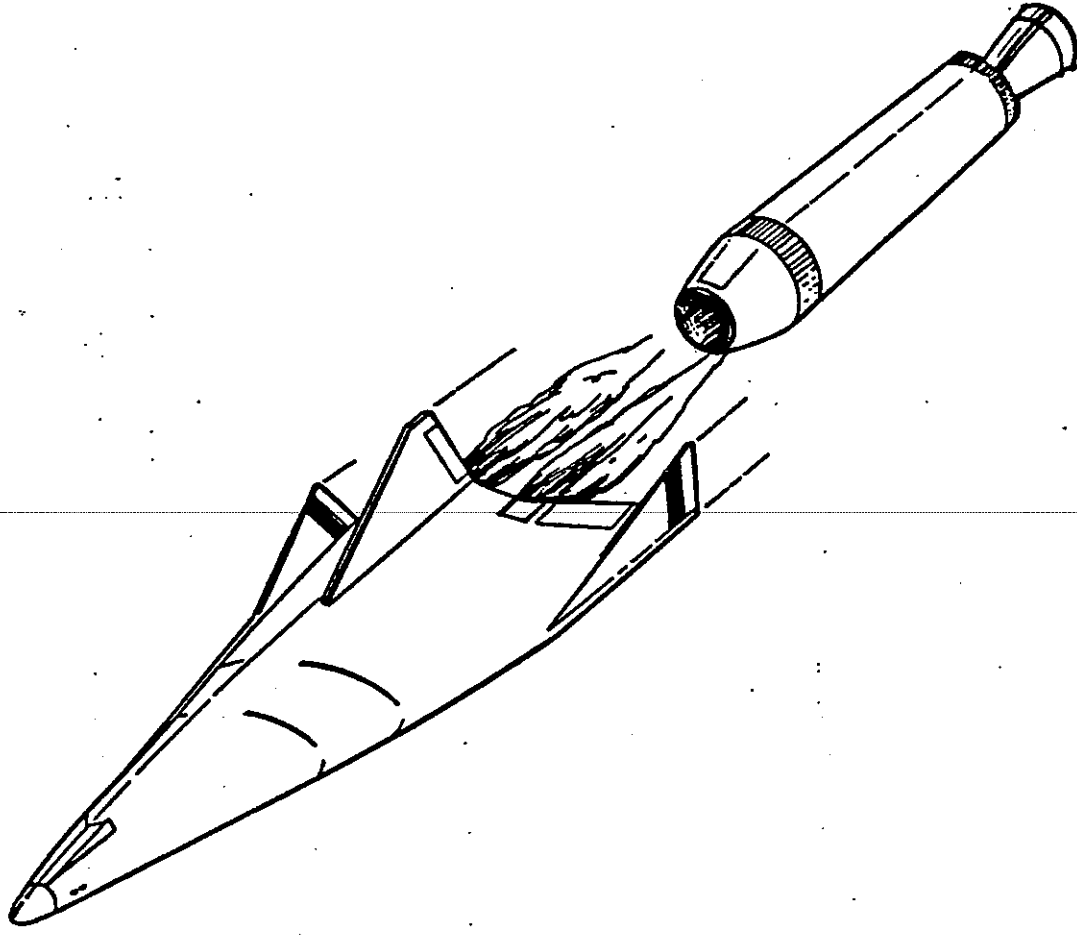


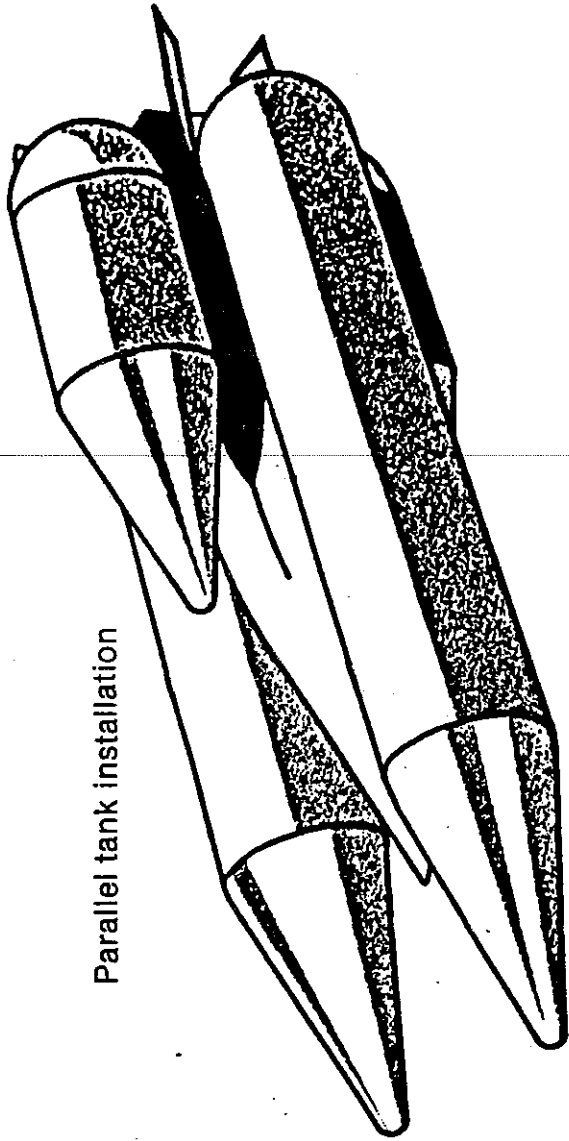
Figure 4-1.- Expendable launch vehicle with reusable spacecraft.

4.1.2 Partially Reusable Vehicles (Concept II)

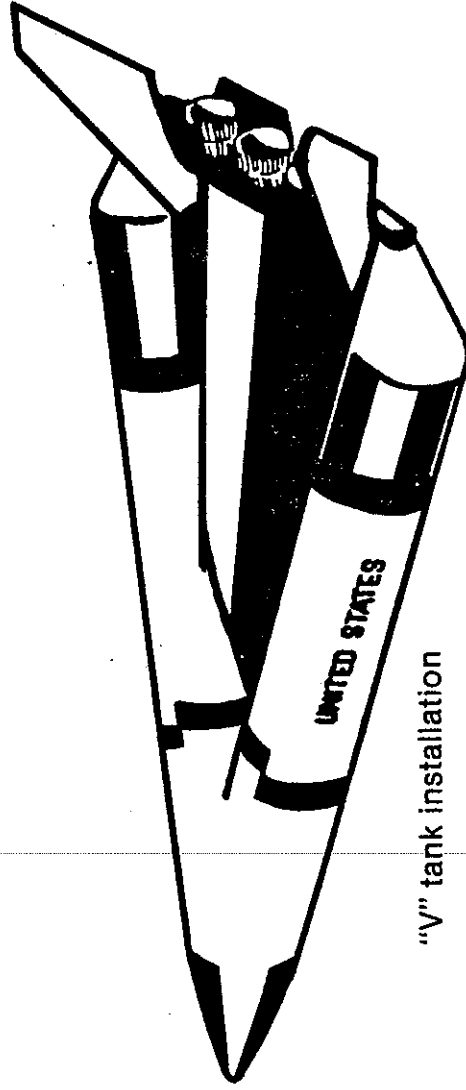
The 1-1/2-stage (drop tank) concept was an integral launch and reentry vehicle which attempted to reduce recurring costs by incorporating all expensive items in the reusable core vehicle and which contained the first-stage fuel in low-cost, expendable tanks. This concept is depicted in figure 4-2.

Some of the major design alternatives for this concept are illustrated, including the basic body shape for the reusable core, parallel vs. inclined arrangement of the expendable tanks, and the possible use of a second set of drop tanks to reduce the internal volume and size of the reusable core vehicle.

A typical vehicle was powered by 4 to 8 high-performance, liquid hydrogen-oxygen rocket engines with up to 4 million pounds of total thrust. High-pressure bell and Aerospike engines were design alternatives. The expendable propellant tanks were to be of simple and low-cost design and construction, within the weight constraints essential to acceptable payload performance.



Parallel tank installation



"V" tank installation

Figure 4-2.- 1-1/2-stage concept (drop tank).

4.1.3 Fully Reusable Systems (Concept III)

Prior to and during Phase A, many varied concepts for a fully reusable, round-trip, low-Earth-orbit transportation system were analyzed. The study results narrowed the field of reusable concepts to those involving boost stages with a vertical takeoff mode and a horizontal aircraft-type terminal landing mode. The results also indicated the desirability of using an all-rocket-boost engine system as opposed to combining airbreathing engines with rockets to boost. In June 1969, there was no preferred concept. However, the following five concepts held the greatest promise; their typical configurations are depicted in figure 4-3.

1. Triamese Concept - a parallel-stage arrangement with parallel-stage thrusting and propellant crossfeed wherein the first stage was divided into two identical elements as similar as possible to the orbital second stage.
2. Parallel/Parallel Concept - a parallel-stage arrangement with parallel-stage thrusting and propellant crossfeed resulting in two similar but different size stages.
3. Parallel Stage/Sequential Burn - a parallel-stage arrangement with sequential-stage thrusting.
4. MSC Concept - a parallel-stage arrangement with sequential-stage thrusting and using fixed wings and a high angle-of-attack reentry mode.
5. Langley Research Center (LaRC) Concept - a tandem-stage arrangement with sequential-stage thrusting and fixed geometry lifting planform shape.

Each of these concepts attempted to achieve unique economy in development and operations by capitalizing on concept characteristics such as stage and engine arrangement, aerodynamic shape, terminal landing requirements, sequential or parallel thrusting, cargo arrangement, and reentry mode and thermal protection systems. Within this framework, however, the final configuration design having the best combination of tradeoff features and resulting in the greatest overall savings to program cost without reduction in safety would be selected.

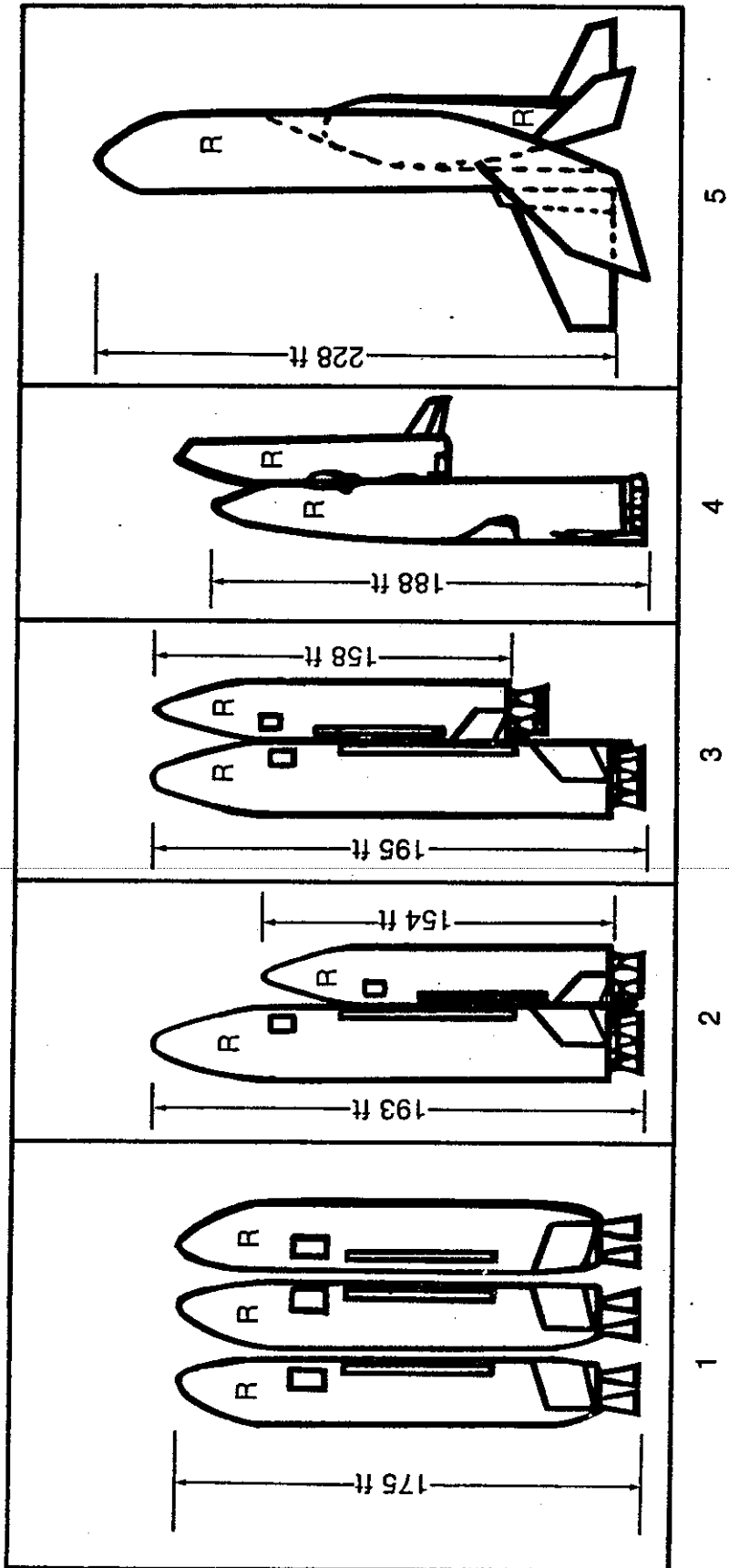
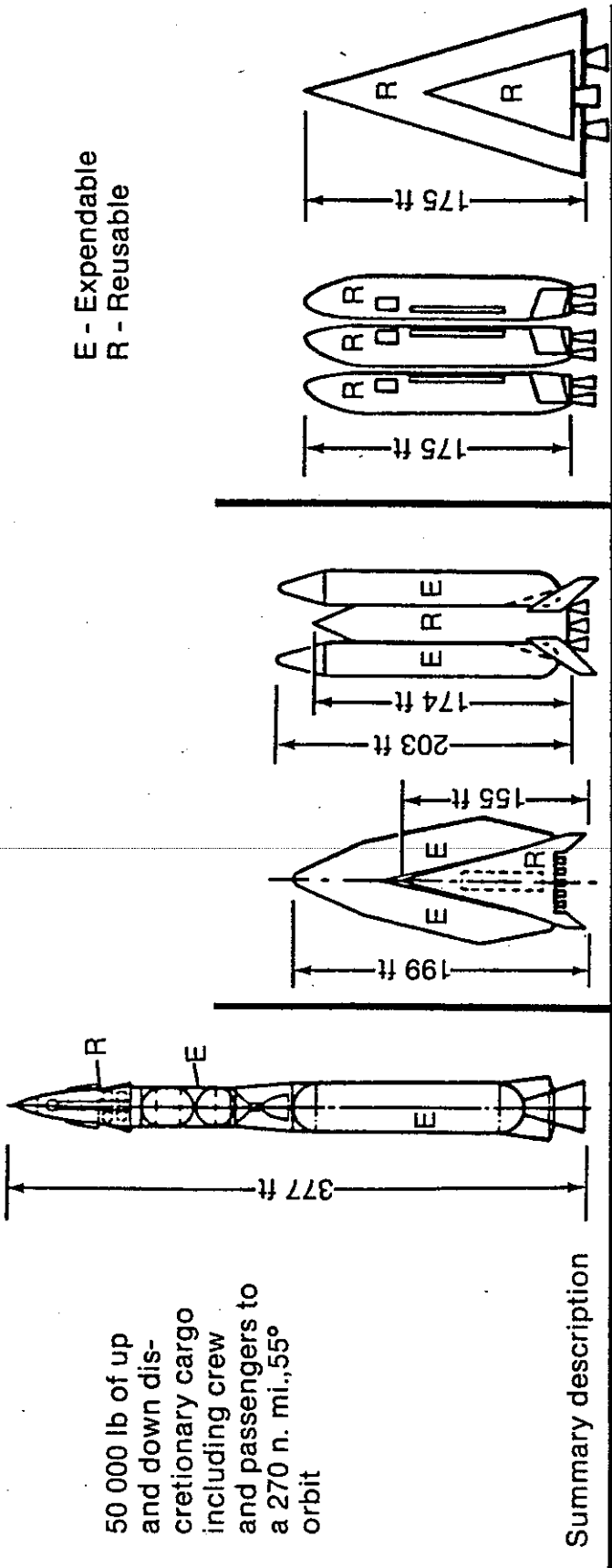


Figure 4-3.- Fully reusable concept (two stage).

4.1.4 Concept Summary and Comparison

Basic dimensions, weights, and thrust levels for three representative concepts have been summarized in figure 4-4. The spacecraft/stage weights at orbit insertion range from 150,000 to 300,000 pounds. Lift-off weights for the 1-1/2-stage and fully reusable vehicles were on the order of 3 to 4 million pounds for 50,000 pounds payload and 10,000 ft³ payload volume.

As an indication of weight sensitivities for highly reusable vehicles, a 1000-pound increase in inert weight of the orbital stage would require on the order of 15,000 to 20,000 pounds increase in total vehicle weight.



50 000 lb of up and down discretionary cargo including crew and passengers to a 270 n. mi., 55° orbit

E - Expendable
R - Reusable

Summary description

| | | | |
|---------------------------------|-----------|-----------|-------------|
| Total launch wt | 7.13 M lb | 2.92 M lb | 3.64 M lb |
| Total launch thrust | 9.75 M | 3.65 M | 5.50 M |
| 1st stage wt | 5.21 M | 2.39 M | 1.30 M (x2) |
| 1st stage propellant wt | — | 2.30 M | 1.05 M (x2) |
| 2nd stage wt | 1.63 M | 0.474 | 0.98 M |
| 2nd stage propellant wt | — | 0.264 | 0.74 M |
| 3rd stage (spacecraft) wt | 0.24 M | N/A | N/A |
| 3rd stage propellant wt | 0.13 M | N/A | N/A |
| Spacecraft (Orbiter) landing wt | 0.16 M | 0.26 M | 0.28 M |
| Spacecraft lift to drag (hyp.) | 1.7 | 2 | 2 |

Figure 4-4.- Space Transportation System configuration concepts.

4.1.5 Operational Modes

The basic operational aspects of the three classes of vehicles are compared in table 4-1. It will be noted that aircraft-type operations are adopted for the lifting-entry, reusable elements of each vehicle, from launch, abort, entry/landing, and recycle for subsequent flights. The opportunity to take advantage of these features was progressively greater with increased degree of vehicle recovery and reuse.

TABLE 4-1.- OPERATIONAL MODES

| Item | Concept | Reusable spacecraft plus new expendable launch vehicle | 1 1/2 stage | Fully reusable |
|-----------------------------|--|--|--|--|
| Launch | Similar to Saturn/Apollo | Vertical launch - aircraft-type launch activities | Vertical launch - aircraft-type launch activities | Fully reusable |
| Launch azimuth restrictions | Similar to Saturn/Apollo | Similar to Saturn/Apollo | Limited by disposal of expendable drop tanks | No expended hardware |
| Abort | Similar to Apollo, except must reach land for an aircraft-type landing | Similar to Apollo, except must reach land for an aircraft-type landing | Intact abort of complete stage, including payload. No dependence upon rocket-type "escape" devices | |
| Staging | Similar to Saturn/Apollo | Similar to Saturn/Apollo | Requires jettison and disposal of drop tanks, for abort during ascent | |
| Reentry | Lifting reentry | Lifting reentry | Staging of large, depleted tanks. Tanks not under active control. | Parallel staging of two or three large vehicles. Stages return to launch site. |
| Terminal landing | Aircraft-type approach and landing | Aircraft-type approach and landing | Aircraft-type approach and landing | Capability provided |
| Refurbishment | Refurbish spacecraft, and mate with new launch vehicle | Refurbish spacecraft, and mate with new launch vehicle | Replace tip tanks | |
| | | | Refuel and minimum refurbishment, only | |
| | | | No replacement of ablatives required | |

4.2 MISSION REQUIREMENTS

The major portion of the space exploration and application missions defined in early 1969 involved placing either men, equipment, propellants, other consumables, and instruments, or some combination of these, in low Earth orbits. This was true not only for those missions which were to be conducted from low Earth orbits but also for higher Earth orbits, such as geosynchronous, lunar, and escape missions. For all these missions, low Earth orbits would serve either as a staging, buildup area, or as parking orbits, while waiting to achieve the proper orbital conditions for continuation of a particular mission. A versatile space shuttle system that could effectively transport a varying mix of personnel and cargo to low Earth orbits and return could be the keystone to the success and growth of future space flight developments such as exploration and exploitation of near and far space. The salient characteristics and requirements of six basic missions of major interest in NASA's space program planning are summarized in table 4-2. While these requirements are stated in transportation properties, they were based upon the requirements formulated by the space science and applications communities. The requirements are listed below.

- Space Station/Base Logistics Support
- Placement and Retrieval of Satellites
- Delivery of Propulsive Stages and Payloads for High-Energy Missions
- Delivery of Propellants
- Satellite Service and Maintenance
- Short-Duration Orbital Missions
- Space Rescue

TABLE 4-2.- MISSION CHARACTERISTICS

| Orbital characteristics | Space Station/ base logistics support | Placement and retrieval of satellites | Delivery of propulsive stages and payload | Delivery of propellants | Satellite service and maintenance | Short duration orb. mission |
|------------------------------------|---|---|---|-------------------------------|---|-----------------------------------|
| Altitude (n. mi.) | 200 - 300 | 100 - 800 | 100 - 200 | 200 - 300 | 100 - 800 | 100 - 300 |
| Inclination (deg) | 28.5-90 | 28.5-sun syn. | 28.5-55 | 28.5-55 | 28.5-sun syn. | 28.5-90 |
| On-orbit ΔV (1000ft/s) | 1 - 2 | 1 - 5 | 1 - 1.5 | 1 - 2 | 1 - 5 | 1 - 2 |
| On-orbit stay time (days) | 7 | 7 | 7 | 7 | 7 - 15 | 7 - 30 |
| Crew | 2 | 2 | 2 | 2 | 2 | 2 |
| Passengers (min.) | Rotate 50 men/qtr | 2 | 2 | 2 | 4 | 10 |
| Ascent discretionary payload | | | | | | |
| Weight (1000 lb) | *70/qtr | 1 - 50 | 25 - 50 | 50 | 5 - 15 | 25 - 50 |
| Volume (1000 ft ³) | | 5 - 10 | 10 | 10 | 15 - 10 | 4 - 6 |
| Critical dimension diam (ft) | 15 | 15 | 15 | 15 | 15 | 15 |
| Return discretionary payload | | | | | | |
| Weight (1000 lb) | *45/qtr | 10 - 50 | -- | -- | 15 | 50 |
| Volume (1000 ft ³) | -- | 5 - 10 | -- | -- | 5 - 10 | 4 - 6 |

* Include passengers

4.3 TRAFFIC MODEL

A nominal traffic model for the space shuttle is shown in table 4-3. This model was based on June 1969 planning data for future missions and included flights in support of a space station leading to a space base in 1980, the unmanned planetary program, the unmanned satellite program, and the lunar program. This model shows less than 20 flights per year for the first 3 years followed by an increase to 60 flights per year with the initiation of an expanded lunar exploration program in 1978. The dominant missions were in support of the space base and the lunar program. While future events would require changes in this nominal traffic model, it represented June 1969 projected plans.

TABLE 4-3.- NOMINAL SPACE SHUTTLE TRAFFIC MODEL

| Program year | 1975 | 1976 | 1977 | 1978 | 1979 | 1980 | 1981 | 1982 | 1983 | 1984 | 1985 |
|---|------|------|------|------|------|------|------|------|------|------|------|
| Space Station 12 men, rotate entire crew per quarter | 7 | 7 | 7 | 7 | 7 | | | | | | |
| <u>Space Base</u> 50 men, rotate entire crew per quarter (10 men/flight) | | | | | | 23 | 23 | 23 | 23 | 23 | 23 |
| <u>Unmanned planetary</u> | 7 | 1 | 8 | 3 | 4 | 6 | 5 | 2 | 7 | 5 | 3 |
| <u>Unmanned satellites</u> | 2 | 2 | 2 | 2 | 2 | 2 | 2 | 2 | 2 | 2 | 2 |
| Lunar Program 6-man orbital station and 6-man base | | | | 48 | 48 | 34 | 34 | 34 | 34 | 34 | 34 |
| Total Space Shuttle flights/year | 16 | 10 | 17 | 60 | 61 | 65 | 64 | 61 | 66 | 64 | 62 |

4.4 TECHNOLOGY REQUIRED FOR PHASE A CONFIGURATIONS

Critical subsystems and design requirements are summarized in table 4-4 for the three representative vehicle concepts. Requirements for reusable thermal protective systems and provisions for terminal approach and landing applied to all three concepts. In addition, the 1-1/2-stage and fully reusable concepts required development and integration of large, high-performance, reusable, hydrogen-oxygen rocket engines. New integrated electronics requirements arose in two areas: onboard checkout and malfunction detection and display systems, and provisions for highly autonomous flight operations during ascent, on-orbit, and return phases. Most of the objectives for autonomous operation in real time have been achieved; however, the capability for "onboard checkout" has not. Therefore, we have seen a higher-than-planned reliance on turnaround testing and processes.

TABLE 4-4. TECHNICAL RISK AND TECHNOLOGY REQUIREMENTS

| Item | Reusable spacecraft plus new expendable launch vehicle | 1 1/2 stage | Fully reusable |
|------------------------------------|--|---|--|
| Technology status and requirements | <p>Launch vehicle is current state of the art.</p> <p>New developments required for reusable spacecraft, including thermal protection and approach/landing provisions.</p> | <p>Comparable overall risk</p> <p>Propulsion integration more difficult.</p> <p>Large engine weights in base add to difficulties in configuration/balance.</p> <p>Entry heating due to higher wing-loadings imposes somewhat greater demand upon thermal protection system.</p> <p>Low-cost tank development.</p> | <p>Fully reusable, more sensitive to uncertainties in reusable design features.</p> <p>Uncertainty in ability or desirability of common stage design for booster and Orbiter elements.</p> |
| Development and test approach | <p>Must use rocket test and development approach</p> | <p>"Aircraft" approach for core with tanks</p> | <p>"Aircraft" approach for all elements</p> |
| New technology requirements | <p>Development of low-cost, pump-fed liquid rocket engines required</p> | <p>Requires development of high-performance hydrogen-oxygen engine in the 500,000 to 1-million-pound thrust class.</p> <p>Long life, reusable throttling. Must be adaptable to boost and upper-stage use.</p> | |
| Engines | <p>Development of large solid stage required</p> <p>Technology for both at hand</p> | | |
| Thermal protection system | <p>Reusable, all radiative thermal protection system required (2000° to 2500° F) for minimum refurbishment</p> | | |
| Configuration development | <p>Requires aerodynamic and thermodynamic characteristics compatible with entire ascent, entry, and landing phases of flight</p> | | |
| Astrionics systems | <p>Parallel staging of large bodies</p> <ul style="list-style-type: none"> ● Integrated into reusable spacecraft ● Onboard checkout | <ul style="list-style-type: none"> ● Autonomous operation during ascent and return ● Onboard checkout and malfunction detection | |

4.4.1 Tradeoff Between Recurring and Development Costs

The tradeoff between recurring and development costs is displayed in figure 4-5a. A high development cost for this class of vehicles was expected to result in lower operational and refurbishment costs. The chart also shows an approximation of the range of recurring costs anticipated for concept II (1 1/2 stage) resulting from uncertainties in external tank cost estimates and gives a rough idea of the uncertainties anticipated in other costs.

Estimates available in 1969 indicated that programs of 100 or more flights per year were desired. This criterion made it very difficult for concept I to be competitive. Concept II was not competitive for programs of more than approximately 70 flights (for the assumed refurbishment rates and external tank costs) as is shown in figure 4-5b. Further analysis indicates that if external tank costs were reduced from \$75/lb to \$45/lb, concept II would remain competitive through 100 flights. At \$25/lb, concept II was competitive through 175 flights. Changes in development and vehicle costs of plus or minus 10 percent for both concepts II and III would make concept II competitive with concept III from 1 flight to more than 150 flights. Reductions in external tank cost coupled with changes in development cost produced even wider ranges where concepts II and III were competitive.

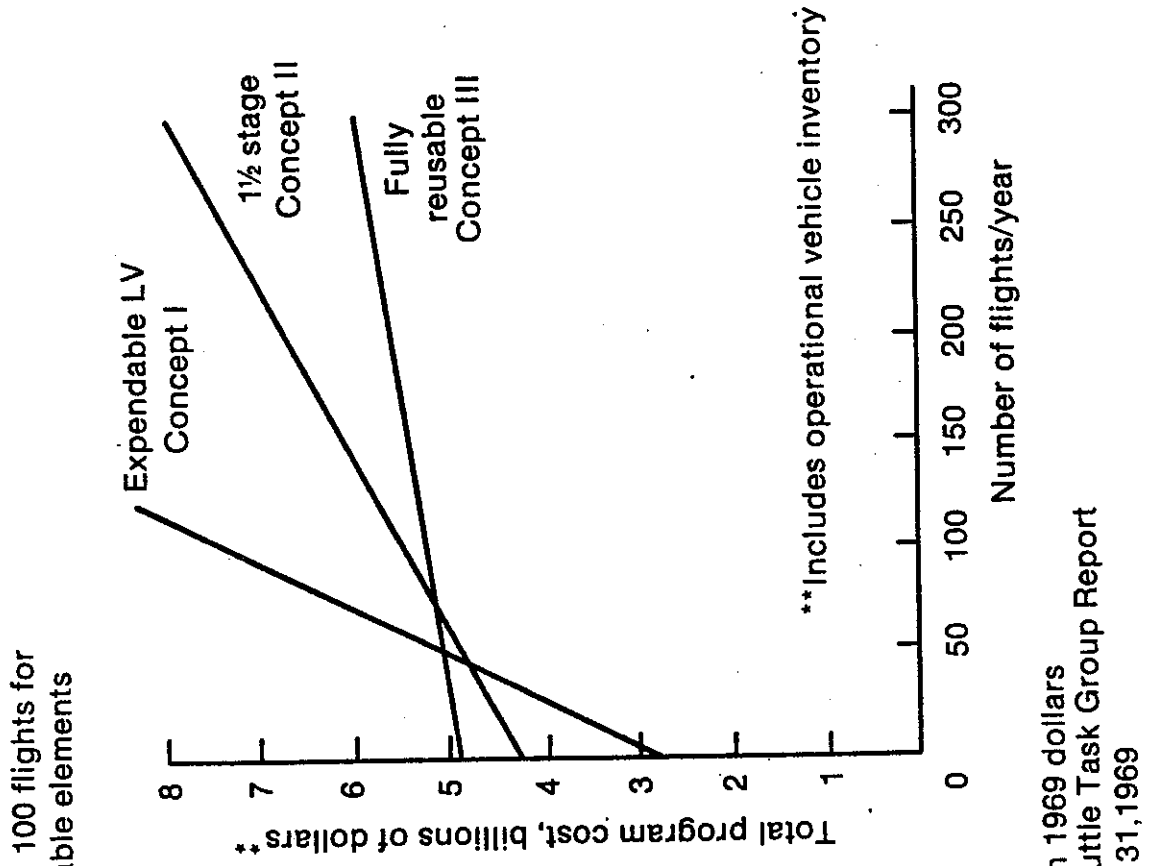


Figure 4-5a.- Tradeoff between development and operational costs.

Note: 100 flights for reusable elements

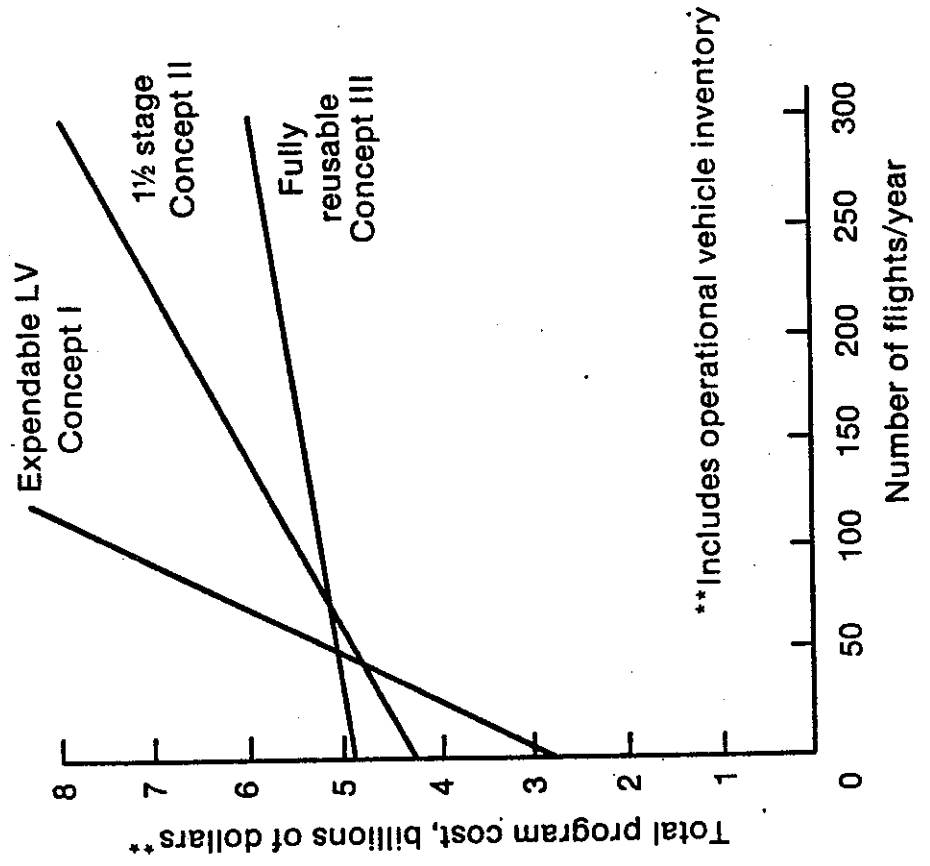


Figure 4-5b.- Total concept cost comparison (nominal cost values).

4.4.2 Refurbishment Cost Analysis

4.4.2.1 Refurbishment Rate Effects

The refurbishment requirements of reusable vehicles substantially affected the cost of payload to orbit as illustrated in figure 4-6a. The effect of refurbishment for various initial vehicle costs is reflected for various refurbishment rates to illustrate the effect on payload cost per pound. For concept III vehicles, 1 percent refurbishment could contribute approximately \$45/lb delivery cost, indicating the significance of this parameter.

4.4.2.2 Reusability of Rocket Engines

A major subsystem which substantially affected refurbishment costs was the propulsion system, including the engine. The effect on payload cost of mandatory system replacement is shown in figure 4-6b. It is readily apparent that if transportation costs were to be kept minimal, a high degree of confidence that the engine system could be reused without replacement was mandatory.

If propulsion system replacement is required at 50 reuses, payload mass transport costs are increased by \$7 to \$8/lb over what they would be with infinite reuses. Replacement costs at 100 reuses drop to a \$3 to \$4/lb penalty. If it is possible to perform major overhaul of the engine systems to obtain more than 100 reuses, this cost penalty would be reduced.

A propulsion overhaul was expected to be performed for 25 to 30 percent of the cost of a replacement unit. Rocket engines for reusable launch vehicles and spacecraft must be designed for an economical number of reuses with a minimum of servicing operations between uses; up to that time [1969], few rocket engines had been designed for reusability. The YLR 99-RM-1 engine for the X-15 research airplane had a design operational life of 1 hour and/or 100 starts (whichever occurred first). The LR-121-NA-1 engine, designed for thrust augmentation on the F-104 aircraft, had a design life of 2 hours with a major overhaul at the 1-hour point. Almost all other U.S. rocket engines had been designed for a single operational use of a few hundred seconds, with an allowance for calibration, acceptance, and preflight testing. Some engines designed to this criterion had minor service problems when used for a large number of static firing tests, but only after the design life was exceeded. There was no fundamental reason why rocket engines could not be designed for lifetimes longer than those projected for any current [1969] operational requirement for reusable launch vehicles, if operational life was an initial design criterion.

Although current [1969] rocket engines had been designed for a short operational life commensurate with use in an expendable vehicle, most current [1969] engines could have actual life far beyond the design requirement. A typical example is the J-2 engine, with a specification firing duration of 470 seconds per firing; 3750 seconds firing time and

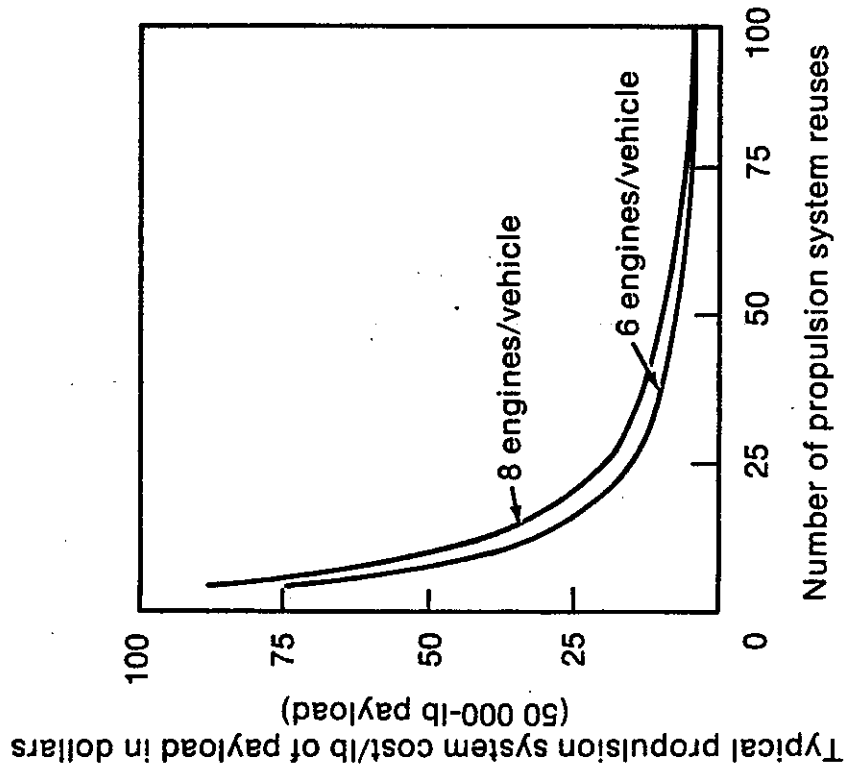


Figure 4-6b.- Effect of propulsion system reuses on cost/lb payload.

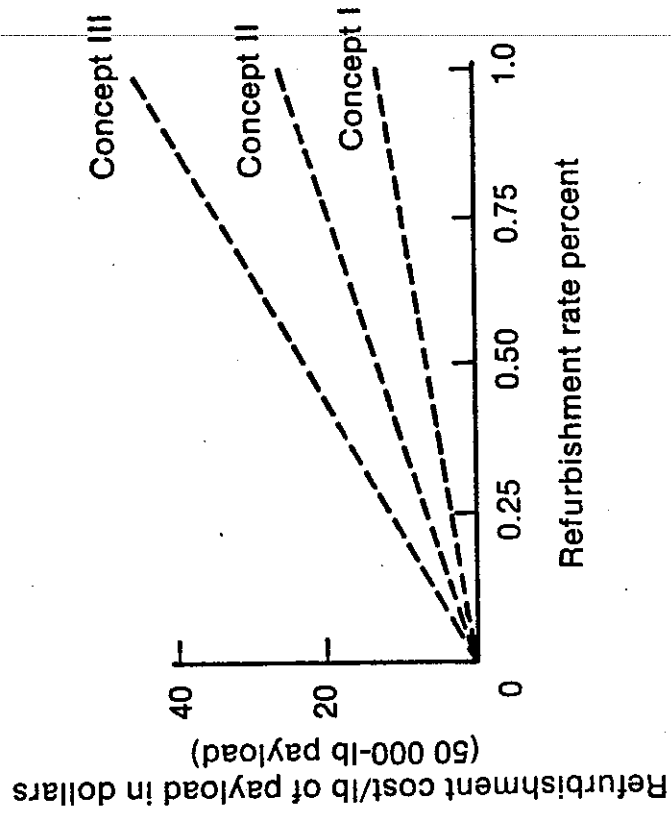


Figure 4-6a.- Effect of refurbishment rate on cost/lb of payload.

4.4.2.2 Refurbishment Rate Effects (Concluded)

30 starts were accumulated on a single engine during qualification. Subsequently, 20,094 seconds firing and 103 starts had been accumulated on a single engine, with a longest firing duration of 646 seconds determined by the largest available tankage.

Qualification life requirements for the F-1 engine are 2250 seconds and 20 starts, with very minor maintenance permitted during this life. This was adequate for a life of 14 flights of 150 seconds per flight, plus one full-scale acceptance test. Research and development tests of the engine in qual configuration had demonstrated 5000-second life, achieved through LO₂ pump impeller and turbine manifold replacement at 3500 seconds. Changes to these components to give the 5000-second life without replacement were incorporated in the projected low-cost simplified F-1 engine program. All other components had a demonstrated life above 5000 seconds. This would provide for a life of 30 flights. Many components had demonstrated life above 10,000 seconds. With an appropriate overhaul and flight schedule, the engine was expected to be usable for 60 flights.

The Pratt and Whitney high-pressure, reusable rocket engine had a design-life requirement of 10 hours between overhauls, 100 reuses, 300 starts and thermal cycles, and 10,000 valve cycles. The engine was designed for ease of maintenance, good inspectability, and minimum servicing between flights. An engine control unit, diagnostic system, and operational history recorder together would provide data to eliminate the requirements for extensive between-flight checkout operations.

All components of this engine which operate in life-limiting modes (wear, fatigue, creep, thermal cycling) had been identified and component-life tests had been or would be made. For example, oxidizer shutoff valves had demonstrated the specification requirements on leakage for the 10,000 cycle life, 25 percent of these at cryogenic temperatures, and 75 percent at room temperature. Full-scale tests on the roller bearings used in the turbine and pumps had demonstrated more than the specified life under full load and speed conditions in bearing rigs. A lifetime of 300 thermal cycles had been demonstrated on the regenerative and dump cooling tubes used in the nozzle beyond an area ratio of 6. Turbine blades, which were subject to creep, were designed to the same criteria and design standards used for commercial jet engines. On the RL-10 LO₂/LH₂ engine, Pratt and Whitney had run as many as forty-one 10-minute firings on a single engine. One run was of 27 minutes duration and was terminated only when fuel was exhausted. Also, 70 RL-10 engines had been run for more than 3600 seconds each.

These data were the basis of predicting that an LO₂/LH₂ engine which could be reusable for 50 flights without overhaul could be developed.

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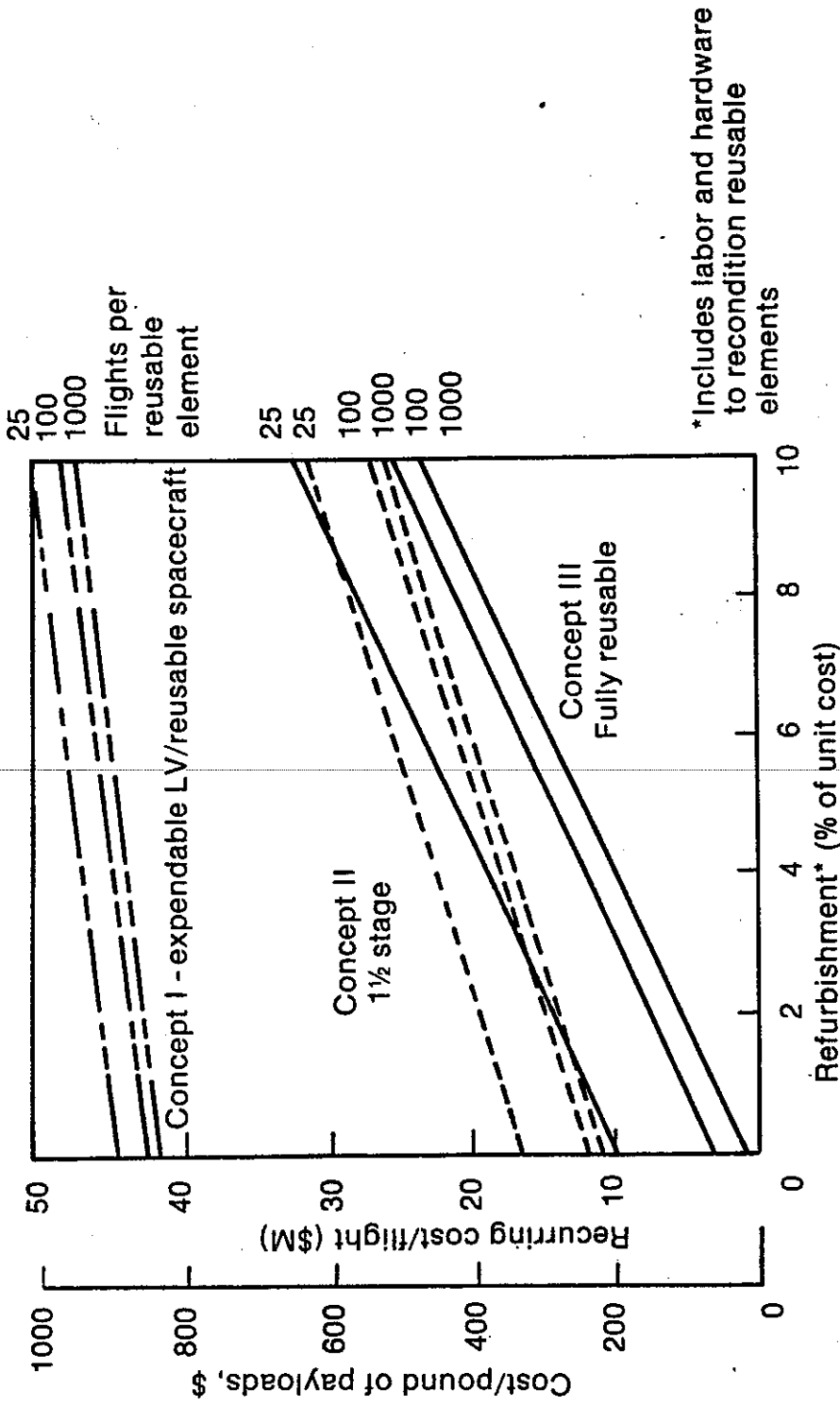
4.4.3 Sensitivity of Recurring Costs to Major Assumptions: Total Number of Uses, Refurbishment Rates, and External Tank (ET) Costs

An investigation was made to determine how recurring operational turnaround costs of each vehicle were affected by the number of flights and assumed refurbishment rates (fig. 4-7). There is a large advantage to increasing vehicle uses from 25 to 100 flights (fig. 4-7), but for flights per vehicle of 100 to 1000, the price advantage decreases rapidly. The high expendable booster cost per flight for concept I prevented it from being competitive for lower refurbishment percentages.

The effects of the ET costs on the competitiveness of concept II is shown in figure 4-8. The expendable ET's made this concept more costly than concept III at the lower refurbishment rates, but at approximately 10 percent refurbishment, the concept became competitive for reasonably expected values of ET costs. If tank costs could be reduced from \$75/lb to \$20/lb, the point where concept II became competitive was changed from 10 to 2 percent refurbishment.

In 1969, refurbishment for concept III was estimated to be under 1 percent. Therefore, for concept II to be competitive on an operational basis, the ET cost had to be reduced to \$20/lb or less. This was considered to be an unrealistic goal.

Note: The cost per pound of the current Space Shuttle ET is approximately \$321 in 1982 dollars.

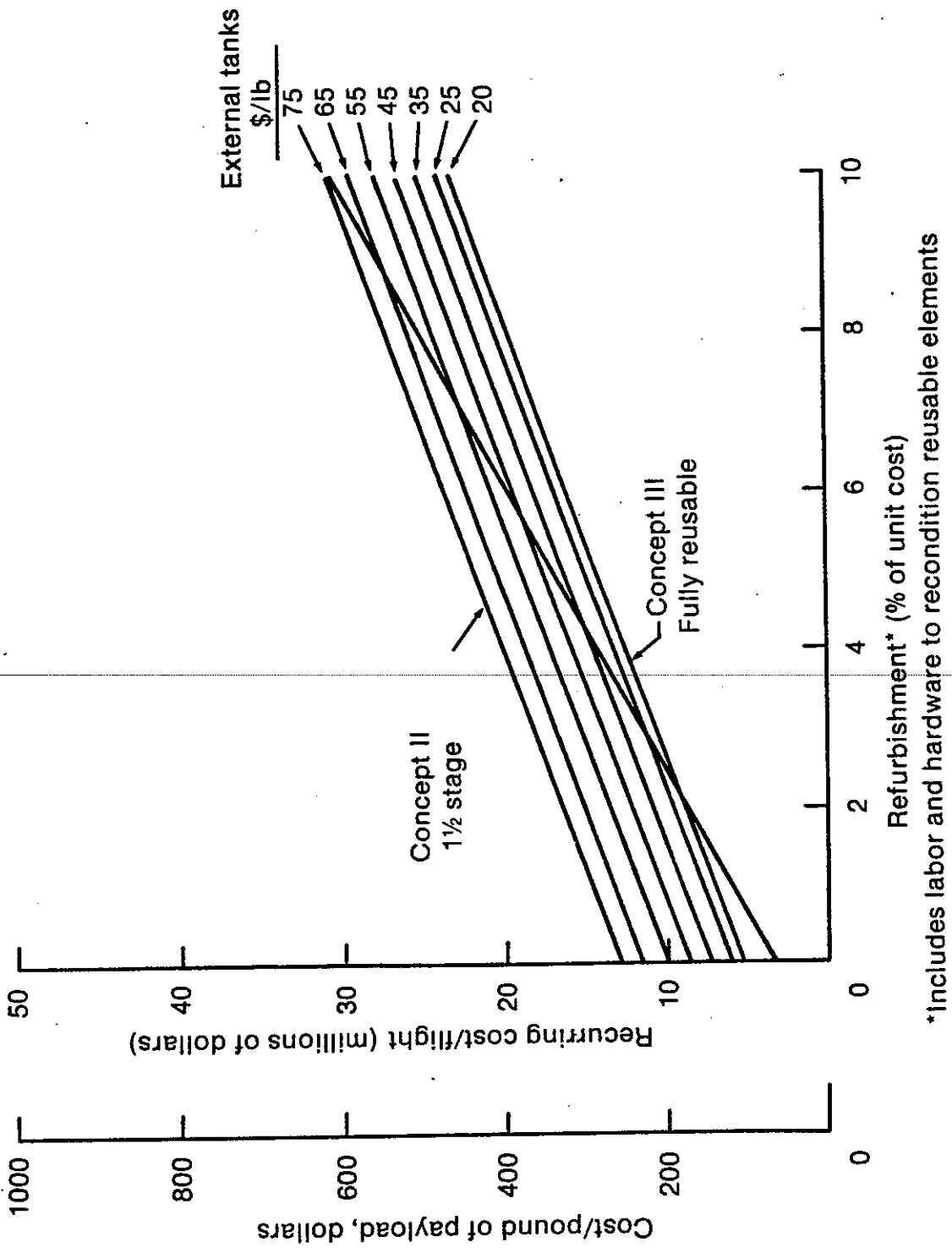


*Includes labor and hardware to recondition reusable elements

| Assumptions | I | II | III |
|---|--------------|-----------------|----------|
| Entry vehicle | \$67.0M | \$137.0M | \$225.0M |
| Expendable items/launch | \$40.0M (LV) | \$10.0M (Tanks) | -0- |
| Launch and flight operations, propellants | \$2.2M | \$0.8M | \$0.8M |
| Cost of expendable tanks/lb | - | \$75.00 | - |

Figure 4-7.- Recurring cost comparison.

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*Includes labor and hardware to recondition reusable elements

Figure 4-8.- Recurring cost comparison, effects of external tank cost at 100 flights/vehicle.

4.4.4 Vehicle Reuse Life

The vehicle operational life could be determined from the vehicle refurbishment cycle, nominal mission cycle, and the number of flights desired per vehicle; the relationship involved is shown in figure 4-9. The turnaround time for the concept III vehicle had been estimated at 10 working days (14 calendar days). (Concept II was estimated to require 18 days.) With a 7-day mission cycle and a 10-year program life, each vehicle could make only 173 flights. It would require 60 years to complete 1000 flights. Refurbishment/turnaround time became extremely important when attempting to increase the number of uses of each vehicle; by reducing the turnaround time to 5 working days (7 calendar days), the number of flights per vehicle could be increased to 260 flights in the 10-year period. Therefore, it appeared that a logical design reuse life might fall within the range of 173 to 260 flights per vehicle.

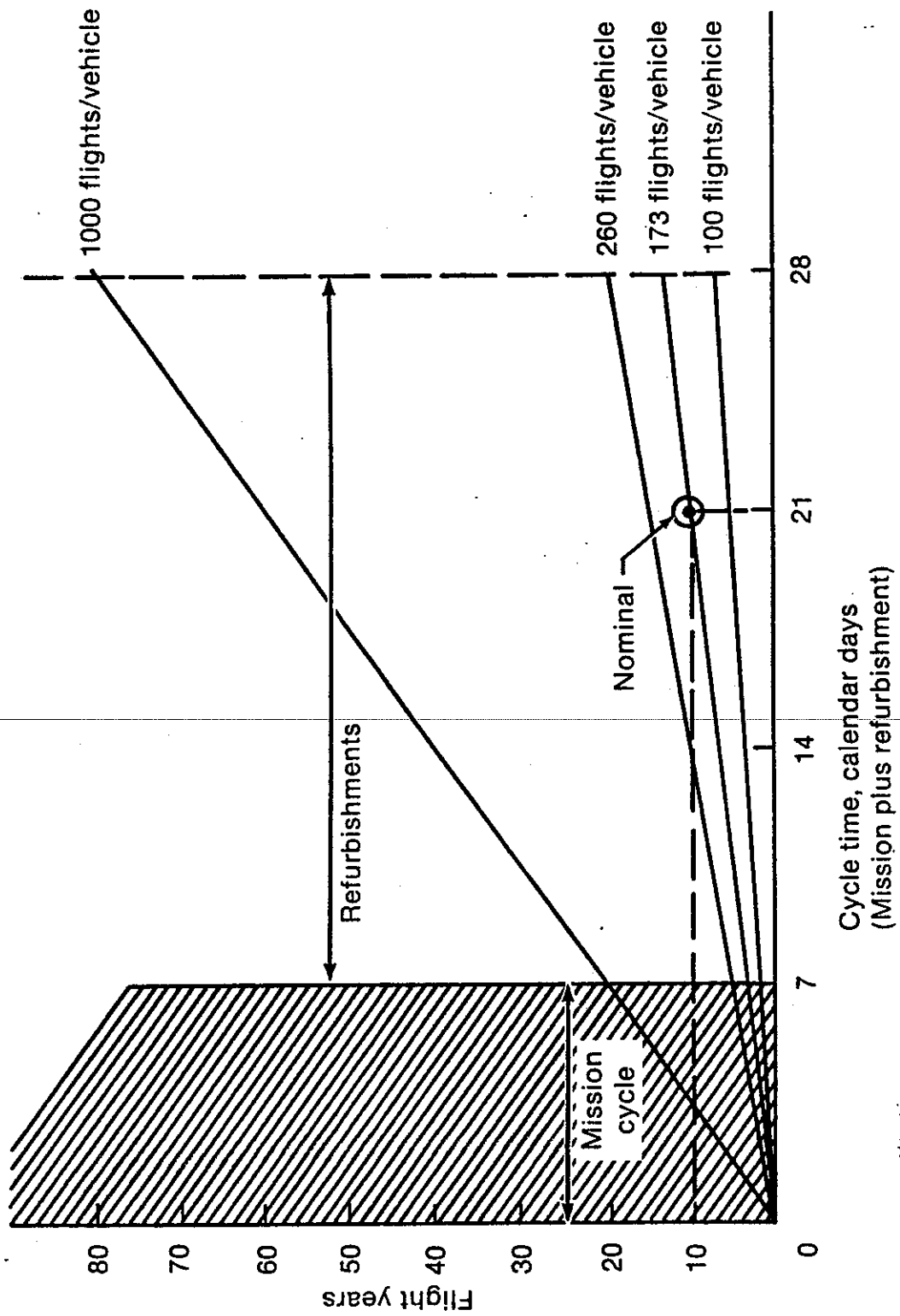


Figure 4-9.- Derivation of vehicle reuse life.

4.4.5 Costs

Because of the lack of detailed engineering and operational definition of the three basic system concepts, the approach to the cost analysis was to estimate a most probable or nominal value and then to employ parametric curves to illustrate the potential ranges within which possible real values lay. Therefore, only first approximations of concept costs, tradeoffs, and sensitivities are presented. The following three parameters were of primary interest.

- Refurbishment, treated as a function of first unit cost; a range of 0.1 to 10.0 percent of the first unit cost was investigated.
- Recurring costs per flight, including launch, flight operations, recovery, checkout, refurbishment, and ground turnaround operations (ranges for these costs are noted on the bar chart in figure 4-10).
- Equivalent number of system reuses, for which a range of 25 to 1000 was investigated. A design life of 100 to 300 flights was determined to be adequate for the traffic models studied.

While emphasis in the definition and development of the Space Shuttle was the reduction of the recurring cost of transportation to and from Earth orbit, significant cost reductions were considered by the adoption of an "aircraft-like" approach in the design and development phase, i.e., by accomplishing qualification testing in flight and by using limited-life development subsystems.

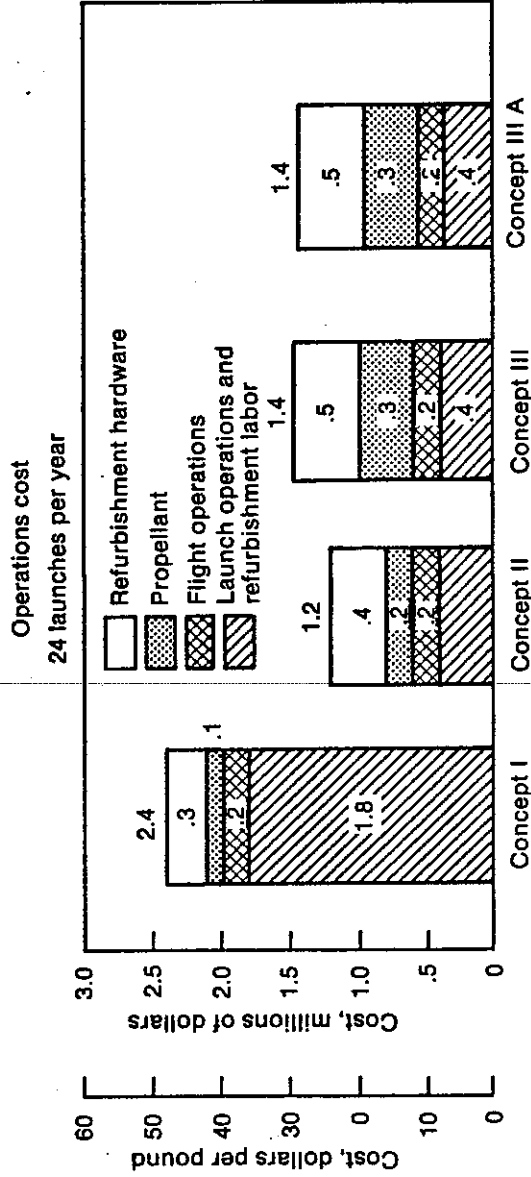
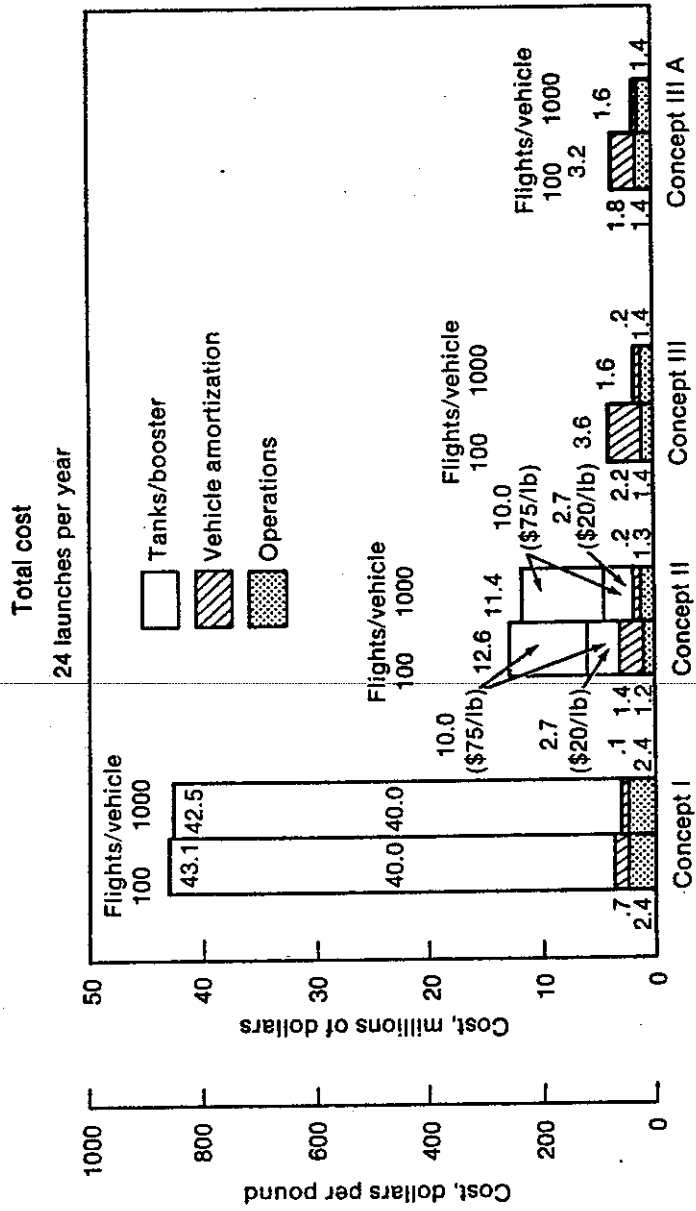


Figure 4-10.- Recurring cost per launch.

SECTION 5
PHASE B - 1970 TO 1972

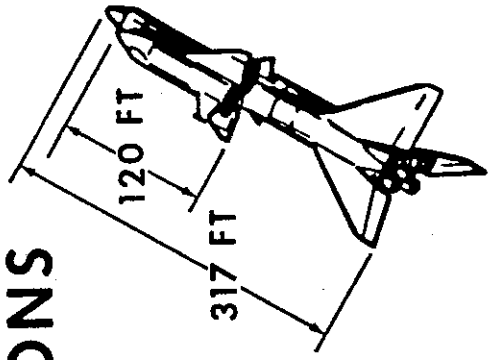
SECTION 5
PHASE B - 1970 to 1972

As manned space flight progressed through the Mercury, Gemini, and the beginning of the Apollo and Skylab programs, efforts continued within both the public and private sectors to finalize objectives for the post-Apollo period. Safe, low-cost transportation to low orbit continued to be recognized as a key element. This resulted in the study of a wide variety of potential logistic vehicle configurations. These studies, in conjunction with technological advances and the objective of low costs per flight, pointed toward a fully reusable, flyback booster and orbiter system as specified in the Orbiter Phase B contracts of 1970. This focal point then drove configuration studies primarily toward series versus parallel burns and liquid versus solid boosters.

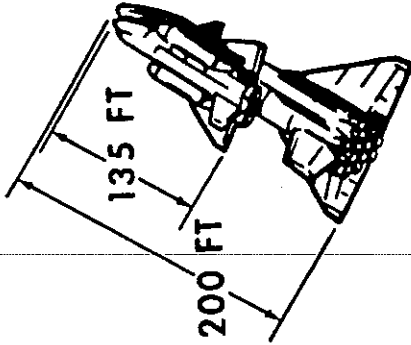
As the Phase B studies progressed, it was recognized that not only was cost-per-flight a primary driver, but that DDT&E and peak-year funding had to be addressed. The totally reusable approach with high development risks and costs, therefore, became oriented toward meeting the funding constraints while retaining a space shuttle system that would fulfill required objectives. This series of events resulted in the selection of the current vehicle configuration as being optimum.

The following sections document the key Phase B decisions that were driven by technology issues.

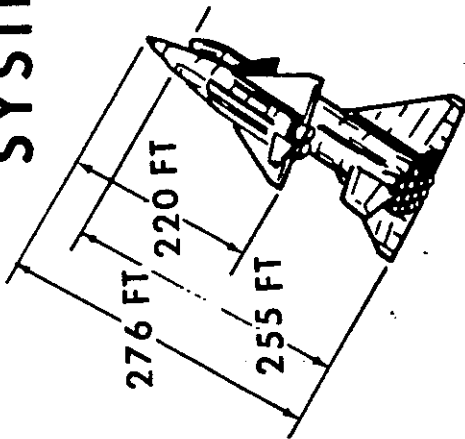
SOME PHASE B SPACE SHUTTLE SYSTEM SIZE COMPARISONS



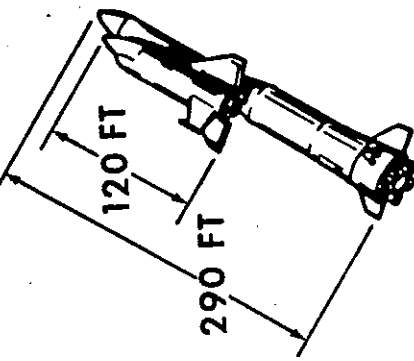
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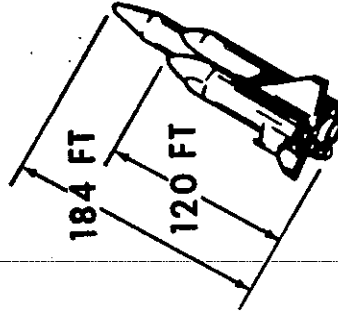
EXTERNAL LH2 TANKS



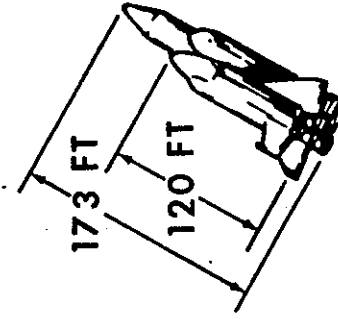
FULLY REUSABLE



F-1 FLYBACK



PARALLEL LIQUID



PARALLEL SOLID ROCKET MOTOR

Figure 5-1.- Flight vehicle configuration evolution, Phase B and Phase B extension - 1970 to 1972.

5.1 ORBITER PROPELLANT TANKS REMOVED - APRIL 1971

The original concept of the fully reusable space shuttle included an orbiter with internal liquid oxygen and liquid hydrogen propellant tanks. These pressure vessels were built into the airframe to support the tank and propellant weight but were to be supported in such a way as to allow shrinkage during propellant loading and airframe expansion during entry. Structurally, this represented a significant challenge which was never addressed in detail because other reasons indicated that the tanks should be removed from the orbiter. These reasons included the following:

- Orbiter sizing was being driven by tank volumes, which steadily increased as detailed propulsion analyses progressed.
- Program guidelines to reduce development cost resulted in trade studies indicating that the external tank (ET), parallel-burn, solid rocket booster system had about half the development cost of the fully reusable concept.
- Significant technical and safety reasons dictated the removal of the tanks from the orbiter; these reasons will be discussed further in the following section.

The decision was made on April 1, 1971, when NASA shifted emphasis of the Space Shuttle Phase B studies from "fully reusable" to "orbiter with external expendable hydrogen tanks." By May 1971, the LO₂ tank was also considered for removal from the orbiter as evidenced by MSC Orbiter 023 drawings showing a small orbiter with belly-mounted LO/LH tank. In early June 1971, Dr. Maxime Faget briefed the space community on an alternate space shuttle concept - a two-stage space shuttle with drop tank orbiter. On June 16, 1971, NASA Administrator James C. Fletcher sent a letter to the U. S. Senate identifying a two-stage, delta wing, reusable system with jettisonable ET's as the preferred configuration.

In the internal tank design, the propellant tanks were to be constructed from aluminum alloys. The tank design and construction would have to react to bending and torsion loads from the airframe as well as react to pressure, slosh, and other primary tankage loads. The tanks would have to be heavily insulated to prevent ice formation and liquefaction of air within the airframe. The insulation system represented a challenge in that it had to be very efficient and yet be durable enough to last the life of the orbiter; these dual requirements presented a significant development issue.

| Approximate dimensions - ft | |
|---|-----------|
| Orbiter | Wing span |
| Internal tanks | 107 |
| External LH ₂ | 97 |
| External LH ₂ /LO ₂ | 80 |

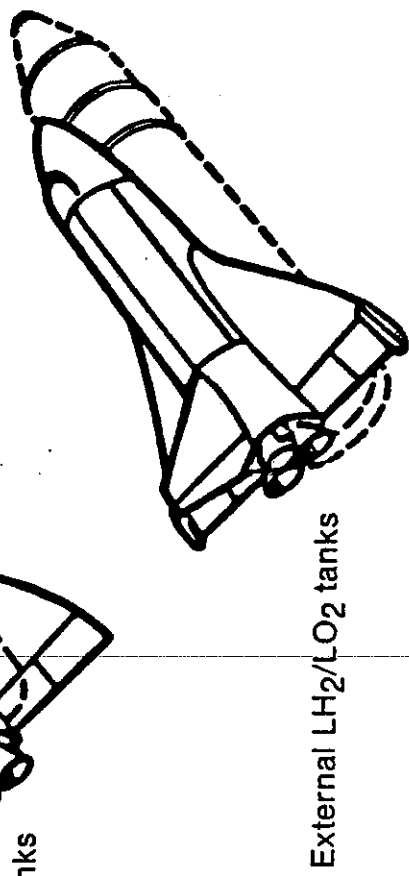
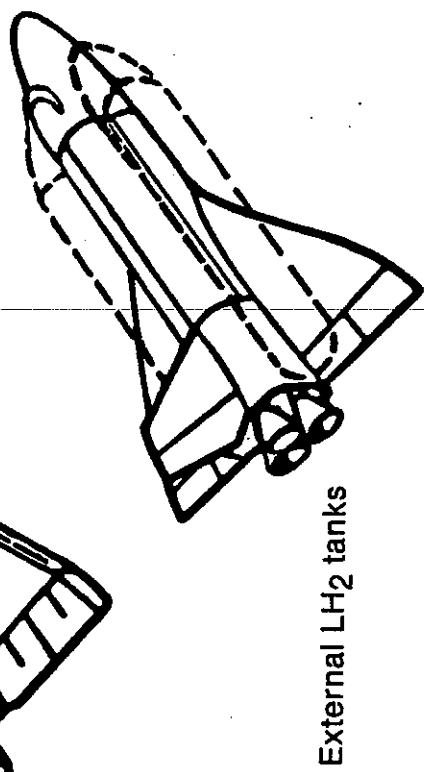
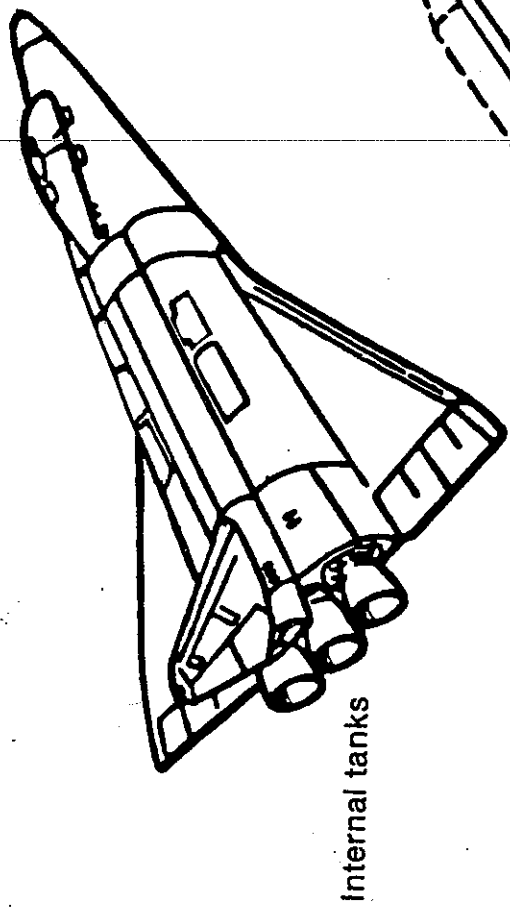


Figure 5-2.- Orbiter comparison.

5.1 ORBITER PROPELLANT TANKS REMOVED - APRIL 1971 (Concluded)

Perhaps the most significant technological issue that proved the wisdom of the expendable-external-tank approach was the need to satisfy the requirements that the propellant tanks be "safe-life" structure - a completely reliable structure not subject to the hazards of fracture or even leak development over the life of the vehicle. This was, and still is, a daunting technological challenge.

The use of linear elastic fracture mechanics analyses to verify integrity of pressure vessels was introduced by NASA into the Apollo Program following serious test failures on the service module propellant tanks and other pressure vessel failures in the Saturn and Apollo development programs. This analysis required that the pressure vessel be subjected to a pressure test that would screen potential flaws large enough to create a service hazard over a finite-use life cycle. An alternative approach to the pressure test would require a complete inspection periodically to prove that the tank was free of flaws large enough to cause leak development during use. Neither of these approaches could be satisfied with a propellant tank that could not be pressure tested above operating pressure without significant hazard of destroying the orbiter should the tank fail during test. Nor could they be satisfied with a tank that was largely inaccessible for inspection without removal of the insulation, the tank, or both.

Analysis indicated that, for the following reasons, an external oxygen/hydrogen tank was the better approach:

- Payload could be increased.
- Off-line test and inspection of the tank allowed safe-life analyses by fracture mechanics methods.
- Insulation complexity was greatly reduced.
- Tank resizing was easier as detailed analyses dictated needs for change.
- Manufacturing cycles for orbiter and tank were decoupled.
- Reduced ground lift-off weight (GLOW) and DDT&E costs indicated lower total program cost.

Although this approach violated the philosophy of a "fully reusable" vehicle concept and the cost-per-flight was increased by the use of an expendable tank, the decision in favor of the expendable tank was technically sound.

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5.2 SPACE SHUTTLE MAIN ENGINE (SSME)

The stringent procurement requirements for the SSME were seen in 1969 as a model for the requirements placed on the Space Shuttle as a whole (ref. 5-1). The key technical issues faced by this development program to replace "one shot," expendable rocket vehicles with a reusable space transportation system promised to be a turning point in liquid rocket engine history (ref. 5-2):

- The requirement for high performance in delivering more payload to orbit at reduced cost focused main engine performance requirements on the following (ref. 5-2):
 - The selection of the highly energetic propellant combination of hydrogen/oxygen
 - Achievement of a high thrust-to-weight ratio
 - Achievement of a high thrust-to-engine exit area (reduced vehicle drag due to base area)
- The requirement to reduce vehicle acceleration through the point of maximum dynamic pressure required thrust control by throttling.
- The requirements for reuse with a minimum refurbishment and turnaround cycle and for long life with minimum maintenance introduced the technology challenge to develop components to operate in the extreme temperatures and pressures imposed by the performance requirements (ref. 5-2).

Original Space Shuttle Main Engine Proposal Requirements (SSME-70-1, 1 March 1971)

- High performance
- Bell nozzle configuration
- Reusability
- Low-cost operation
- Long service life
- Propellant mixture ratio control
- Throttleability
- Minimum maintenance
- Criteria for safety, reliability, maintainability, and low-cost manufacturing developed and incorporated early into the hardware design
- Common procedures and tooling developed for use in manufacture of both R&D and production engines

5.2 SPACE SHUTTLE MAIN ENGINE (SSME) (Continued)

Equally ambitious, as compared to the technology challenges, was the programmatic challenge to accomplish the design, development, and certification with the utilization of resources substantially less than required in previous, comparable development programs. A measure of the resources is represented by the engine test programs shown in figure 5-3. As can be seen, the projected number of tests and development engines were reduced by some 40 percent (ref. 5-2).

Technology issues critical to the development of the SSME centered on the following:

- Development of high-temperature-tolerant, low-creep materials
- Predictability of performance of high-temperature-tolerant materials under repeated stress (fatigue life)
- Development of an understanding of fluid system flow dynamics at high-pressure, high-mass flow

Mr. Robert Thompson, in his Von Karman Lecture Series presentation of 1984, characterized the SSME development program as difficult but successful with the problems encountered being more in the nature of mechanical engineering-type issues (bearings, vibrations, cooling, materials) rather than combustion or combustion stability issues, which were among the postulated predevelopment concerns. Work still remains to certify the engines at full-power level, and lifetime/maintainability effects on engine operating costs remain unknown (ref. 5-3).

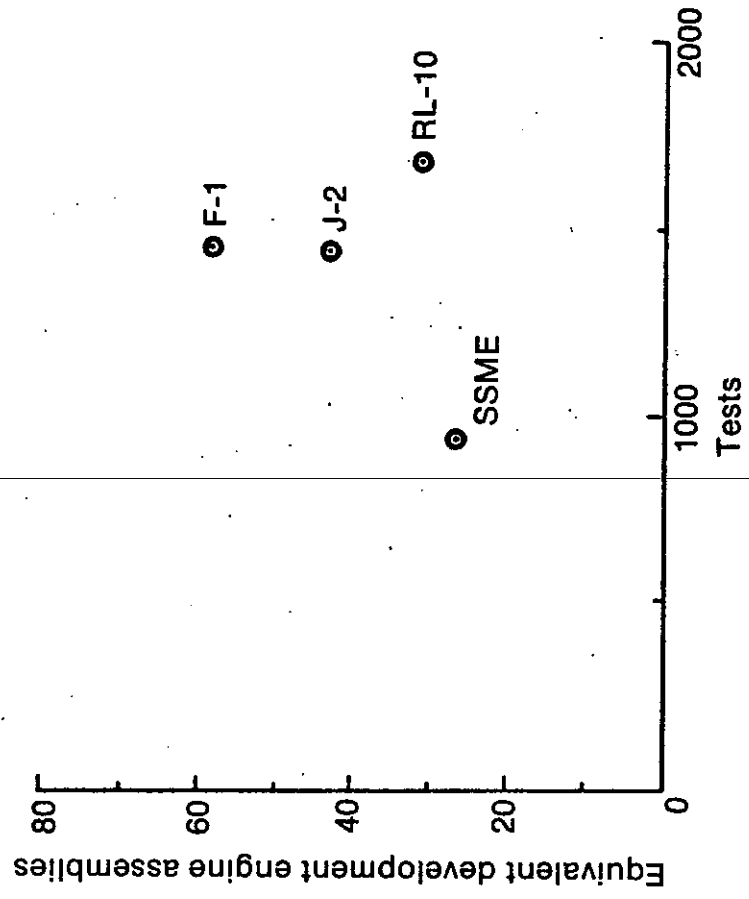


Figure 5-3.- Development resources.

5.2 SPACE SHUTTLE MAIN ENGINES (SSME) (Continued)

Certain decisions in the SSME program are instructive because of their applicability to future vehicle planning and because of the interaction with the remaining challenge of increasing the operating life and reducing field maintenance of the existing engine and future reusable propulsion systems. These decisions include

- Selection of the bell engine
- Selection of the staged-combustion cycle
- Concept of reusability

The selection of the bell engine design concept over the Aerospike or plug nozzle concept was a pivotal decision as NASA "from the beginning" had planned to carry out parallel engine development programs. The objective was "to be able to use either of these two engines whichever proved out best downstream with any one of the selected airframes." Conceding that the Aerospike had significant potential advantages, notably lower weight and reduced base drag, Mr. L. E. Day, Manager, NASA Space Shuttle Task Group, described the reasons for dropping the Aerospike engine (ref. 5-4):

- The Aerospike engine was not interchangeable between airframe designs because of the engine-airframe integration problem, while the integration of the bell engine was less difficult.
- An assessment of the technology development status of both concepts indicated the bell-type engine was slightly ahead.

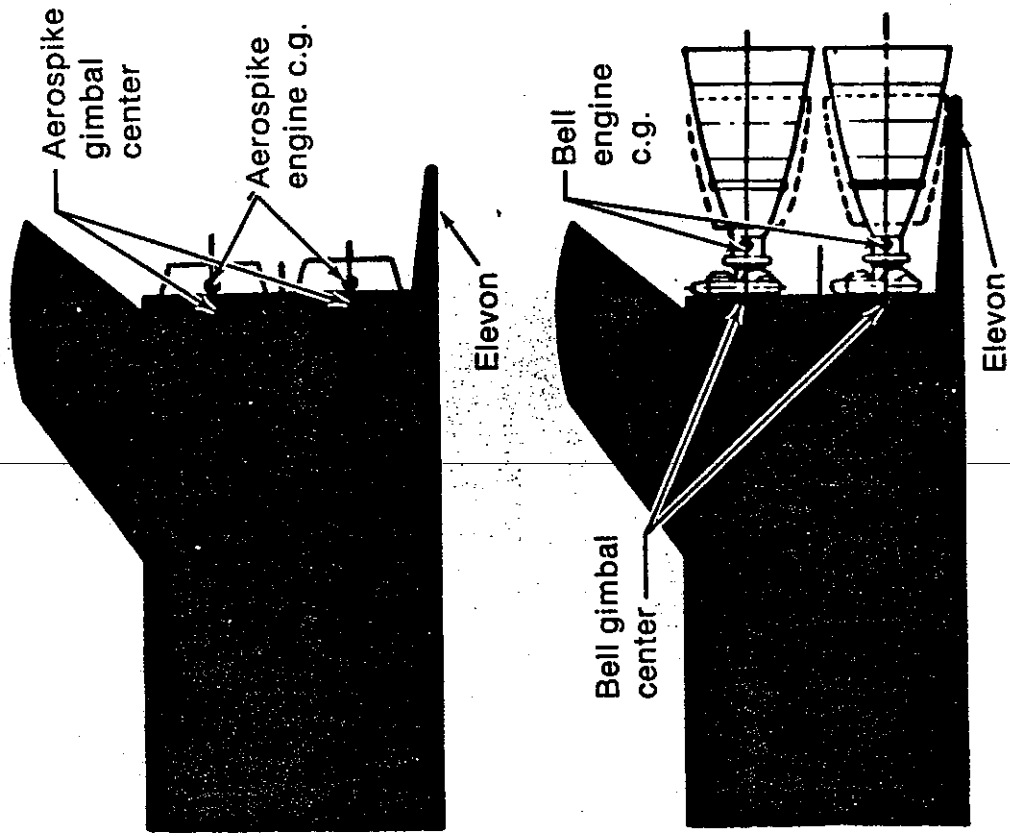


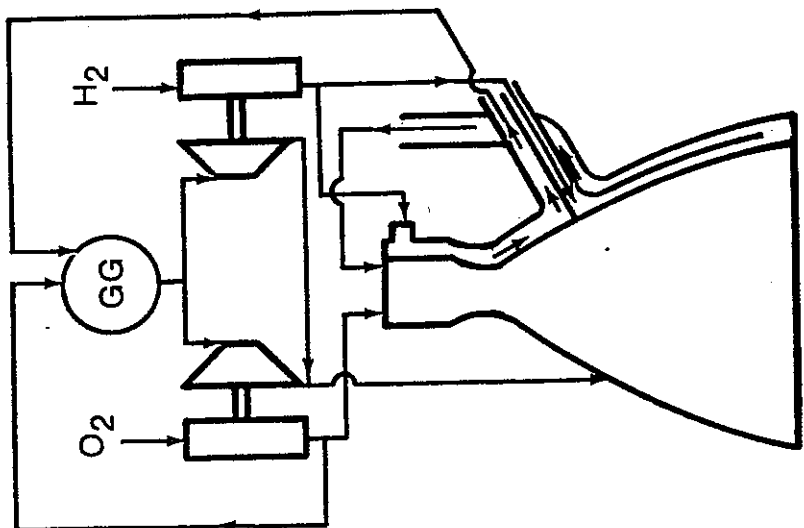
Figure 5-4.- Comparison of Aerospike and bell engine installations.

5.2 SPACE SHUTTLE MAIN ENGINES (SSME) (Continued)

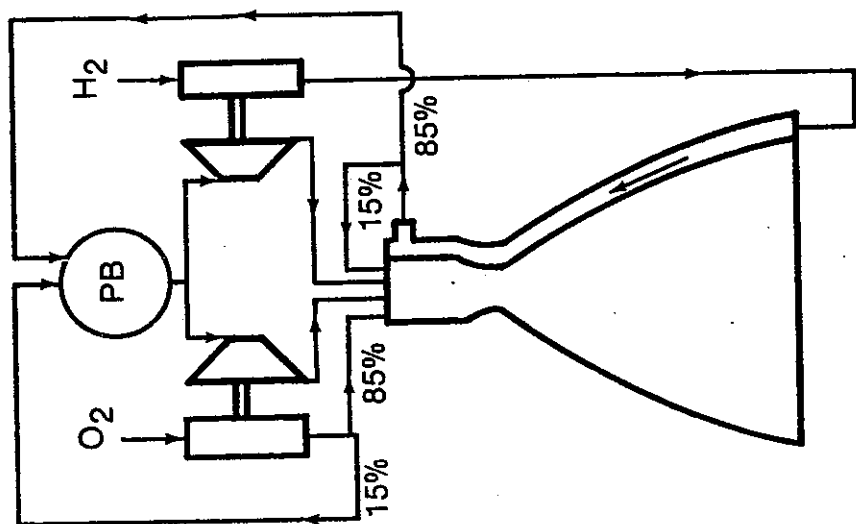
The selection of the staged-combustion-cycle engine over the high chamber pressure low pump discharge pressure (high Pc low Pd) engine was made by Dale D. Myers on October 12, 1970, after lively debate by MSC and MSFC personnel over the merits of the two designs. The MSC position was that the gas generator (high Pc low Pd) engine was competitive from a performance standpoint and early development phases should be funded along with the already funded programs by Aerojet, Pratt and Whitney, and Rocketdyne to develop the staged-combustion-cycle engine (ref. 5-5). In addition to weight and performance advantages, the MSC staff favored the high Pc low Pd design because the pump turbines were driven by a separate gas generator which allowed decoupling the development of the injector/thrust chamber from the pump set. The only area of technology in which the high Pc low Pd engine was considered by MSC to be at a disadvantage was the requirement for regenerative cooling at 4000-psi chamber pressure. MSC explained, however, the best flux encountered was only slightly greater than with the staged-combustion engine (ref. 5-6).

The MSC position recognized the difficulty of overcoming airplane instability by placing the heavier staged-combustion engines in the tail. The alternatives to solve the stability problem were as follows (ref. 5-7):

- Use a lighter engine such as the high Pc low Pd.
- Move the wing back to the center of gravity which eliminated the straight wing concept.
- Use bigger tail surfaces with attendant payload sensitivities.
- Use ballast in the nose which reduces payload capability.



High Pc low Pd
(gas generator)



Staged combustion

Figure 5-5.- Space Shuttle engine concept comparison.

5.2 SPACE SHUTTLE MAIN ENGINES (SSME) (Continued)

MSFC rejected the high Pc low Pd engine as a competitor concluding the design suffered from an immature technical base and had some inherent features (chamber cooling, chamber operations at higher mixture ratio, and turbine efficiencies) which cast serious doubts on engine manufacturer promises of substantially lower engine weight and only slightly reduced I_{sp} . Even assuming an optimistic break-even position on weight versus I_{sp} , MSFC saw no discernible payload or cost advantage for the high Pc low Pd engine configuration. MSFC recommended that no further studies of the design should be pursued because the diversion of valuable resources from the more promising staged-combustion concept would not be justified (ref. 5-7).

Mr. Myers concurred in the MSFC recommendation that the limited manned-space-flight resources should be concentrated on the staged-combustion-cycle engines for the Space Shuttle Program. He directed termination of all activities related to the gas generator, high Pc low Pd engine (ref. 5-8).

TABLE 5-1.- ORBITER MAIN ENGINE COMPARISON*

| | <u>Gas generator design</u> | <u>Staged-combustion design</u> |
|--|-----------------------------|---------------------------------|
| ● Chamber pressure, psia | 4000 | 3000 |
| ● Pump discharge pressure, psia | 5293/4703 | 6288/5904 |
| ● Envelope | | |
| - Diameter (OD), in. | 128 | 128 |
| - Length, in. | 242 | 242 |
| - Area ratio | 220 | 157 |
| ● Thrust (vacuum), lb | 474.2k | 475.8k |
| ● Specific impulse (vacuum), sec | 455.0 | 461.1 |
| ● Chamber cooling | | |
| - Heat flux (at throat) | 90 | 72.5 |
| - Coolant flow rates | 20.8 | 26.0 |
| ● Engine accessories weight | | |
| - Dry, lb | 4952 | 6240 |
| - Wet, lb | 5606 | 6940 |

*(Ref. 5-9)

5.2 SPACE SHUTTLE MAIN ENGINE (SSME) (Concluded)

Basic to the Space Shuttle concept of reusability was the extension of design life typically required for expendable engines (10 starts and 3600 seconds) which was sufficient for acceptance tests and single flight to that required for the SSME (55 starts and 27,000 seconds) which should be sufficient for 50 to 55 missions after acceptance tests. Field maintenance with minimum between-flight activity required advances in rocket engine technology to include identification and design of line replaceable units that are interchangeable without system recalibration; establishment and verification of effective inspection and automatic checkout procedures to facilitate the short turnaround goals; and integration of development experience, field maintenance records, and flight data analysis to extend the time between component replacements and engine overhaul (ref. 5-2). Further, complete engine interchangeability simplifies engine replacement time.

Internal visual inspection of critical parts replaces the engine disassembly method for routine inspection of parts. Routine maintenance tasks include automatic checkout, external inspection of engine hardware, turbomachinery torque checks, and "life" inspections with internal inspection of key components using borescopes. Routine use of fibrous optic devices is common practice for SSME (ref. 5-2)

The major SSME development challenges focused on developing sufficient turbomachinery horsepower to meet the high-pressure performance demands and to maintain turbine operating temperature both transient and steady state within life-limit practicality. The high-pressure turbopumps still have the most demanding field maintenance requirements. Borecope inspections of both high-pressure pumps are made after every flight along with shaft torque and travel checks. The number 3 bearing of the high-pressure oxidizer pump (HPOTP) is also inspected by borescope after each flight. The high-pressure fuel pump (HPFTP) is partially disassembled after every third flight for inspection of the turbine blade root sections. The engine life is currently limited by the pump changeout requirements of 1375 equivalent seconds at 109 percent power or 10 flights for the HPFTP and 3500 seconds of operational time or 7 flights for the HPOTP. Current development testing is being focused on these areas to extend life and reduce operational maintenance. Testing and development will continue to keep pace with operational use and extend the operational life goal to 55 starts and 27,000 seconds, as shown in figure 5-7. The testing will define components not capable of full life, and spares requirements will be adjusted accordingly. The redesign of short-life components will be considered only if clearly economical to the program (ref. 5-2).

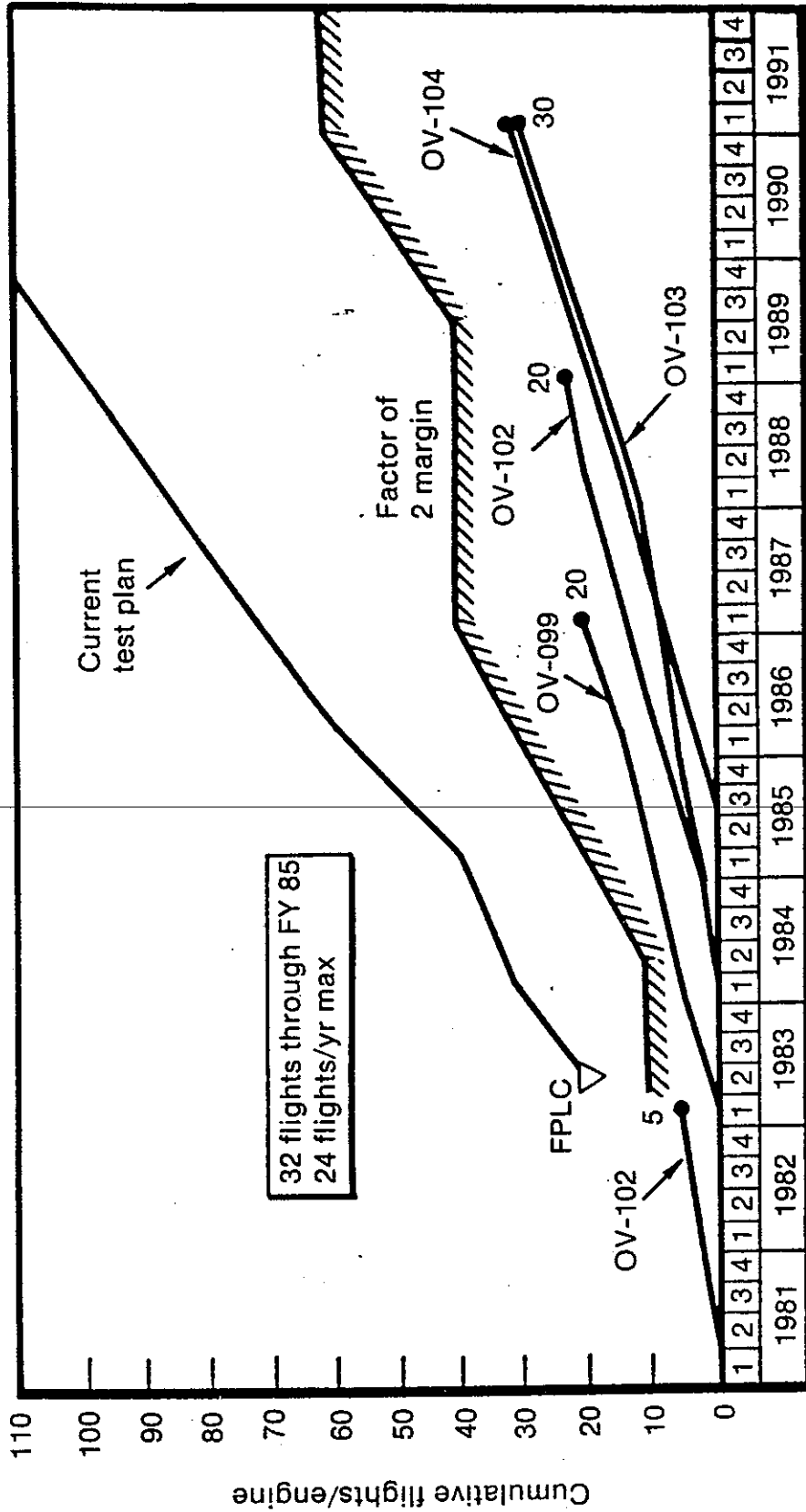


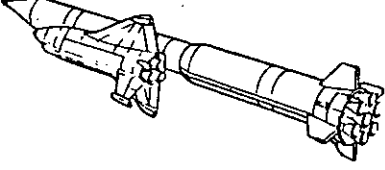
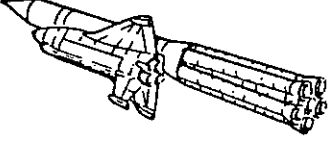
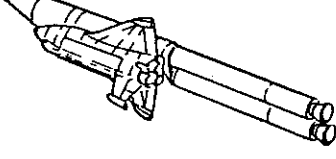
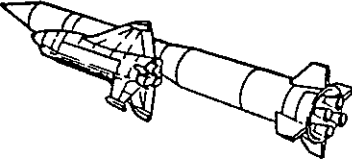
Figure 5-6.- Representative flight certification extension testing plan as described in 1983 Space Shuttle Technical Conference (ref. 5-2).

5.3 PARALLEL BURN SELECTED - 1972

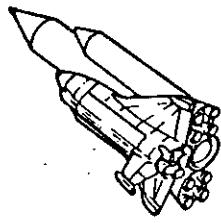
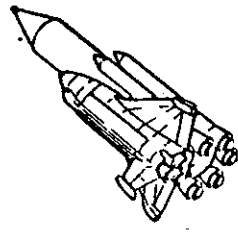

The series-burn configuration was preferred through the Phase B studies, primarily because the stacking arrangement and ignition sequence were existing technology and a familiar arrangement with which to go forward. The series-burn concept also had the advantage of placing the Orbiter higher on the stack, reducing thermal, acoustic, and vehicle damage due to debris such as ice or insulation. The series burn also had less complex separation requirements.

With the advent of the ET and the development of the high-performance main engines, the potential for a different modular configuration came into focus. Although unfamiliar to manned space flight, the parallel-burn concept came into being for a number of reasons. First, it allowed the use of a relatively low-performance, low-development-cost booster that was refurbishable to be used in parallel with the orbiter main engines. Secondly, it allowed verification of main engine ignition prior to booster engine ignition and subsequent release for launch.

There were drawbacks. The parallel-burn, clustered arrangement had never been used in prior manned spacecraft, which increased the technical risk. Also, the parallel configuration led to a highly integrated stack, coupling all elements and making change in one element more difficult. Of primary concern were the structural/dynamic interactions, and aerodynamic, heating, and controllability effects of such an arrangement. Through a lengthy series of tests and analyses and the addition of gimbal provisions to the solid rocket booster, these concerns were mitigated.

| | |
|---|----------------------|
|  | Reusable PFB |
|  | 120-in. SRM's |
|  | 156-in. SRM's |
|  | Reusable F-1 booster |

- Series burn
 - Higher weight/cost per flight
 - Familiar design (better load path)
 - Reduced heating, acoustics, and debris problems on Orbiter

| | |
|--|---------------|
|  | Reusable PFB |
|  | 120-in. SRM's |
|  | 156-in. SRM's |

- Parallel burn
 - Reduced costs
 - Reduced GLOW
 - Difficult structural interactions
 - All engines verified at lift-off
 - Control modes and separation
 - Highly integrated configuration

Figure 5-7.- Parallel burn selected.

SECTION 6
PHASE C/D (1972 TO PRESENT)

SECTION 6
PHASE C/D (1972 TO PRESENT)

With the decision to proceed with a staged-combustion, high-performance engine, the removal of the liquid hydrogen/oxygen primary boost propellant to an expendable ET, and the use of refurbishable solid rocket boosters (SRB's), the focus on the initial Phase C/D activity shifted to modular shapes, sizes, and stacking arrangements. These are depicted in figure 6-1.

Significant decisions made in Phase C/D involving technology issues are discussed in the following sections.

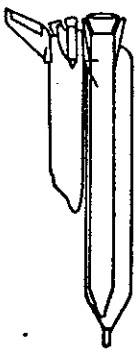

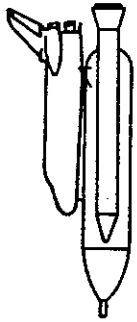
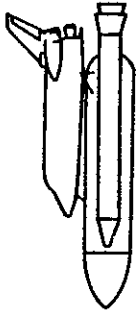

| Configuration | Vehicle | Orbiter | | External tank | Solid rocket boosters |
|---|------------------------------|--|---|---|--|
| | | Location and incidence | Design effects | | |
|  | ATP (3/4/72) | <ul style="list-style-type: none"> 972 in. aft of ET nose -1.2° incid. | <ul style="list-style-type: none"> Straight wing T.E. | <ul style="list-style-type: none"> 38 in. diam 30° nose cone | <ul style="list-style-type: none"> 156 in. diam SRB nose fwd. of ET shoulder Large cant nozzle No TVC |
|  | PRR (10/24/72) | <ul style="list-style-type: none"> 1063 in aft of ET nose +0.5° incid. | <ul style="list-style-type: none"> ASRM's removed OMS pods moved to shoulder Swept T.E. | <ul style="list-style-type: none"> 304 in. diam Ogive nose w/retro pkg. | <ul style="list-style-type: none"> 162 in. diam SRB nose at ET shoulder Pre-cant reduced TVC added |
|  | 2A (12/22/72) MCR 0074 | <ul style="list-style-type: none"> 768 in. aft of ET nose | <ul style="list-style-type: none"> Increased nose radius | <ul style="list-style-type: none"> 324 in. diam | <ul style="list-style-type: none"> 142 in. diam SRB aft of ET shoulder Reduced aft skirt flare No pre-cant |
|  | 3, 3A MCR 0200 (5/73) | <ul style="list-style-type: none"> 680 in. aft of ET nose | <ul style="list-style-type: none"> Reduced wing incid. Body 38 in. shorter Decreased nose radius | <ul style="list-style-type: none"> 324 in. diam Removed retro | <ul style="list-style-type: none"> 142 in. diam Decreased aft skirt size 7:1 nozzle in-line sep. motors |
|  | 4, 5 MCR 3570 | | <ul style="list-style-type: none"> Blunt OMS pods Uncovered RCS ports | <ul style="list-style-type: none"> 333 in. diam Biconic Ascent Air Data System (AADS) | <ul style="list-style-type: none"> Attach ring area reduced Eliminated 2 of the 4 hoop stress rings 4 abreast sep. motors |

Figure 6-1.- Flight vehicle configuration evolution, Phase C/D - 1972 to 1973.

6.1 NONMETALLIC ORBITER HEAT SHIELD MATERIAL SELECTED - FEBRUARY 2, 1973

For 2 years (1970-72), NASA studied metallic heat shields and nonmetallic heat shields by contract and in the NASA laboratories to determine the relative weights, costs, and technical risks. Metallic heat shields were studied as radiative panels as well as hot structural approaches.

Although weights of metallic panels were predicted to be competitive with external insulation systems, coatings for the refractory metals, oxidation, attachment methods, and creep behavior of all metallic materials were key concerns that were still being examined when NASA selected the nonmetallic, external insulation for the Orbiter baseline.

After selection of the nonmetallic system, two candidate materials, silica and mullite (aluminum silicate), were evaluated by an independent test source (Battelle Mem. Inst., Contract NAS9-10853); silica was superior in thermal performance.

The density of the reusable silica insulation was reduced from 15 lb/ft³ to 11 lb/ft³ and later to 9 lb/ft³. As density decreased, the strength also decreased and the fragility increased. Eventually, the material density was varied to meet strength requirements in various locations of the vehicle. Bond line failures were remedied by gradation of the density through the tile thickness, and strain compatibility with the Orbiter airframe was achieved by a nonmetallic pad between the ceramic and the aluminum structure.

Two contractors, McDonnell Douglas (NAS9-11223) and LTV (NAS9-1124), developed versions of the reinforced carbon-carbon (RCC) system; at the completion of the studies, LTV was selected to provide the leading edge and nose cap hardware. Metallic materials considered for the leading edge and nose cap were not acceptable because of poor reliability of coatings and poor performance in an oxidizing environment.

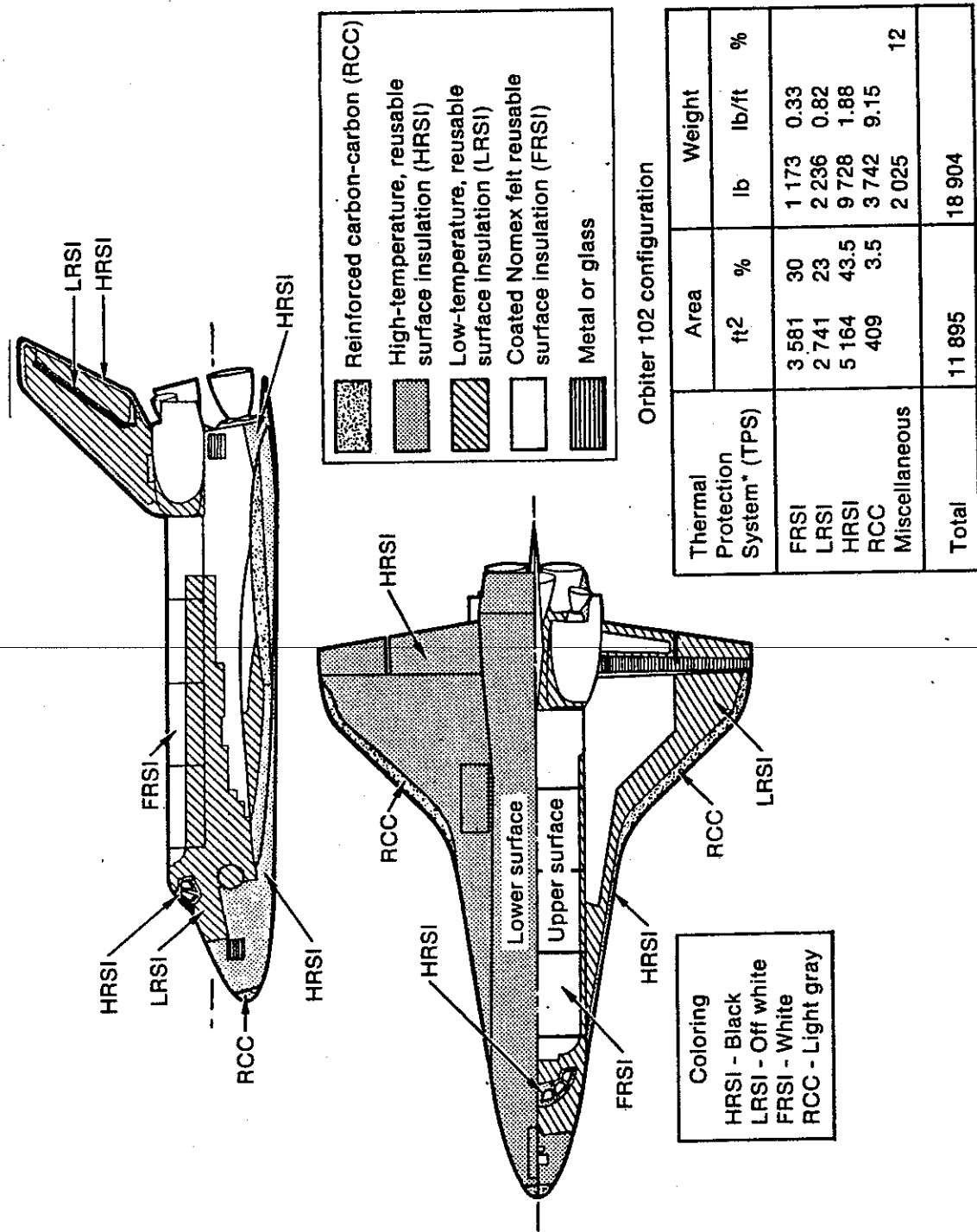


Figure 6-2.- Nonmetallic heat shield material selected.

6.2 SOLID ROCKET MOTOR PROPELLANT SELECTED: NOVEMBER 1973 - POLYBUTADIENE ACRYLONITRILE (PBAN) PROPELLANT FAMILY SELECTED; JUNE 1974 - PROPELLANT FORMULATION FINALIZED

The NASA decision to use solid rocket motors (SRM's) for the Space Shuttle booster was based on the low development cost of SRM's relative to the development cost of liquid propellant boosters. Program risks and costs were to be held down by adopting technology proven in Air Force programs such as Minuteman and Titan. For propellants, this meant scaling up the size of the PBAN family of propellants.

In November 1973, Thiokol had processed some 200 million pounds of PBAN propellant for first-stage Minuteman and had extensive experience with controlling the propellant mechanical properties and burn rate. The PBAN propellant was a low-cost composite propellant with a Class II (nondetonable) explosive classification and proven performance.

During the study phase and the initial part of the development program, tradeoffs were made between propellant mechanical properties and the increased performance available by replacing up to 4 percent of the PBAN polymer with 4 percent ammonium perchlorate and aluminum. Air Force programs had demonstrated increased performance and adequate mechanical properties with solids loadings (ammonium perchlorate and aluminum) as high as 90 percent. However, because of the concerns that might be introduced about propellant cracking with lowered mechanical properties, it was decided to finalize the propellant formulation with 76 percent solids loading to provide a higher grain safety factor.

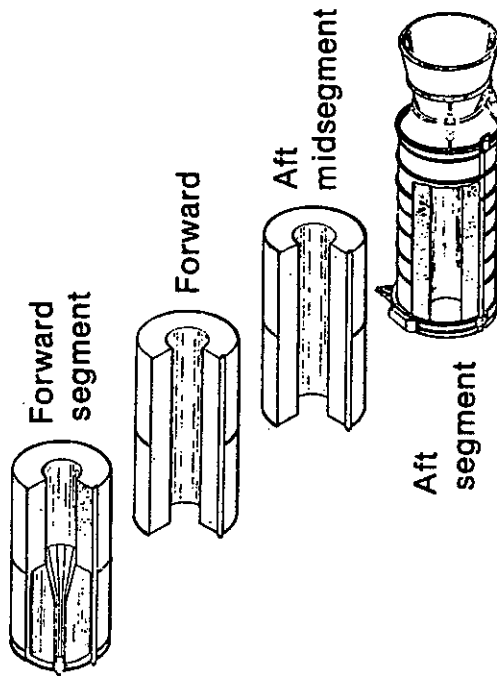
Several alternatives were studied for achieving burning-rate control by varying the ground to unground oxidizer ratio, varying iron oxide content, and by the addition of other burning rate modifiers. The final selection for burn-rate control was the variation of iron content with the ground to unground oxidizer ratio fixed. Some concerns were raised during the development program about acid rain fallout from the booster plume but these were not deemed serious enough to dictate more than a paper study of propellants that could cure the acid rain "problem" but with a large performance penalty and a total loss of the propellant experience base (ref. 6-1).

TP-H1148 propellant formulation

| <u>Material</u> | <u>Weight (%)</u> |
|-----------------------------------|-------------------|
| HB Polymer | 14.0 |
| ECA Type II | 16.0 |
| Aluminum (nonspherical 25 μ) | 0.3 |
| Iron oxide Type II | 69.7 |
| Ammonium perchlorate | |

Propellant characteristics

| | |
|---|---------|
| Burn rate at 1000 psia - r_b (in/sec) | — 0.435 |
| Burn rate exponent - m | — 0.35 |
| Density (lbm/in ³) | — 0.064 |
| Temperature coefficient of pressure (π_k) - %/°F | — 0.11 |



Weight data

| | <u>Forward segment (lb)</u> | <u>Center segment (2 req)</u> | <u>Aft segment (lb)</u> |
|---------------------|-----------------------------|-------------------------------|-------------------------|
| Ignition system | 472 | | |
| Case | 26 182 | 22 445 | 30 286 |
| Propellant | 299 471 | 272 155 | 266 344 |
| Ins/inh/liner | 5 537 | 2 179 | 7 369 |
| Systems tunnel | 102 | 102 | 150 |
| Instrumentation | 3 | | |
| Nozzle | | | 22 451 |
| External insulation | | | 138 |
| | 331 767 | 298 881 | 326 738 |

| | |
|-----------------------------|------------------|
| | 1 252 267 |
| Attach provisions | 289 |
| Uncontrolled inerts | 240 |
| Inert contingency | 2 256 |
| <u>Total nominal weight</u> | <u>1 255 052</u> |

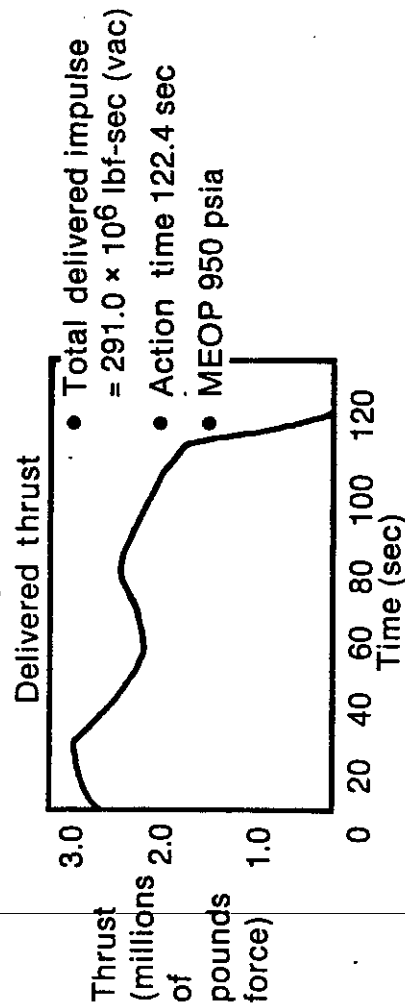


Figure 6-3.- Solid rocket motor propellant selected.

6.3 ICE/DEBRIS PROTECTION

In 1971, when the parallel-burn Space Shuttle configuration was selected, neither NASA nor any of the Phase B contractors recognized that ice would form on the external LO₂ tank, dislodge at engine ignition, and damage the Orbiter TPS. The fact that ice will form on cryogenic propellant tanks and will dislodge at rocket ignition or during ascent has been known ever since cryogenic propellants have been used in liquid rockets. In late 1973, North American Rockwell engineers realized that ice from the LO₂ ET was to be a serious issue when they began to analyze the capability of the Orbiter to fly through rain. Even after this issue was recognized, the ET procurement specification did not indicate that insulation would be necessary to prevent ET ice formation. The MSFC ET RFP, released April 2, 1973, did not specify the need for insulation to prevent the formation of ice. In the contractor proposals received for the ET DDT&E, only McDonnell Douglas recognized and addressed the need for insulation to prevent ice from damaging the Orbiter TPS. Martin Marietta, the winner of the ET competition, did not address the issue because it was not an RFP requirement.

To date no complete solution has been implemented to eliminate all ET ice formation and this problem still plagues the program. Significant program resources and Space Shuttle lift capability have been expended to develop means to minimize the ice and to analyze the impact of ice debris procedures to assure minimal damage at Space Shuttle engine ignition. The Space Shuttle will continue to be faced with launch constraints until a satisfactory means can be implemented to completely eliminate ice formation regardless of the weather conditions. The selection of the parallel-burn configuration with the Orbiter located on the lower portion of the ET contributes to this issue.

A chronology of events related to this overlooked, ongoing, and troublesome unknown is shown on page 6-9.

LEVEL II ICE/DEBRIS REQUIREMENTS

- Insulation (spray-on foam insulation (SOFI)) requirements: (1974)

Protect LH₂ and LO₂ tanks and other cryogenic surfaces with a minimum of 1 inch (nominal) SOFI (some limited ET areas don't meet these requirements).

- Protuberance requirements: (1976)
 - Minimize ice for all protuberances
- Ice minimization requirements: (1979)

"The Shuttle system, including the ground systems, shall be designed to preclude the shedding of ice and/or other debris from the elements during prelaunch and flight operations that would jeopardize the flight crew and/or mission success."

Note: Ice is defined as frozen water of 18 lb/ft³ or greater density formed on the outside exposed surface(s) of any element. Frozen water of < 18 lb/ft³ is considered to be frost and is of no concern.

"The Shuttle system shall be designed so that 'launch holds' due to ice formation shall not occur more than 5 percent of the time based on atmospheric conditions at the launch pad in the proximity of applicable launch vehicle surfaces."

"The Shuttle system shall provide the capability to monitor the local atmospheric conditions and parameters and to be able to determine the quantity of ice formation upon the Shuttle vehicle surfaces to ensure acceptable ice conditions for launch."

- Post-ST_S-2 residual issue: (1981)
 - Eliminate or reduce ice/debris areas on lightweight tank.

6.4 DELETION OF ET DEORBIT SYSTEM AND INCORPORATION OF SUBORBITAL JETTISON CONCEPT - 1973

On June 15, 1972, H. W. Tindall, Director of Flight Operations, expressed his concern about the technical complexity and risk involved in using a solid rocket motor to deorbit the ET. He predicted that this approach would have a major adverse operational cost impact. He recommended that the Mission Planning and Analysis Division (MPAD) analyze the feasibility of suborbital ET jettison as an alternative approach. This led to a series of MPAD studies which refined the concept.

This approach of suborbital jettison has been used operationally in what is now known as direct insertion with the external tank being reentered into the Pacific Ocean south of Hawaii. The ET reentry velocity is, for all practical purposes, identical to velocities encountered by orbital reentry of the ET. The breakup characteristics are quite similar.

The final proposal by W. R. Woodis, of Martin Marietta Corporation, on January 23, 1973, to jettison the ET at a suborbital velocity to obtain safe ocean disposal provided for a significant simplification of the ET systems, lower development and recurring cost, and a Space Shuttle payload weight improvement of approximately 2750 pounds. Prior to the proposal by W. R. Woodis, North American Rockwell was involved in the preliminary design of SRM systems and procedures for the safe and accurate disposal of the ET in the Indian Ocean and South Pacific Ocean. The advantages and disadvantages are listed below.

ADVANTAGES

- NO EXPENSIVE DEORBIT SYSTEMS
 - SOLID ROCKET MOTOR
 - ATTITUDE REFERENCE ON ET
 - RF LINK WITH ORBITER
 - POWER SUPPLY TO IGNITE SRM
- IMPROVED SAFETY AND RELIABILITY
 - NO DEORBIT SYSTEM TO FAIL AND DAMAGE THE ORBITER OR CAUSE REMOVAL OF SEVERAL SOURCES OF IMPACT DISPERSIONS
- SIMPLICITY
 - NO MOTOR FIRING OR CONDITIONING
 - NO PRELAUNCH ACCESS REQUIRED AT ET NOSE
 - ALLOWED LO₂ TANK NOT TO BE DESIGNED BY RETRO LOADS
 - RESIDUAL PROPELLANT DUMP SYSTEM ELIMINATED
 - REDUCED TPS REQUIREMENT
- COST REDUCTION - ESTIMATED 1972 DOLLARS UNCONTROLLED ET IMPACT

| | | |
|----------------|-------|--------------|
| - NONRECURRING | ----- | 8.3M |
| - RECURRING | ----- | 52.2M |
| | | <u>60.5M</u> |

DISADVANTAGES

- REDUCED ORBITER MAXIMUM ALTITUDE
40 NAUTICAL MILES

The concept was to target main engine cutoff (MECO) for a specific Indian Ocean or South Pacific Ocean ET impact point; following MECO, separate the ET and increase the Orbiter velocity with the orbital maneuvering system (OMS) engines until the desired orbital velocity is achieved. This concept, used on the first orbital flight, is depicted in figure 6-4. The concept was approved for incorporation in the program by the Space Shuttle Program Director at NASA Headquarters on March 6, 1973.

The Space Shuttle Program cost savings were calculated in fiscal year 1986 dollars to possibly exceed \$250 million.

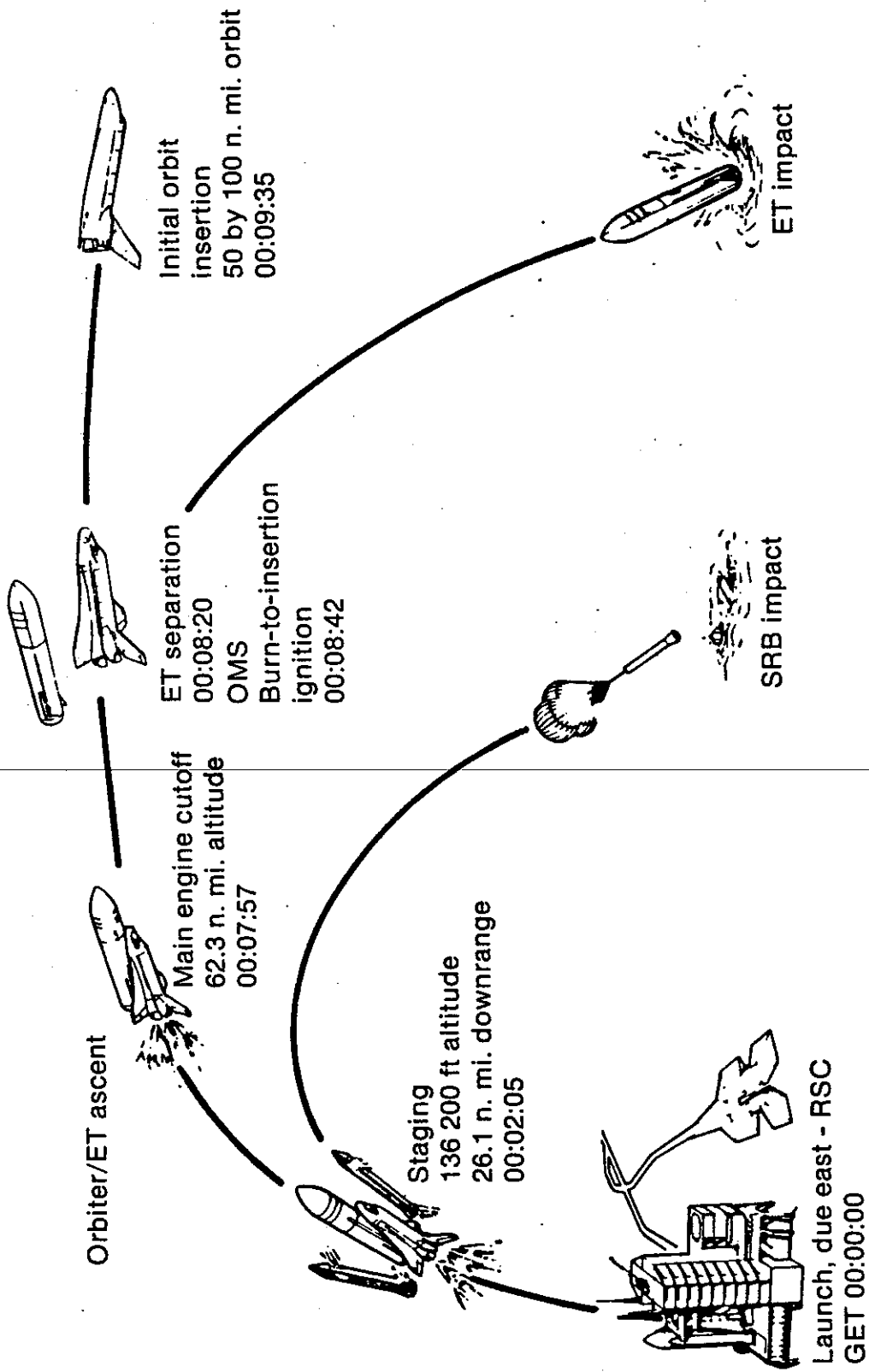


Figure 6-4.- NSTS mission 1 profile: due east - payload delivery/retrieval.

6.5 PERFORMANCE IMPROVEMENTS

Historically, the space programs have been faced with spacecraft inert weight growth, as have all prior transport and orbital spacecraft. Solutions have included spacecraft weight reduction and booster performance enhancement programs. The Space Shuttle has been no exception. The problem, however, was compounded because there was weight growth in the Orbiter and a loss of performance with the ascent LO₂/LH₂ propulsion SSME's.

The following sections outline the more significant performance improvement efforts undertaken to regain a portion of the lift-capability losses.

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6.5.1 ET Weight Reductions - 1977

During the late 1976 and early 1977 time period, there was an increasing concern relative to the loss of Space Shuttle payload lift capability because of inert Orbiter weight growth, increased ET insulation weight to reduce ice formation, constrained ascent trajectories to limit Orbiter wing loads, and installation of additional helium tankage to provide SSME bearing purge. Performance improvement and weight-reduction efforts were implemented across the program to arrest the degradation in the Space Shuttle lift capability.

Following analyses by MSFC and Martin Marietta, the ET Level III MSFC project manager proposed to the Level I and Level II program managers that 4000 to 6000 pounds could be removed from the baseline ET. This proposal was approved July 1, 1977, and the lightweight ET became effective on ET 26 and subsequent tanks.

The weight-reduction effort on the ET was highly successful and, in fact, 2000 pounds more weight reduction was achieved than was originally planned. This ET extra weight reductions to some extent offset the planned Orbiter and SRB weight-reduction goals that were not achieved.

A chronology of the events leading to the final approval of the lightweight tank, the objectives, and the methodology are depicted on page 6-15.

● CHRONOLOGY

- July 22, 1975, MSFC initiated Technical Directive 1.6.2.4-13A to study potential weight reduction design options.
- March 4, 1976, letter from Dr. Malkin to Mr. R. F. Thompson; Dr. Fletcher suggested creating an independent group to explore ET weight reductions.
- October 7, 1976, ET quarterly review; statused ET weight reduction.
- November 6, 1976, letter from Dr. Malkin to Mr. R. F. Thompson; Space Shuttle weight and performance review resulted in a request to develop a 4,000-pound weight-reduction plan.
- January 21, 1977, Lindstrom/Kingsbury review of ET project weight-reduction implementation plan.
- July 1, 1977, Approval of 4,000-pound ET weight reduction by Dr. Malkin as a result of a special reference (PRC8D) number 203655 ARI.

● OBJECTIVE

- Define weight reduction candidates with acceptable technical risk to accomplish a 4,000-pound dry weight reduction considering

1. Total program cost
2. Current state of the art in structures/materials
3. Availability of STA data to
 - Verify load paths
 - Verify structural margins
4. Flight loads data

● METHODOLOGY

- MMC identified and scoped a total of 31 candidates

| | | |
|-----------------|---------|-----------|
| - 23 Structures | (8692) | |
| - 1 TPS | (500) | (10,000) |
| - 6 Propulsion | (700) | |
| - 1 Electrical | (38) | |
- 4,000-POUND SELECTION

| | | |
|----------------|---------|------|
| - 4 Structures | (3025) | |
| - 2 TPS | (800) | 4285 |
| - 1 Propulsion | (420) | |
| - 1 Electrical | (40) | |

6.5.2 High-Performance Motor (HPM) - 1980

As the Space Shuttle system evolved through the development process, it became evident that the payload delivery commitment made at program initiation could not be met because of growth in system weight. In surveying the alternatives for improving performance versus the cost and schedule impact of providing the increases, the high performance solid motor appeared attractive as one of the routes to pursue resulting in the issuance of program authority to proceed in October 1980.

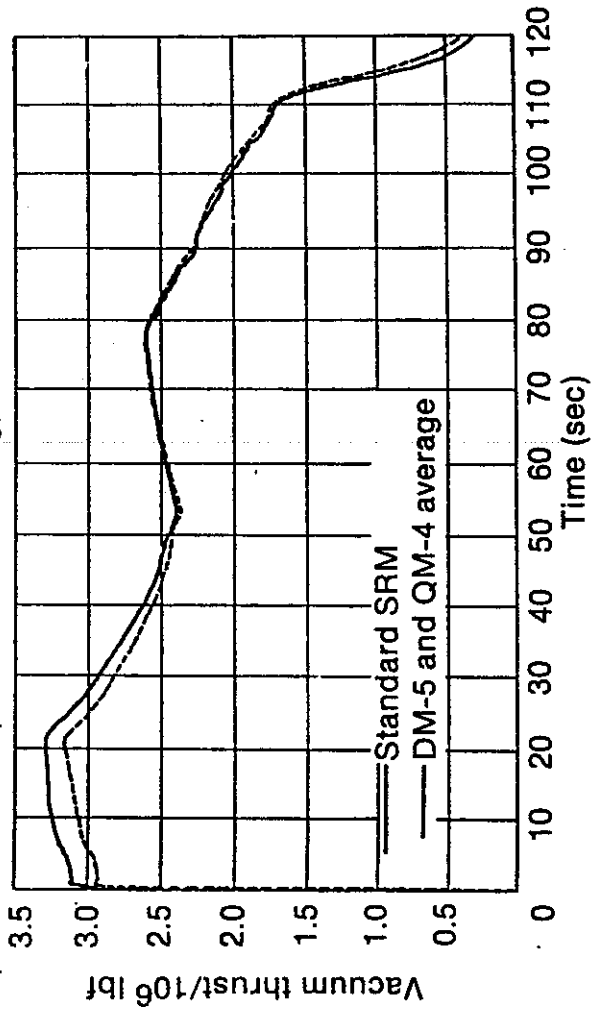
The HPM was designed to provide 3000 pounds of added payload by tailoring the thrust time trace and by increasing the expansion ratio from 7.16 to 7.72. The higher thrust during the first 20 seconds of propellant burn was provided by reducing the height of castable insulator by 20 inches on the center segments exposing more propellant at motor ignition. Some added insulation was required at that end of the center segments to protect against the longer exposure time for that area.

The increased expansion ratio was achieved by simultaneously extending the exit cone 10 inches and by decreasing the nozzle throat diameter from 54.43 to 53.86 inches. The nozzle throat diameter change was possible through changes in the wrapping mandrel.

The weight of the HPM was reduced by changing the case wall thickness.

The HPM changes were verified in successful firings of DM-5 on October 21, 1982, and QM-4 on March 21, 1983 (ref. 6-2).

Comparison of HPM (DM-5 and QM-4 avg) and standard SRM thrust traces (1)



(1) Corrected to 0.368 in/sec burn rate

HPM nozzle changes

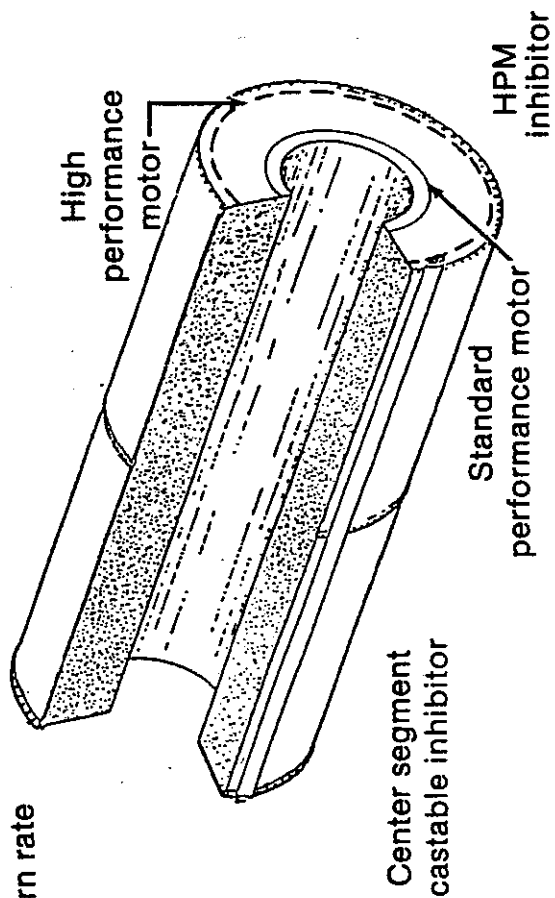
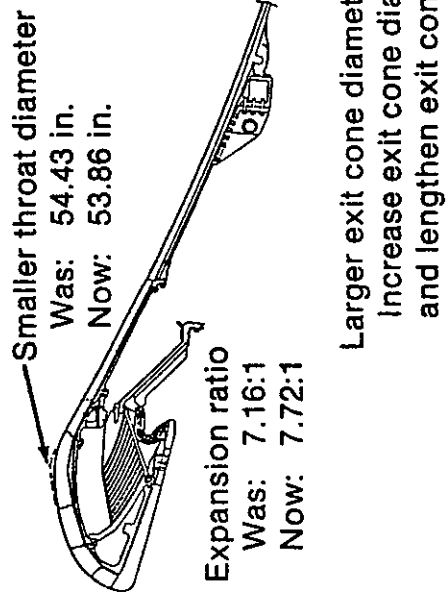


Figure 6-5.- HPM changes.

6.5.3 Graphite Case SRM - 1982

A need for increasing the Space Shuttle payload capability for polar missions (Mission 4 from 24,000 to 32,000 pounds) resulted in the study of several options for achieving this payload increase in late 1979.

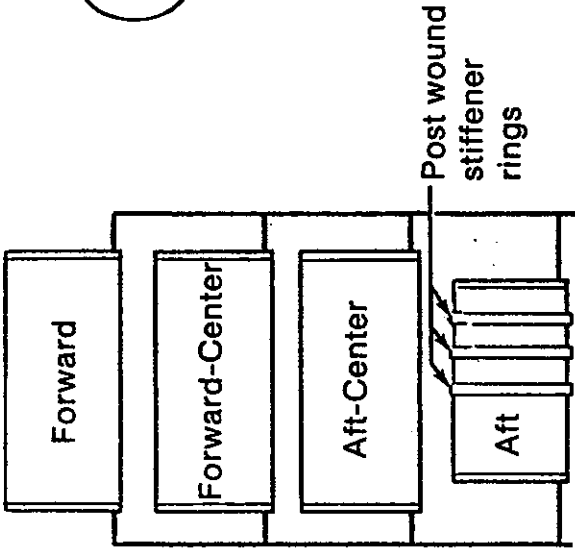
Options included:

- Liquid boost module - use of Titan components and engines mounted under the ET
- SRM's (shortened) mounted under the ET
- Short-duration, high-thrust solid motors mounted on the SRB's
- Increasing SSME thrust to 115 percent

Each option had many disadvantages and so 6 months later when Hercules presented a proposal to achieve the 8000-pound payload increase by substituting a filament-wound case (FWC) for the steel case, it was favorably received by NASA Headquarters and MSFC was directed to initiate a feasibility program to see if such a case could be developed. In February 1981, MSFC competitively selected both Hercules and Morton Thiokol to demonstrate the feasibility of FWC's. During the feasibility programs, both contractors selected graphite fibers as opposed to Kevlar for the FWC.

In early 1982, MSFC proceeded with the competition for the full-scale FWC. By that time, it had been decided that the steel forward and aft domes and the ET attach rings would be retained to preserve the interfaces with the forward and aft skirts and the ET attach interfaces. During the source selection, both contractors proposed an FWC which also preserved the SRM field joints by using steel rings attached to the ends of the graphite cylindrical segments. Hercules was chosen to develop the FWC in May 1982 (ref. 6-3).

- FWC cylinders from Hercules/Bachus



- SRM steel flight hardware

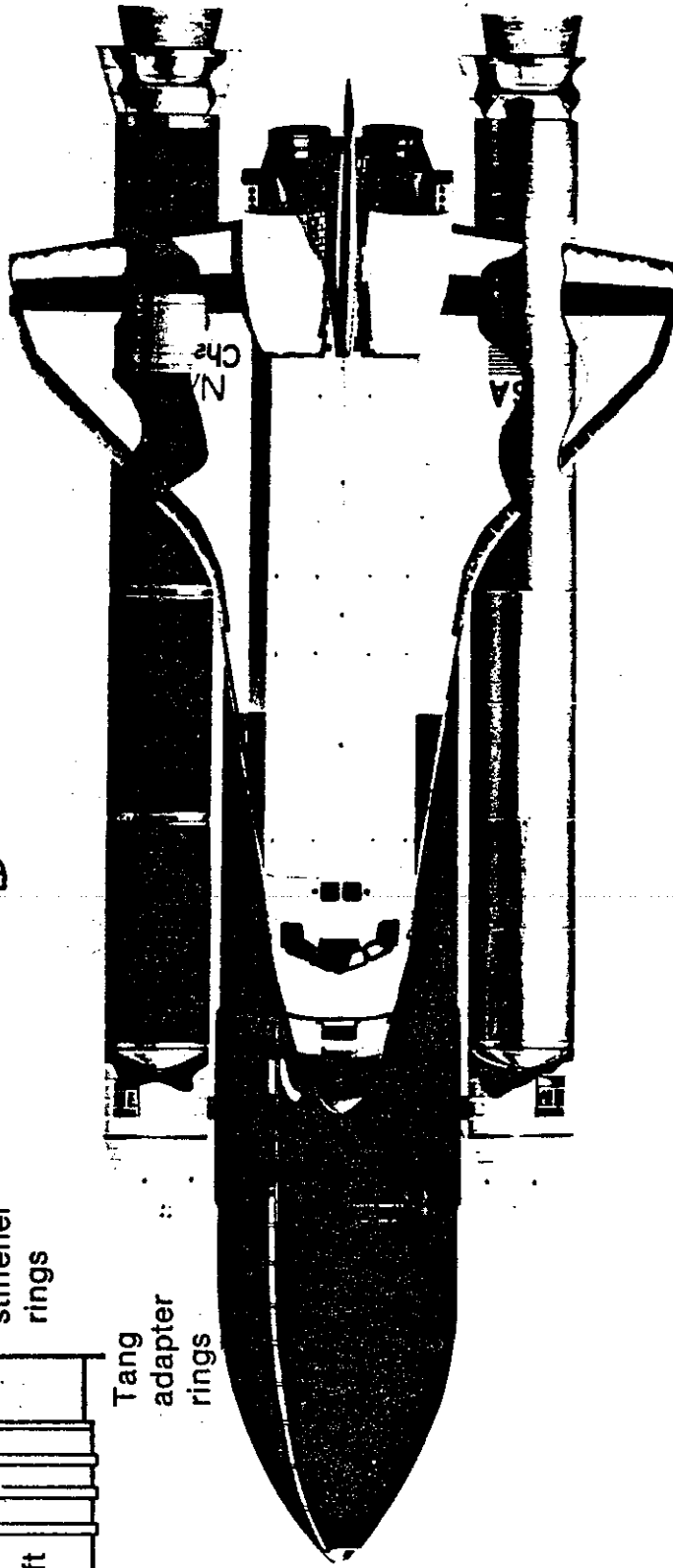
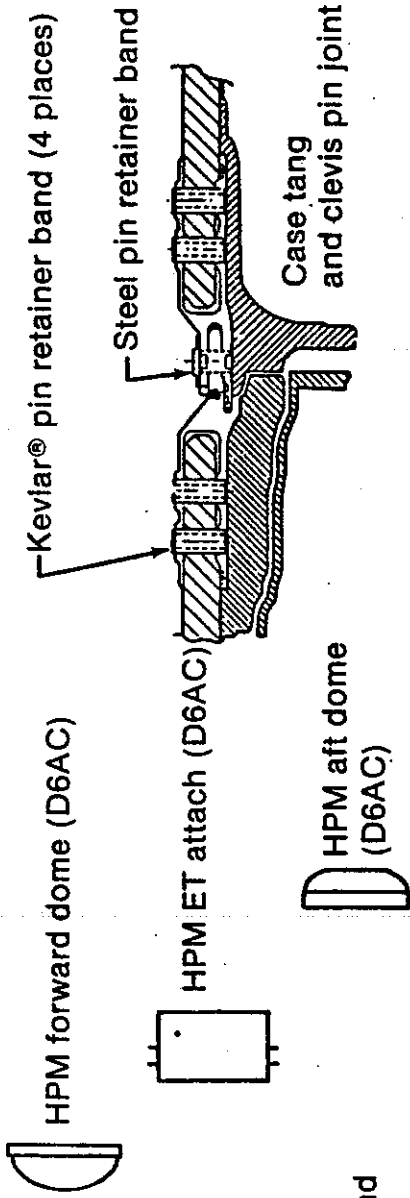


Figure 6-5.- FWC-SRM

6.6 AVIONICS

At the completion of the Space Shuttle Phase A studies, the Space Shuttle Task Group identified several capabilities and technology requirements that would be needed to support the program objectives. These could be classified as follows:

- Capabilities

Autonomous Operation (ascent and return)

Onboard system management, including malfunction detection and onboard prelaunch checkout

- Hardware Development
 - Standard Subsystem Interfaces
 - Multiplex Data Bus
 - Multipurpose Displays
 - Integrated Avionics

The need was also identified to develop an integrated approach to avionics to successfully implement the desired features.

6.6.1 Autonomous Guidance, Navigation, and Control (GN&C) Operation (Ascent)

As distinct from previous manned space programs, the Space Shuttle has no separate ascent guidance and control system; therefore, the Orbiter GN&C system performs all mission functions.

To meet the autonomous requirement, the system is implemented using a redundant set (3) of inertial sensors. These inertial measurement units (IMU's) are capable of providing the desired insertion conditions with no updates or backup from the ground.

Between lift-off and SRB staging, there is no closed loop guidance; the system is driven by a stored pitch and roll profile. After SRB separation, a powered explicit guidance algorithm is invoked which guides the vehicle to the desired trajectory conditions at MECO.

No significant problems have been experienced with ascent GN&C and the system meets the objective of fully autonomous operation. Several inertial sensors are available off the shelf with the main trade being the choice between strap down and gimballed.

6.6.2 Autonomous GN&C Operation (Return)

To meet the requirement for return to a specific runway, an inertial system is inadequate because of time-related errors. Although a number of concepts were investigated, no sensors or systems capable of meeting the autonomous definition were found to be available. Accordingly, the system design incorporated tactical air navigation (TACAN) and microwave scanning beam landing system (MSBLS) to supplement the onboard information. The TACAN data were checked for accuracy by ground tracking radar before being added to the onboard navigation data.

Provisions have been made for adding a global positioning system (GPS) when the GPS becomes operational.

6.6.3 Onboard Systems Management

6.6.3.1 Fault Detection, Identification, and Reconfiguration (FDIR)

Redundant systems were in common use in aircraft and previous space programs but the function usually involved the manual recognition of faults and crew-initiated switching to an alternative system. In most cases, the alternate system was a degraded mode of operation. Two driving requirements on the Space Shuttle avionics were the FAIL operational, FAIL operational, FAIL safe (FoFoS) requirement (which could not be achieved by a primary/backup concept) and the determination that during certain flight phases, failure detection and reconfiguration required faster response times than a human operator could perform.

The solution was the use of multiple units in parallel operation giving high mean time between failure (MTBF) and tolerance to two failures. Considerable effort was required to develop redundancy management software, devise logical end-to-end redundant parts at hydraulic and electrical system interfaces, and eliminate "sneak path" single point failures in both hardware and software. To provide assurance against generic software single point failures, a fifth computer was added to form a backup system with degraded capability. Based on the performance to date, the multiple redundancy and FDIR techniques appear to satisfy all requirements, and it is probable that the backup system concept can be dropped from future designs.

6.6.3.2 Onboard Checkout

Because of the effort required to achieve a redundant system meeting all requirements, insufficient emphasis could be applied to onboard checkout. However, most of the basic elements of onboard checkout were implemented in the onboard software; examples are the redundant set launch sequence, the test supervisory routines used in conjunction with the launch processing system (LPS), and the backup caution and warning system, etc. The full capabilities of the Space Shuttle data management system (DMS) are still to be exploited when program resources can be made available. No technology advances are required to achieve onboard checkout in future programs; however, adequate priority must be given to the requirement to ensure successful completion.

6.6.4 Hardware Implementation

6.6.4.1 Multiplex Data Bus and Standard Interface

Data bus technology was being applied to aerospace programs at the start of the Space Shuttle Program but was confined to data acquisition for instrumentation systems only. No program had used such techniques for critical command and control functions. The Space Shuttle pioneered the use of data technology for this type of application to the extent that it is now accepted practice on most vehicles where copper path weight savings are an important design factor. The Space Shuttle Program also successfully managed the difficult problem of applying a standard interface to all subsystems by employment of the multiplexer/demultiplexer (MDM) and multiplexer interface adapter (MIA) concepts with the latter component being a government-furnished-equipment item to subsystem suppliers.

Multiplexed data buses and standard interfaces are now available as off-the-shelf components, some designed to military standards such as MIL STD 1553. Because of the wide availability of components (IMU's, etc.) with a built-in standard interface at a low price, the approach should be considered for future programs.

Optical buses are now being considered for future vehicles with the main advantage being greater immunity to electromagnetic interference (EMI) problems and higher data rates. It is important to note that no EMI problems were encountered on the Space Shuttle Program and copper path data buses have an adequate data rate for manned vehicle application. Optical buses, however, are needed for transmission of experiment data, video applications, etc.

6.6.4.2 Multipurpose Display

Multipurpose displays using CRT were in their infancy at the beginning of the Space Shuttle Program. Small nonclave displays qualified for airborne use were available in this country but Europe had the only airborne color display available. As color was considered important for an adequate representation of the fault detection, annunciation, and identification (FDAI) and other functions, a hybrid-approved was adopted with multipurpose displays being backed up by dedicated electro-mechanical instruments. (A significant system penalty was exacted by the requirements to convert the avionics digital information into synchro-type information for driving these displays.)

With the rapid advancements in color display technology, using techniques other than CRT, future systems can be expected to fully utilize multipurpose display technology and dedicated displays used only where adequately justified.

6.6.5 Integrated Avionics

The identified requirements for onboard system management, onboard checkout, and redundancy management dictated a need to consider the onboard electronics systems as a total integrated system and thereby fully capitalize on the potential advantage of commonality, standard interfaces, and data formats.

The considered approach was to request the prime contractor to organize with an avionics integration group heading up the system mechanization and develop software requirements, with the Government providing verified software to their requirements. The design was based on a centralized computer system with the intent of keeping the software sufficiently modularized that fixes regressive testing, etc., would not perturb the overall software release.

However, because of continued growth in software requirements, the modularity concept was diffused to the extent that software changes of any significance created a requirement for total system reverification, this escalating the total software and avionics facility costs.

Notwithstanding problems of this nature, the avionics integration was accomplished successfully with no show stoppers during the development and operational phases. Most current aerospace programs are conceived with an integrated avionics approach.

The current trend is, however, towards distributed processing systems which more fully utilize the capabilities of the major subsystem supplier and by enforcing the disciplines of software modularity can lessen the overall verification costs.

6.7 ORBITER BRAKES

In the early 1970's, NASA's goal to develop a completely reusable spacecraft generated the requirements for several new technology subsystems. The brakes were considered one of the systems that could benefit from new technology. This understanding was reached after a detailed historical review of operational aircraft braking systems and their performance. The study covered aircraft in general, commercial and military aviation. The Department of Transportation, Federal Aviation Administration, and military services were requested to provide data on all accidents involving brakes. Both organizations did this on a monthly basis. These data revealed many accidents were attributed to brake malfunctions. It then became evident that Orbiter brake development was going to be a difficult technology problem. In an attempt to get an early understanding of potential new brake technology, JSC initiated a \$10,000 contract to the B. F. Goodrich Company to design, build, and test a beryllium carbon brake. This brake design investigation resulted from the good experience reported by the Air Force on a similar design for the C-5A airplane. The B. F. Goodrich Company was claiming this brake developed equal energy for stopping when compared to conventional carbon, carbon, or sintered-iron brakes at a big savings in weight and volume. By this time, the Orbiter's design requirements were becoming well defined which further indicated the desirability for a new high technology braking system. Two of the major design driver requirements were for a system capable of delivering 50 million foot pounds of energy and an operational life use of 5 landings. As the Orbiter system began to mature, the landing weight began to increase. The brake system, along with all the other systems, was constrained to redesign to the lightest weight possible. At this point, it appeared the only braking system that could possibly meet all the Orbiter landing requirements was the beryllium carbon brake being developed by B. F. Goodrich Company. This braking system was recommended for the Orbiter and was approved as the design approach.

Rockwell contracted with B. F. Goodrich Company for the Orbiter brakes. The brake development was a significant challenge. Many problems both small and large were encountered and solved. The solutions at times were questionable because of the lack of an adequate test method. Examples of problems were pad clip failure and cracking of beryllium. The situation was further complicated by the following: weight growth which increased landing velocities; how to test a full-up system when there was no way to taxi an Orbiter; and how to interpret the test data obtained on a dynamometer. Funding problems caused delays in the test program and deletion of a much needed automatic braking feature. This feature is standard on a lot of commercial aircraft.

Many analyses and a lot of effort were expended in developing means to test the brakes directly on the Orbiter. Examples of proposed methods were towing the Orbiter with 747 aircraft, truck towing, mounting solid rockets in rear of the Orbiter for thrust, and use of a Navy catapult launcher. Generally, all of these approaches were feasible; however, they were deleted because of cost and complexity. Finally, it was realized that brake testing would be limited to static and dynamometer tests. The problem, as stated earlier, was understanding the data obtained by this method and what it meant when the brakes were actually used to stop the Orbiter.

An off-limits test program was initiated at Bendix. Even though Bendix had a better test facility, the use of the Bendix facility caused many problems. Bendix claimed proprietary rights and would not allow B. F. Goodrich to witness or directly inspect brake damage as it occurred in the test. The data obtained from these tests, if properly understood, could have possibly foretold the problems that were experienced on actual flights. The final plan was to obtain braking test data during the Orbiter approach and landing test. We learned from the approach and landing tests; however, this was not considered the most desirable way to qualify a new technology braking system.

There are other sideline issues such as excessive brake pressure (3000 psi) which was corrected by installation of a 1500 psi pressure limiter. The flight crew was concerned with pedal track and forces. Design changes were made to include their requests.

In summary, all the brake anomalies and corrective action are now known; therefore, a redesigned brake system having adequate safety margins to meet the now known actual Orbiter landing requirements is feasible. In retrospect, this was one system that probably would have been more successful with old technology. The ingredients required to correct the brake anomalies are resources and an adequate test program.

SECTION 7
CURRENT TECHNOLOGY ISSUES

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CURRENT TECHNOLOGY ISSUES

A significant objective of this task was identification of the technology areas that limited the development of the Space Shuttle system and the technology areas that will be beneficial to future generations of space transportation systems. This section identifies those technologies applicable for future space transportation systems and describes them in detail with appropriate rationale.

7.1 MAIN ENGINE REUSABILITY

Continue the SSME Flight Certification Extension Testing to increase the life of high-pressure turbomachinery elements such as bearings, seals, and turbine blades and to improve the technology associated with airline-type maintenance concepts such as trend analysis.

Only test results can become part of the mechanized information system being developed to analyze trends in tracking critical hardware for operational exposure and life-limiting conditions. Such a system can provide fingertip information with respect to component, remaining available life.

For a reusable system to be competitive with emerging expendable systems, it is mandatory that the engine has a reuse life in excess of 50 flights.

7.2 REFURBISHMENT TECHNOLOGY TO REDUCE OPERATING COST

Low-refurbishment rate is essential if low-cost/flight goals are to be achieved. To date, the Space Shuttle has not achieved these goals and the contribution is evident in the cost/flight results. Maintainability standards and practices must be designed into the system. Maintainability has been built into aircraft turbofan engines and aircraft airframes design. To date, we have not been able to get adequate maintainability designs into rocket engines and spacecraft design.

7.3 ADVANCED TPS/STRUCTURAL CONCEPTS

The TPS was one of the major development risks in the STS Program. There had been no prior experience of reusable TPS. During the early portion of program definition, several different concepts were examined and while none of them had the desired level of maturity, a system utilizing cold structures with ceramic refractory insulation was chosen as the best approach for the Space Shuttle Orbiter high-temperature areas. This decision allowed the design

of a conventional aluminum structure for the vehicle to proceed in parallel with the development of a suitable insulation system. The development of the chosen insulation material and the attachment concept was a major constraint to the overall development schedule of the Orbiter. In addition, the fragile nature of the existing tiles imposed severe operating limitation on the vehicle, which will continue throughout its life span.

During the development cycle of the Orbiter, improved materials were developed for use in a large portion of the low-temperature areas of the vehicle. These materials, such as flexible batting, have proved to be much easier to install, more reliable, and much less subject to damage in the operational environment. While researchers seek improved concepts for high-temperature areas, no concept has been brought forward which is clearly superior in all the required aspects.

Research and development should be conducted to explore both proposed and new concepts to provide the best and most durable integrated structural TPS for the next-generation transportation system.

7.4 AEROTHERMAL ENVIRONMENT PREDICTION CAPABILITY

The use of computational fluid dynamic (CFD) codes to provide accurate detailed prediction of the aerothermal environment induced during both ascent and descent was in its infancy at the start of the STS Program. During STS development, CFD technology advanced rapidly and the accuracy over a period of time was improved by correlation with wind tunnel and flight data. During the latter portion of the Space Shuttle development, CFD was used to enhance the aerothermal data base and proved to be a very valuable tool.

Further improvements in CFD codes are needed to accommodate some of the more complex configuration details of aerospace vehicles. In view of the cost effectiveness and schedule advantages available, work in CFD should be extended to accommodate as many of the configuration details as possible.

Lack of availability of math model and simulation tools early in the program inhibited or prevented an orderly accomplishment of systems engineering and integration (SE&I) and design tasks. On past programs, this, to a large extent, resulted from a lack of thorough SE&I prior to Phase C/D, forcing the development of at least simplified versions of the needed tools.

In the early portion of Phase C/D, significant efforts should be applied to the development and/or enhancement of the synthesis tools. By requiring participation from all of the contractors, a better understanding of system characteristics will be spread throughout the program.

7.5 CONTROL CONFIGURED VEHICLE TECHNOLOGY

At the start of the Space Shuttle Program, control configured vehicle technology was rapidly emerging from research and development; however, there was very little demonstration of its use in actual flight vehicles. While the Space Shuttle control system was implemented in a way that control configuration technology could be readily applied, it was decided to provide an aerodynamically stable configuration for the ascent atmospheric portion of flight. During the Orbiter development cycle, it was decided to accept an unstable aerodynamic configuration at low speeds for a portion of the center of gravity envelope. The flight tests and operational experience have shown this to be a technically acceptable mode and have eliminated the necessity of adopting significant changes to the Orbiter which would have otherwise been required.

During the same time period, high-performance control aircraft systems proceeded through development and on into operation. The advantages of adopting control configuration concepts to aerospace vehicles with large air speed/altitude envelopes are obviously significant. More analytical test work must be done to assure that viable control concepts and effectors are available for incorporation prior to the start of new vehicle designs.

7.6 LARGE CRYOGENIC STORAGE VESSELS

In any reusable space transportation system, the accommodation of large quantities of cryogen is a major configuration driver. During the STS-concept definition, the decision was made not to integrate fuel tankage into a reusable element of the system. This decision was at least partially based on the difficulty of proving flight worthiness over a life span of many missions. Techniques and actual mechanics involved both quality control inspection techniques which were in or beyond the state of the art of production line usage and pressure testing that endangered the safety of the entire vehicle in the event of a failure.

New materials are available for use in the next-generation vehicle and new concepts may be available which would allow integration of the propellant tankage internal to reusable elements.

New materials and concepts for both tankage and insulation should be pursued in an effort to provide the capability for the next-generation transportation system to take advantage of the cost effectiveness which is accrued by reusing the tank.

One of the goals of the next vehicle development program will be to significantly reduce cost per flight. Achievement of this goal will probably require use of reusable lightweight hardware. Based on the Space Shuttle design, the ascent propellant tank is a prime candidate for further research to develop lighter and less expensive cryogenic propellant tanks. The feasibility of balloon tanks as used on the Atlas and Centaur should be assessed. Other considerations are aluminum tanks with a composite overwrap. One of the major technical reasons for the Space Shuttle tanks being expendable was concern over risk associated with ensuring that the tanks were acceptable

for reuse on the next flight. Achievement of the reduced cost/flight goal will also require lighter weight hardware. Also, emphasis should be placed on identifying candidate metallic and nonmetallic materials for tanks and the materials should be characterized under cyclic propellant loading conditions.

7.7 LONG-DURATION CYRO STORAGE CAPABILITY IN SPACE

The next-generation transportation system may well have requirements to conduct missions of significantly longer duration than the current STS. This capability could be achieved in a more efficient manner if better long-duration cryogenic storage systems could be provided as an energy resource for various vehicle systems (i.e., reaction control systems, life support systems, and electric power generation). This is also a very significant need for the Space Station Program. Work presently contemplated to support the Space Station will, to a large extent, provide the information base necessary for the next-generation transportation system. The work being done for the Space Station should, however, be examined to assure that it adequately covers the requirements and duty cycles anticipated for the transportation system.

7.8 ELECTRO-MECHANICAL ACTUATORS

Goals of the next-generation vehicles will be the reduction of cost/flight and ground turnaround time between flights. Achievements of these goals will require use of lighter weight/smaller volume systems and the reduction of servicing requirements between flights. Although a hydraulic-mechanical actuation system which used a hydrazine-fueled auxiliary power unit for power has performed successfully on the Orbiter, an electro-mechanical system which uses electrical power could be a lighter weight, smaller volume, and lower risk system. It would probably also have the potential to be more compatible with control configured vehicle concepts; therefore, emphasis should be placed on the development of high-torque/fast-response electro-mechanical actuators.

When the current Space Shuttle actuators were selected in the early 1970's, the electro-mechanical actuator technology was not up to the task. Considerable advancement has since occurred in this area of technology with the development of reliable, small, high-torque motors.

7.9 AERODYNAMIC/JET PLUME INTERACTION PREDICTION CAPABILITY

A significant problem that developed during the definition of the STS configuration was the lack of ability to adequately predict the effects of interaction between free-stream aerodynamic flow and rocket exhaust. This became particularly important on the orbital vehicle during early phases of reentry when the reaction control system is used to augment the aerodynamic controlled systems. In some cases, the interaction had large effects on the overall vehicle aerodynamic characteristics and these effects continued to be very hard to predict analytically.

Research and development should be conducted to allow more accurate predictions during the conceptual design of future vehicles in as much as they can be a significant influence on the overall vehicle configuration. It is anticipated that this activity will require effort both in advancing the capability of CFD and correlation with experimental and flight test data.

7.10 OPTIMIZATION OF THE COUPLING OF SPACECRAFT/ASCENT SYSTEMS

Mercury, Gemini, and Apollo were highly decoupled spacecraft/ascent systems while the Space Shuttle was at the opposite extreme with a highly coupled spacecraft/ascent system. Both systems have their advantages and disadvantages. It is obvious that neither approach is optimum for the next-generation system. An indepth analysis to optimize the amount of coupling would be advantageous.

The Orbiter contained all crew safety, orbital, and entry systems that the Mercury, Gemini, and Apollo vehicles contained in addition to ascent guidance and control, launch element power, launch element telemetry and command ascent liquid propulsion, and a highly integrated environment and structural attachment system.

Once on orbit, all Orbiter weight-bearing ascent support hardware is a burden which limits the orbital, entry, and landing capability envelope. This is expensive in propellant requirements, which is equivalent to payload. The Space Shuttle approach makes the task of upgrading or modifying either the ascent system or the spacecraft system highly difficult, if not impossible. The Apollo command module could be launched on either a Saturn IB or Saturn V and with or without a lunar module. What is the optimum approach considering economics, mission capability, growth capability, and flight safety?

A reassessment, since we now have experience with both ends of this spectrum, appears to be in order.

SECTION 8
REFERENCES

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REFERENCES

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SECTION 9
ACRONYMS

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ACRONYMS

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| AADS | ascent air data system |
| ABES | airbreathing engine system |
| APU | auxiliary power unit |
| ASRM | abort solid rocket motor |
| ATP | authority to proceed |
| BIS | British Interplanetary Society |
| BSM | booster separation motor |
| CDR | Critical Design Review |
| CFD | computational fluid dynamic |
| c.g. | center of gravity |
| DDT&E | design, development, test, and evaluation |
| DMS | data management system |
| DOD | Department of Defense |
| E&I | engineering and integration |
| ECS | environmental control system |
| EMI | electromagnetic interference |
| EPL | emergency power level |
| ET | external tank |
| FDAI | fault detection, annunciation, and identification |
| FDIR | fault detection, identification, and reconfiguration |
| FPL | full power level |
| FRSI | felt reusable surface insulation |
| FWC | filament-wound case |
| GET | ground elapsed time |
| GG | gas generator |
| GLOW | ground lift-off weight |
| GN&C | guidance, navigation, and control |
| GPS | global positioning system |

| | |
|-----------------|---|
| HPM | high-performance motor |
| HPFTP | high-pressure fuel turbopump |
| HPOTP | high-pressure oxygen turbopump |
| HRSI | high-temperature, reusable surface insulation |
| IMU | inertial measurement unit |
| ILRV | integrated launch and reentry vehicle |
| ISP | initial specific impulse |
| JSC | NASA Lyndon B. Johnson Space Center |
| KSC | NASA John F. Kennedy Space Center |
| LaRC | Langley Research Center |
| L/D | lift to drag |
| LH ₂ | liquid hydrogen |
| LM | lunar module |
| LO ₂ | liquid oxygen |
| LPS | launch processing system |
| LRSI | low-temperature, reusable surface insulation |
| LV | launch vehicle |
| LWC | lightweight case |
| MCR | Master Change Record |
| MDM | multiplexer/demultiplexer |
| MECO | main engine cutoff |
| MEOP | maximum expected operating pressure |
| MIA | multiplexer interface adapter |
| MMC | Martin Marietta Corporation |
| MPAD | Mission Planning and Analysis Division |
| MSBLS | microwave scanning beam landing system |
| MSC | Manned Spacecraft Center |
| MSFC | NASA George C. Marshall Space Flight Center |
| NPSH | net positive suction head |
| NS1-1 | NASA standard initiator - type 1 |
| NSTS | National Space Transportation System |

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|-------|--|
| OD | outside diameter |
| O/F | oxidizer to fuel |
| OMI | operational maintenance instruction |
| OMS | orbital maneuvering system |
| OMSF | Office of Manned Space Flight |
| OPF | Orbiter Processing Facility |
| OV | Orbiter vehicle |
| PBAN | polybutadiene acrylonitrile |
| Pc | chamber pressure |
| Pd | pump discharge |
| PDR | Program Design Review |
| PFB | pressure-fed booster |
| PLBD | payload bay door |
| PRCBD | Program Requirements Control Board Directive |
| PRR | Program Requirements Review |
| PSAC | President's Science Advisory Committee |
| RCC | reinforced carbon-carbon |
| RCS | reaction control system |
| R&D | research and development |
| RFP | request for proposal |
| RPL | rated power level |
| RSC | range safety command/control |
| RSI | reusable surface insulation |
| RTLS | return-to-launch site |
| SAMSO | Space and Missile Systems Organization |
| SE&I | systems engineering and integration |
| SIR | Systems Integration Review |
| SL | sea level |
| SOFI | spray-on foam insulation |
| SOW | statement of work |
| SRB | solid rocket booster |
| SRM | solid rocket motor |
| SRR | System Requirements Review |
| SSME | Space Shuttle main engine |
| SSPO | Space Shuttle Program Office |
| STS | Space Transportation System |

TACAN
T.E.
TPS
TVC

tactical air navigation
trailing edge
thermal protection system
thrust vector control

VAB
VLS

Vehicle Assembly Building
Vandenberg Launch and Landing Site