MATHEMATICA

ECONOMIC ANALYSIS OF THE SPACE SHUTTLE SYSTEM

Prepared for

National Aeronautics and Space Administration

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CHAPTER 4.0

SPACE MISSIONS AND PAYLOADS, 1979-1990

In this chapter man's space activities are first reviewed historically beginning in 1957 (although a period of interesting stirrings and essential accomplishments preceded that year); next space activities of the several nations during the 1970's as seen from the present are discussed; then an effort is made to estimate the prospects for international cooperation in space during the 1980's and beyond. Man's activities in space through the 1980's in terms of the various missions, payloads and traffic schedules are delineated as provided by the NASA and the DoD for the purposes of this study, and by the study contractors, and others up to the time of this report. MATHEMATICA has taken the liberty of extending the activities and discussing possible changes in emphasis in several areas. The scenarios, i.e., variations on the mission model, that were costed represent reasonable alternatives as seen at the present time.

4.1 Introduction

4.1.1 Historical Review of Space Activity

The United State Space Program as we have experienced it over the past ten years did not really exist until our society was challenged by the Soviet Union with Sputnik I on 4 October 1957 and we failed to orbit a grapefruit sized, six kilogram satellite with Vanguard TV-3 later in the same year. A manned landing on the moon and safe return of the astronauts to Earth was declared to be the nation's goal in space for the 1960's by President Kennedy in the spring of 1961. This was widely approved by the American people and supported by the U. S. Congress.

Explorer I, the United States' first satellite, was successfully orbited on 31 January 1958 and since then almost 200 civil spacecraft and over 400 military payloads have been placed in orbit by the United States with an overall success rate of about 75% and a total expenditure of approximately $50 billion.

The Soviet program has continued actively and many notable "firsts" have been recorded. In addition to the first successful artificial satellite of Earth, the Russians orbited the first animal, the dog Laika, and the first man, Yuri Gagarin. They were also first to hit the moon, photograph its backside and soft-land a payload on it. They have maintained a strong planetary program and their Cosmos Series of military and other satellites have exceeded 450 in number. Their program has also included weather and communications satellites as well as manned proto-space stations.

France, England, Canada, Italy, Japan and the People's Republic of China have successfully orbited spacecraft and the United Nations has shown a significant interest in outer space affairs.

During the first decade the United States Space Program was the most strikingly successful of any country, but it was also by far the most expensive. Its impact on the people of the United States was substantial but its remarkable achievements were offset by domestic difficulties and the war in Southeast Asia; however, in the world community it has been a great success and eagerly followed by people everywhere. The most impressive accomplishment was the attainment of two Apollo manned lunar landings and safe returns within the decade of the 1960's and within the estimated cost of $25 billion; however, it has not been possible to sustain the established level of overall space activity or resolve acceptable new goals as of the present time. The Apollo Lunar Module shown in Figure 4.1 characterizes the United States Space Program of the 1960's [1].

4.1.2 Space Activity in the 1970's

United States space activities in the first half of the 1970's are clearly seen and necessarily already programmed by the DoD and NASA. Manned activities feature continuation of the Apollo Program through 1972 and Skylab A, a precursor space station, in 1973. The unmanned planetary program includes: a Mariner Mars orbiter in 1971, a Venus swingby to Mercury in 1973, Pioneer flybys of Jupiter in 1973 and 1974 (to be launched in 1972 and 1973), and the Viking soft landings on Mars in 1975. Considerable

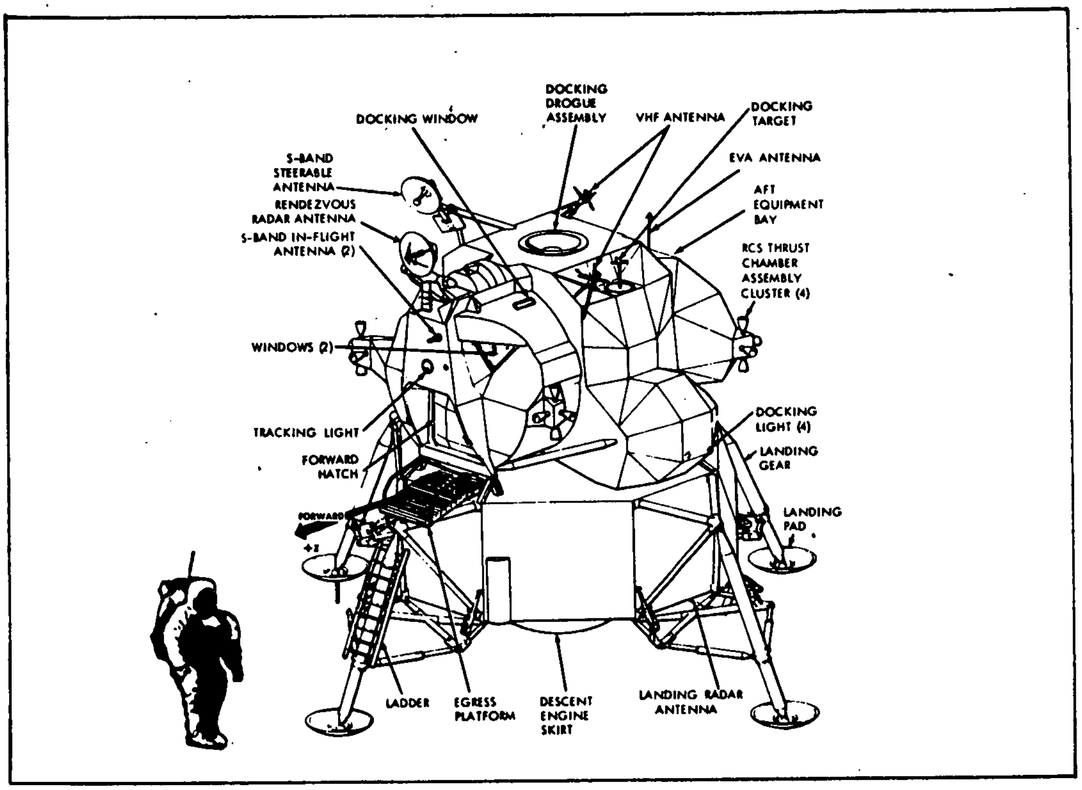


Figure 4.1 Apollo Lunar Module
Reference [1]

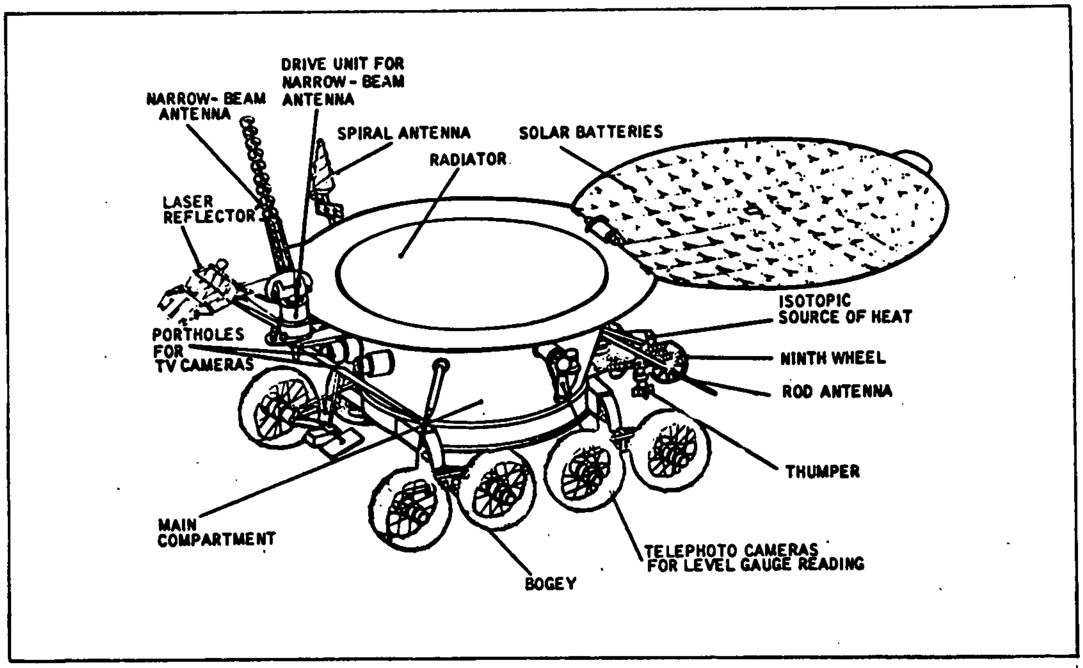


Figure 4.2 Lunakhod
Reference [2]

activity in near Earth orbit is programmed for the 1970's in both space science and applications: astronomy with orbiting astronomical observatories will be continued and earth resources test satellites will be launched in 1972 and 1973; weather satellites and communications satellites will be used ever more extensively; navigation satellites will find widespread use for both sea and air travel; and military uses will continue at near the present level of activity.

The Soviet Space Program is expected to challenge us anew, especially in the area of manned space stations and planetary exploration. Their very successful unmanned lunar rover, Lunakhod, shown above in Figure 4.2 [2] characterizes the space vehicles of the 1970's. This automated spacecraft survived six months of lunar days and nights and was directed by commands from Earth for over five miles across the lunar landscape. It was subsequently retired to "reduced activity". The deliberate Soviet pace and increasing success provide a clear challenge that cannot be ignored. On the other hand, it is to be hoped that increased cooperation between the two leading space-faring nations will make it possible in the near future to assure success and reduce costs of many missions and that efforts to bring the benefits of space to the entire Earth, with the participation of other countries and through the United Nations, will prosper. The development of the Space Shuttle System can be an important step in bringing this about.

In the second half of the 1970's, although the scene is clearing in various mission areas, much uncertainty is evident and new planning is needed in view of changing circumstances; however, some missions can be seen more or less definitely. There is a considerable agreement that a continuing and "balanced" space program is desirable (although not everyone agrees) but at any rate its balance is subject to much difference of opinion [3].

Specific planning relating to space stations and other manned flight in Earth orbit in the last half of the 1970's is indefinite at the present time. Continuation of the unmanned planetary program can be expected with hope for increased activity dependent on findings and Soviet competition.

Space science will continue and strengthen, especially astronomy from Earth orbit. Space applications activity will be led by the communcations satellites, especially from synchronous orbit including the direct broadcast of television -- voice and data. Navigation satellite systems will be used extensively and observation of the Earth's surface for weather, resources -- agricultural and mineral, mapping and other purposes will become widespread.

4.1.3 One World in Space

It is to be hoped that the international initiatives currently being pursued will prosper and that during the 1980's man's space activities will be coordinated or carried on, outside the military sphere, in a spirit of cooperation and of equable benefit to all the peoples of the Earth. It is hoped that the present Soviet-U.S. efforts to cooperate will develop successfully [4] and be sustained, and that the other nations will be included as they may be interested. The developing countries are already being introduced to the benefits of space through the efforts of the United Nations [5].

In the 1980's science in space, including astronomy, physics and solar system explorations, should provide important and exciting results and contribute to a great surge in Man's understanding of the Universe and his place in it. Most importantly space applications, including various manned stations, should contribute substantially to life on Earth. Commercial enterprise will find many opportunities that will contribute to space activity in the 1980's.

4.2 United States Space Missions 1979-1990

Although the U.S. Space Program in the period 1979-1990 will necessarily evolve from the spacecraft and launch vehicles employed throughout the 1970's [6], the prospect of a new Space Transportation System, the maturing of various space technologies and the cumulative results of space activities can give sufficient reason to take a broader and more extensive view of space in the 1980's. Although the mission model provided by NASA is presented here, a broader and deeper view may more surely identify the directions and activities of the present decade and project into the 1980's a better understanding of the worth of space activities as a continuing program of vital interest to the Nation.

For purposes of the economic study so far undertaken, however, a conservative view of both the space program and traffic level has been taken by MATHEMATICA using scenarios with numbers of flights between 300 and 900 based on the NASA "Fleming" Mission Model of Spring 1971 as presented by the Aerospace Corporation in References [7] and [8].

4.2.1 Department of Defense Missions

The Department of Defense (DoD) is cooperating in Space Shuttle planning and has offered several mission options. In addition to formal committee activity highly placed Air Force officials have publicly stated that they will use the Space Shuttle if it is developed and has the required characteristics. The Space Shuttle Main Engine (SSME) program is generated from earlier USAF research; re-entry technology is supported by the USAF (PRIME and ASSET Programs). DoD spending equalled $4 million in Fiscal Year 1972 to study uses and operational characteristics of the Shuttle. However, a commitment to provide financial support during the development and early operational phases and to undertake accommodation of payload design for low cost and other effects within the expected DoD traffic projection has not been publicly announced.

One of the major difficulties in considering DoD missions is security classification, but aside from that a more definite understanding of operational and other problems in integrating spacecraft and system design and development is needed if the overall most effective and economical space capability is to be realized for the United States in the 1980's.

DoD missions are assigned primarily to the Air Force but the Navy and Army have programs of particular concern; they include surveillance and other reconnaissance, warning, communications -- strategic and tactical, navigation, and other special space activities in addition to weapons. There is at present no manned DoD space activity. The termination of the Manned Orbital Laboratory (MOL) appears to be a constraining factor in DoD consideration of manned flight for the future. The DoD space budgets that have appeared in published sources during recent years range between $1 2/3 and $2 billion per year [9] and will probably be continued at the same level through the 1970's and into the 1980's.

Quite aside from ballistic missiles and similar weapons, a number of DoD missions in the1980's will surely require quick reation times or orbits that demand expendable launch vehicles. Such special requirements may result in a Space Transportation System that is a mix of reusable and expendable vehicles.

It would appear that a more certain understanding and more extensive coordination are needed to establish the details of the DoD involvement in the use of any new Space Transportation System.

4.2.2 National Aeronautics and Space Administration Missions

The National Aeronautics and Space Administration (NASA) missions in the 1980's are discussed in this section under the following categories:

Space Science
Space Applications
Solar System Exploration
Space Shuttle Sortie
Space Station
Lunar

MATHEMATICA's primary sources of mission and payload information are References [7] and [8] although other sources, such as National Academy of Sciences Space Science Board publications (e.g., [10]), and individuals have been consulted.

4.2.2.1 Space Science

Space science missions are generally of two categories -- space astronomy and space physics. Space astronomy, freed from the atmosphere and other disturbances over the full band of electromagnetic radiation, is expected to yield discoveries and understandings in both the Solar System and the Universe beyond that will advance man's knowledge to a major extent. A fully developed and extensive program has been proposed and high hopes are held for its realization.

The missions identified as Space Physics are related to man's determination of the physical phenomena in space near the Earth and elsewhere in the solar system especially those emanating from the Sun. Basic physical theories such as General Relativity, that can be studied in space, are also of interest.

4.2.2.2 Space Applications

NASA space applications activity encompasses primarily Earth observation and communications and navigation missions. The Earth observation activity includes continuing research and development of meteorological satellites as well as newer forms of earth observation interest such as mapping, resources -- hydrological, plant, fish, and mineral, crop disease, pollution, etc. Communication satellites are well established world-wide but NASA will probably continue to handle further development of these capabilities and the operation of NASA peculiar systems. Navigation (and traffic control) satellites have also found operational usefulness and further development of these capabilities is being pursued with a much broadened applicability to surface vessels, aircraft, and even land transport.

4.2.2.3 Solar System Exploration

The area of solar system exploration will become of major importance in the future space program, but it is essentially related to the available Space Transportation Systems because of the high energy requirements of the missions. The extent of the solar system and the bodies and phenomena to be explored are discussed in References [10] and [11] and are shown on Figure 4.3.

The NASA mission model used in the present study and shown on Figure 4.4 recognizes the accomplishments to date and continues the program to include all the planets except Pluto and also includes the asteroids and comets, but is, in fact, not well conceived for the 1980's.

It is reasonable to expect that many interesting things will be learned from solar system exploration in the 1970's that will strengthen this area of the space program in the 1980's. In particular, solar system exploration has much to gain from both the advent of the shuttle or other new Space

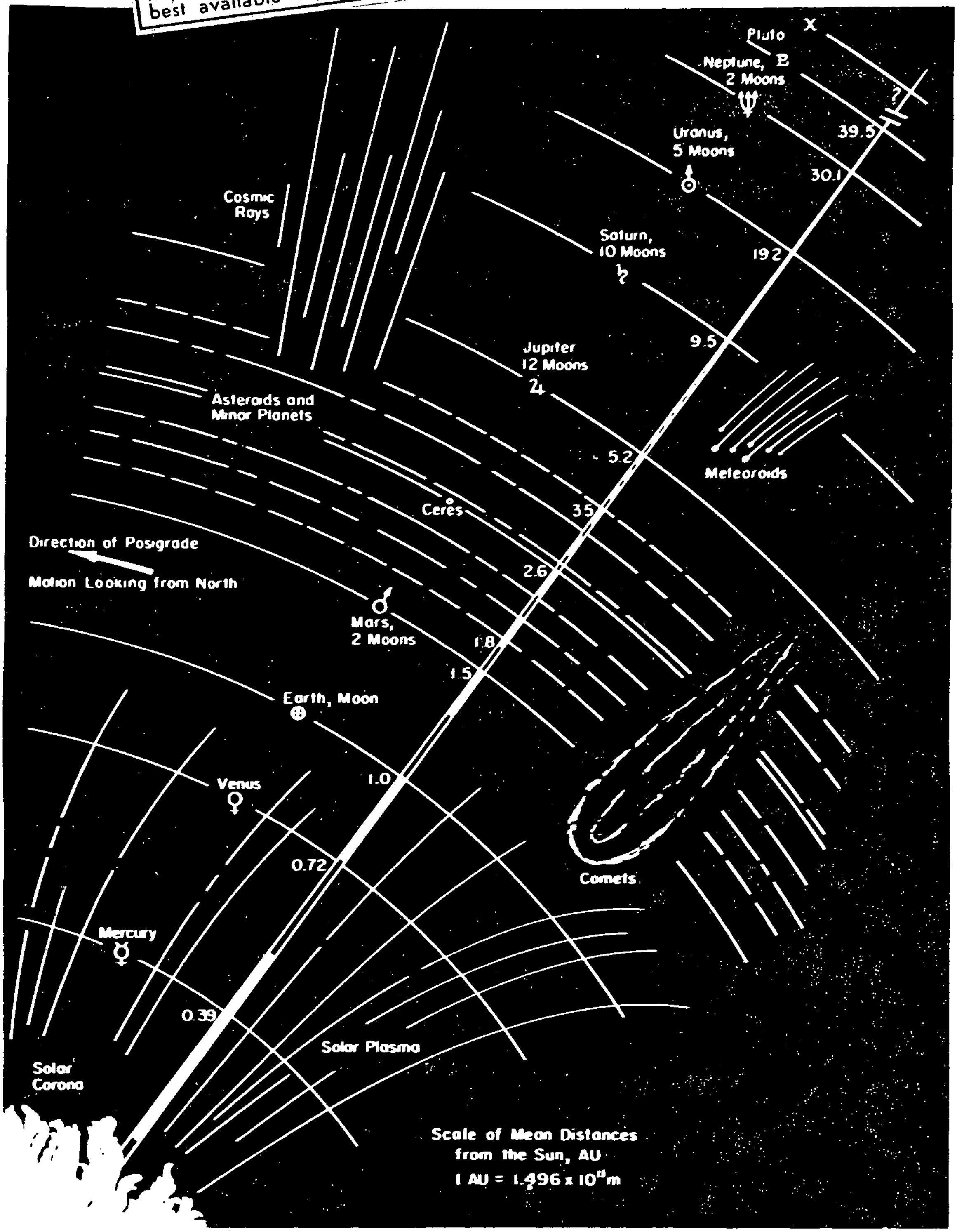


Figure 4.3 Bodies and Phenomena of the Solar System
Reference [10]

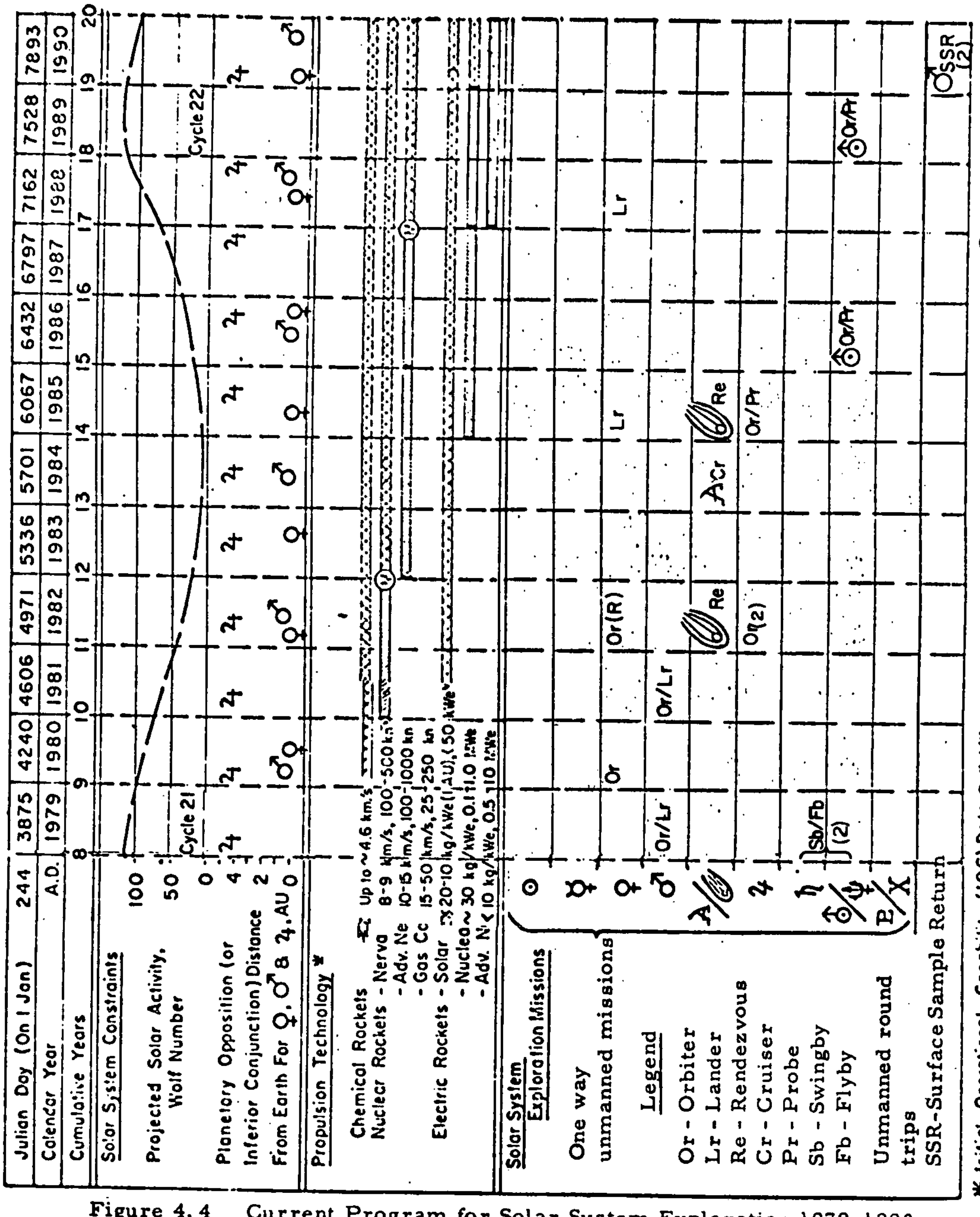


Figure 4.4 Current Program for Solar System Exploration 1979-1990
Reference [8]

Transportation System capabilities and advanced space propulsion and other technology so that the entire complexion of the missions and the overall program in this area will be greatly strengthened.

A well developed and integrated program of solar system exploration during the 1980's that recognizes fully the possibilities of exploring the bodies and phenomena of the solar system is considered to be an essential part of the United States Space Program during that period. A program is foreseen that extends the present planning substantially and exploits the kinds of missions that the Space Shuttle and other advanced systems (especially using nuclear propulsion) can provide beyond the present mission model. Such a program would include more substantial attention to the major planets and their satellites using orbiters and landers, fast trips to the outermost planets, and a number of atmospheric and surface sample return missions.

4.2.2.4 Space Shuttle Sortie

The Space Shuttle Orbiter can operate in orbit in a "sortie" mode for periods up to about two weeks. This relieves the necessity for carrying the Space Station development in parallel and permits development of modules with certain capabilities while attached to the orbiter (although they may be rotated out of the payload bay) that will contribute directly to the later establishment of the Space Station and its activities. Both manned experiment and pallet-type sortie modules have been identified as described in Reference [12] and presented in more detail below.

4.2.2.5 Space Station Missions

One of the major regions for man's activities in space during the 1980's will be Earth orbit around 500 km altitude where manned-space operations with spacecraft and stations of several possible configurations will permit the performance of a wide variety of missions. Space Station configurations and systems deriving from Skylab A and also from new technology have been studied in considerable detail under the Apollo Applications Program (AAP), Manned Orbiting Research Laboratory (MORL) and other studies, and at the present time, the Shuttle Orbital Applications and Requirements

(SOAR) and Research and Application Modules (RAM's) studies. It is believed that this will be an important and active area in the Space Program of the United States and the Soviet Union and involve other nations as well.

A particularly interesting concept for the 1980's is the modular Space Station described in Reference [13] and shown on Figure 4.5. The station would be assembled from essentially self-sufficient modules and would be capable of a very wide variety of missions [14]. All of the modules and other parts of such a station would be carried into orbit in the payload bay of the Space Shuttle Orbiter and following accomplishment of their missions would be returned to the Earth's surface in the same way.

The Space Station will be assembled over a period of time with increasing capability and crew capacity from 3 to 6, 12 and eventually 24 or more persons. Vacations in Earth orbit probably lie beyond the 1980's.

In addition to near Earth missions, the Space Station can serve in the 1980's as an orbital terminal for spacecraft returning from interplanetary space, especially the unmanned-sample-return missions which will probably be an important feature of planetary activity in the 1980's. Samples can be quarantined in Earth orbit and given preliminary analysis before they enter the atmosphere.

4.2.2.6 Lunar

Although no lunar missions are included in the present mission models and it appears untimely to "sell" them, it is very likely that the 1980's will see both unmanned and manned activity on the moon. Whether this will be carried out on a competitive or a cooperative basis, a la Antartica, remains to be seen. It will probably not be studied seriously until after the results of the Apollo Program are digested.

Aside from further explorations one of the most attractive uses of the moon would seem to be a very large radio telescope on its backside with

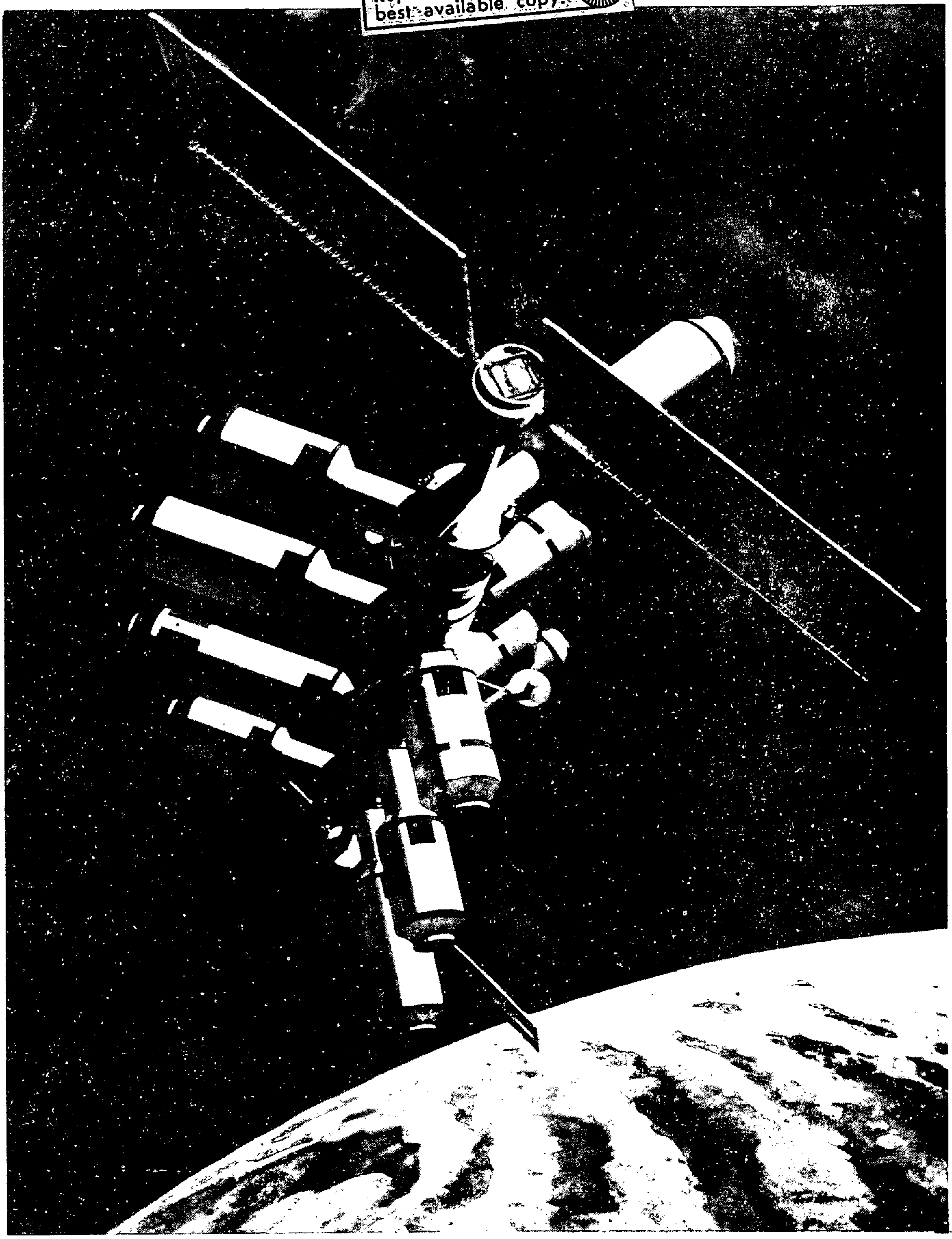


Figure 4.5 Modular Space Station Concept for the 1980's
Reference [13]

communications maintained by a lunar satellite in a "halo" orbit at the farther quasi-stable Lagrangian point. Other important uses may be identified.

4.2.3 Non-NASA Operational Missions

A considerable increase in United States space activity in the 1980's outside of NASA and the Defense Department is foreseen whether the Space Shuttle is developed or not. This activity will come from other government agencies who have been identifying uses of space that will be proven during the 1970's,and from commercial enterprises -- privately and governmentally sponsored or supported. A number of payloads and a considerable number of missions are identified in this category in the present mission model as described below.

4.2.3.1 Governmental Agencies

A number of departments of the Federal government, in particular, the Departments of Commerce, the Interior, Agriculture, and Housing and Urban Development, and various other agencies are actively pursuing a definition of their activities in space in the 1970's and projecting them into the 1980's. For example, the Coast and Geodetic Survey, Department of the Interior has produced strikingly detailed and accurate maps from Apollo photography of the Southwest United States and are projecting mapping of the entire Country from space which will represent the first up-to-date, complete mapping at reasonable cost that has ever been possible.

Meteorological measurements -- not only of clouds and cloud layers, but actual and accurate ambient temperatures, humidity, wind velocities, etc., at various layers in the atmosphere -- will be made routinely around the world and other Earth observations of interest to various agencies will be made to determine new information about the Earth we live on -- resources, especially water, plant, animal, including fish, and mineral, population, pollution, etc. Special measurements such as snowfall, locations of icebergs, sources of thermal energy, etc., can be made. Considerable governmental activity from state and local levels to national, regional and worldwide agencies can be seen.

4.2.3.2 Commercial

The communications satellites are currently the greatest commercial success in space and their usefulness and the variety of U. S. and foreign domestic applications both privately supported and government sponsored are expected to increase greatly during the 1970's and to be exploited during the 1980's. The Communications Satellite Corporation (COMSAT) will continue to play a significant role but competition within the United States and from foreign nations will increase very significantly.

4.2.4 New Missions

Any new Space Transportation System will offer advantages that combined with the ongoing technological and social scene will spawn new missions. The revisit and reuse possibilities of the Space Shuttle offer changed circumstances for prospective missions. One novel but probably feasible mission that has not been analyzed considering the new advantages and changed circumstances is a large solar boiler power system for Space Station supply with local transmission of power to neighboring spacecraft by microwave or laser with superimposed control signals or to the Earth's surface to meet special needs.

4.3 Foreign Missions

Space missions by foreign nations are expected to represent an increasingly important activity in space throughout the 1970's and particularly in the 1980's as space technology and applications mature and the costs of payloads and transportation into space reduce. Retrieval, refurbishment and reuse as offered by the Space Shuttle should emphasize this trend if it offers realizable economies and is made available.

4.3.1 European Missions

The prospect of coordinating space activities with the European community in the 1980's or sooner is of considerable interest and importance to the United States. One aspect of European involvement involves the Space Tug and other aspects include cooperative and competitive payloads on missions of considerable variety as described in Reference [15].

4.3.2 Soviet Missions

The Soviet Space Program has continued to strengthen (in every year except 1969) while the United States Program has declined since 1966 in level of activity [15]. The relative success of the Soviet Program has improved although they are known -- and in some measure have admitted publicly -- to have suffered major reverses. Hope is held high for increased cooperation in space activity between the United States and the Soviet Union and some arrangements, particularly in manned space operations, including rescue, have already been made. Some missions will remain competitive but a general willingness to cooperate, bilaterally or through the United Nations, in space activities, including military, has been evident in recent years and hopefully will be strengthened.

4.3.3 Other Foreign Missions

In the arena of space, other foreign nations are being seen more frequently and the activity of both developed and developing countries will surely show remarkable increases in the 1980's. Canada, Japan and the People's Republic of China have already orbited satellites while the developing countries have provided spacecraft or taken a role in launches made by the space powers. As the utility of space becomes increasingly apparent both of these kinds of activity can be expected to find further exercise so that by the 1980's the entire world community will be truly involved in man's space endeavors and adventures.

4.4 United States Payloads 1979-1990

In this section, United States' payloads of a new Space Transportation System in the 1979-1990 period are: first, described according to their general kinds; then, the types identified from payload effects analysis; next, their reliability, retrieval, refurbishment/updating and reuse possibilities; and last, the individual payloads that constitute the current mission model given to MATHEMATICA for this economic analysis.

4.4.1 Kinds of Payloads

4.4.1.1 Unmanned Spacecraft

By far the largest number of space payloads in prospect for the 1979-1990 period are unmanned spacecraft. Although these spacecraft will derive from those flown in the 1960's and 1970's several new directions are evident. They will be larger and more complex as typified by the Large Stellar Telescope shown in Figure 4.6. This major spacecraft will be completely automatic in its function, while being operated from a ground station and largely unattended during its operating lifetime although retrieval and refurbishment will be provided for. It will fill the very important function of probing the deepest reaches of space and is fully described in Reference [16].

The Synchronous Equatorial Orbiter shown in Figure 4.7 is of Lockheed Missiles and Space Company low cost design and typifies a number of spacecraft that will be used for Earth applications missions. It represents a step in the standardization and modularization of classes of spacecraft to obtain initial low development and production costs while retaining the benefits of refurbishment and reuse. It is described in considerable detail in Reference [17].

4.4.1.2 Space Tugs and Teleoperators

Although in orbit the Space Tugs and Teleoperators may be considered as propulsion stages and auxilliaries for the Space Transportation System, they are as much payloads as the spacecraft that require their services in the performance of a mission. A typical mission configuration involving a Space Tug with a payload and teleoperator in orbit adjacent to a Space Shuttle Orbiter is shown in Figure 4.8. Space Tugs and Teleoperators are described in more detail in Chapter 5.

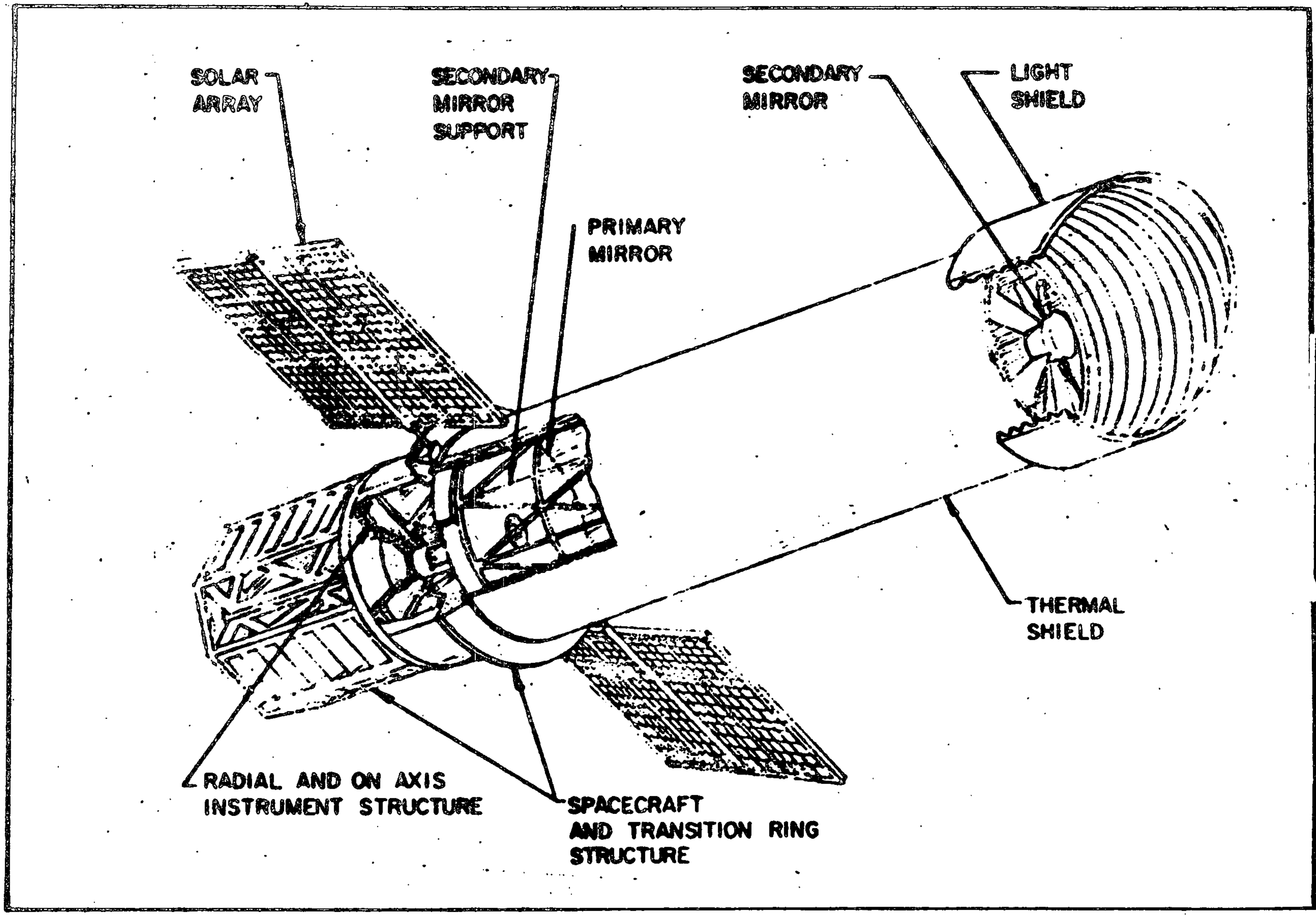


Figure 4.6 Large Stellar Telescope
Reference [16]

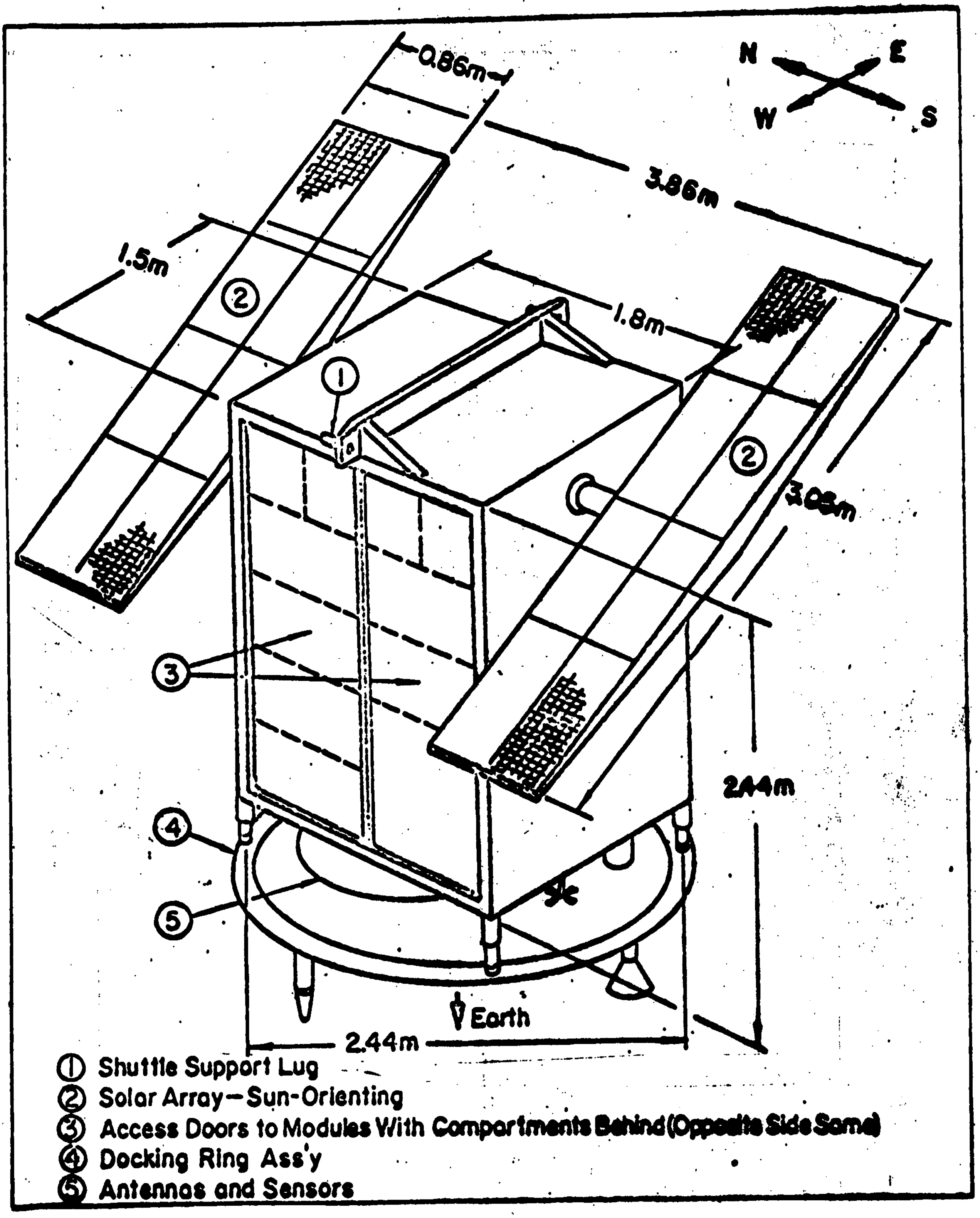


Figure 4.7 Synchronous Equatorial Orbiter—Lockheed Low Cost Design Reference [17]

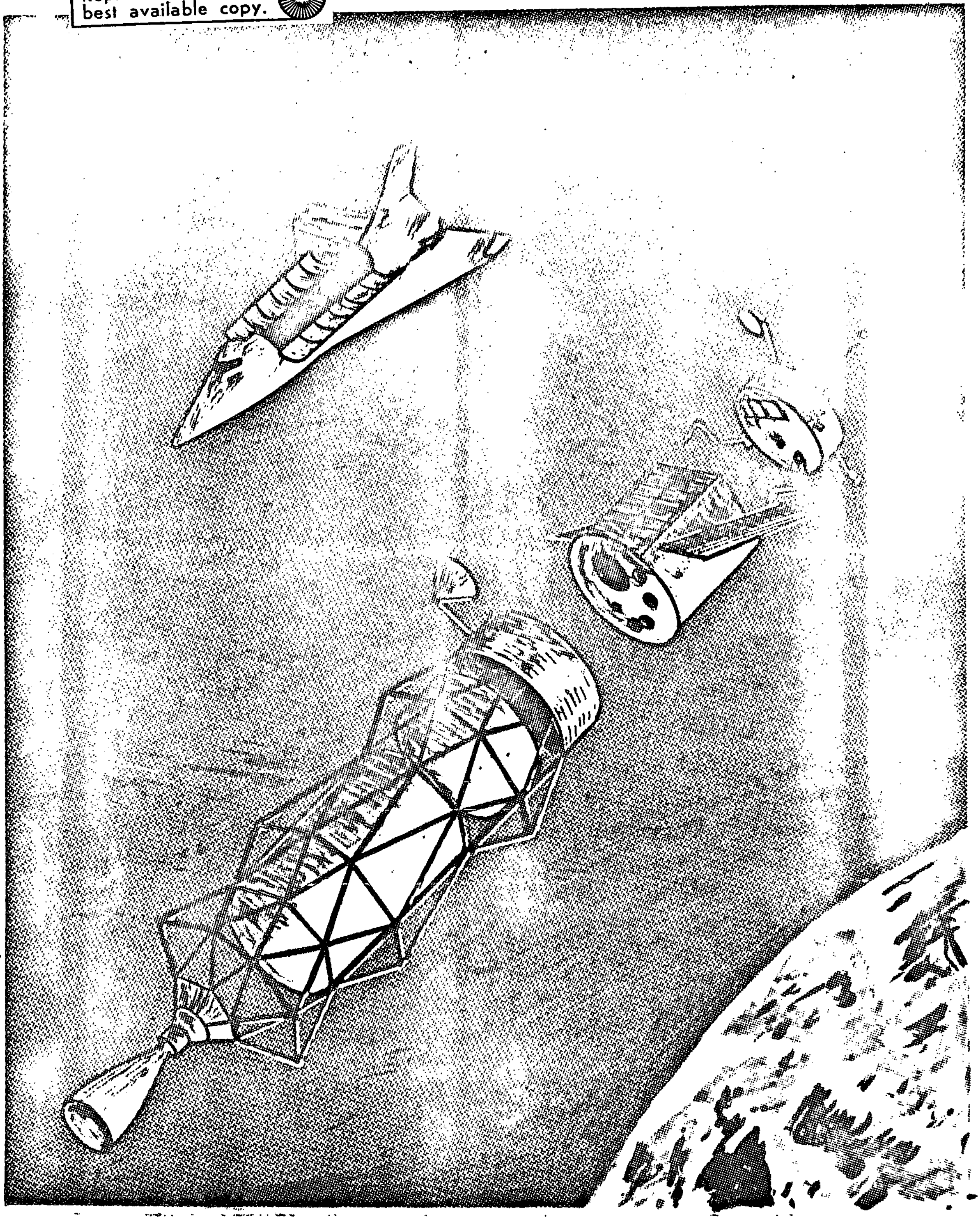


Figure 4.8 Space Tug with Payload and Teleoperator in Orbit Adjacent to Space Shuttle Orbiter

4.4.1.3 Nuclear Vehicles

One of the important kinds of payloads in the 1980's for a new Space Transportation System is the nuclear vehicles that will be used for high energy and deep space missions. A typical modular nuclear rocket propelled space vehicle configured for a major solar system exploration mission is shown in Figure 4.9. The propulsion and propellant tank modules are each configured to be carried in the Space Shuttle Orbiter payload bay. A number of modular nuclear rocket vehicle configurations are shown in Figure 4.10 and their performance with various payload masses is given graphically in Figure 4.11.

The integration of two nuclear-electric rocket propelled spacecraft with two typical power levels into the payload bay of the Space Shuttle Orbiter is presented in Figure 4.12.

4.4.1.4 Shuttle Sortie and Space Station Modules

Important kinds of payload configured for the Space Shuttle Orbiter are the sortie modules and the space station modules. The Manned Experimental Module delineated in Figure 4.13 is significant because it permits use of the Space Shuttle Orbiter in the sortie mode. This and other modules will be employed for up to two weeks in orbit attached to the Space Shuttle Orbiter but with a capability for being rotated out of the payload bay as shown in the figure.

The considerable variety of space station modules that can be derived from three common modules to perform a number of missions in presented in Figure 4.14. These modules will be attached to the space station core modules as shown above in Figure 4.5 or may be detached to function in orbit near the station. Details of the various modules are given in Section 4.4.5.7 below. They will all fit the Space Shuttle Orbiter payload bay.

4.4.2 Types of Payloads

Two primary types of payloads have been identified by the Lockheed Missiles and Space Company Payload Effects Analysis [20, 21]. Although

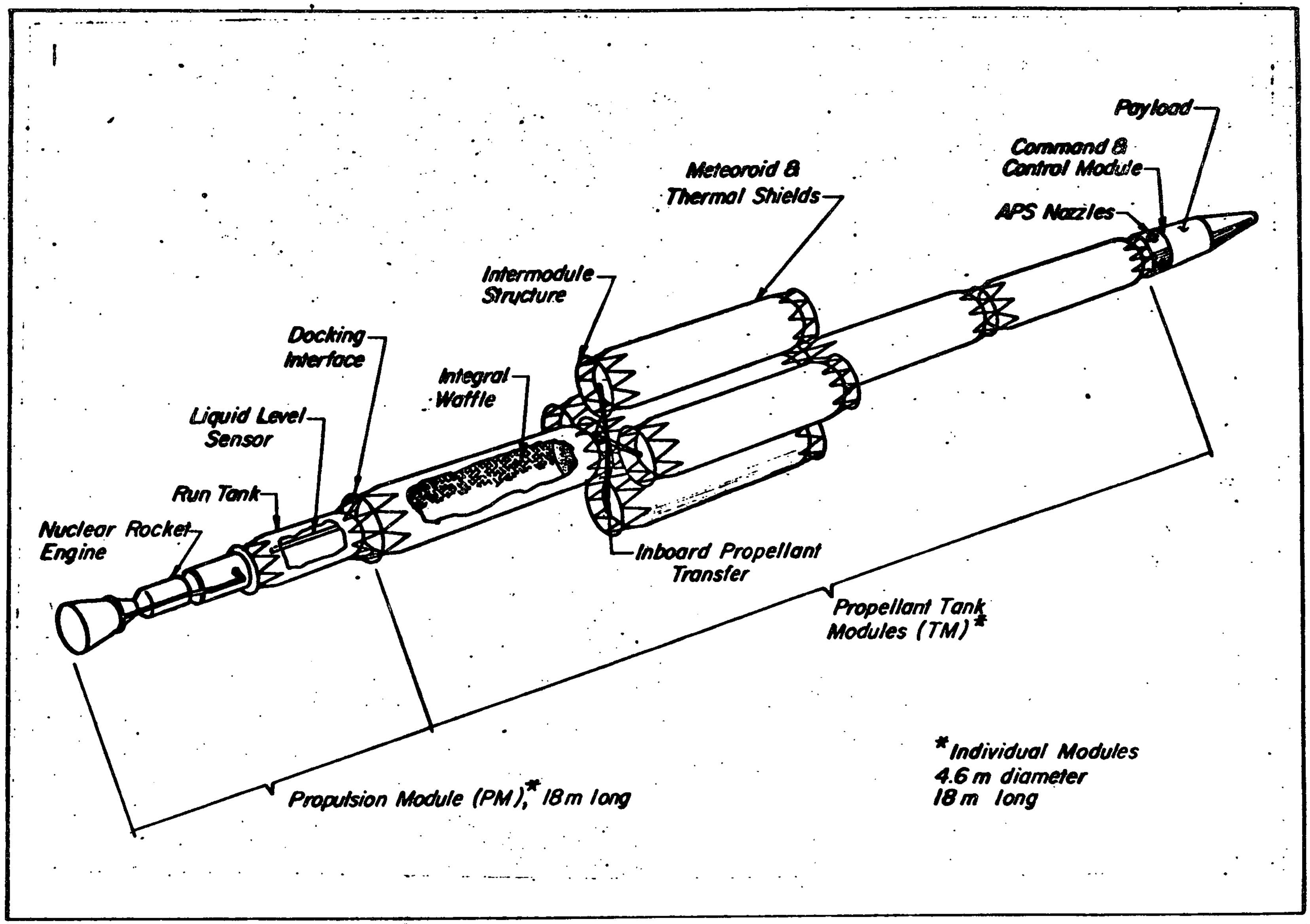


Figure 4.9 Modular Nuclear Rocket Propelled Space Vehicle Reference [18]

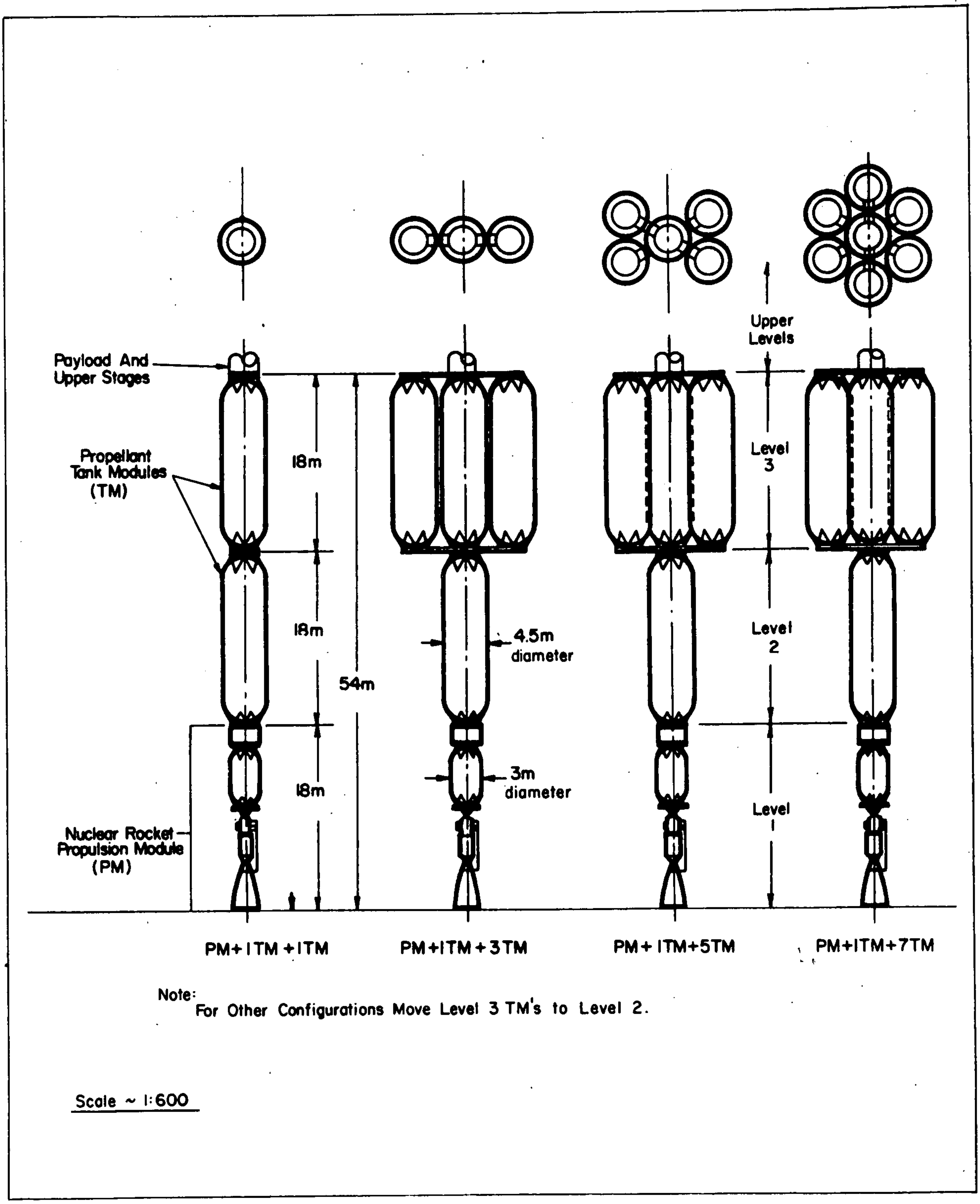


Figure 4.10 Nuclear Rocket Vehicle Configurations
Reference [18]

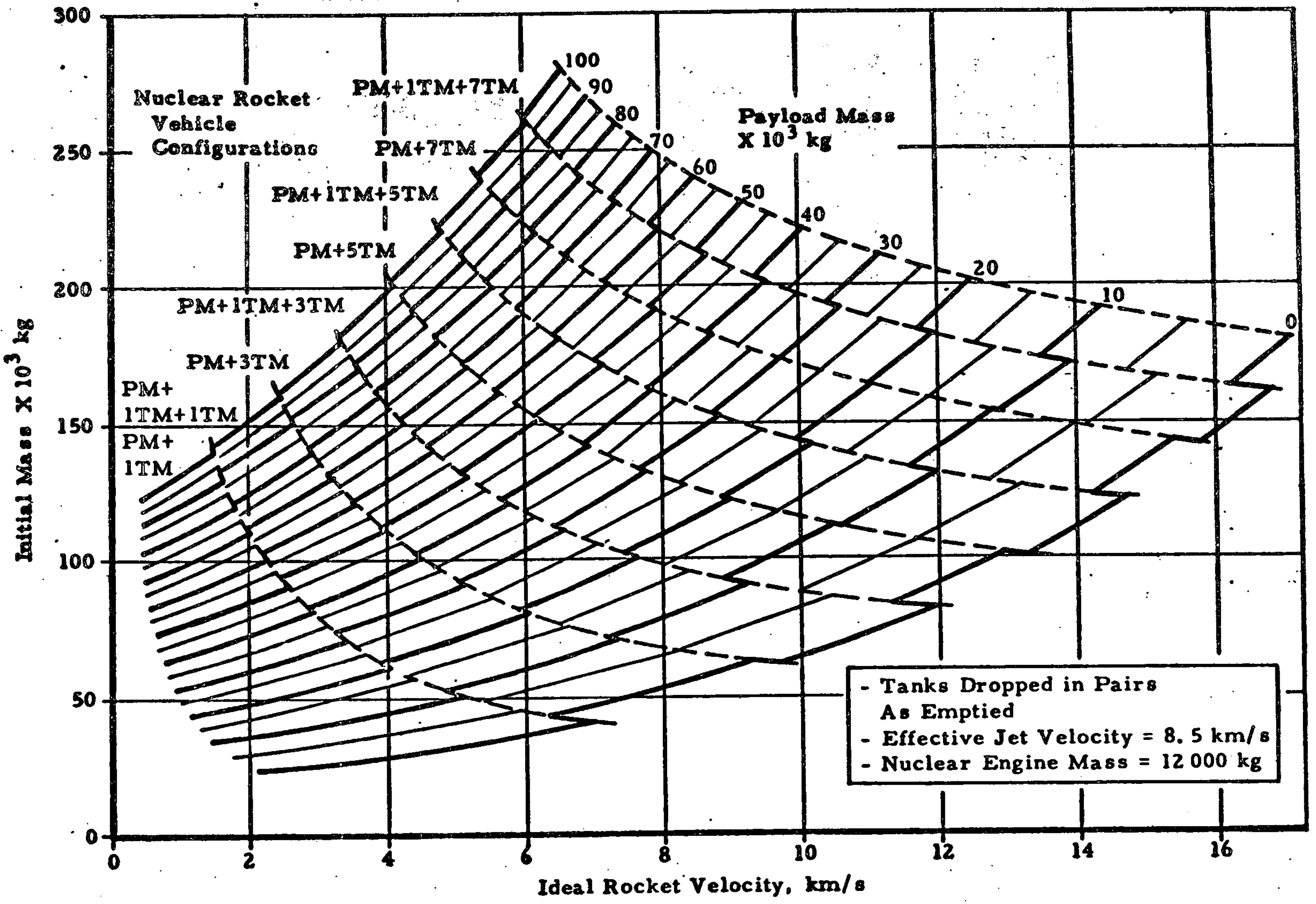
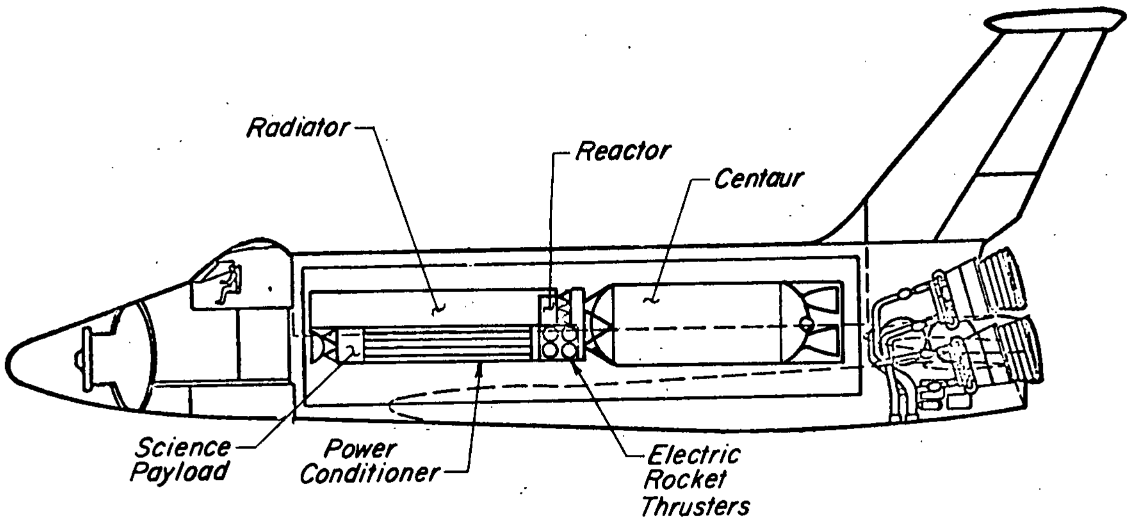
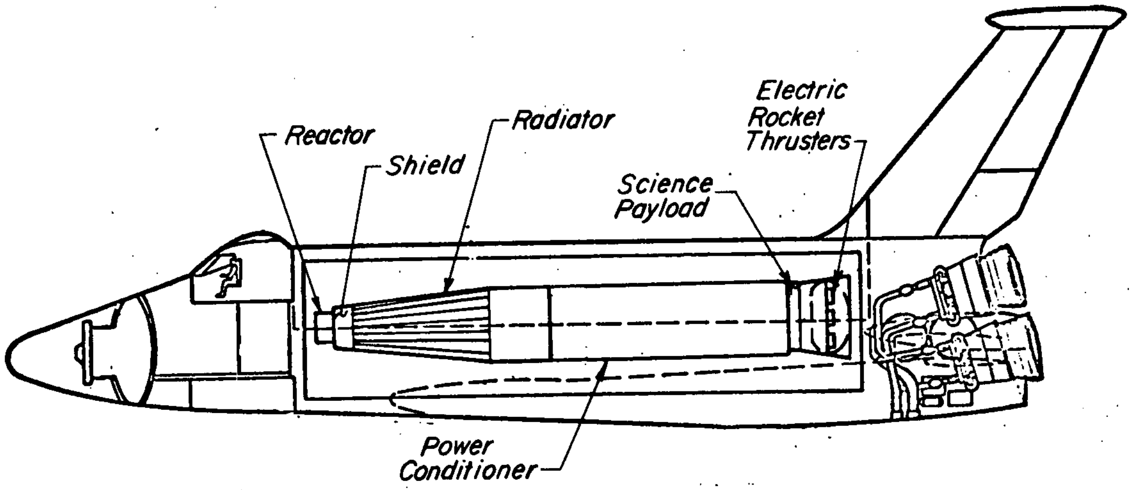


Figure 4.11 Nuclear Rocket Vehicle Initial Mass vs Ideal Rocket Velocity for Various Payload Masses
Reference [18]



a. 120 kWe Nuclear Electric Rocket Propelled Spacecraft/ Centaur in Space Shuttle Orbiter Payload Bay



b. 250 kWe Nuclear Electric Rocket Propelled Spacecraft in Space Shuttle Orbiter Payload Bay

Figure 4.12 Nuclear Electric Rocket Propelled Spacecraft/ Space Shuttle Orbiter Integration Reference [19]

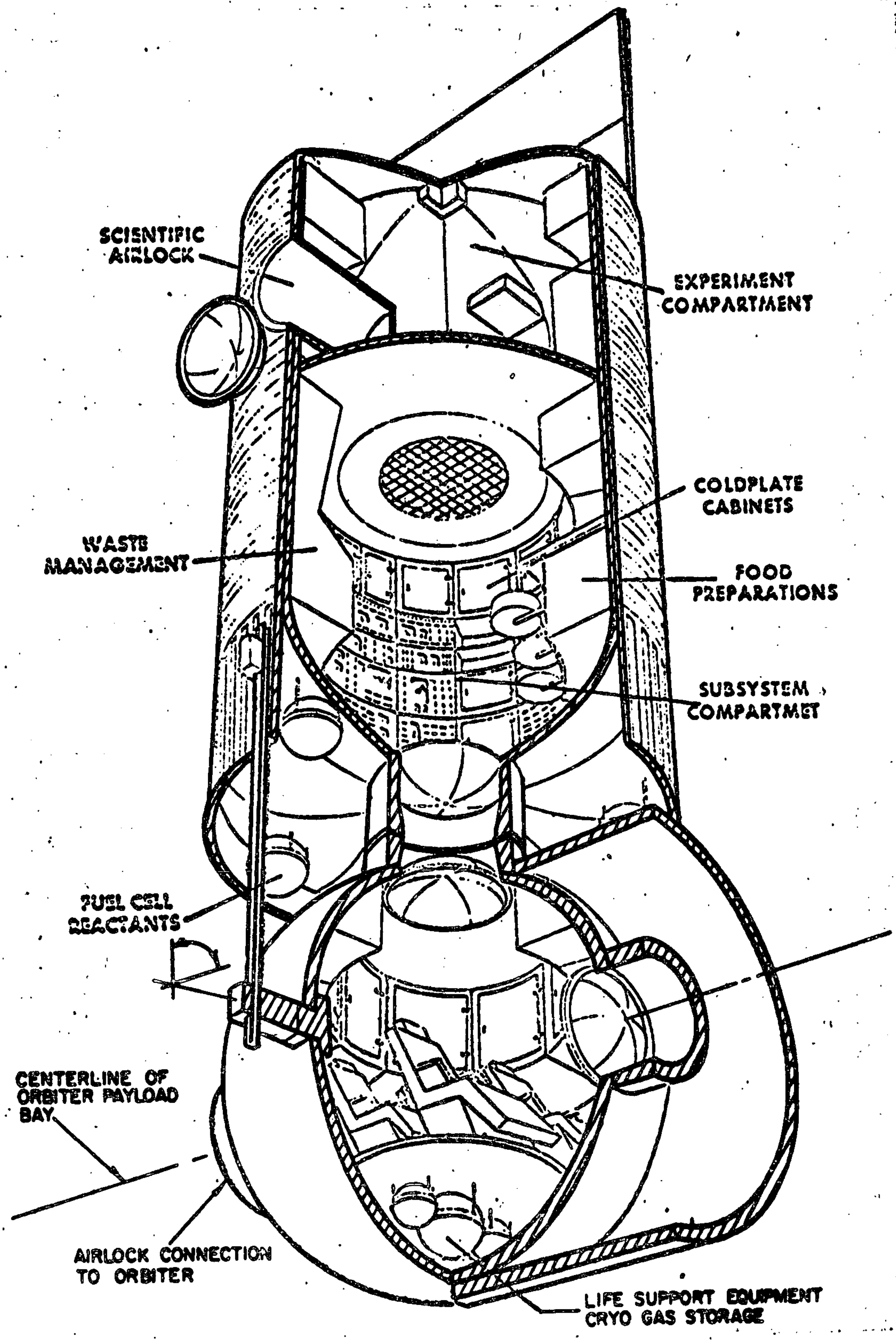


Figure 4.13 Space Shuttle Sortie Manned Experiment Module Reference [8]

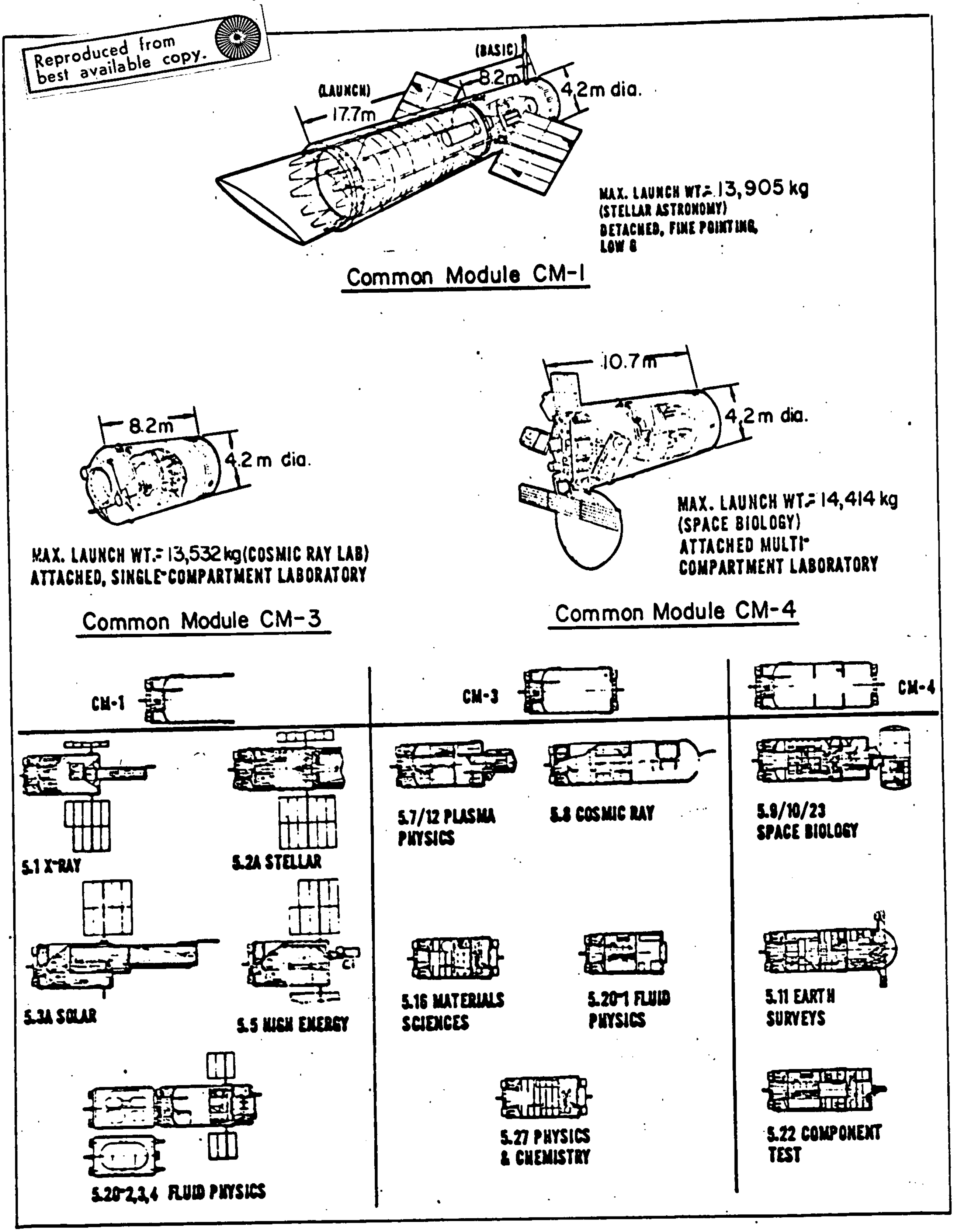


Figure 4.14 Space Station Modules
Reference [14]

further work is necessary to better define the new possibilities for payload design the reduced costs resulting from standard spacecraft and modules and the prospects for reuse of spacecraft offered by the Space Shuttle represent important results from the LMSC studies.

4.4.2.1 Low Cost Payloads

The LMSC low cost design of a synchronous equatorial orbiter has already been shown above in Figure 4.7 and other typical spacecraft have been similarly designed. A further generalization of these design principles is shown in configurations of a standard spacecraft and module in Figures 4.15 and 4.16. Although such spacecraft will usually be larger and more massive than conventional these factors are more than offset by the reduction in initial cost and refurbishment costs resulting from standardized subsystems, etc. These ideas need further development against real designs for actual payloads and classes of payloads in the mission model and in conjunction with the Space Transportation System before the benefits can be certified; however, the current Design Guide for Space Shuttle Low-Cost Payloads [21] contains some interesting and important considerations.

4.4.2.2 Reusable Payloads

The importance of payload reusability is closely associated with other considerations, such as lifetime/reliability, retrieval, refurbishment/updating and replacement of the payload in orbit. The design of payloads for reuse has also been analyzed by LMSC and presented in References [20 and 21]. These payloads differ as a class from expendable payloads and their feasibility for various missions and the cost savings to be realized require further definition in an overall space program.

4.4.3 Payload Reliability, Retrieval, Refurbishment/Updating and Reuse

The interactions between considerations of payload lifetime, reliability, retrieval, refurbishment/updating and reuse with other Space Transportation System aspects are so complex that the ultimate cost savings remain to be shown. While there are strong indications that considerable

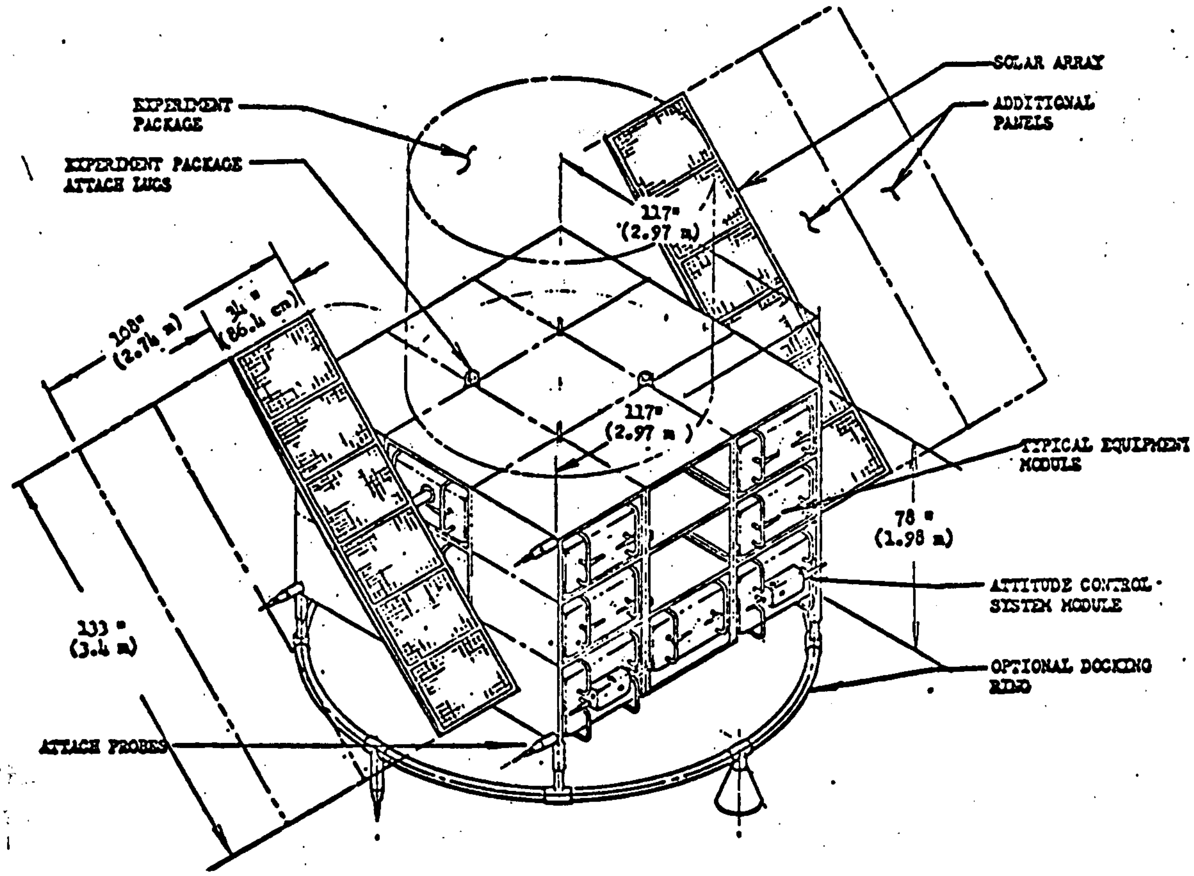


Figure 4.15 Standard Spacecraft General Configuration
Reference [20]

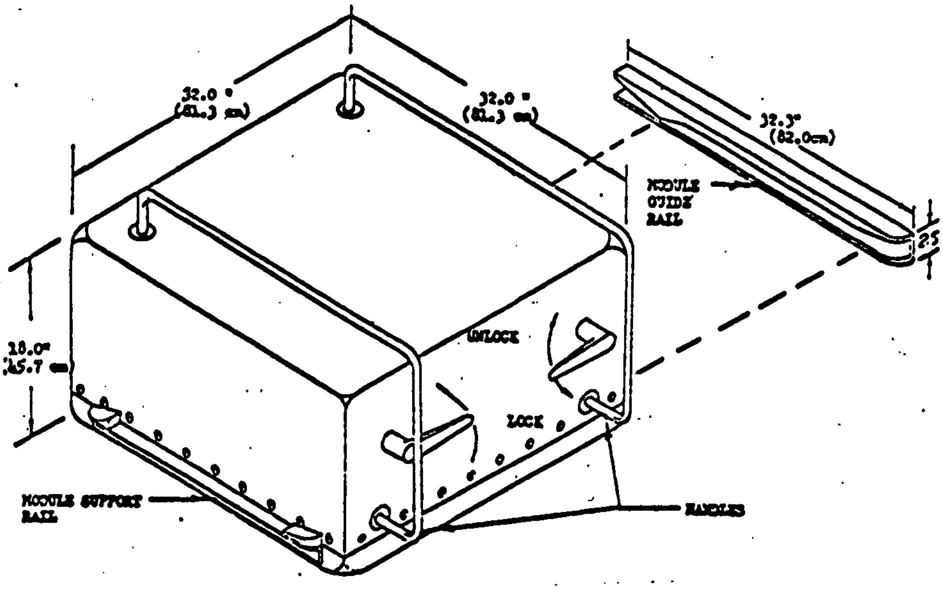


Figure 4.16 Typical Standard Spacecraft Module
Reference [20]

savings will result, a definitive analysis remains to be performed and although this may not be practicable in the ultimate sense continued efforts will be helpful in realizing the most economical result possible.

4.4.4 Department of Defense Payloads

Although classified data on DoD payloads have been provided no details are available for presentation in this report. Some unclassified information is available in Reference [9].

4.4.5 National Aeronautics and Space Administration Payloads

The "Fleming" mission model of the Spring 1971 from NASA Headquarters has been used as the primary source of the payload data by The Aerospace Corporation in addition to their own detailed compilations [7, 8]. The Aerospace Case A (Baseline) mission model was used in identifying the payloads presented below with some additions. The primary source of the payload information was Reference [7] which contains more detail, especially a computerized Payload Data Bank.

4.4.5.1 Astronomy

There are twelve astronomy payloads in the mission model. They represent the largest and more massive and complex group of spacecraft. Payload characteristics are given in Table 4.1.

4.4.5.2 Physics

The five space physics payloads in the mission model are characterized in Table 4.2. They are small spacecraft since larger experiments will be conducted in the Space Shuttle sortie manned experiment modules or the space station modular laboratories.

4.4.5.3 Earth Observation

The NASA Earth observation missions include three R&D and four systems demonstration payloads. Their characteristics are given in Table 4.3.

Table 4.1

Current Expendable Payload Characteristics—NASA Astronomy Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
15	NAS-1	Large Stellar Telescope	Extend space astronomy capability to diffraction limited 3 meter dia. optical technology. Determine universe curvature stellar and galactic composition and evolution.	10140	4 x 13.7	648/648/28.5	10/2	1981
17	NAS-2B	Large Solar Observatory	Conduct high resolution visual and UV studies of solar granular structure and areas of high solar activities. Continue UV and X-ray observations.	12600	4.6 x 17.4	648/648/30	10/2	1983
19	NAS-3	Large Radio Observatory	Understand physical processes in the corona and in the magnetospheres of the planets, especially Jupiter and Earth.	9090	4.3 x 9.2	648/648/30	10/2	1985

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.1 (cont'd-2)

Current Expendable Payload Characteristics—NASA Astronomy Missions 1979-1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
13	NAS-4	High Energy Astronomy Observatory	Conduct long duration observations to characterize the high energy, but lower flux radiation of importance to astrophysicists and cosmologists. Locate and describe stellar sources of high energy photons and particles.	9770	3.4 x 15	370/370/30	10/2	1979
10	NAS-7	Solar Orbit Pair (A) (ea)	To monitor all of the solar sphere simultaneously and to continuously provide information on flares, sun spots, solar winds, etc.	860	3.1 x 3.7	35786/ 35786/30	10/5	1984

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.1 (cont'd-3)

Current Expendable Payload Characteristics—NASA Astronomy Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
11	NAS-8	Solar Orbit Pair (B) (ea)	To monitor all of the solar sphere simultaneously and to continuously provide information on flares, sun spots, solar winds, etc.	1 140	3.1x 3.7	1 A. U. Helio/28.5	10/5	1984
12	NAS-9	Optical Interferometer (A)	To measure stellar diameters and IR spectra. This is achieved by two spacecraft, (A) and (B), separable up to 300 meters and each with 2 ft diameter mirror	1 410	2.1x 3.1	35786/ 35786/30	5/5	1988
12	NAS-10	Optical Interferometer (B)	To measure stellar diameters and IR spectra. This is achieved by two spacecraft, (A) and (B), separable up to 300 meters and each with 2 ft diameter mirror	1 410	2.1x 3.1	35786/ 35786/30	5/5	1988

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.1 (cont'd-4)

Current Expendable Payload Characteristics—NASA Astronomy Missions 1979-1990

"Fleming" Model, Spring 1971

Reference 7

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
9	NAS-11	Radio Interferometer	To measure radio spectra & radio diameter and velocities of space objects. One leg of the interferometry will be earth based.	4730	4.3 x 7.6	71572/ 71572/28.5	3/3	1981
1	NAS-14A	Astronomy Explorer	Independent investigations of solar and stellar behavior in the UV, X-ray and radio spectral regions.	409	1.4 x 1.2	500/500/28.5	3/3	1979
2	NAS-14B	Astronomy Explorer	Independent investigations of solar and stellar behavior in the UV, X-ray and radio spectral regions.	409	1.4 x 1.2	35786/ 35786/0	3/3	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.1 (cont'd-5)

Current Expendable Payload Characteristics—NASA Astronomy Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
6	NAS-15	Orbiting Solar Observatory	Monitor temporal variations of the Sun's brightness in the UV, X-ray and gamma-ray regions	909	2.1 x 3.1	648/648/28.5	1/1	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.2

Current Expendable Payload Characteristics — NASA Space Physics Missions 1979-1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life - Sys/Pl, yr/yr	IOC, yr
3	NSP-1	Lower Magneto-sphere	To conduct investigations of the environment of the lower magnetosphere, neutral air chemistry and density, and ionospheric behavior.	545	1.2 x 2.4	333/33 336/ 90 & 28.5	3/1	1979
4	NSP-2	Middle Magneto-sphere	To measure ionospheric current systems and behavior with respect to solar activity. Also neutral atmospheric studies.	455	1.8 x 2.4	1852/37 040/ 90 & 28.5	3/1	1979
5	NSP-3	Upper Magneto-sphere	To monitor "space weather" and the boundary of the geomagnetic field as it interacts with the solar wind.	273	1.2 x 1.8	1 A.U. Helio/ Ecliptic	3/1	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.2 (cont'd -2)

Current Expendable Payload Characteristics — NASA Space Physics Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass kg	Dia & Lgth, mxm	Orbit - peri/apo/incl. km/km/deg	Life Sys/Pl yr/yr	IOC, yr
7	NSP-6	General Relativity	To experimentally test Einstein's general relativity theory. Gyroscopes in an earth orbiting satellite will experience two relativistic precession effects.	682	1.5 x 2.1	555/555	1/1	1984
8	NSP-7	General Relativity	To experimentally test Einstein's general relativity theory. Gyroscopes in an earth orbiting satellite will experience two relativistic precession effects.	227	1.2 x 1.5	1 A.U. Helio/ 28.5	1/1	1981

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.3

Current Expendable Payload Characteristics—NASA Earth Observation Missions 1979 - 1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit-peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			R&D					
21	NEO-2	Polar Earth Observation Satellite	To design, develop and operate a space observatory system to perform meteorological and earth resources surveying by advanced remote sensing techniques	1180	3.7 x 4.6	925/925/100	2/2	1979
22	NEO-3	Synchronous Earth Observation Satellite	Research satellite to investigate and develop remote sensing techniques for measurement of the Earth's surface and atmosphere from synchronous altitude	455	1.8 x 1.2	35 786/ 35 786/0	2/2	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.3 (cont'd -2)

Current Expendable Payload Characteristics—NASA Earth Observation Missions 1979 - 1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
23	NEO-5	Earth Physics Satellite	To make precision measurements of the Earth's land and sea areas to determine (1) continental drift, (2) mass distribution, (3) surface strain, and (4) variation of gravity, sea altitude, and mass	273	1.1 x 3	740/740/90	2/2	1980
			SYSTEMS DEMONSTRATION					
27	NEO-4	Synchronous Earth Resources Satellite	To design, develop and operate a satellite system for remote sensing of the Earth's surface and the lower regions of the atmosphere from synchronous orbital altitude	468	1.2 x 1.8	35 786/ 35 786/0	2/2	1981

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.3 (cont'd -3)

Current Expendable Payload Characteristics—NASA Earth Observation

Missions 1979 - 1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			SYSTEMS DEMONSTRATION (cont'd)					
24	NEO-8	Synchronous Meterology Satellite	Develop and operate a synchronous meterological satellite for DOC/ESSA	468	1.5 x 2.4	35 786/ 35 786/0	2/2	1982
25	NEO-6	TIROS	System demonstration of the 4th generation series of operational meteorological satellite for DOC/ESSA	468	1.5 x 3.0	1300/1300/100	5/5	1981
26	NEO-17	Polar Earth Resources Satellite	To design, develop and operate a space observatory system to perform meteorological and Earth resources surveying by advanced remote sensing techniques	1180	3.6 x	925/925/100	2/2	1986

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

4.4.5.4 Communications and Navigation

Nine NASA payloads are assigned to five R&D, three systems demonstration and one operational missions. The operational mission uses the tracking and data relay satellites for space mission control. The payload characteristics for these missions are provided in Table 4.4.

4.4.5.5 Planetary

NASA currently includes 13 different payloads in the planetary missions. Their characteristics are given in Table 4.5.

4.4.5.6 Shuttle Sortie Modules

Four manned experiment modules and eight pallet type modules are listed for the Space Shuttle sortie missions. The characteristics are shown in Table 4.6.

4.4.5.7 Space Station Modules

Five basic space station modules are needed to make up the Shuttle orbited station; while four Big Gemini modules are identified if the space station is lofted by an expendable launch vehicle. Six different experiment modules are listed at the present time in support of the various experimental undertakings in the mission model. Characteristics of these modules are given in Table 4.7.

4.4.6 Non-NASA Operational Payloads

The nine non-NASA operational missions and their payloads are also presented in the "Fleming" model. A wide variety and number of missions are seen for these payloads whose characteristics are presented in Table 4.8.

4.5 Foreign Payloads 1979-1990

Foreign payloads in the 1980's will be oriented toward space applications except for the Soviet Union where their Space Program will include the widest variety of space activity. Both developed and developing countries in the

Table 4.4

Current Expendable Payload Characteristics—NASA Communications and Navigation Missions 1979 - 1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			R&D					
28	NCN-1	Application Technology Satellite	Earth to geo-stationary orbit communication power, high gain multi-beam satellite antenna, general application technology (meteorology, earth observations, etc.)	3 740	4.6 x 6.4	35 786/ 35 786/0	5/5	1979
30	NCN-2A	Small Application Technology Satellite	To design, develop, launch, and operate a series of small R&D satellites for the experimental application of research and technology developments in spacecraft and sensor subsystems	282	2 x 3.7	555/5 550/90	1/1	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

Table 4.4 (cont'd -2)

Current Expendable Payload Characteristics—NASA Communications and Navigation Missions 1979 - 1990

"Fleming" Model, Spring 1971 — Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit-peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			R & D (cont'd)					
29	NCN-2B	Small Application Technology Satellite	To design, develop, launch, and operate a series of small R&D satellites for the experimental application of research and technology developments in spacecraft and sensor subsystems	282	2 x 3.7	35 786/ 35 786/0	1/1	1979
31	NCN-3A	Cooperative Applications	Communication satellite to be flown in partnership with other nations who will provide corresponding technical and funding assistance	386	2 x 3.7	35 786/ 35 786/0	2/2	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.4 (cont'd -3)

Current Expendable Payload Characteristics—NASA Communications and Navigation Missions 1979 - 1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			R & D (cont'd)					
32	NCN-3B	Cooperative Applications	Communication satellite to be flown in partnership with other nations who will provide corresponding technical and funding assistance	386	2 x 3.7	555/5 550/90	2/2	1982
			SYSTEMS DEMONSTRATION					
33	NCN-11	Medical Network Satellite	Facilitate applications of space technology and satellite systems for medical data transmission purposes	941	3.7 x 4.6	35 786/ 35 786/0	5/5	1979
34	NCN-12	Education Broadcast	Facilitate application of space technology and satellite systems for educational broadcast purposes	1 600	3. x 7.6	35 786/ 35 786/0	5/5	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.4 (cont'd -4)

Current Expendable Payload Characteristics—NASA Communications and Navigation Missions 1979 - 1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit- peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr.
SYSTEMS DEMONSTRATION (cont'd)								
35	NCN-13	Follow-on Systems Demonstration	Systems demonstration satellites for law enforcement, air traffic control, land traffic control type missions	941	3.7 x 4.6	35 786/ 35 786/0	5/5	1981
OPERATIONAL								
36	NCN-5	Tracking and Data Relay	Develop and operate a command, tracking and data relay of low orbiting satellites from a synchronous satellite to a few centrally located mission control centers	1082	3 x 5.2	35 786/ 35 786/0	12/3	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.5

Current Expendable Payload Characteristics—NASA Planetary Missions 1979 - 1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit-peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
50	NPL-1	Mars Viking	To provide information regarding the possible existence and nature of life on Mars, the atmospheric and surface characteristics of the planet and the nature of the planetary environment	3 509	3 x 3.7	Interpl. Helio. + Mars Plan Planeto.	1/1	1979
52	NPL-5	Venus Explorer Orbiter	Measure plant magnetosphere, magnetosheath, detached bow shockwave, and tail and wake region. Investigate internal composition, structure and magnetic field	455	1.5 x 3.7	Interpl. Helio. + Venus planeto	1/1	1980
53	NPL-6	Venus Radar Mapping	Detailed surface mapping of Venus to a resolution of 50 meters using radar imaging	3 591	3 x 7.6	Interpl. Helio + Venus planeto	1/1	1982

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.5 (cont'd - 2)

Current Expendable Payload Characteristics—NASA Planetary

Missions - 1979 - 1990

"Fleming" Model, Spring 1971 — Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth,	Orbit-peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
54	NPL-7	Venus Explorer Lander-1	Analysis of surface properties and environment of Venus. Measurement of atmospheric properties during descent. Surface mapping by orbiter	3 373	3 x 7.6	Interpl. Helio + Venus planeto	1/1	1985
54	NPL-7	Venus Explorer Lander-2	Orbiting microwave and IR spectral instruments for surface, atmosphere and cloud studies. Landed seismometer, X-ray diffraction, composition measurement, environmental dynamics	2 159	3 x 7.6	Interpl. Helio + Venus planeto	1/1	1988

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.5 (cont'd - 3)

Current Expendable Payload Characteristics—NASA Planetary Missions - 1979 - 1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit-peri/apo/incl, km/km/deg	Life Sys, Pl, yr/yr	IOC, yr
56	NPL-10	Grand Tour	Obtain first-generation flyby data of Uranus and Neptune. Correlate spatial effects in cosmix flux and solar wind with JSP mission	687	3 x 3.7	Interpl. Helio + Jupiter Swingby + Uranus and Neptune Fly-by	9/9	1979
55	NPL-11	Jupiter Pioneer Orbiter	Measure particles and field environment to 5AU, particle density of asteroid belt, magnetic and radiation fields of Jupiter, and to provide Jupiter imaging	432	3 x 4.6	Interpl. Helio + Jupiter planeto	2/2	1982
57	NPL-13	Jupiter TOPS Orbiter/ Probe	Monitor particles and field environment, measure atmospheric composition, characteristics and profiles	1 495	3 x 4.6	Interpl. Helio + Jupiter planeto	3/3	1985

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54

Table 4.5 (cont'd -4)

Current Expendable Payload Characteristics—NASA Planetary Missions - 1979 - 1990

"Fleming" Model, Spring 1971 | Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit-peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
58	NPL-14	Uranus TOPS Orbiter Probe	Mapping, composition analysis, and time dependent measurements of the atmosphere. Determine the extent and intensity of planetary fields	1682	3 x 4.6	Interpl. Helio + Uranus planeto	7/7	1986
59	NPL-15	Asteroid Survey	Define micrometeoroid, particle and field environment in asteroid belt. Prove solar electric propulsion over long duration	864	3 x 6.1	Interpl. Helio	4/4	1984
60	NPL-18	Comet Rendezvous	Close range, long duration examination of comet. Determine physical state, structure, composition and mode of interaction with the planetary environment	941	3 x 6.1	Interpl. Helio	4/4	1982

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.5 (cont'd -5)

Current Expendable Payload Characteristics—NASA Planetary Missions - 1979 - 1990

"Fleming" Model, Spring 1971 — Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth m x m	Orbit- peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
51	NPL-19	Mars Surface Sample Return-A	The exploration of Mars and the return of physical samples of the planetary surface to Earth. Orbiter 1 Bus vehicle.	4 818	4.3 x 4.9	Interpl. Helio + Mars planeto	3/3	1990
(Mated with B in Earth orbit for flight to Mars)								
52	NPL-20	Mars Surface Sample Return-B	The exploration of Mars and the return of physical samples of the planetary surface to Earth. Lander 1 Return Probe	5 182	4.3 x 7	Interpl. Helio + Mars planeto + Earth planeto	3/3	1990
(Mated with A in Earth orbit for flight to Mars)								

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

Table 4.6

Payload Characteristics—NASA Shuttle Sortie Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			MANNED EXPERIMENT MODULES					
38	NSO-1	General Scientific Research	Provide manned research module to conduct astronomy, space physics, life science, and/or contamination monitoring experiments while attached to shuttle	12500	4.3 x 16.5	370/370/55	5/.04	1981
39	NSO-2	General Applications	Provide research module for man to conduct earth observations, communications and navigations, and/or material science experiments while attached to shuttle	13636	4.3 x 16.5	185/185/165	9/.04	1981

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.6 (cont'd-2)

Payload Characteristics—NASA Shuttle Sortie Missions 1979-1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			MANNED EXPERIMENT MODULES (cont'd)					
40	NSO-3	Dedicated Scientific Research	Provide a dedicated research module for man to conduct Earth observations, communications and navigations, and/or material science experiments while attached to shuttle	13409	4.3 x 16.5	370/370/55	7/.04	1984
41	NSO-4	Dedicated Applications	Provide module for man to conduct short-term Earth observations while attached to the shuttle	10227	4.3 x 12.5	185/185/75	7/.04	1984

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

Table 4.6 (cont'd-3)

Payload Characteristics—NASA Shuttle Sortie Missions 1979-1990

"Fleming" Model, Spring 1971 — Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC, yr
			PALLET TYPE MODULES					
42	NSO-5A	Earth Observation	Provide a test bed to conduct scientist-astronaut training and automated experiments	2727	4.3 x 11.3	231/231/90	5/.04	1980
43	NSO-5B	Bioscience Research	Provide a test bed to conduct scientist-astronaut training and automated experiments	1955	4.3 x 11.3	370/370/28.5	5/.04	1979
44	NSO-5C	Astronomy	Provide a test bed to conduct scientist-astronaut training and automated experiments	2591	4.3 x 11.3	370/370/28.5	5/.04	1980
45	NSO-5D	Fluid Management	Provide a test bed to conduct scientist-astronaut training and automated experiments	3227	4.3 x 11.3	370/370/28.5	5/.04	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) 2 0.54 n.mi.

Table 4.6 (cont'd-4)

Payload Characteristics—NASA Shuttle Sortie Missions 1979-1990

"Fleming" Model, Spring 1971 — Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl km/km/deg	Life Sys/Pl. yr/yr	IOC, yr
			PALLET TYPE MODULES (cont'd)					
46	NSO-5E	Teleoperator	Provide a test bed to conduct scientist-astronaut training and automated experiments	2273	4.3 x 11.3	370/370/28.5	5/.04	1979
47	NSO-5F	Manned Work Platform	Provide a test bed to conduct scientist-astronaut training and automated experiments	3045	4.3 x 11.3	370/370/28.5	5/.04	1981
48	NSO-5G	Large Telescope Mirror Test	Provide a test bed to conduct scientist-astronaut training and automated experiments	5909	4.3 x 16.5	370/370/28.5	5/.04	1979
49	NSO-5H	Astronaut Maneuvering Unit (AMU)	Provide a test bed to conduct scientist-astronaut training and automated experiments	1727	4.3 x 11.3	370/370/28.5	5/.04	1980

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.7

Payload Characteristics—NASA Space Station Modules 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
			BASIC SPACE STATION MODULES					
61	NSS-2A	Core Module	Long-term manned space operations, core module	9090	4.3 x 12	500/500/55	10/.33	1981
62	NSS-2B	Power Module	Long-term manned space operations, power module	9090	4.3 x 9.2	500/500/55	10/.33	1981
63	NSS-2C	Crew Module	Long-term manned space operations, crew module	9090	4.3 x 9.2	500/500/55	10/.33	1981
64	NSS-2D	Control Module	Long-term manned space operations, control module	9409	4.3 x 9.2	500/500/55	10/.33	1981
65	NSS-2E	General Purpose Laboratory	Long-term manned space operations, general purpose laboratory	9409	4.3 x 9.2	500/500/55	10/.33	1985

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

E

Table 4.7 (cont'd-2)

Payload Characteristics—NASA Space Station Modules 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
			BIG GEMINI MODULES					
	NSS-3A	Min-Mod Big Gemini -Crew Module	Resupply of orbiting space station and transportation of 9 men, crew module	10888	4.6 x 15	500/500/55	10/.04	1981
	NSS-3B	Min-Mod Big Gemini -Cargo Propulsion Module	Resupply of orbiting space station and transportation of 9 men, cargo/propulsion module	9891	4.6 x 7.6	500/500/55	10/.04	1981
	NSS-4A	Advanced Big Gemini -Crew Module	Resupply of orbiting space station and transportation of 12 men, crew module	11098	6.6 x 16	500/500/55	10/.04	1986
	NSS-4B	Advanced Big Gemini -Cargo/Propulsion Module	Resupply of orbiting space station and transportation of 12 men, cargo/propulsion module	47586	6.6 x 13	500/500/55	10/.04	1986

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.7 (cont'd-3)

Payload Characteristics—NASA Space Station Modules 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Flem-ing	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
			EXPERIMENT MODULES					
66	NSS-9	Exp. Mod. -1 Life Science	Support of space biology experiments	12594	4.6 x 17.7	500/500/55	3/.33	1981
67	NSS-5A	Exp. Mod. -1 Earth Obs.	Support of earth observation experi-ments	12594	4.6 x 17.7	500/500/55	3/.33	1981
68	NSS-5B	Exp. Mod. -1 Space Mfg.	Support of material science and process-ing experiments	12594	4.6 x 17.7	500/500/55	3/.33	1990
64	NSS-7A	Exp. Mod. -3 Phys. Lab.	Support of space physics experiments	11310	4.6 x 12.5	500/500/55	4/.33	1983
65	NSS-7B	Exp. Mod. -3 Cosmic Ray Lab.	Support of cosmic ray experiments	11310	4.6 x 12.5	500/500/55	2/.33	1988
68	NSS-7C	Exp. Mod. -3 Comm. & Navig.	Support of communi-cation and navigation experiments	11310	4.6 x 12.5	500/500/55	/.33	1981

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

Table 4.8

Current Expendable Payload Characteristics—Non-NASA Operational Missions 1979-1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
70	NCN-7	Communication Satellite	Provide operational services in information networks and navig.	677	2.7 x 6.7	35 748/ 35 748/0	12/5	1979
71	NCN-8	US Domestic Com. Satellite	Provide operational services in communication networks, cable TV, broadcast TV, radio, telephone, teletype, etc.	1611	4.6 x 7.6	35 748/ 35 748/0	12/7	1979
72	NCN-9	Foreign Domestic Comm. Satellite	Provide operational services in comm. networks for S. Amer., Can., Australia, ESRO, S. Africa, India, and neighboring countries	468	1.2 x 3.7	35 748/ 35 748/0	11/5	1980
73	NCN-10A	Navig./ Traffic Control Satellite	To gather data from remote mobile platforms and scattered transmitters and centralize the outputs into a common data center	330	1.5 x 2.4	35 748/ 35 748/5	12/5	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

Table 4.8 (cont'd-2)

Current Expendable Payload Characteristics—Non-NASA Operational Missions 1979-1990

"Fleming" Model, Spring 1971 Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
74	NCN-10B	Navig. / Traffic Control Satellite	Navig. data over oceans and domestic areas	330	1.5 x 2.4	29 600/ 55 500/29	12/5	1979
75	NEO-7	TOS Meteorological Satellite	Observe global clouds, day and night, cloud top heights, heat balance, vertical temp. and water vapor profiles	468	1.5 x 1.8	1295/1295/100	12/4	1979
76	NEO-15	Synchronous Meteor. Sat.	Operational meteorological satellite operating from synchronous alt. for ESSA	470	1.5 x 2.4	35 748/ 35 748/0	12/2	1979
77	NEO-16	Polar Earth Resources	Operational satellite to continually survey Earth resources and to perform meteorological survey with high resolution sensors and transmit data to Earth	1177	1.8 x 1.8	925/925/100	12/2	1979

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n.mi.

Table 4.8 (cont'd-3)

Current Expendable Payload Characteristics—Non-NASA Operational Missions 1979-1990

"Fleming" Model, Spring 1971

Reference [7]

Ident. Nos. Fleming	Ident. Nos. Data Bank	Payload	Mission	Launch mass, kg	Dia & Lgth, m x m	Orbit - peri/apo/incl, km/km/deg	Life Sys/Pl, yr/yr	IOC yr
78	NEO-11	Synchronous Earth Resources	Operational remote sensing and measurement of the Earth's resources and lower atmosphere	468	1.8 x 1.8	35 748/ 35 748/0	6/3	1985

Conversions: 1 kilogram (kg) = 2.2 pounds, 1 meter (m) = 3.28 feet, 1 kilometer (km) = 0.54 n. mi.

world will be very active in attempting to realize the benefits of space.

4.5.1 European Payloads

It is hoped and expected that the European space activity will be well coordinated with that of the United States. The primary types of activity will include: communications, air traffic control, meteorology, Earth resources, scientific and planetary spacecraft that will provide payloads for a new Space Transportation System if cooperative agreements can be satisfactorily worked out.

4.5.2 Others

The developing countries will have or share payloads that are oriented toward Earth applications especially resources, mapping, communications including education, etc. A considerable competition could develop in providing the spacecraft and transport and in particular the associated ground activity to the developing countries.

4.6 Projected Traffic 1979-1990

The projected traffic for a new Space Transportation System from all sources during the 1979-1990 period used in the MATHEMATICA economic analysis is taken from the "Fleming" mission model of Spring 1971 as it appears in Reference [8]. Scenarios have again been used that range from 300 to 900 missions during the 12-year period as presented and discussed briefly below. It is believed that the numbers of missions are conservative in certain respects and that a new system with attractive performance will itself generate new missions and traffic but no basis exists for including this belief in the analysis.

4.6.1 United States Traffic 1979-1990

A summary of United States traffic -- DoD, NASA and non-NASA operational missions as provided by The Aerospace Corporation in Reference [8]is shown in Table 4.9. For the MATHEMATICA economic analysis the traffic was subsequently modified as described in Chapter 6.

Table 4.9

Mission Traffic Summary 1979-1990

"Fleming" Model, Spring 1971 Reference [8] (Table 2-1)

	1979	1980	1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	Total
NASA													
Physics and Astronomy*	6	8	10	10	10	14	13	13	14	15	16	14	143
Earth Observations	1	3	4	6	4	2	3	4	7	4	2	3	43
Comm. and Nav.	7	6	6	5	7	7	4	5	6	6	6	4	69
Planetary	3	1	1	4	0	1	3	1	1	1	1	2	19
Space Station	0	0	9	6	8	7	12	11	10	9	8	10	90
Sorties	2	6	8	10	8	10	10	9	7	9	9	9	97
NASA TOTALS	19	24	38	41	37	41	45	43	45	44	42	42	461
NON-NASA													
Communications	3	5	8	3	6	3	3	6	7	6	4	4	58
Navigation	3	2	3	0	2	0	2	0	2	0	2	0	16
Meteorology	2	2	2	2	2	2	2	2	2	2	2	2	24
Earth Resources	4	0	4	0	4	0	8	0	0	4	6	0	30
NON NASA TOTALS	12	9	17	5	14	5	15	8	11	12	14	6	128
TOTAL	31	33	55	46	51	46	60	51	56	56	56	48	589
DOD	24	25	19	21	29	25	22	24	25	22	22	23	281
TOTAL:	55	58	74	67	80	71	82	75	81	78	78	71	870

* Includes Revisits

The United States traffic projection is still quite uncertain and more work is needed to clarify the missions from both the recognized and newer sources. Their interactions mutually and with the Space Transportation System need to be studied to better ascertain the likely traffic demand and further effort can be spent to considerable advantage in this direction.

4.6.1.1 Department of Defense Traffic

As discussed earlier the DoD missions and traffic rationale are security classified, but the traffic is identified with a modification of DoD Option B and is discussed in Reference [8]. The traffic shown in Table 4.9 has been further modified for the MATHEMATICA economic analysis as discussed in Chapter 6.

4.6.1.2 National Aeronautics and Space Administration Traffic

The NASA traffic in the "Fleming" model as provided by Reference [8] is shown in Table 4.9. Further modification of this mission model in response to recent and continuing results from the Space Transportation System alternative concepts studies and developments in other programs should be made early in 1972 when NASA's overall program, hopefully, has clarified somewhat.

4.6.1.3 Non-NASA Operational Traffic

The traffic from non-NASA operational missions both governmental agencies and commercial institutions as currently seen is also shown in Table 4.9, however, a considerable amount of traffic is assigned to relatively few spacecraft. Direct contact needs to be made with the prospective user agencies so this traffic can be better defined. Increased variety and numbers of flights are anticipated in this category, but considerable effort is foreseen to improve the accuracy of the projection and this is believed to be warranted at the presnt time.

4.6.2 Foreign Traffic

It is in the important area of foreign traffic that the need for a new

United States Space Transportation System needs to be carefully defined for both developed and developing countries as space activity is expected to develop strongly from these sources of traffic. The United Nations is currently engaged in identifying space applications, especially for the developing countries, and should be quite helpful in defining the traffic possibilities.

4.6.2.1 European Traffic

European space traffic during the period of concern as presently identified is shown in Table 4.10 which was taken from Reference [15]. Further identification of this traffic and its possible assignment to a new United States Space Transportation System should be studied.

4.6.2.2 Other Foreign Traffic

This source of traffic for a new Space Transportation System as well as the payloads and their associated ground activity in the developing countries has had relatively little attention in preparing the mission models or in overall projections of the Earth's space activity in the 1980's. More attention by all interested parties should be given to this possible traffic and it associated concerns.

4.6.3 Traffic Uncertainty and Space Program Planning

It is, of course, not possible to project space traffic for almost twenty years in the future with any real accuracy; however, it is considered necessary to continue to project mission models with flexibility in kinds of payloads and level of activity. This necessitates further efforts to develop more rational space program plans and more effective leadership in convincing the American people of the desirability -- even, the necessity -- of carrying them through during the next twenty years and beyond.

Table 4.10

Estimated European Space Activity in the Period 1979-1990

Reference [15] (Figure 10)

type	calendar year													total in 1978-1990 period
	78	79	80	81	82	83	84	85	86	87	88	89	90	
communications		2		2		2			3		2		3	14
air traffic control			2					2		1	1		2	8
meteorology	2			1			2			1		2		8
earth resources		1		1		1				2				5
scientific	3	3	4	4	4	4	4	5	5	5	5	6	6	58
planetary	1		1		1		1		1		1			6
TOTAL	6	6	7	8	5	7	7	7	9	9	9	8	11	99

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ECONOMIC ANALYSIS OF THE SPACE SHUTTLE SYSTEM

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National Aeronautics and Space Administration
under Contract NASW - 2081

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Volume II

CHAPTER 5.0

SPACE TRANSPORTATION SYSTEM ALTERNATIVE CONCEPTS

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CHAPTER 5.0

ALTERNATIVE CONCEPTS OF SPACE TRANSPORTATION SYSTEM

5.1 Introduction

The purpose of this chapter is to introduce and summarize the various Space Transportation System concepts as they have evolved in terms of cost, reliability and performance for the systems presently in use and the system concepts proposed for future use. This chapter outlines the framework within which the economic analysis was carried out. If dramatically new concepts of space transportation were to arise, then the economic analysis would have to be extended to include these.

While cost may be a major factor in the selection of the most desirable Space Transportation System for the period 1970-1990, it should by no means be the only factor. That new systems for this period may be conceived is a reflection of significant technological advances in materials, thermal protection systems, propulsion systems, structures and electronic systems. The risk associated with committing a development program toward a particular system is largely related to the gap between existing and required technologies. The payload capability of the various transportation systems is of equal import and operational subtleties exist that can cause significant performance differences between systems that have seemingly similar capability. The reliability of the various transportation systems can have a great effect on total program costs, particularly considering new families of very expensive payloads. The partial or complete failure of a payload is not to be neglected as such events may require either payload replacement or the acceptance of a reduced payload benefit. The proposed space shuttle systems promise a substantial reduction of these risks and uncertainties in the 1980's.

A considerable portion of this chapter is devoted to a description of the various Space Transportation System alternatives and their relative performance capability. However, substantial attention has also been given to the technological aspects described above. In particular, the various technologies are reviewed in some detail to highlight the technical assumptions underlying the economic analysis as reported in here with emphasis on elements of risk associated with further pre-programmed developments. Also, factors contributing to the payload capability of transportation systems are discussed.

5.2 Current Expendable Launch Vehicles

This section provides descriptions and performance data for currently available expendable launch vehicles. These vehicles consist of the latest versions of vehicles that have been used in the recent past to deliver payloads to orbit, as well as some newer vehicles that have not as yet been launched but are based on proven technology and are considered by NASA to be part of the current fleet [1, 2, 3].* The vehicles which comprise the current expendable fleet are listed in Table 5.1 with some of their characteristics. The Atlas family of launch vehicles has been deleted since it received no mission assignments in the economic analysis for the period 1979-1990.

5.2.1 Small Payload Class

Two vehicle families, namely the Scout and Thor, comprise the current small payload class of launch vehicles. The Scout vehicle is available in both four stage and five stage configurations for launch of very small payloads. It has been in use since 1960 and has evolved into a reliable and versatile launch system. The cost per flight is low; however, because of its low payload capability, the cost per kilogram** (kg) in orbit is high. The basic Scout (Figure 5.1) is a multi-stage guided booster using solid rocket motors. The ALGOL III version, introduced in 1971, increased the performance of the Scout vehicle to more than 227 kg payload into low earth orbit, and the heat shield (payload shroud) diameter has been increased to 0.86 meters (m).

---

*Numbers in brackets identify References listed at the end of this Chapter.

**Metric units are used throughout this report. For conversion factors from metric to English units, see Appendix 5.1

Table 5.1

Characteristics of Current Expendable Launch Vehicles

N.B.: All data are approximate — References 1, 2, 3, 4

Launch Vehicle	GLOM(1) MT	Length(2) m	Dia., m	Payload Volume(3) m^3	Reliability
Scout (1971), Four Stages	18	23	1	0.5-1	~0.95
Scout (1971), Five Stages	18	23	1	0.3-0.4	>0.9
TAT (3C)/Delta	91	32.4	2.4	12	> .85
TAT (3C)/Delta/TE 364-3	91	32.4	2.4	12	> ,85
TAT (3C)/Delta/TE 364-4	118	32.4	2.4	12	> .85
Titan IIIB/Agena	167	47.5	3	40	> .95
Titan IIIB/Centaur	178	47.5	3	40	(> .9)
Titan IIIC - Veh. 26	644	44	3	85	> .95
Titan IIID	631	47.3	3	110	> .95
Titan IIID/Centaur	650	50	3	85	(> .9)
Titan IIIM	723	55	3	85	(> .95
Titan IIIF/Centaur	840	50	3	85	(> .9)
Titan IIIF/Centaur/Burner II	828	50	3	85	(> .9)

(1) 1 metric ton (MT) = 1000 kilograms (kg)
(2) Nominal overall with typical payload fairing
(3) Maximum with largest currently available shroud

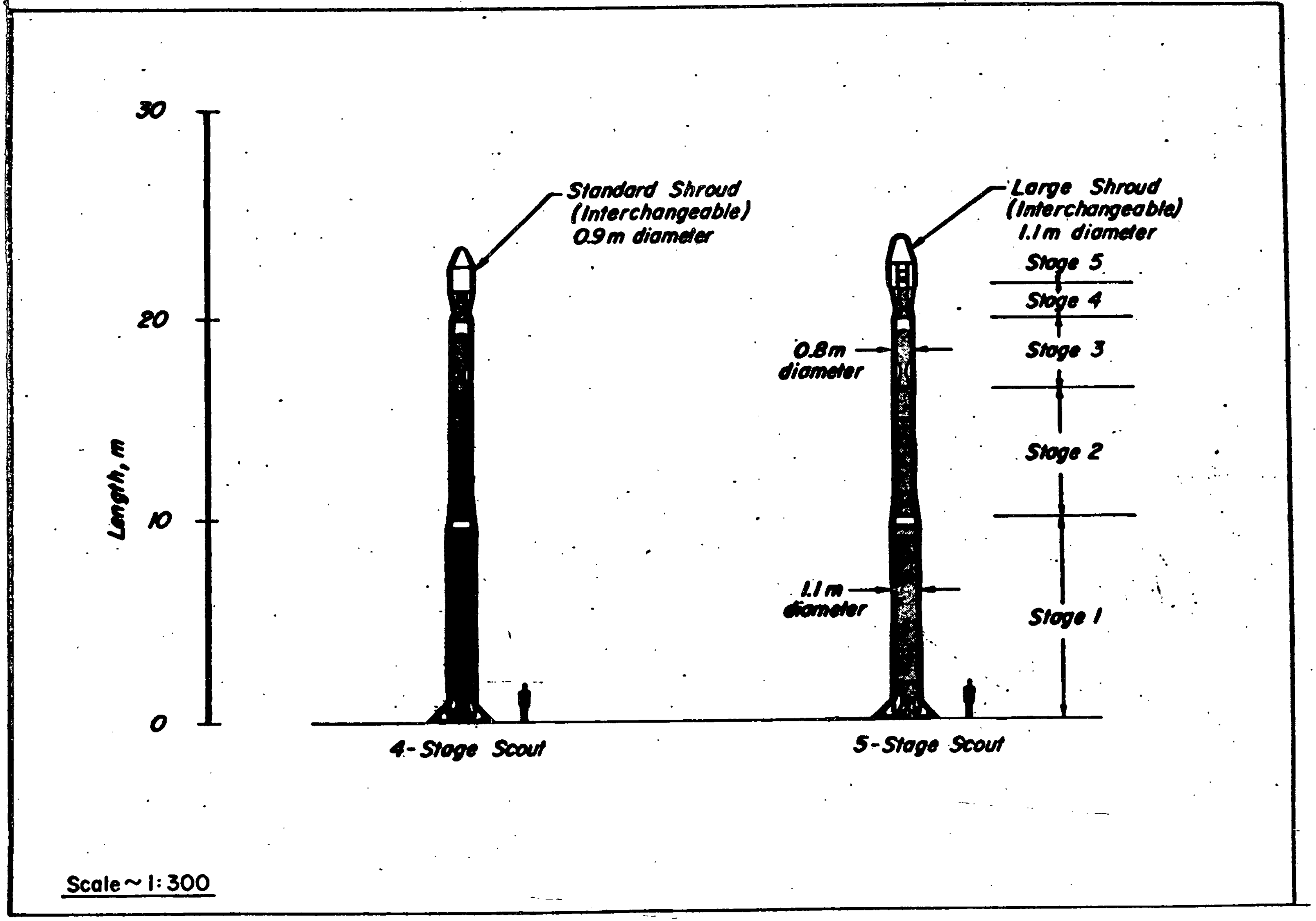


Figure 5.1 Current Expendable Launch Vehicles - Scout

Thor Family

The Thor family of launch vehicles was introduced in 1960 and by 1970 had successfully orbited 74 out of 80 payloads. The latest growth configuration of the first stage is the long tank vehicle (THORAD) which NASA refers to as the Thor Delta when mated with a Delta second stage. The McDonnell Douglas designation is DSV-2L or 3L. Another version employed by the USAF uses an Agena second stage with the Thor. In addition, NASA utilizes the designation TAT to refer to a Thrust Augmented Thor booster with Castor II Solid Rocket Motor (SRM) strap-ons. Third-stage solid rocket motors are available for use of the Delta vehicles and are designated as TE-364 and FW-4. The TE-364 and FW-4 are spin-stabilized high reliability stages adaptable to a wide variety of missions. Also, NASA model number designators such as Delta 300, Delta 602, Delta 903 and Delta 904 are frequently utilized. These identify the same multi-stage Thor Delta launch vehicle where, for example, Delta 903 is the Thor Delta vehicle with nine Castor III Solid Rocket Motor strap-ons and a TE 364-3 third stage.

The Castor strap-ons and the third stages use solid propellants and the Thor first stage Rocketdyne engine uses liquid oxygen as oxidizer and kerosene as fuel (LO_2/RP-1). The Delta second stage has used inhibited red fuming nitric acid and unsymmetrical di-methyl hydrazine (IRFNA/UDMH) propellants, whereas the latest Delta second stage uses nitrogen tetroxide as oxidizer and a fuel mixture of 50 percent UDMH and 50 percent Hydrazine, (N_2O_4/A-50).

Drawings representative of the Thor family are shown in Figure 5.2 and additional characteristics are given in Table 5.1.

The payload capabilities of the small payload class vehicles are shown in Figure 5.3. This figure gives payload (kg) as a function of ideal rocket velocity in kilometers per second (km/s). The payload mass for these and successive vehicles includes the spacecraft mass plus necessary adapters and accounts for a payload shroud mass, typical of that required for the payloads associated with the respective vehicles. The shroud, the mass of which is not included in the payload mass, is generally assumed to be jettisoned when the vehicle is sufficiently out of the atmosphere. This is typically taken to be at an altitude of about 120 km.

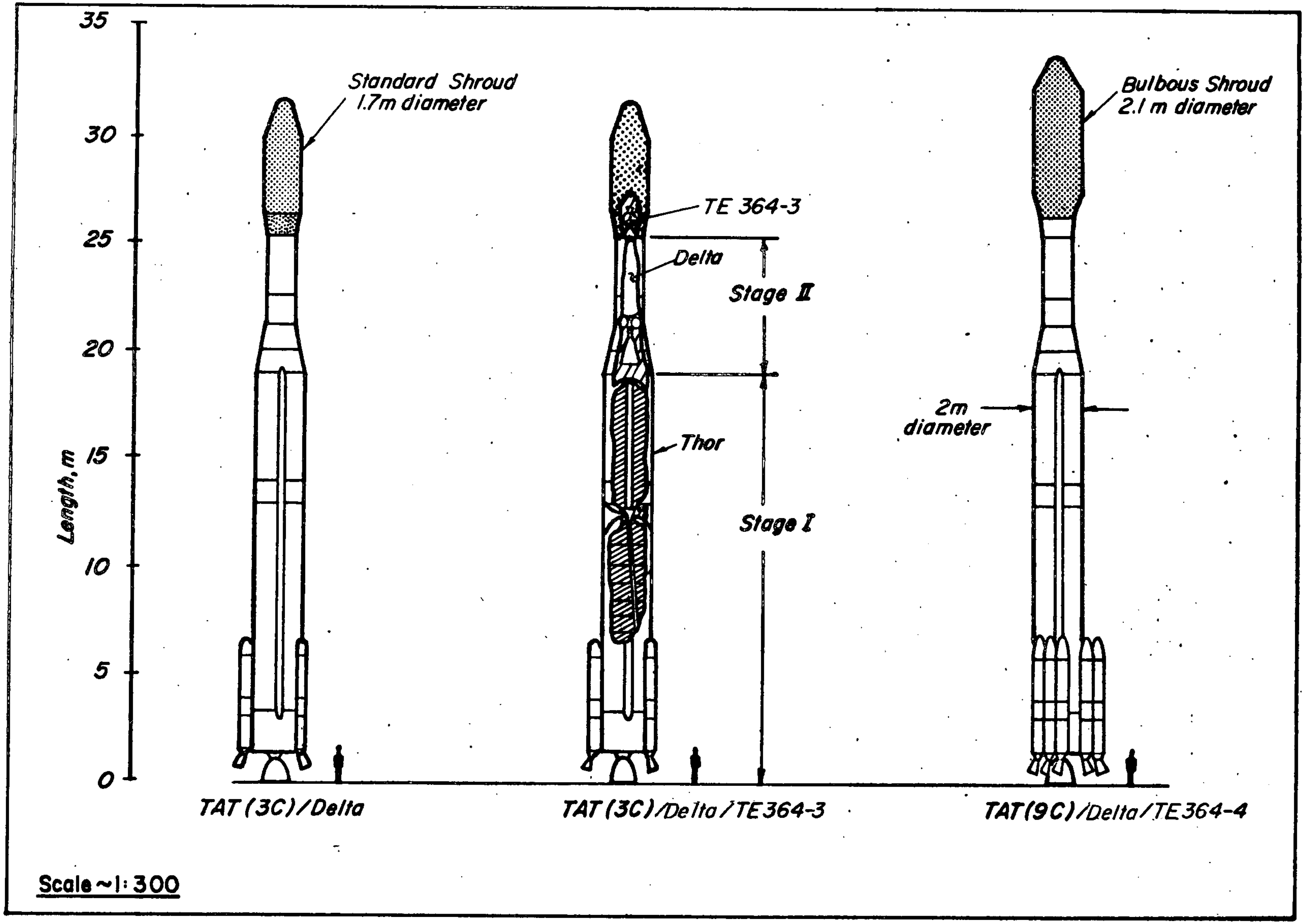


Figure 5.2 Current Expendable Launch Vehicles—Thor Family

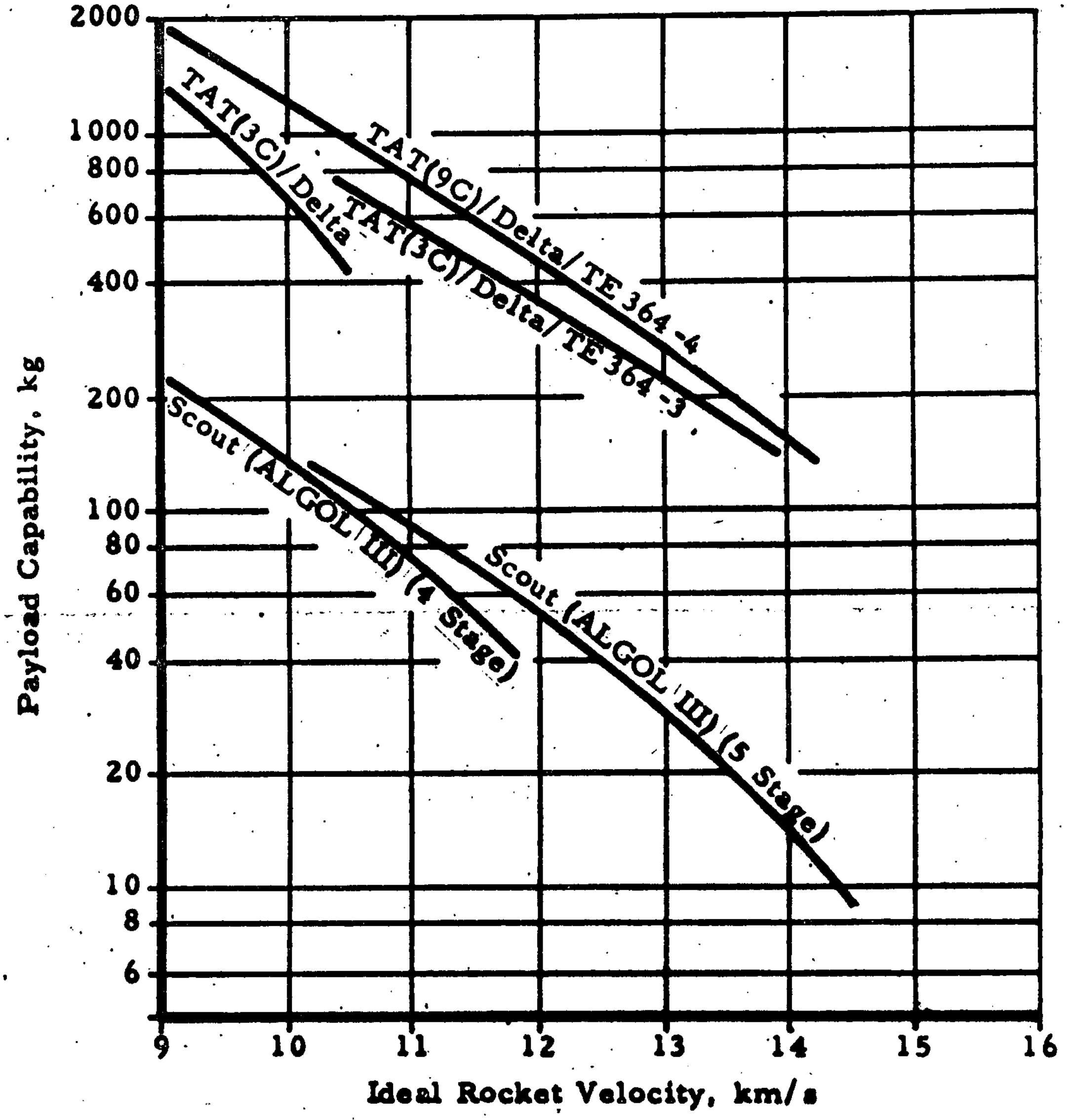


Figure 5.3 Scout and Thor Family Performance

The ideal rocket velocity is the velocity to which the rocket vehicle could accelerate a given payload from rest assuming that the flight occurs in a vacuum, in the absence of all gravitational forces and that the vehicle flies in a straight line. The payload capability of a rocket vehicle can be expressed as a function only of the ideal rocket velocity, independent of a mission. To relate the ideal rocket velocity to a particular mission it is necessary to determine the ideal mission velocity--the ideal, drag-free, infinite thrust, minimum velocity change required to perform a mission-- and the associated velocity loss for the mission. Then the ideal rocket velocity required to perform a mission equals the ideal mission velocity plus the velocity loss. A detailed discussion of ideal mission velocity requirements for a variety of missions is given in Section 5.8.1 together with a discussion on the velocity loss. This approach is taken partly to emphasize the extreme variability in vehicle performance due to variations in mission mode and orbital requirements. However, for typical missions that do not require orbital plane changes (also referred to as dog-leg maneuvers), the ideal rocket velocity required to achieve low earth orbit is between 9 and 10 km/s.

5.2.2 Medium Payload Class

The medium payload class of current expendable launch vehicles that receive payload assignments for the period 1979-1990 are comprised entirely of the Titan family. These vehicles were developed to satisfy various mission requirements of the United States Air Force. Development of the initial family vehicle, the Titan III C, began in 1962 and was followed by the III B, III M, and III D launch vehicle systems [1,5]. The above programs progressed to the point where various "building blocks" of the current Titan III vehicle were developed; the standard-two-stage core, the stretched core, the solid rocket motor strap-ons, the Transtage and from other development programs - the Agena, Centaur, and other upper stages. Combinations of the core, strap-ons and the various upper stages permit the building of many and varied configurations for a wide range of launch vehicle applications. Drawings representative of the Titan family of vehicles are shown in Figures 5.4, 5.5 and 5.6. The status of various building blocks (August, 1970) is given below [5] and the payload capability of the Titan

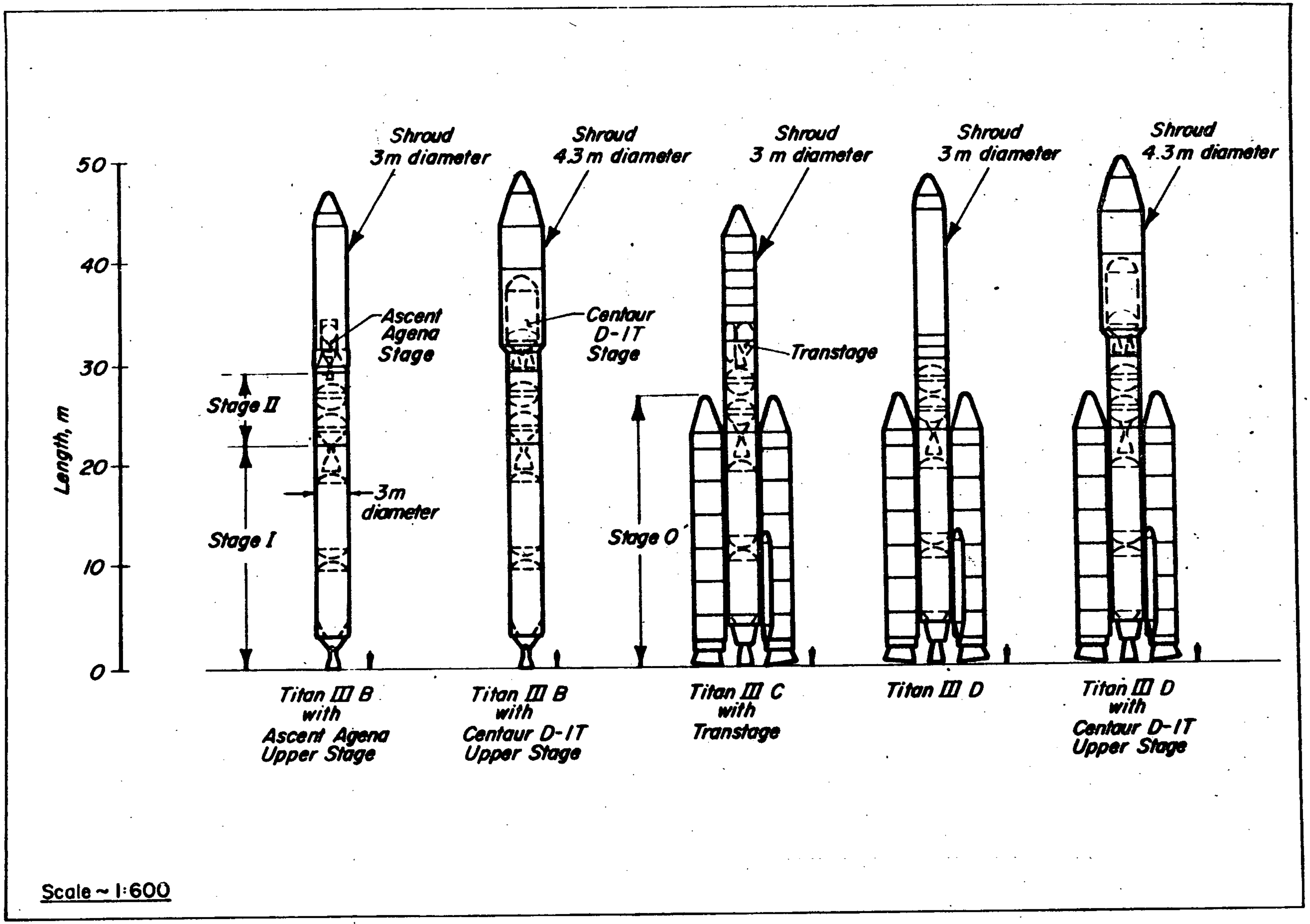


Figure 5.4 Current Expendable Launch Vehicles - Titan III B, C and D Family

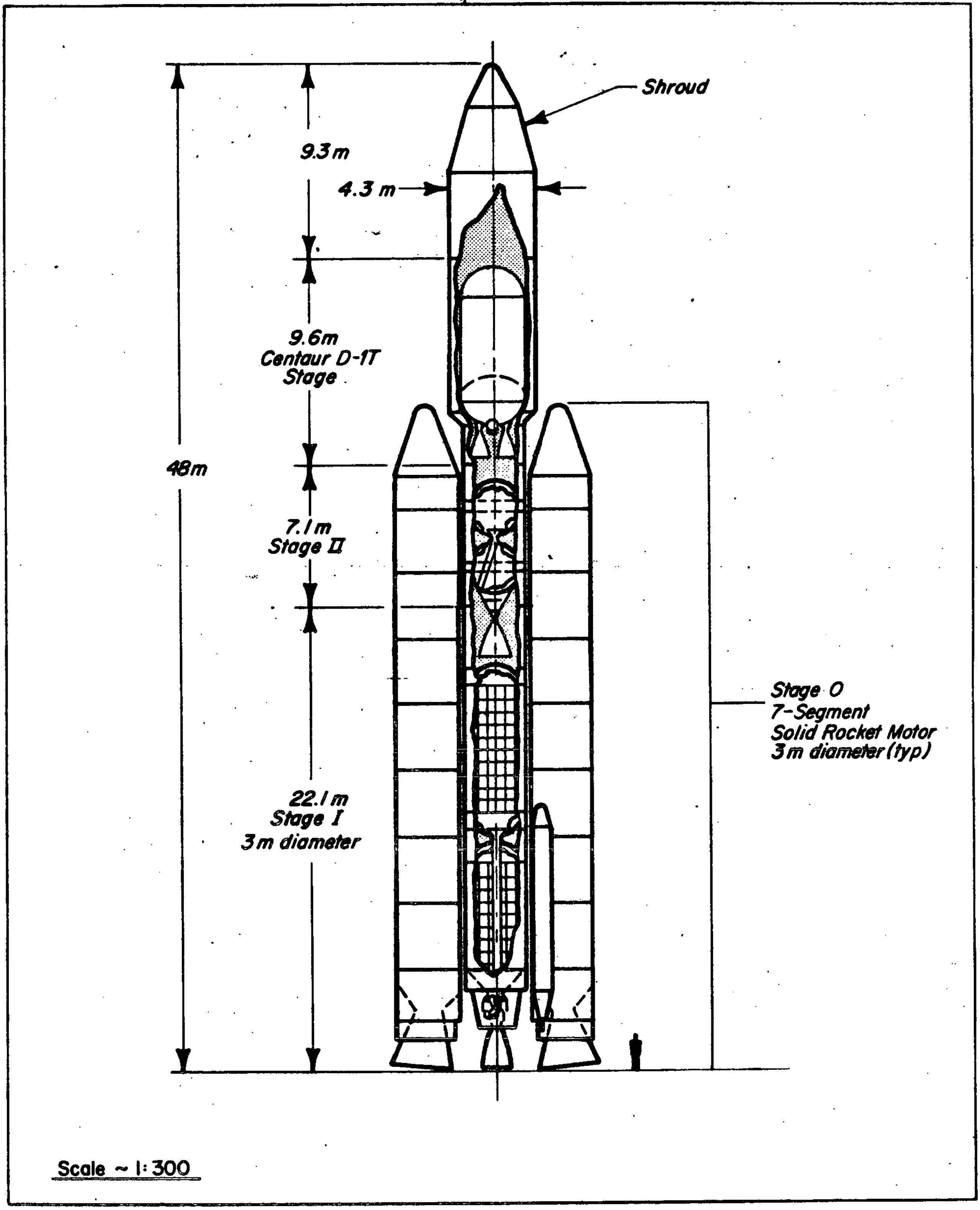


Figure 5.5 Current Expendable Launch Vehicles - Titan III F/Centaur

G

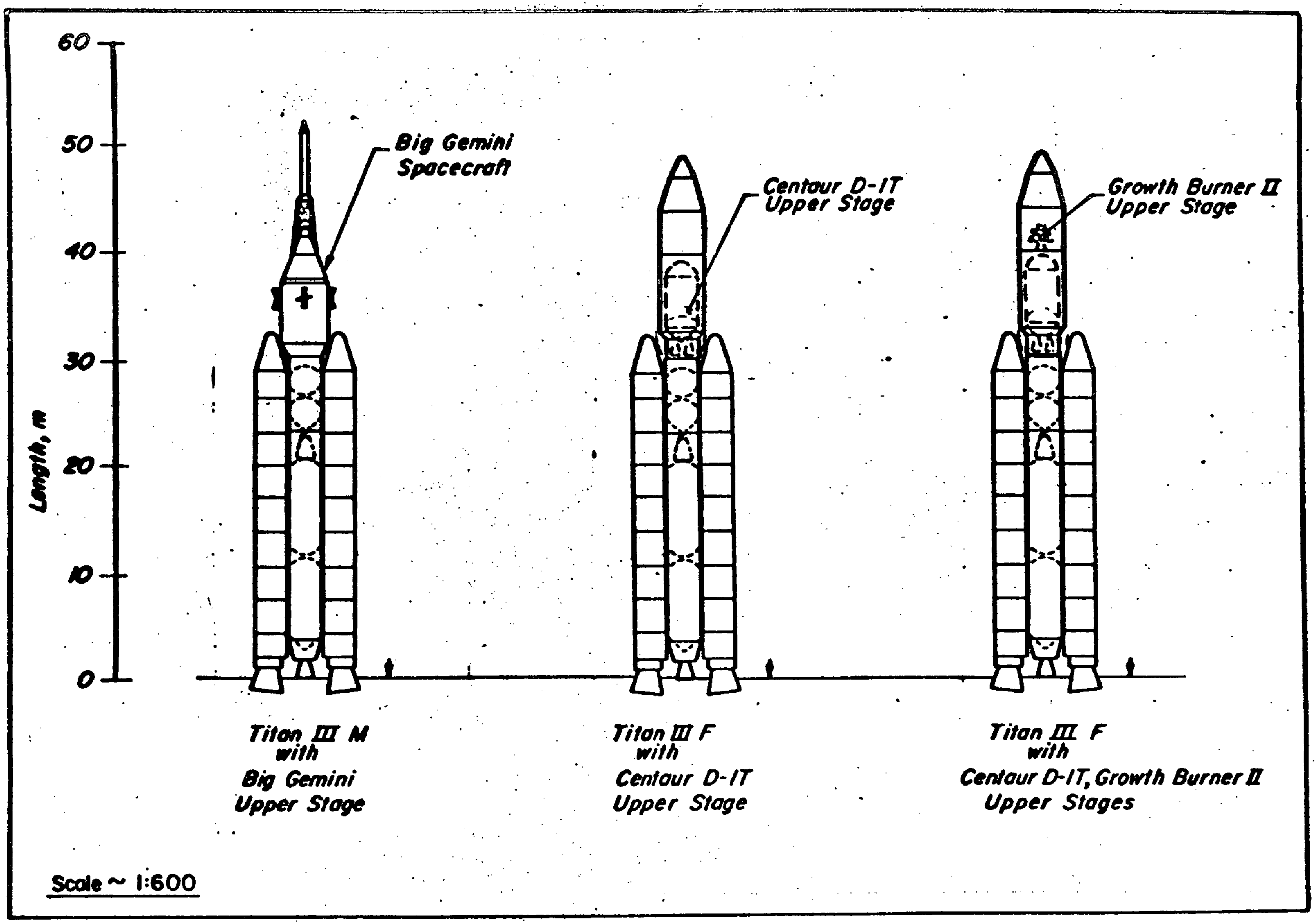


Figure 5.6 Current Expendable Launch Vehicles—
Titan III M/Big Gemini, F/Centaur and F/Centaur/Burner II

family is given in Figure 5.7.

Solid Rocket Motors, SRM Strap-ons (Stage 0)

The five-segment, three-meter diameter SRM is operational and is being flown on the Titan III C and Titan III D vehicles. Two five-segment motors are attached to these vehicles and provide approximately 10.2 meganewtons (MN) of sea level total thrust from liftoff to separation. Development of the seven-segment SRM, which is to be used for the Titan III M, is in progress. At least three demonstration test firings have already been completed. The ALGOL III SRM is currently being developed and may be used with the Titan III for some missions. Two static firings were made in 1970, and completion of the development program is scheduled in 1971.

Cores (Stages I and II)

In general, the Titan family of launch vehicles can be grouped according to their "core" cpnfiguration. The "standard core" group of Titan III vehicles includes the Titan III B, Titan III C and the Titan III D. The two stage, three meter diameter cores for these standard core vehicles are essentially the same except that the core structures for Titan III C and Titan III D have provisions for the attachment of solid rocket motors (SRM's). Also, the Stage II top structure is common for the Titan III B and III D but is unique for Titan III C due to upper-stage interface requirements. All Titan III cores use propellants that can be stored in a launch-ready vehicle for extended periods of time. The oxidizer is nitrogen tetroxide and the fuel is a 50-50 mixture of hydrazine and unsymmetrical dimethylhydrazine (UDMH). The core propulsion systems of both stages are essentially the same in the three vehicles and the engines (two LR-98's in Stage I and a LR-91 in Stage II) are gimballed for thrust vector control.

Standard core propulsion for Stage I has approximately 2.07 MN sea level thrust for the Titan III B and 2.33 MN vacuum thrust for the Titan III C and Titan III D. Standard core propulsion for Stage II has approximately 454 kilonewtons (kN) vacuum thrust for all three Titans.

Upper Stages

Although these vehicles may be launched without upper stages,

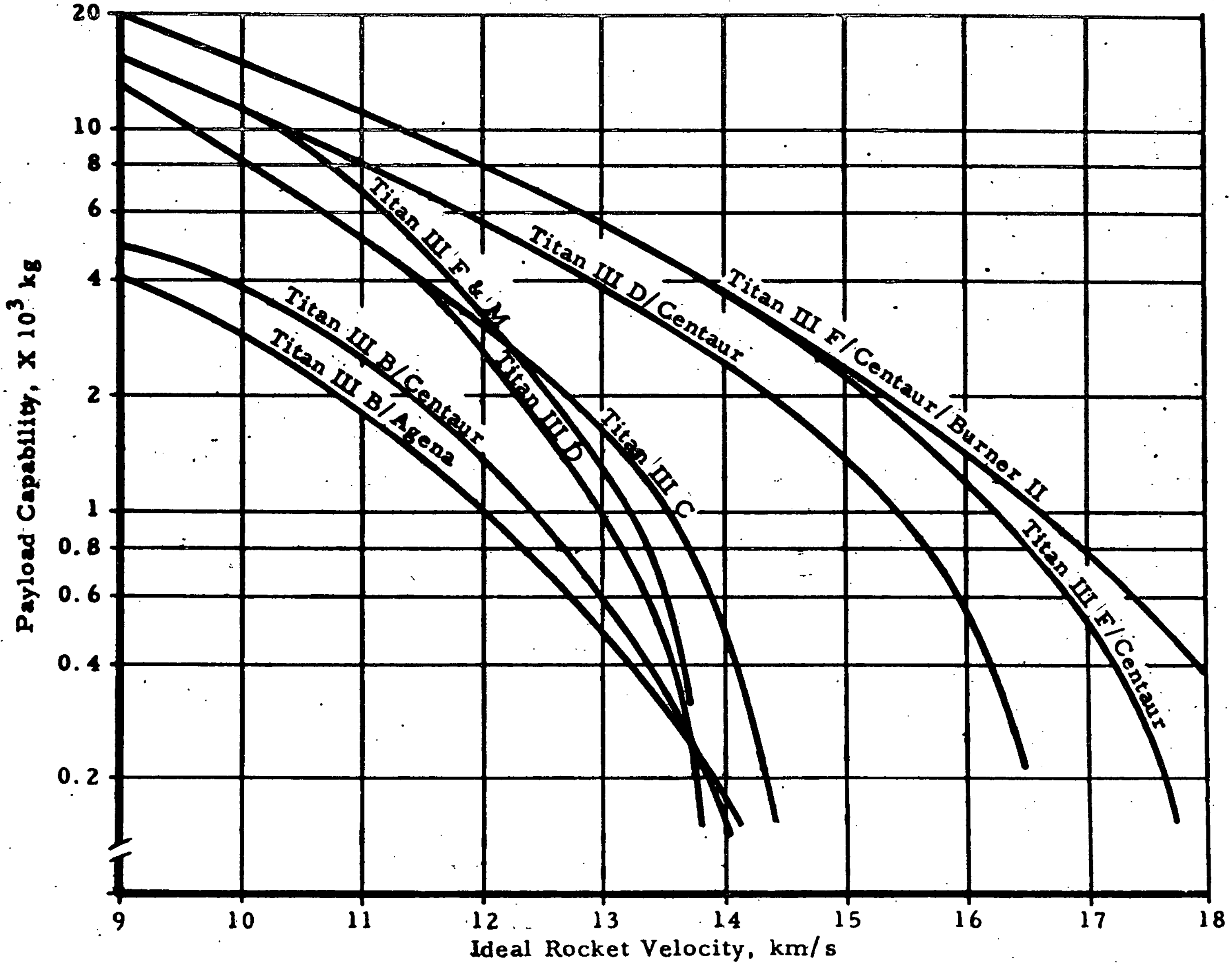


Figure 5.7 Titan Family Performance

various "building block" upper stages are available.

The Transtage is an integral stage of the Titan III C and consists of two major components. The first is a propulsion module that uses the same propellants as the core vehicle and the second is a control module which includes a gimballed three-axis inertial unit, computer and guidance and control equipment. The Transtage is specially designed to provide maximum flexibility in delivering a wide range of payloads including those scheduled for synchronous orbit.

The Agena vehicle is currently operational and has been integrated with the Titan III B for military missions at WTR. This upper stage has an improved version, the Ascent Agena, which is utilized to provide higher altitude performance.

The Centaur upper stage is operational and is presently being integrated with the Titan III D for the NASA Viking program. This will result in integrated Titan III/Centaur launch facilities at ETR.

The standard Burner II upper stage is operational and utilizes a TE 364-3 spherical SRM. The Growth Burner II will use a TE 364-4 SRM that is currently under development. This motor is a growth configuration of the TE 364-3 motor and can be integrated with Centaur to serve as a kick stage for placing a payload into final orbit.

The Tandem Burner is a modification of the Growth Burner II giving a two stage vehicle with increased performance. Use of the Tandem Burner would require integration with the standard or stretched core or with a Centaur upper stage.

The three "standard core" vehicle configurations are listed below and additional data are presented in Table 5.1 above.

<u>Titan III B</u>

The Titan III B is a standard two-stage core vehicle. Third (and fourth) stage options include Burner II, Tandem Burner, Centaur, Centaur/TE 364-4, Centaur/Burner II and Agena. Liftoff mass (less payload) varies from 156,038 kg for the two-stage vehicle to over 173,729 kg for the four-stage vehicle.

Titan III C

The Titan III C is a four stage launch vehicle consisting of three stages with liquid propellant motors (Stages I, II, and Transtage) and an initial stage (two SRM's, five segment) of solid propellant motors (Stage 0). Also, a Growth Burner II fifth stage can be used to obtain additional performance. Liftoff mass (less payload) for the Titan III C is approximately 631 metric tons (MT).

Titan III D

The Titan III D (which has no Transtage) is a three-stage launch vehicle with two five-segment SRM's (Stage 0) and a standard core (Stages I and II). Optional upper stages are about the same as the Titan III B options, except that the Tandem Burner or Growth Burner II may be used with a Centaur upper stage to make a five stage vehicle. Liftoff mass (less payload) for the Titan III C varies from 618 MT, with no upper stages to 637 MT for five stages.

The "stretched core" Titan series of launch vehicles include the three stage, Titan III M and Titan III F and are similar to the Titan III D except that Stage I is lengthened to hold about 14,515 kg of additional propellants and new Stage zero attachment points for the longer seven-segment SRM's have been provided.

Titan III F

The Titan III F three stage vehicle (also referred to as the Titan III D7 in Reference 1 may be used with or without upper stages. An optional fourth stage is the Centaur and the Burner II may be used as a fifth stage. The liftoff mass varies from 804 MT for no upper stage, to 822 MT for the five stage launch vehicle.

Titan III M

The Titan III M is identical in appearance to the Titan III F, except that it uses no upper stages and is specifically designed for use with manned spacecraft as in the Big Gemini (Big G) configuration shown in Figure 5.6. The Titan III M gross liftoff mass is about 823 MT.

The Titan III M was carried well through the development stage by the U.S. Air Force for their Manned Orbiting Laboratory (MOL) program. The proposed Big Gemini capsule can carry nine men to orbit and return and it also contains a cargo compartment. The characteristics of the Titan III M/Big Gemini configuration are listed in Table 5.2.

5.2.3 Large Payload Class

The large payload class of launch vehicles consists of those vehicles capable of placing over 20000 kg into low earth orbit. Only one vehicle of the current expendable launch vehicles of the large payload class has been assigned any payloads. This is the Int. 21 which is based on the Saturn family of vehicles and can place 115000 kg into 185 km orbit via an easterly launch from ETR or 99000 kg into a 185 km polar orbit via a southerly launch from WTR. Based on the very limited flexibility and applicability of this vehicle it is quite unlikely that it will be used in the 1980's.

5.3 Space Transportation Systems Related Technology Status

Successful development of the space shuttle depends on the status of several technologies. This section deals with those technologies which are most important.

5.3.1 Materials

The development of a new Space Transportation System provides a number of opportunities in the research and development of new materials and their processing that will surely have wide impact on future technology. Although the STS will cause focusing of efforts on specific materials with concomitant shifting of interest in some directions, the problems are sufficiently broad and essential that the directed attention and funding will result in an overall gain to high technology.

Improvements in materials and in their availability in the following broad categories can be expected: high temperature structural and protective materials; composite materials including ceramics, cermets and filaments; high performance insulations; optical materials; high and low temperature lubricants and hydraulic fluids; and specialty materials.

Table 5.2

Characteristics of Titan III M/Big Gemini Configuration

Reference 5-6

Big Gemini Spacecraft		
Spacecraft Crew Size (Number aboard)		9
Spacecraft Mass Breakdown:		
Crew/Passenger Module		6 000 kg
Cargo/Propulsion Module		7 000 kg
Cargo (up)		3 000 kg
Total Liftoff Mass		16 000 kg
Launch Vehicle		Titan III M
Stage 0:		2 SRM's (3.05 m)
Propellant Mass, each	269 000 kg	
Loaded Mass, each		316 000 kg
Stage I:		2/LR-87
Propellant Mass	132 000 kg	
Loaded Mass		140 000 kg
Stage II:		1/LR-91
Propellant Mass	31 000 kg	
Loaded Mass		34 500 kg
Liftoff Mass		806 000 kg
GROSS LIFTOFF MASS		822 000 kg

In addition, space activities will permit research and development on classes of materials in zero gravity and hard vacuum that could never be produced on the Earth's surface. Precise metallic shapes, especially spheres, and very large single crystals are examples that will certainly receive early attention.

It is metal technology that will probably receive the greatest early boost from the development of a new STS and substantial efforts should be continued. Superalloys, dispersion-strengthened alloys (particularly thoria dispersed nichrome), and refractory alloys should be strongly advanced toward common usage. In this general category are alloys of titanium, tantalum, niobium, beryllium, molybdenum, rhenium, tungsten and others that are at present insufficiently developed and/or highly expensive.

Associated technology in research and development testing techniques, production methods and non-destructive inspection will certainly benefit greatly.

5.3.2 Propulsion Systems

The development of new reliable and economical Space Transportation Systems relies heavily on the availability of high performance and/or low cost propulsion systems ranging from high chamber pressure O_2/H_2 motors through advanced, recoverable solid or liquid propellant rocket motors. Generally, the candidate systems reflect the present state-of-the-art adapted to new uses with a correspondingly high confidence level in RDT&E as well as initial fleet and operating cost estimates. However, certain concepts, for example, large pressure fed liquid rocket motors and recoverable stages, involve the initial usage of significantly new technologies and techniques. The candidate propulsion systems reviewed in this section are grouped according to task and, where applicable, comparisons are made with existing systems.

5.3.2.1 Orbiter Propulsion

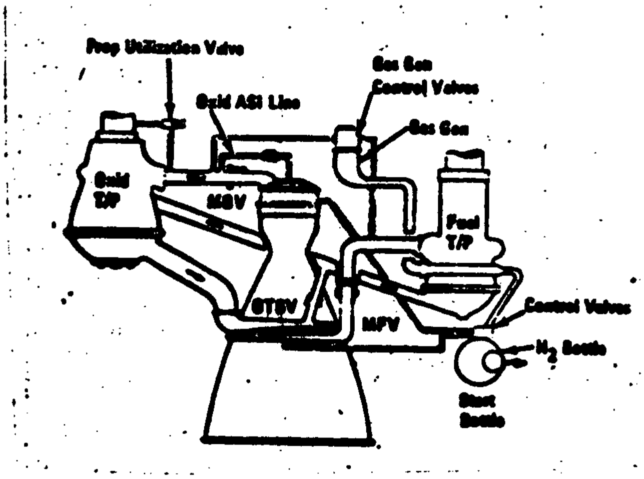
Main Propulsion System

In the STS alternate concepts under consideration until November, 1971, the orbiter main propulsion requirements cover a range of thrust and involves both standard and high chamber pressure designs. All typical designs in contention are based on oxygen-hydrogen technology and include the J-2 [7] (second and third stage propulsion for Saturn V), the J-2S [7] (an upgraded version of the J-2), the XLR 129 (a high chamber pressure engine that was being developed for the Air Force [8, 9] and the SSME, a high chamber pressure engine designed for the originally conceived orbiter (with the possibility that it may be somewhat downrated in thrust for the present orbiter [10]). Figures 5.8 and 5.9 illustrate these four candidates.

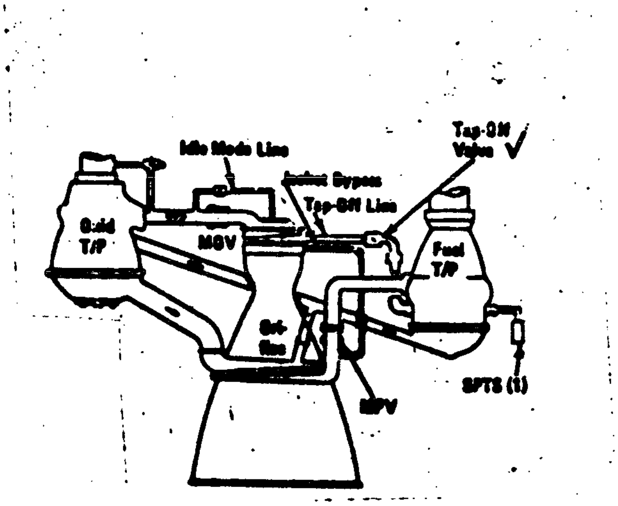
The J-2 engine, since it represents the minimum modification to an existing design, results in the lowest RTD&E cost with the least thrust and performance (see Table 5.3). The J-2S raises this thrust and vacuum effective jet velocity through an increase in chamber pressure. The several modifications shown point to performance approaching that of the last two candidates. The two high chamber pressure designs shown in Figure 5.9 bring about a substantial increase in the vacuum effective jet velocity (Table 5.3) and since each ten meters per second of exhaust velocity represents an allowable orbiter burnout mass increase of approximately 500 kg [9, 10], the additional cost required to develop such engines has a considerable payload capability benefit in the 1980's.

Another important cost consideration for the 1980's is the number of reuses, and the interval between overhauls, for each of the candidates. The choice of materials for critical components in the candidate engines is made with reusability as a criterion. Only the SSME and the XLR 129 engines were originally conceived with the important requirement of 100-mission capability.

In addition to performance considerations, the internal envelope and operational requirements of such orbiter designs as the 040A pose specific problems of interfacing the engines to the vehicle. As seen in Table 5.3 the J-2S versions have nozzle area ratios from 40 to 105 and hence size is affected. Gimbaling requirements, feed line configurations and other details of the interface design can result in important changes in the positioning of the engines and thus influences such factors as the re-entry heating of the



J-2



J-2S

Figure 5.8 Schematic Diagrams of the J-2 and J-2S Rocket Engine Systems

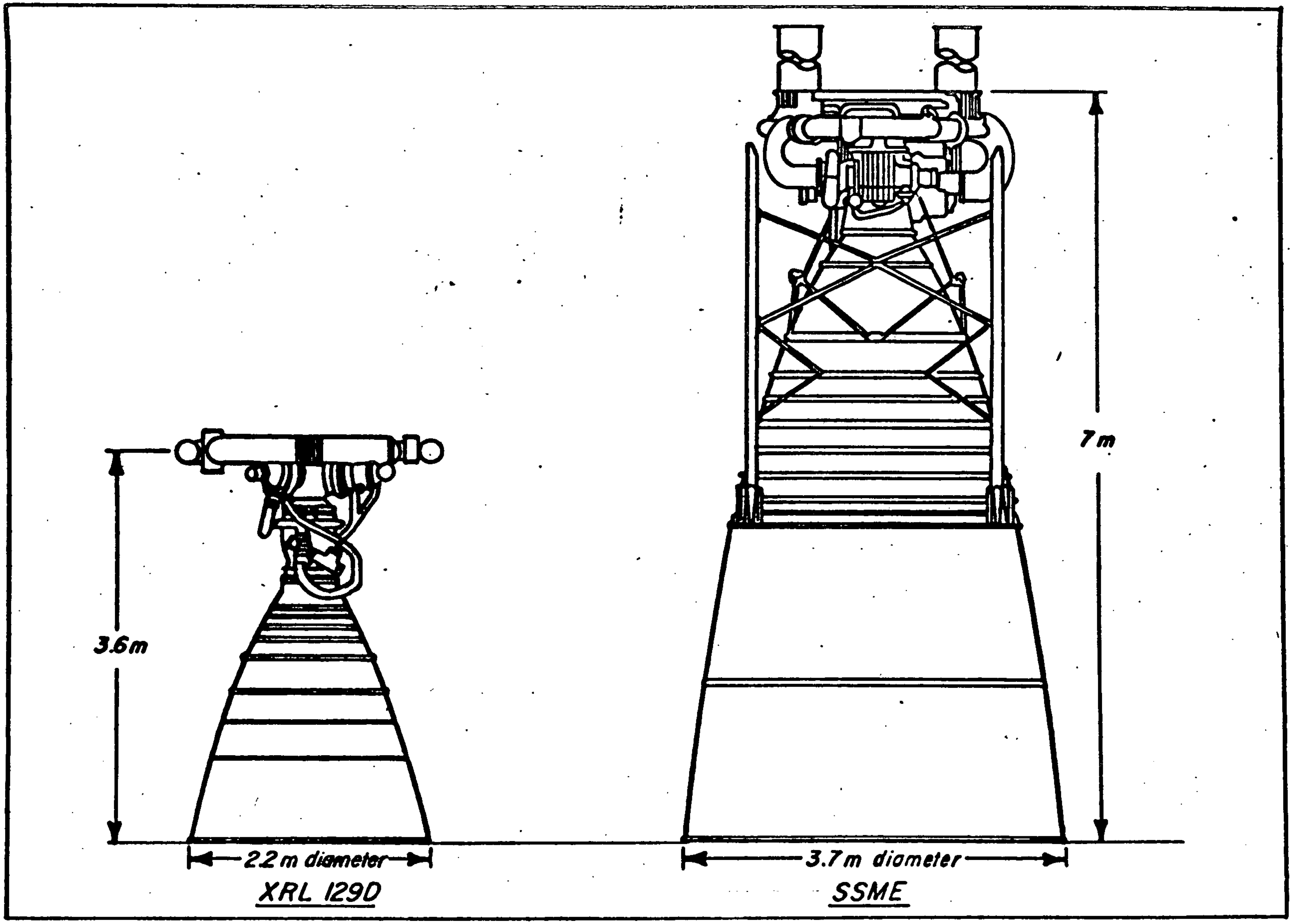


Figure 5.9 High Chamber Pressure Rocket Engines

Table 5.3

OMS Main Engine Candidate Data

Engine	J-2	J-2S			XLR 129D		NASA Spec. SSME
Thrust, MN	1.023	1.179	1.212	1.223	1.200	1.334	2.81
Chamber pressure N/cm^2	493	807	807	807	2110	2070	2070
Nozzle area ratio	27.5:1	40:1	80:1	105:1	120:1	105:1	150:1
Propellants	Liquid oxygen/liquid hydrogen						
Exhaust jet vel, m/s	4,165	4,270	4,393	4,437	4,470	4,459	4,498
Engine mass (dry), kg	1570	1,727	1,707*	1,707*	1,629	1,629	4000
Engine length, m	295	295	437	498	355	355	704
Engine diameter, m	203	203	284	325	221	221	373
Reliability	9.99% at 50% confidence level						not specified

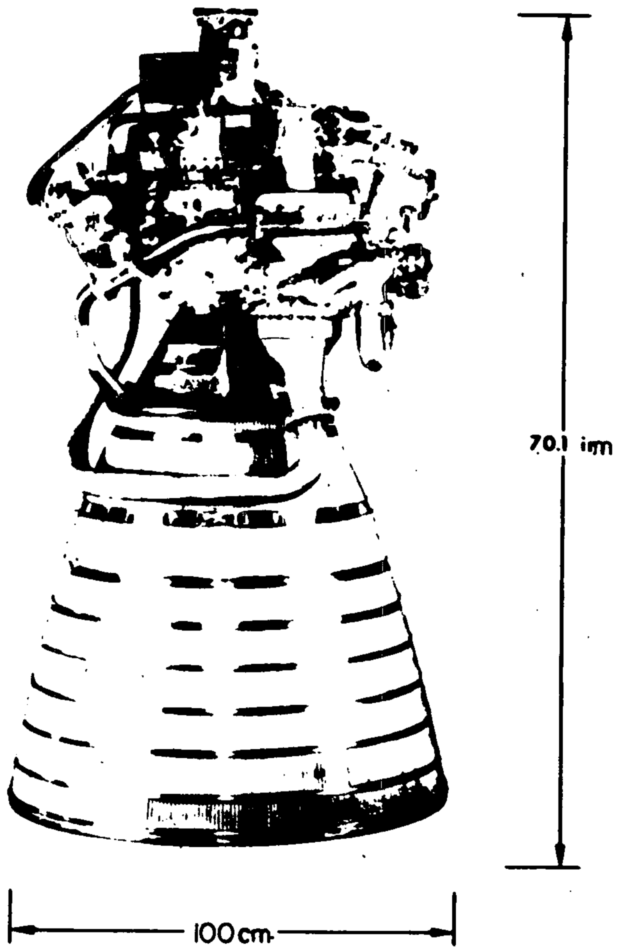
*Mass reduction based on redesigned engine nozzle [7, 9]. The nozzle designs, including both retractable - extendable nozzles and jettisonable nozzle extensions (for low altitude abort) have been investigated to meet performance and safety requirements [7, 8; 9]. Designing from an engine standpoint, all these factors directly influence the vehicle engine compartment size.

Table 5.3 also compares the Pratt and Whitney XLR-129 and Rocketdyne SSME. The former data represent a lower thrust scale than the latter NASA specification for the SSME engine which Rocketdyne has exceeded [7]. The choice to proceed with the high-chamber pressure engine development has important implications for the booster as well as the orbiter performance in the longer term. The availability of such an engine will permit application to such concepts as a future high performance booster core. Strap-on, lower performance boosters (either SRM or LRM designs) for heavier payloads could then be added. Also, the future commitment could readily change to the high performance completely reusable concept.

In viewing the reliability of these orbiter main propulsion candidates the test history varies widely. The J-2 has seen some 410,000 sec. of operation as a single engine plus 419 additional vehicle-cluster tests resulting in a 99.9 percent reliability at the 50 percent confidence level [7]. J-2S and XLR-129 testing has amounted to several hundreds of seconds, whereas the SSME has undergone several short duration tests. The advancements incorporated in the J-2S should eliminate some potential failure modes resulting in still higher reliability. The higher RDT&E cost estimates used in the economic analysis for the SSME reflect the need for a longer testing program to gain the same expected reliabilities as the J-2, J-2S and XLR-129 engines.

Orbit Maneuvering System (OMS)

As in the APS section that follows, the STS alternate concepts have drastically changed the OMS propulsion requirements. A prime OMS engine candidate for the larger versions of the STS orbiter has been the RL-10. That engine, together with its performance characteristics, is shown in Figure 5.10. Designed to use oxygen-hydrogen propellants with resulting high performance, this mature engine has been fired more than 9000 times,



Vacuum Thrust, N	66,720
FSpective Jet Velocity, m/s	4,351
Expansion Ratio	57
Thrust to Mass Ratio, m/s^2	510
Mixture Ratio	5
Restarts	Minimum 3
Mass, kg	132
Service Life, s	4,000

Figure 5.10 RL10A-3-3 Performance Characteristics

C-2

and 74 engines have been flown on Saturn and Centaur vehicles with 100 percent flight reliability. The RL-10 has been proposed for use in the space shuttle orbiter in its present or a modified form [11]; however, it has recently lost favor since almost all cryogenic propellants have been removed from the orbiter in its current baseline concept.

The current leading choice to satisfy the revised OMS propulsion requirements is the LM ascent engine. Table 5.4 summarizes the performance of this unit for both the C and E models [7]. The propellant combination is the earth storable nitrogen tetroxide and the 50-50 blend of hydrazine and UDMH run at low chamber pressure. The chamber and nozzle are of ablative design. Instability suppression devices are incorporated in the injector. As in the case of the RL-10, the LM ascent engine is a thoroughly tested, highly reliable unit.

Reaction Control System (RCS)

Originally the plans for the reusable STS called for a large number of gaseous oxygen and gaseous hydrogen auxiliary propulsion system units aboard the orbiter and booster. The required number of units, the thrust, and the propellant selection have all been changed in considering alternate STS concepts. Typical of the present thinking on the performance requirements for this rocket (now termed reaction control system, RCS) are the specifications [12] shown in Table 5.5. A representative rocket engine assembly is shown in Figure 5.11. Technology for such designs is firmly based on applicable past experience such as the main engine for the Mars Mariner '71, and the RS-14 engine for the Minuteman II [12]. The beryllium rocket engine has been qualified for both NASA and Air Force use, that includes an "off limits" test history encompassing propellant flooding, saturated propellants, cold starts, throttle down to 15 percent thrust, and bomb rating for combustion instability. In this engine class, hundreds of units have been produced with thousands of starts and hundreds of thousands of seconds operation [13].

Because of the long service life and the number of operating cycles required of the RCS, two competitive designs rely on columbium rather than beryllium based on superior ductility with temperature cycling [14, 15].

Table 5.4

OMS Engine Data (LM Ascent)

Model Number	RS-1801C	RS-1801E
Thrust (vacuum), N	15,600	15,800
Chamber pressure, N/cm^2	82.5	82.5
Nozzle area ratio	45.6:1	80:1
Propellants	N_2O_4/50% N_2H_4 — 50% UDMH	
Exhaust jet velocity, m/s	3041	3079
Engine length, cm	132	180
Engine diameter (nozzle), cm	84	107
Engine mass (dry), kg	77.7	98.2
Reliability	99.8% at 50% confidence level	

Table 5.5

Performance Requirements for the Current Baseline Orbiter Reaction Control System Engine

Thrust	3340 N
Chamber pressure	120 N/cm^2
Expansion Area ratio	40:1
Propellants	N_2O_4/MMH
Mixture ratio (O/F)	1.6
Exhaust jet velocity	2,840 m/s
Minimum Impulse Bit	220 N/s
Response	50 m-s
No. of pulses/mission	3,000
*Total No. of pulses	100,000
Accumlative burn duration/mission	200 s
*Total service life	5 hours
Backwall temp.	700 ^oK

*Added items from References 5-14, 15.

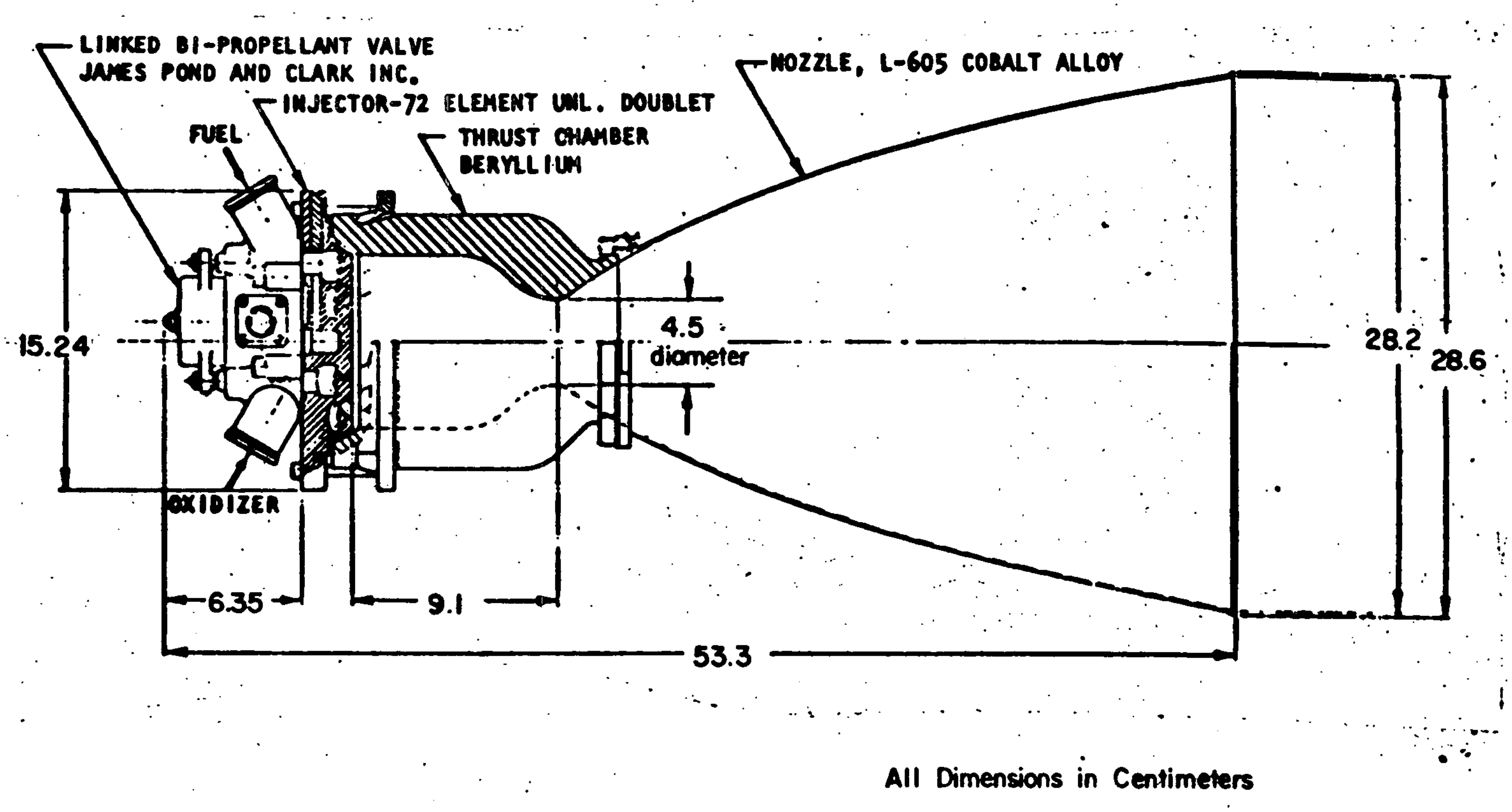


Figure 5.11 Space Shuttle Current Baseline Orbiter Auxilliary Propulsion System

Also being considered is a substitution of the 50-50 blend of hydrazine and UDMH for MMH. Still another approach would utilize a monopropellant RCS design; however, here one problem appears to be a question as to the catalyst life with this mission cycle. Final selection of the RCS design awaits the conclusion of a number of studies now in progress.

Airbreathing Engine System (ABES)

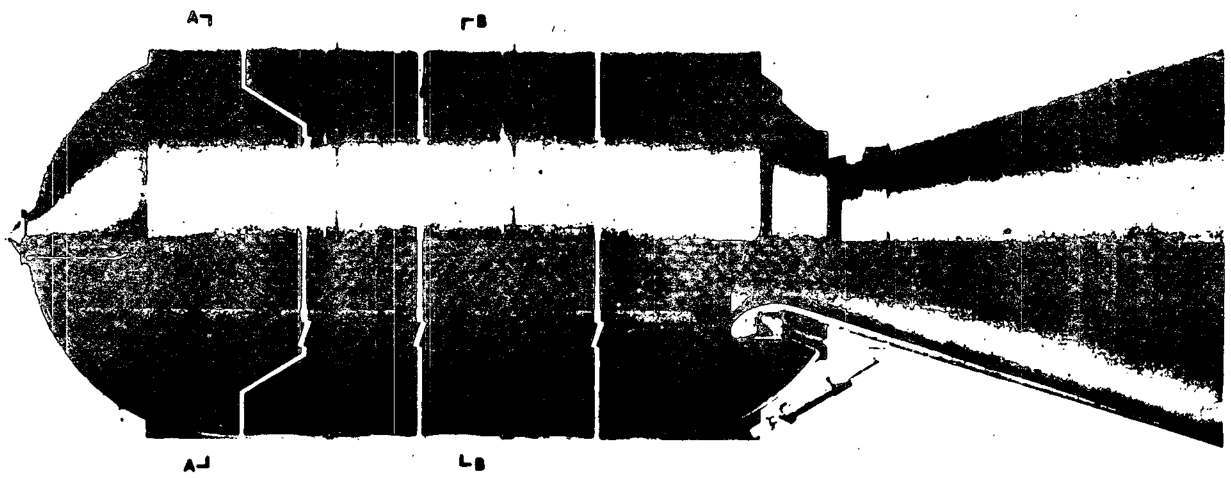
In the STS alternate designs the ABES plays a lesser role than in the all reusable shuttle where ABES are in both the orbiter (four engines) and booster (10 engines). In one alternate shuttle design [16] two such engines are suggested for use in the orbiter. Candidate engines are the GE F101/F12B3 and the PWA STF22A-6 (Mod III). Since both engines have been developed for military aircraft, the B-1 and F14B, performance is classified. The emphasis in the ABES application is the space-rating requirement and that units be removable with low support structure mass in the orbiter. On missions that require the ABES, the penalty for engine and propellant mass subtracts directly from the payload.

5.3.2.2 Booster Propulsion

Solid Rocket Motors (SRM)

The SRM family of booster designs is representative of proven and available state of the art propulsion. Data from the four large solid rocket propulsion companies on the performance of 3.04 and 6.6 meter diameter boosters has recently been assembled [17]. The purpose of the study was to consider configuration and programming options that would lower peak annual funding requirements through an interim expendable SRM booster. SRM performance and design data assembled in the study were in close agreement.

A typical SRM design is shown in Figure 5.12 where the simplicity of solid rockets is quite evident [18]. Mass fraction (the ratio of propellant mass to overall motor mass) averaged close to 0.9 for the three sizes of boosters. Effective jet velocity was approximately 2646 m/sec for vacuum conditions and 688 N/cm^2 chamber pressure. Each size booster has been tested by one or more of the four companies, with the majority of



Overall Length	33.89 m
Maximum Case O.D.	4.01 m
Gross Mass	677,373 kg
Propellant Mass	621,500 kg
Mass Fraction	0.916
Average Thrust	11.56 mN
Burn Time	140.0 s
MEOP	690 N/cm^2

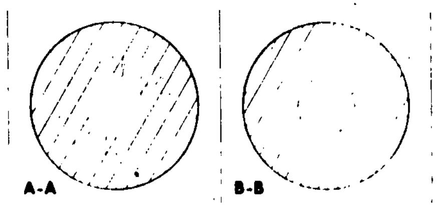


Figure 5.12 Typical Large Solid Rocket Motor—4 Meter Diameter

experience gathered with the highly reliable three meter designs. In the current design study [17] PBAN (polybutadiene/acrylic acid/acrylonitrile) type propellant was employed by three companies, whereas the fourth company used HTPB (hydroxyl-terminated polybutadiene). The thrust vector control approach of using a flexible seal nozzle represents one area where additional development is necessary to optimize the final designs. One aspect of best utilizing SRM boosters, covered only briefly in the report [17] was that of recovery of a unit from the sea after a parachute descent. A high confidence level of retrieval without damage is the conclusion of two previous studies [19, 20]. Reference [20] predicted cost saving of 35 percent over the cost of a new unit. One factor of importance in SRM costing is the price of the processed propellant. This is cited as four or more dollars per kg [17, 18], compared to the cost of raw materials (in large quantity purchases) averaging 66 cents per kg [21]. Table 5.6 summarizes the SRM designs using Reference 18 data.

Liquid Rocket Motors (LRM)

Of the various LRM designs the one that most closely rivals the SRM in simplicity is the pressure-fed booster. Shown in Figure 5.13 design eliminates the sophistication of many LRM units through minimum cost design [22]. Number of components are minimized and simple; conservative design allows maximum use of industrial fabrication methods. In contrast with the full thrust test history of the SRM, the LRM pressure-fed booster to date has been fired at one-fourth of the 5.34 mN thrust level. In those tests no instability problems were evident.

Performance for the LRM is somewhat higher than the SRM designs even though chamber pressure is designed to be only 172 N/cm^2 (one-fourth that of the SRM. see Table 5.6). For the liquid oxygen/liquid propane propellant combination vacuum effective jet velocity is 2770 m/s at 90 percent efficiency [23, 24] and 6.78 mN thrust. This value improves by one percent when thrust is doubled or drops slightly if an ablative chamber design is used rather than duct cooling approach. Duct cooling [22] provides an internal duct open at the nozzle end of the chamber to achieve chamber cooling and downstream barrier gas protection. Such departures from the more delicate regenerative cooling designs are deemed necessary to insure intact

Table 5.6

Booster Motor Data

TYPE	SRM			LRM		
MODEL	3m	4m	6.6m	Pressure-fed	Pump-fed F-1	
Thrust, MN (vac)				6.68	7.40	7.77
Thrust, MN (ave)	6.70	11.56	32.03			
Thrust, MN (S. L)				5.34	6.77	6.77
Chamber pressure, N/cm^2	690	690	690	172	677	677
Nozzle Area Ratio	10:1	10:1	10:1	6:1	10:1	16:1
Propellants	PBAN			LOX/Propane	LOX/RP-1	
Exhaust jet vel (vac) m/s	2,644	2,649	2,622	2,770	2,836 (*2,881)	2,987
Exhaust jet vel (S. L.) m/s				2,181	2,595 (2,636)	2,601
Engine mass (dry) kg***	282,109	677,373	1,981,984	~ 6,590	7,559	8,427
Engine mass (burnout) kg	24,836	55,873	163,773		8,376	9,234
Mass fraction	.912	.916	.916	.846	not applicable	
Reliability				**	99.8% at 50% confid.	

* Considers new 80% nozzle design

** Presently under study

*** Total motor mass (with propellant) for SRM recovery from the sea of the LRM booster. As Figure 5.13 illustrates, valving and components are minimized to make the unit better able to withstand impact and eliminate water damage.

Mass fraction is approximately 0.85 in the pressure-fed LRM design, which is less than the SRM. Propellant costs, including on-site delivery, are less than five cents per kg. The LRM pressure-fed concept is now in the third month of a four month study to provide additional data on materials and fabrication, injector scaling, dynamic combustion stability and recoverability.

Considering pump-fed LRM boosters, F-1 designs fit into the thrust range demanded by the STS. The F-1 booster experience on Saturn V has been one of high reliability and performance. Table 5.6 summarizes the basic data on the F-1 engine [24]. The flyback version represents approximately a 10 percent mass increase over the data listed. The engine is also being considered by NASA for the recoverable mode of operation. The F-1 engine has been tested approximately 3000 times with some quarter million seconds of operation.

5.3.3 Structure and Thermal Protection Systems

The current space shuttle baseline orbiter has a mass in excess of 100,000 kg, a length greater than 30 m and a total wetted surface area of over 1000 square meters (m^2). These factors coupled with a 2000 km cross range requirement and the necessity for multiple reuse place stringent demands on the orbiter's thermal protection system (TPS) and require that careful consideration be given to this area of technology.

When contrasted with previous manned and unmanned re-entry systems, e.g., Dynasoar, X-15, Apollo, Gemini, Mercury, and strategic warheads, it has been estimated [25] that the requirements placed on the orbiter thermal protection system (TPS) materials and associated structure are considerably more severe than have heretofore been encountered. Consequently the orbiter TPS represents a major area of technology which will have a significant impact on the description of the orbiter system and its economic viability. The purpose of this section is to document the status of the orbiter TPS, materials, to place in perspective the various

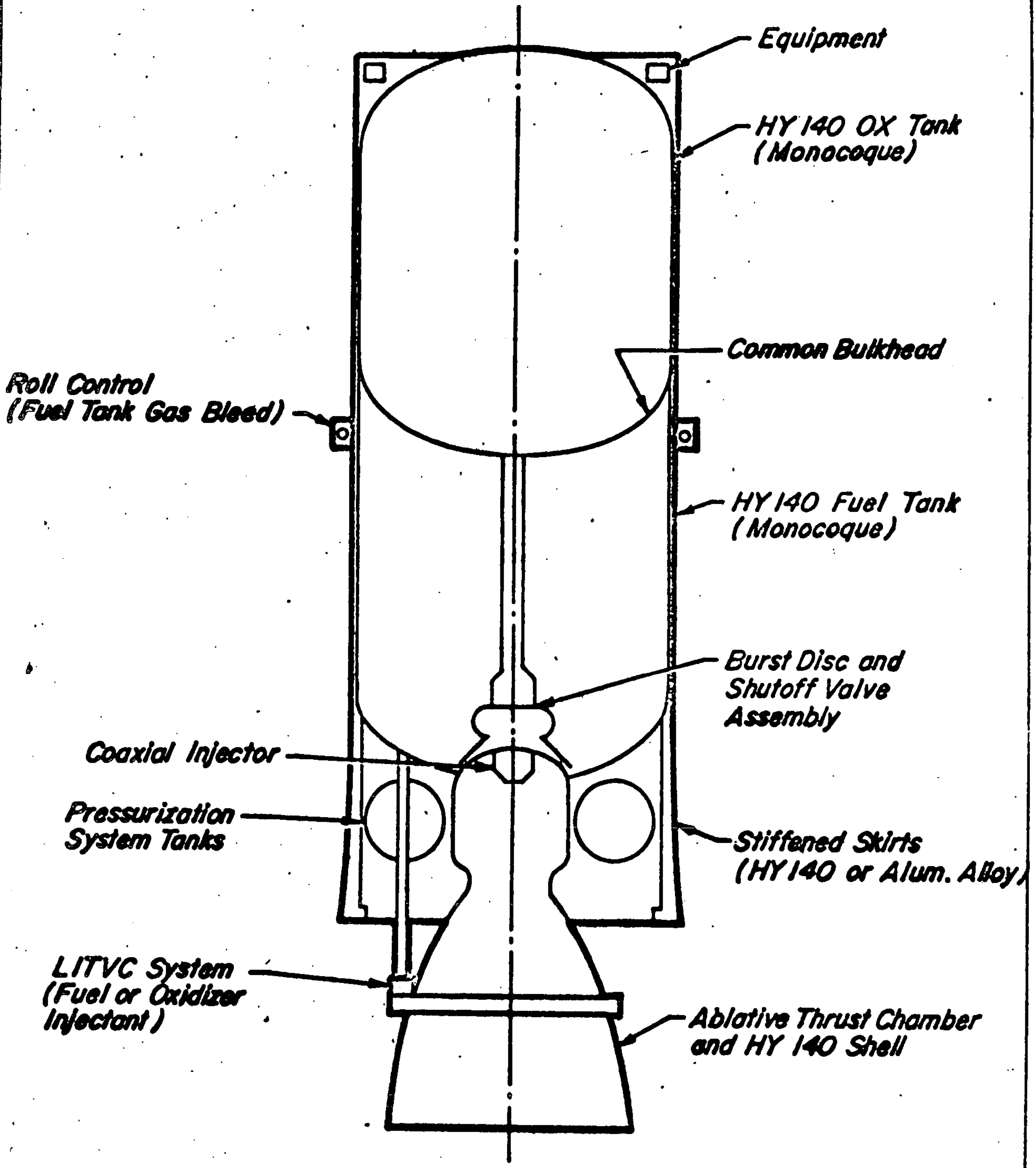


Figure 5.13 Pressure Fed Liquid Rocket Motor

options that are available, and to present an evaluation of the systems and materials best suited to the overall operation of the orbiter as part of an economically sound system.

5.3.3.1 General Considerations

Configuration

As shown in Figure 5.24 the current baseline orbiter is essentially sized by the 4.6 m dia x 18 m long payload bay and the smaller propulsion and crew sections. With an overall length of 36.8 m and a delta wing with a span of 38.2 m, aspect ratio of 1.7 and a 60^{o} leading edge sweep, the orbiter has a total surface area of $\sim 1500\ m^2$ and a dry mass (exclusive of payload) of 59,100 kg [26]. In general the orbiter will be coupled with one or more expendable hydrogen/oxygen drop tanks which accompany it into orbit, and are later jettisoned.

A comprehensive discussion of the various booster concepts currently under consideration is included in Section 5.5; however, the design of the orbiter TPS is relatively independent of the selected booster configuration.

Because of the close coupling that exists between the characteristics of the orbiter's entry trajectory and the thermal environment experienced by the orbiter and its TPS, considerable attention has been devoted to the analysis of the re-entry trajectory. A trajectory consisting of four phases [27] has been shown to offer a number of advantages for control of the thermal environment of the orbiter. During the initial phase, upon re-entering the atmosphere, a constant altitude trajectory is maintained. This is followed by the second phase wherein the heat rate (Q) is the controlling element. Phase three involves a transition from Q control to control of trajectory parameters by means of the acceleration level experienced by the orbiter (G control). Phase four of the entry trajectory is flown under G control until a final velocity and position is attained in the vicinity of the preselected landing field. Landing occurs at a speed of ~ 75 m/sec in a high approach angle energy management mode.

Significant savings in the TPS system mass can be effected by proper trajectory shaping [28]; for example, by employing a bank-modulated trajectory at constant angle of attack and with a sufficient ratio of lift to drag to satisfy performance, it is possible to restrict the maximum surface temperature reached on the bottom surface. This permits a reduction to be made in the thickness of the heat shield material, particularly if a metallic thermal protection system is employed. The details of the type of trajectory flown are determined in large measure by the type of TPS employed, e.g., ablative, reusable surface insulation (RSI) or metallic.

The following comparison of various thermal protection systems is made on the basis of the optimum trajectory for each of the materials under consideration. This basis of evaluation results in the lightest possible system in each case (see Section 5.3.3.5); however, current TPS requirements have not fully allowed for this potential saving.

<u>Aerothermodynamic Environment</u>

The thermal environment encountered by the orbiter TPS can be characterized by the three parameters: maximum heat rate (Q, joules/s) to a particular area, the total heat absorbed by the area in question (Q, joules) and the maximum temperature (T) attained by the TPS material.

These parameters vary as a function of position on the orbiter surface (see Figures 5.14, 5.15). Heating occurs during ascent, staging, abort and re-entry, the most severe effects being evidenced during re-entry. While a number of heating problem areas have been identified as occurring during the staging maneuver [29] particularly with regard to the recoverable booster concept, these effects are overshadowed by the prolonged re-entry phase associated with a 2000 km cross-range requirement. The same comment is applicable to the ascent portions of the trajectory and to abort conditions. Consequently major emphasis has been devoted to the development of suitable computational techniques, and to the collection of pertinent experimental data concerning conditions during hypersonic entry flight.

To provide rapid dissemination of aerothermodynamic data from research programs in support of the shuttle, a data storage and distribution system has been established at Michoud, Louisiana. This system, denoted SADSAC [30] (System for Analysis of Static Aerothermodynamic Criteria),

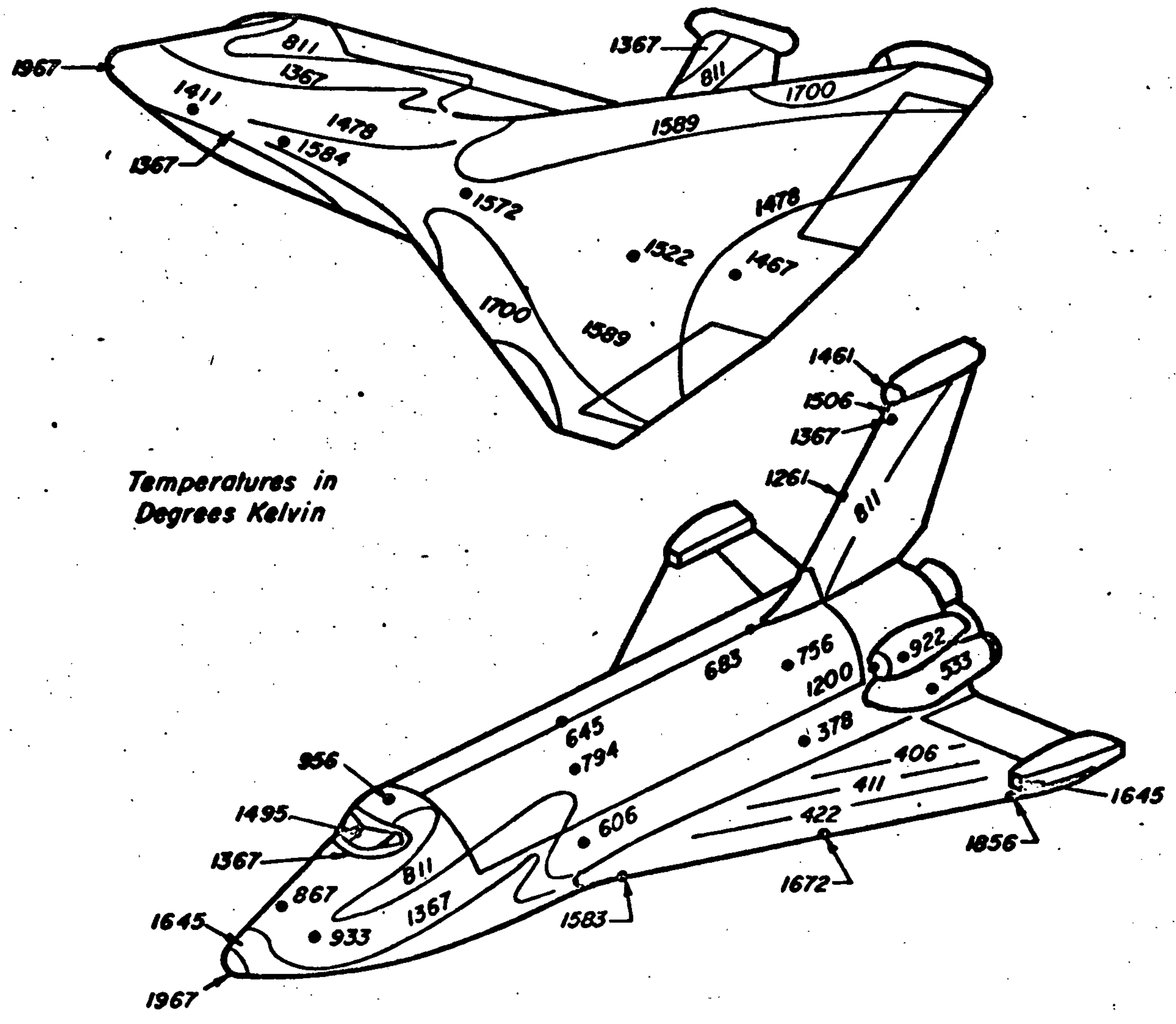


Figure 5.14 Maximum Current Baseline Orbiter Entry Temperature

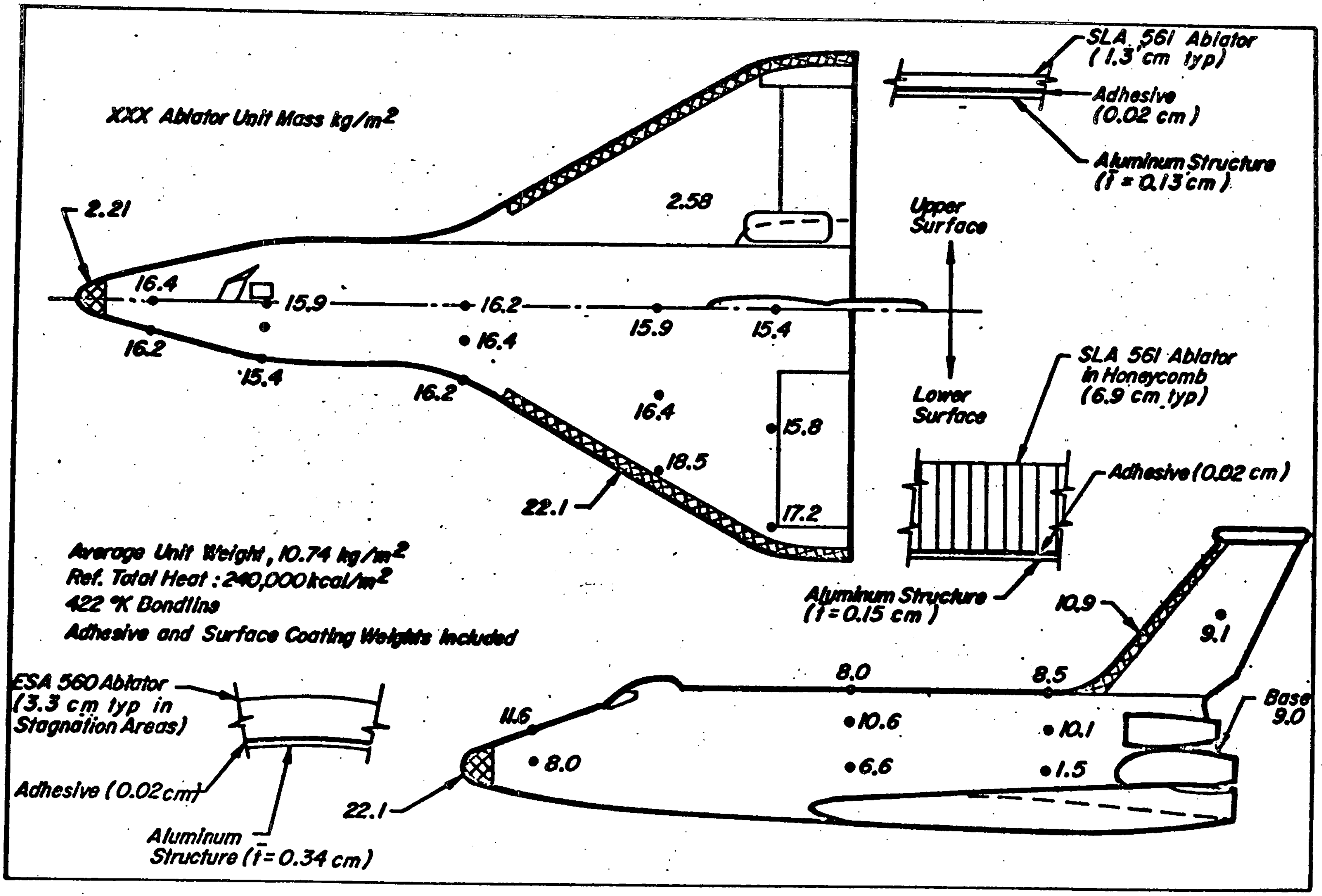


Figure 5.15 Maximum Current Baseline Orbiter Total Entry Heat Load

distributes all contractor-sponsored shuttle configuration force and momentum data, experimental thermocouple and phase-change coating heating data and pressure distribution data, reduced to a common format.

A number of computational techniques are available for calculating heat transfer rate for the high cross range (2000 km) delta wing (i.e., baseline) orbiter [31]. In general the methods employed by both North American Rockwell (NAR) and McDonnell Douglas (MDAC) produce heat transfer predictions that are optimistic compared to available test data [32, 33]. Since the TPS design for the orbiter requires a detailed knowledge of the pressure and heat transfer distribution over the entire flight regime from hypersonic to low subsonic, more work is needed. This is particularly true of the effects of surface roughness arising from, for instance, variable ablation rates for an ablator RSI or from buckling and/or creep effects associated with a metallic TPS [31].

As a first step toward the solution of this problem, it has been proposed that a generalized program to describe the inviscid flow field around the orbiter be developed to be followed by a more comprehensive program incorporating viscous effects [34]. Such a progression of programs is needed to predict the heat load and heating rate on the payload bay due to leeside heating by vortices, and for establishment of reliable estimates for turbulent transition and heating effects.

The complete flow field about the complex orbiter geometry must be considered when calculating the real gas effects caused by the extreme velocities encountered during re-entry [35]. Indeed one of the principal sources of error resulting from the direct application of ground test data to predict flight conditions is the variation in stream-line curvature and the alteration of the velocity gradient, Mach number and pressure fields due to real gas phenomena [36].

While it is generally true that substantial effort still has to be devoted to the development of fully reliable and accurate analytical techniques data show that existing methods are adequate for the calculation of ablator thickness surface recession distribution in the leading edge region [37].

One aspect of the aerothermodynamic environment the orbiter will encounter during operation concerns the manner in which the parameters total heating, heating rate, temperature and pressure will vary about the hypothetical "nominal" profile due to anticipated variations in the atmosphere and trajectory. In order to provide an adequate margin of safety in the design of the TPS it is necessary to properly evaluate such variations. Such an evaluation is currently in progress at NASA Langley Research Center [38]. While definitive results are still in the process of being amassed it is clear that this analysis will influence the detailed design of the EPS currently under study to an extent that cannot be fully assessed at this time (see Section 5.3.3.5).

The thermal aspects of TPS design center primarily about the conditions encountered during hypersonic flight. However, the overall design of the TPS system is determined by conditions both during the hypersonic and the subsonic portions of the trajectory. In general TPS materials are selected by the maximum entry temperature and total heat input while the panels are sized by ascent pressures [28, 30]. In addition, since the orbiter has to exhibit reasonable handling characteristics during the final subsonic phase of the re-entry trajectory, the TPS has to be compatible with operation under adverse weather conditions. Therefore, consideration has to be given to such factors as rain and hail erosion and surface characteristics, e.g., roughness, which could alter the aerodynamic characteristics of the vehicle. Present data support the position that no major problems are likely to be encountered due to subsonic operational conditions for current TPS designs.

5.3.3.2 Orbiter Structure

Design and Materials

With the overall configuration of the orbiter nearing finalization (Figure 5.24) detailed design of the structure and selection of appropriate construction materials are still in the process of being established. A number of design studies conducted using earlier orbiter configurations having substantial internal propellant capacity [39] have shown that integral propellant tanks have lower weights, development investment and total program cost than non-integral tanks. However, if consideration is given to simplicity of design, replaceability and maintenance, a non-integral tank design would be superior.

Aluminum is the likely material choice if it is ultimately decided that a "cold structure" approach (i.e., the structure is thermally protected) is to be adopted. However, if a "hot structure" approach is selected wherein the structure is required to act, to a certain degree, as a heat sink and sustain temperatures in excess of those permissible with aluminum, titanium is probably the appropriate construction material [36].

In comparing the relative merits of aluminum versus titanium a number of factors are worth noting. Aluminum has the advantages of lower design complexity, minimum analysis complexity, reduced testing and fabrication complexity, and lower cost [36]. There is no clear cut weight advantage of one material over the other. Studies are now in progress to define more fully the relative advantages and disadvantages of these materials.

A number of advanced materials have been considered for use in the orbiter structure. It has been shown [40] that considerable cost savings can be affected by the application of advanced structural materials, e.g., beryllium, to the orbiter. The use of such materials would result in reduced system complexity and a reduction in launch weight. A detailed study [39] of boron/epoxy employed in thrust structure tubes and boron/aluminum structural components have indicated weight and cost savings compared to similar structures fabricated of conventional material. Because of the relatively developmental nature of the advanced materials they will doubtlessly be incorporated slowly into the orbiter program, their initial use being restricted to specialized applications.

Structure, TPS Interaction

Recent studies by MDAC conclude that the arrangement of primary and support structure do not determine the type and attachment method of the thermal protection system employed and that a properly designed primary and support structure can accommodate a variety of approaches. Whether the TPS can in fact be treated independent of the structure must be considered a moot point. There is considerable evidence [42] that the TPS and structure in order to be efficient have to be considered as a single entity despite the conveniences that accrue if they are treated separately. This contra view is based on previous experience with re-entry vehicle designs where significant coupling between the TPS and the structure occurred. At this point one can only conclude that coupling between the TPS and structure can occur and should be considered in the design of the structure and TPS.

A problem area that has been recognized and will be discussed in Section 5.3.3.4 involves the limited strength levels in tension and compression of RSI TPS. Such materials require the use of finite thickness flexible adhesives for proper bonding to panels which are then attached to the structure proper. Consequently it is necessary that proper design of the underlying structure and the TPS system be undertaken.

5.3.3.3 Candidate TPS

Ablative

Figure 5.15 displays, in a simplified form, the general details of an ablator TPS. Typically ablative thermal protection systems, as employed in the manned space flight program and as proposed for the shuttle, consist of a honeycomb, usually fiberglass, with cells ~ 1 cm across whose axes are aligned perpendicular to the surface of the vehicle and are filled with an elastomeric material. During re-entry aerodynamic heating causes the surface of the ablator to become hot leading to the formation of a char layer. This char layer, mechanically weaker than the parent material, is restrained from rapid erosion by the presence of the honeycomb. As a consequence the char behaves similar to an insulator and impedes the flow of heat to the interior. With prolonged exposure, additional ablator chars and is erosively removed until at the end of a given mission only sufficient thickness of material remains to prevent the thermal pulse associated with the re-entry maneuver from causing an over temperature condition to exist in the mounting structure.

In general, because the surface material is continually removed during heating, ablators have proven to be virtually insensitive to Q [43]. In addition, ablator TPS are relatively simple to design, are reliable, forgiving of small imperfections and provide a degree of structural vibration damping.

Ablator materials exist in a wide range of densities. A high density phenolic-nylon and filled silicone elastomer (450 kg/m^3) is capable of absorbing $\sim$10-15 x 10^6 joule/kg at heating rates as high as 220 kW/m^2. By contrast a low density ablator of the same material (240 kg/m^3) is capable of absorbing $\sim$15-20 x 10^6 joule/kg at the same rate but requires the presence of a honeycomb structure to retain the char [43].

The test results for ablator materials having small defects are quite comparable to the results obtained from defect free ablators used in the manned space program. This observation is consistent with experience obtained in the Apollo program. Access holes in the Apollo heat shield having dimensions on the order of the basic honeycomb cell size did not conspicuously affect the ability of the system to perform its function either in the vicinity of the holes or overall.

The ability to tolerate defects, e.g., partially filled honeycomb cells, permits relaxed manufacturing and inspection standards and, if permitted on a manned vehicle, would result in substantial cost savings as compared to previous heat shields. This is particularly true when consideration is given to the large area of thermal protection surface required. Slightly in excess of 1000 m^2 of surface area (100 percent of wetted surface) has to be protected, an area so large by usual heat shield standards that the descriptive term "acreage" has come into general use.

A study [44] of one of the most recent orbiter designs (the 040A) revealed the following statistics concerning the characteristics of an ablator TPS: Acreage, SLA-561 ablator in honeycomb (71kg/m^2); Leading Edges, ESA 3560 11A moulded ablator (275 kg/m^2); and Antennae covered with SLA-220 ablator in honeycomb (79 kg/m^2). The total TPS weight from this study was 13,700 kg for an average surface density of 13.5 kg/m^2 and an average area per panel of about one square meter.

The study showed that a total of 1028 separate ablator panels would be required to completely cover the surface, distributed as follows: special shape -149, doubly curved -270, singly curved -473 and flat -136. These figures give some measure of the magnitude of the refurbishment task.

There is very little uncertainty with regard to the question of whether an orbiter using an ablator TPS of the type described could be built and successfully operated. The major question is one of overall economics, in particular the costs involved in refurbishment. Currently it appears that honeycomb ablators, bonded to their panels which are then bolted to the primary structure, would provide ease of refurbishment. The joints between panels would be filled with an easily applied elastomer (RTV) facilitating panel removal and replacement. Allowance for operating cost uncertainties were included in the economic risk analysis in both this report and Reference 5-64.

Until detailed time-motion studies of the refurbishment of actual, full scale test sections are complete by MDAC (test currently in progress at NASA Langley) available cost estimates of the refurbishment process will have to suffice. Current estimates, while disparate, would indicate a panel (1 m^2) will cost in the neighborhood of $1,000 [37, 44] for a total replacement cost per vehicle of $1,000,000 (assuming that the defect free requirement heretofore applied to manned vehicle heat shields will be relaxed). It has been estimated that complete replacement of the TPS would require a total of ~8000 man hours of work per vehicle [44].

Historically, ablator thermal protection systems have been used once and then discarded. It should, however, be recognized that there exists no fundamental reason why a properly designed ablative heat shield could not be reused a limited number of times before its design life is attained. As yet the reuse capability of candidate ablator RPS materials has not been established and is the subject of continuing research [44]. If reasonable reuse can be attained, ablator TPS can be very cost effective.

Ablators represent a well-developed technology, however, several important questions [37] concerning their application to the shuttle remain to be answered. These questions include: what is the effect of shape change and surface roughness upon aerodynamic performance; what is the differential ablation at panel joints; what is the effect of ablation products upon adjacent panels, and what are the effects of prolonged exposure to space. All require further study and evaluation before the ablator TPS can be used. However, the solution to the problems posed by these questions does not appear to require new technology or a prolonged and costly research program.

5.3.3.4 Reusable Surface Insulation (RSI)

Figure 5.16 depicts the manner in which a typical RSI panel would be mounted on the orbiter structure. The generic term reusable surface insulation is used to describe an approach to the requirement that the basic orbiter structure's temperatures remain within a preselected range. This is accomplished in the RSI system by the use of a suitable insulating material, e.g., silica or mullite (aluminum-silicate), which is bonded to a panel, typically titanium sheet or honeycomb, by a suitable flexible adhesive. To prevent scouring of the insulation by the hot gas flow associated with re-entry a suitable protective layer is applied to the outside and edge surfaces of the insulation. It is the ability of this protective surface layer to resist the combined action of aerodynamic forces and high temperature that permits this system to be reused numerous times.

Generally the thickness of the insulation is determined by the temperature tolerance of the adhesive used to affix the insulation to an appropriate backing panel. A temperature of 550°K represents a practical limit [28, 45] (the so-called band-line temperature limit) for insulation mounted on titanium panels.

Since most insulation materials have low strength [46], (on the order of 70 N/cm^2) finite thickness flexible adhesives such as RTU have to be employed to secure the insulation to a backing panel. The unit weight of the insulation used in any given panel or any given location on the orbiter, since the band-line temperature is restricted, is a function of the total heat input (Q) to the panel and the duration of the heat pulse.

Currently, silica and mullite rigidized fiber insulators are the two materials which offer the most promise of satisfying the RPS requirements

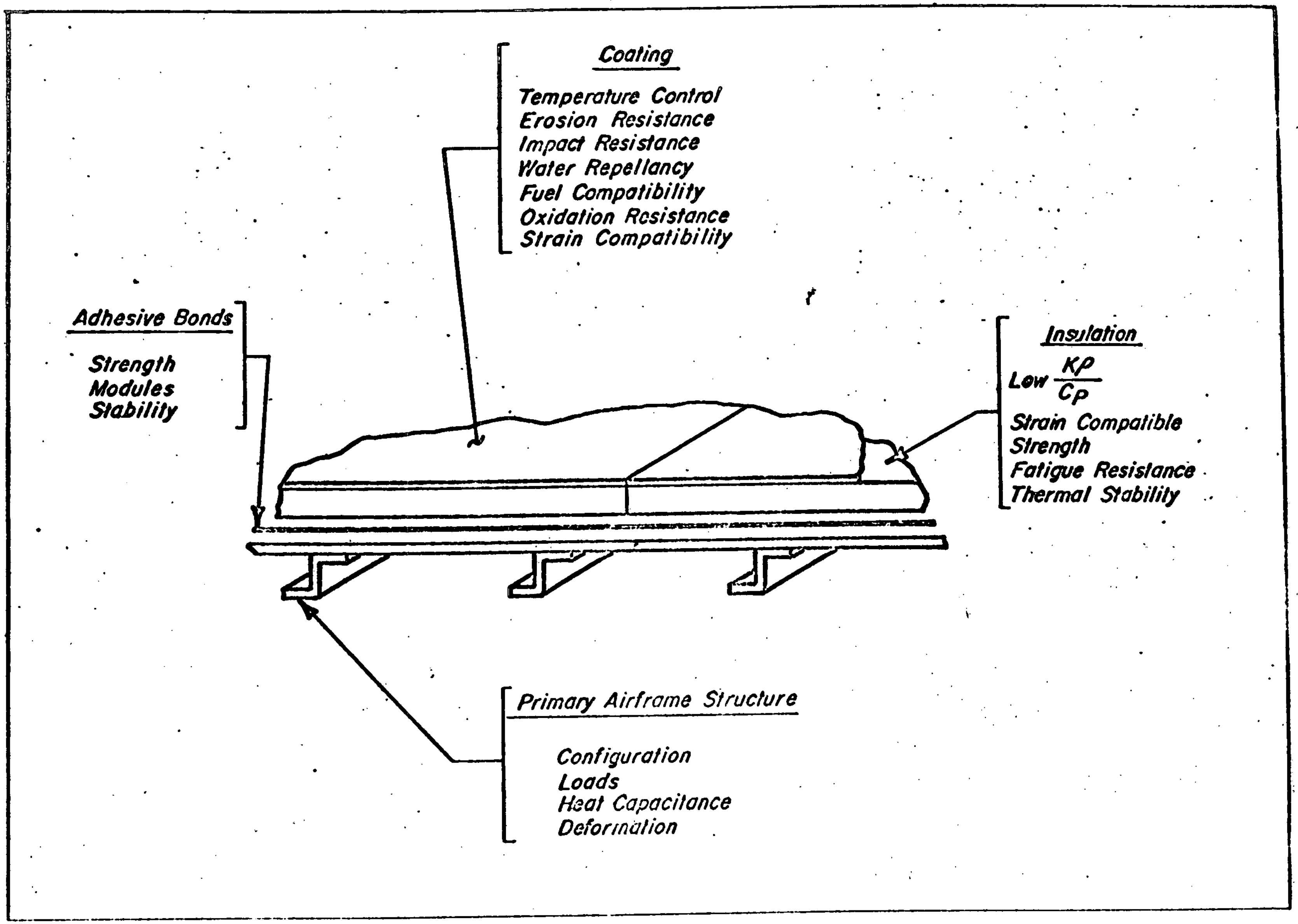


Figure 5.1 6 Typical Removable RSI Panel - Construction and Mounting Details and Structure.

of the shuttle orbiter [45]. A silica system developed by LMSC using a chromium oxide coating has demonstrated reuse capability up to 1530°K and mullite (developed by MDAC) has proven serviceable up to 1640°K [48]. The use of a silicone coating offers the advantage of being non-catalytic and therefore capable of operation at less than the radiative equilibrium temperature of the gas flow. A major problem associated with currently developed reusable surface insulation systems arises from the fact that the protective surface coatings in use are not strain compatible with the insulation, the ultimate strain capability being limited by the coating.

A number of other insulation materials have been considered. Ceramics and some carbon systems have exhibited severe shortcomings as reusable thermally and structurally efficient systems [45]. Ceramics are brittle, have low strength and are sensitive to thermal shock whereas carbon systems are subject to oxidation effects that are small but cannot be neglected. However, new efforts are making these systems more competitive. Oxidation inhibited carbon laminate, a carbon cloth laminate bonded with polymeric resin pyrolyzed to carbon, has demonstrated stability to temperatures as high as 2500°K [45]. Indeed carbon cloth laminates diffusion coated with either silicone or combinations of zinconium, boron or silicone perform very well; siliconized carbon having demonstrated reuse capability at 1800°K.

Repair techniques have been developed to permit reuse of lightly damaged RSI panels that should improve the competitive position of this TPS approach with respect to other methods. Work is currently underway at NASA Langley Research Center directed toward defining the costs involved in repairing a full scale RSI TPS. The actual status of the RSI system should be established during January 1972 when final reports are due from the three contractors involved in this work: LSMC, MDAC and G. E. [49].

LMSC has apparently made a major advance with the development of an amorphous glaze silicone carbide coating. This coating is not only compatible with the base insulation but possesses high temperature stability and is waterproof. The amorphous glaze coating has been shown to be capable of retaining its waterproofing characteristics after prolonged use and to be rain resistant at speeds up to 500 to 650 km/hr. at reasonable angles of attack

($10^{o} > 20^{o}$) [49]. Consequently there is no rain erosion problem.

The thermal shock problems with mullite, wherein the coating cracks exposing the vulnerable mullite to anodynamic forces, does not occur with the new coating material. Consequently, a coating material-insulation material combination may well be available for use on the orbiter. However, until NASA can evaluate the work that is now in the final stages of completion, the exact status of RSI TPS is uncertain.

Metallic

The metallic thermal protection employs a high temperature metallic outer surface which is thermally decoupled from, and therefore shields, the inner load carrying structure. This type of TPS was used in the highly successful X-15 research vehicle which operated at lower velocities and therefore lower heat loadings and temperatures than those associated with the shuttle orbiter.

The metallic heat shield system is temperature limited and therefore sensitive to heating rate [27] but insensitive to the duration of the heat pulse and the total heat load [28]. The following estimates of the fraction of orbiter surface areas below a given temperature: less than 1800°K, 0.95 → 0.98; less than 1600 °K, 0.90→0.98; less than 1400°K, 0.85→0.95; less than 1300°K, 0.65→0.90, show that a significant portion of the acreage can be covered with superalloys. Superalloys (for which a large amount of data exists) are useful up to a maximum temperature of 1300°K.

Above a surface temperature of 1300°K and up to 1400°K cobalt superalloys and coated columbium (element 41, also called niobium) can be used. Above 1400°K coated tantalum shows promise up to 1600 to 1800°K [48]; however, coatings for tantalum are less advanced technically than are those for columbium. Significant reuse of tantalum has been demonstrated at the high temperatures noted but there is a question of the reliability of this material [5-51]. Sylvania has developed a coating (R512E) under Air Force contract that, when used with columbium, shows promise of 100 mission reuse capability in shuttle service at temperatures up to 1600°K. General Dynamics is conducting an extensive proof of concept evaluation of coated columbium employing large scale components to ascertain the detailed characteristics of this material. Preliminary data indicate that defects or cracks in the R512E coating do not lead to catastrophic failure, are relatively easy to detect and are amenable to rapid repair. However, insufficient data are available to assess properly the cost of such a TPS approach [51, 52].

A major problem has been identified with the use of metallic heat shield materials [50, 53, 54]. Such candidate materials as Rene' 41, H5-188, TD-Hi Cr, and Hayes 25 exhibit elevated creep levels when exposed to cyclic variations of temperature and stress similar to those anticipated to be associated with shuttle operation. These creep levels are far in excess of those predicted on the basis of normal continuous creep test data and represent a new effect that was unexpected and is being studied extensively. Once the actual creep characteristics of these metals is established appropriate designs can be made to take this effect into account. A computer model has been developed [53] to predict creep under transient load and temperature conditions permitting design evaluation of these materials to be rapidly conducted.

In addition to the generation of excessive creep rates, tests reproducing the cyclic conditions imposed by the orbiter's operational requirements have shown that TD-Ni Cr develops porosity after prolonged use. An aluminum modified allow (TD-Ni Cr Aly) has been developed [32] which exhibits excellent oxidation resistance and does not become porous under cyclic temperature/stress conditions.

Considerable development work has still to be accomplished before metallic TPS can be judged suitable for use with the shuttle. Obviously there exist many areas on the shuttle surface where the maximum temperature is low enough to permit metallic materials to be used (Figure 5.14). However, even though such a system would offer the advantage of light unit weight [28] (10-12 kg/m^2), extensive proof of concept testing will have to be conducted to establish the basic costs of such a system and to verify the basic design. Such testing is now underway. Consequently, no firm economic analysis of the metallic TPS can be realistically undertaken prior to the time that the results of these tests are evaluated.

Heat Sink

The heat sink concept of thermal protection employs the innate thermal capacity of the vehicle's structure to limit temperatures to safe levels. Obviously such a system is sensitive to total heat loading (Q). An analysis of a heat sink protected shuttle booster stage [55] has shown that an aluminum structure (alloy 2219-787) is capable of surviving entry from staging velocities somewhat in excess of 3 km/s.

This RPS method shows considerable promise for use with the recoverable SRM and LRM concepts now receiving consideration (Section 5.5.2).

Active

The active thermal protection system employs an expendable medium, e.g., water, to provide cooling to the highest temperature areas of the entering vehicle. Work is currently in progress at Langley on this approach so that any evaluation of the so-called "water wick" system would be unwarranted at this time.

LMSC [56] has conducted a study of the advantages and disadvantages such a system would have when incorporated into a delta body orbiter. In the study only the flat, windward, bottom of the orbiter vehicle (36 percent of the total surface area) was actively cooled by a redundant water/glycol, water/NH system. The analysis revealed that a small weight advantage (~2500 kg) could be gained by use of the active system. However, LMSC concluded that the increased cost and reduced reliability of such a system outweighed the advantages.

While this study was essentially inconclusive it would appear that the concept has merit and should be studied in greater depth for specialized applications.

5.3.3.5 TPS Evaluation

On the basis of the comments made in Section 5.3.3.4 it is clear that of the three major TPS approaches considered the ablative system is the most well developed and therefore would represent less of a technological risk than either the RSI or metallic approach. The technology associated with

RSI or metallic TPS has made rapid progress but many areas of uncertainty exist with regard to the application of these materials to the shuttle. These uncertainties are compounded by the uncertainties that exist with regard to the calculation of the orbiter's aerothermodynamic state during re-entry. On the other hand, the ablative system has considerably higher costs per flight than those promised by a reusable TPS.

To a certain extent the success of the metallic and RSI TPS depends on trajectory control to limit surface temperatures and heat loading to acceptable levels. Until the anticipated variations about the nominal trajectory due to guidance and control factors and variations in the normal atmosphere are evaluated in detail these two systems must be designed to provide a sufficient margin to prevent failure. The ablative system has proven to be quite forgiving of not only small defects in fabrication but to variations in maximum surface temperature and heat rate. Consequently, an ablative TPS is being considered for the baseline orbiter since it represents the lowest development cost and lowest risk.

Figure 5.17 based on data presented in Reference 5-43 represents an attempt to evaluate the initial and operating costs of the three major TPS candidates. The lower set of curves is the same as presented in Reference 5-43 (April 1971) while the upper set represents a more recent (November 1971) estimate. Because of the many uncertainties associated with the costs involved in the metallic and RSI systems the data of Figure 5.17 must be viewed as being at best qualitative and indicative of overall trends.

It is clear that, on the basis of the necessarily crude analysis represented by Figure 5.17, the ablative system would offer overall program savings if used as the primary TPS system for the first 20 flights or so. With the development of limited reuse capability it may be possible to use an ablative RPS for a far larger number of flights and still effect a cost savings.

Again, the uncertainties associated with the development of metallic and RSI TPS precludes any firm statement concerning their ultimate application to the orbiter to be made at this time. It seems obvious that ablative TPS will be applied to the earliest orbiters and possibly to a substantial number of subsequent flights as well. The economic analysis of the alternative

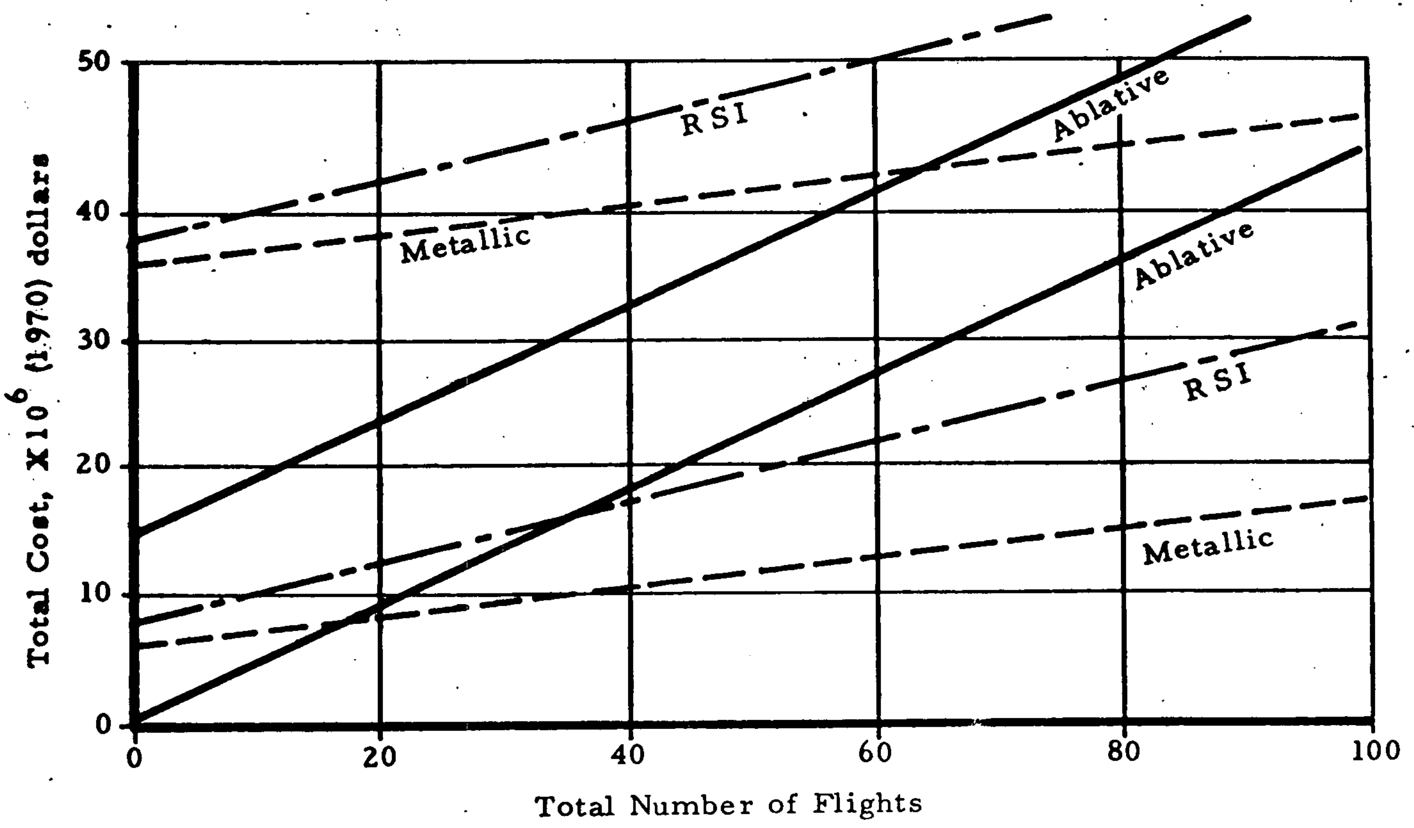


Figure 5.17 TPS Estimated Cumulative Cost Comparison

Space Shuttle Systems was carried out allowing to a considerable extent for these cost uncertainties in the operating phase of the Space Shuttle System. The technical review shows that at least one TPS has an assured performance confidence and the economic analysis has allowed for the implied increase in operating costs. Yet if a reusable TPS proves feasible in time for the 1979 IOC date of the Space Shuttle System, it would increase correspondingly the advantages of a reusable STS. The decision on which TPS to use is therefore a self-contained trade-off question that does not affect the accept/reject decision of the complete system.

5.3.4 Integrated Electronics System

The Integrated Electronics System (IES) provides both the interface and the intelligence linkage between the human operators in the spacecraft and the ground and the vehicle operational systems. The IES implements guidance, navigation, flight control, data management, and communications of the shuttle and booster systems. Also included in the IES are the hydraulic and electric power systems.

5.3.4.1 Requirements and System Organization

It should be noted that the following discussion is specifically applicable to the current baseline shuttle orbiter. The applicability of the IES as defined to the shuttle booster is a function of the yet-to-be-selected booster configuration. Booster configurations under consideration during the study period (July, 1971 to January, 1972) ranged from manned fully reusable to unmanned non-reusable vehicles. At present (January, 1972) the economic choice has narrowed to the (new) baseline orbiter with the following four alternate booster thrust assists: (1) Parallel Burn Solid Rocket Motors, (2) Parallel Burn Pressure Fed Boosters, (3) Series Burn Pressure Fed Booster, and (4) Series Burn Solid Rocket Motor Booster. For the manned fully reusable booster, the booster IES will be similar to the shuttle orbiter IES, with the exception of the deletion of the Rendezvous and Docking Aids. As the configuration moved toward an unmanned non-reusable design, the booster IES requirements will be minimized. Other booster configurations such as partially reusable booster or recoverable will employ scaled down versions of the IES described in this report.

The organization and design of the IES will be shaped by system level requirements for crew safety, turn-around time, and equipment commonality. For example, the requirements that "electronic systems shall be designed to fail operational after failure of the two most critical components and fail-safe for crew survival after the third failure" [57] will play a major role in establishing the requirements for redundancy and cross-strapping in subsystem design. In a similar manner, the requirement for vehicle turn-around in less than two weeks will establish the design requirement for fault isolation, accessibility, and replacement within the electronic subsystems at the chassis or module levels. The requirement for commonality of systems, subsystems, components, and parts between various program elements may cause the subsystem designer to incur penalties in the areas of size, mass, and power, but should lead to program economics through improved procurement and logistics practices.

5.3.4.2 Subsystem Characteristics

The functional organization of the IES is shown in Figure 5.18. The IES is capable of three modes of operation:

- automatically, under control of central computer stored software
- manually, with crew control via computer stored software
- manually, via hardwire control for limited periods during atmospheric flight.

During all modes of operation, the displays and associated controls provide the crew with program decision and intervention capabilities.

Data Management Subsystem

This area and the related figures are very much undetermined at present due to the austerity review underway. The Data Management Subsystem (DMS) consists of the central computer and processor, associated memories, the digital data bus, and digital interface units which provide buffering, analog-to-digital conversion, and formatting for all data to be processed by the DMS [58]. The central computer and processor performs computations required for

- vehicle pre-and in-flight checkout
- navigation, guidance, and control
- payload checkout
- vehicle and payload sensor data processing

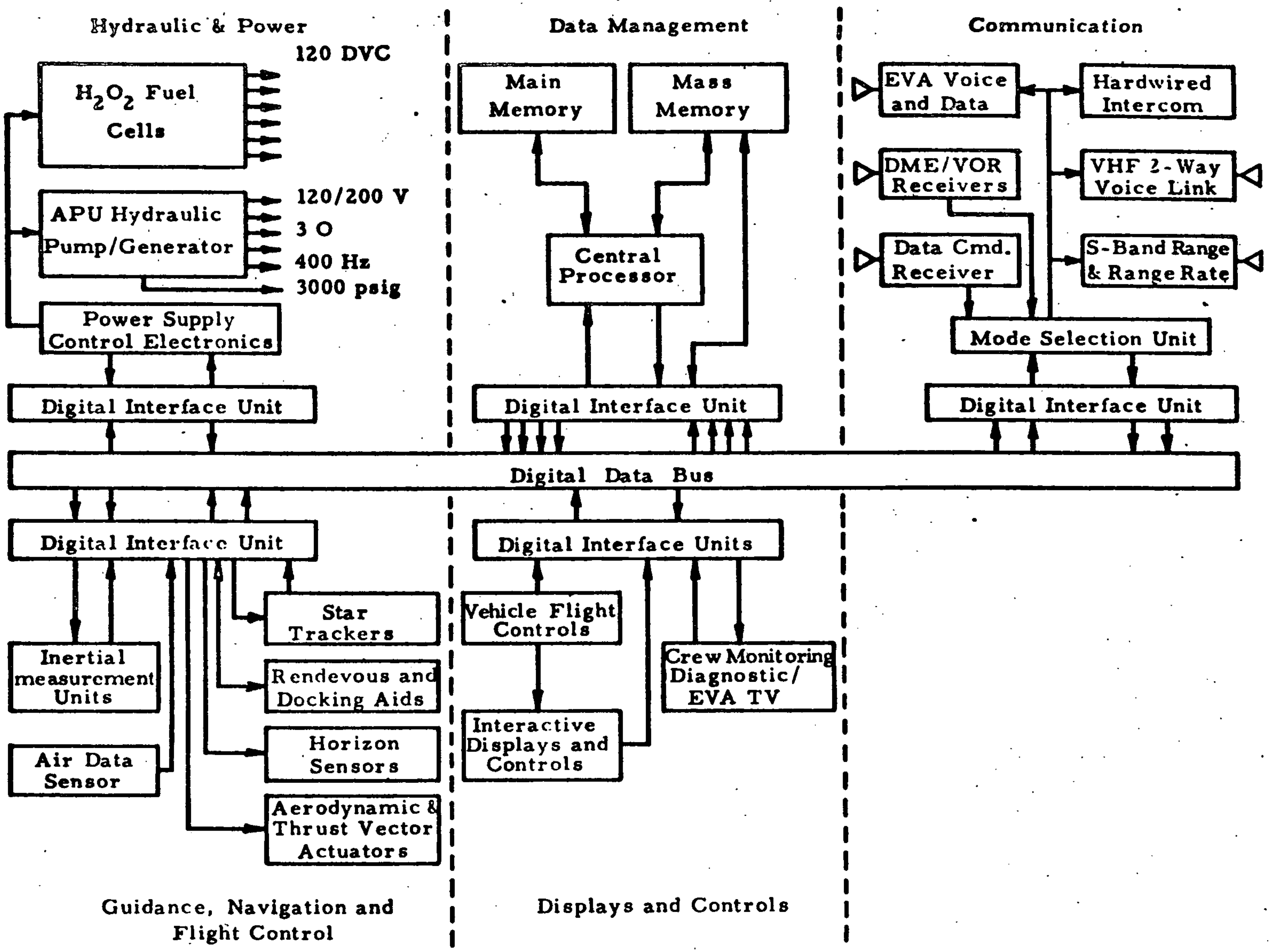


Figure 5.18 Space Shuttle Current Baseline Orbiter Integrated Electronics System

The digital data bus and digital interface units provide for data acquisition from subsystems and sensors, and for control signal transfer for vehicle systems. Additionally, the digital data bus interfaces with ground support equipment to enable the crew to monitor critical ground operations such as fuel loading.

Guidance, Navigation and Flight Control Subsystem

The guidance, navigation and flight control subsystem (GNFCS) enables the on-board determination of vehicle position and velocity during all phases of booster and shuttle flight, and controls the flight path and executes maneuvers as determined by automatic guidance programs or crew generated commands. The major hardware elements of the GNFCS are:

- inertial measurement units
- star trackers and horizon sensors
- s-band ranging transponders
- DME and VOR receivers
- localizer, glide slope, and radar altimeter landing aids
- rate, acceleration, air speed, barometric altitude and air data sensors
- aerodynamic and thrust vector control activators

Communication Subsystem

The communication subsystem (CS) provides voice and data links between the shuttle and booster, and between the ground and flight vehicles. Capabilities provided in the CS include:

- two-way voice between ground and orbiter, orbiter and space station, and crew and passengers (via hardwired intercom)
- unified s-band tracking
- ground-to-flight vehicle data/command link
- shuttle-to-ground data link
- two-way data link shuttle-to-detached payload
- EVA voice and data transfer

Major equipment provisions include the use of VHF for ground-to-flight vehicle voice links, s-band for data links and tracking, and hardwired intercoms for internal communications.

Displays and Controls Subsystem

The displays and controls subsystem (DCS) comprise the crew station avionics necessary to enable the flight crew to interact with the vehicle subsystems. The DCS satisfies mission requirements for:

- vehicle checkout
- flight control and display
- computer access
- subsystem and mission monitor and control

The DCS if designed to provide for full vehicle control by either of the two man crew. In addition to the aforementioned design requirements for crew safety, equipment, maintainability and commonality, the DCS must satisfy crew station functional requirements for human factors such as comfort and controls accessibility.

Several possible applications for television equipment can be foreseen as a result of the successful use of TV in the Apollo missions. As in the case of Apollo, it is readily apparent that TV can be used to broadly disseminate mission status and events for public consumption on a world-wide basis. As a result of the now proven development of color TV for space applications, TV can be used for monitoring crew status and health, and for the remote observation of potentially hazardous mission operations.

Software

The IES software consists of the programs stored in the DMS to organize and control the functioning of the flight vehicle subsystems. Each of the aforementioned IES subsystems, as well as the non-avionics subsystems, has unique software requirements.

Catagories of software to be developed include prelaunch checkout, guidance, navigation, flight control, data management, sensor data processing, display, system status and reconfiguration, memory access and utilization, various computational subroutines, and executive programs for system access and control by the crew.

Orbiter Hydraulic and Electric Power

Shuttle power is provided by H_2O_2 fuel cells and AC generators. The generators are driven by the turbine power units that provide the shuttle hydraulic power. Fuel cells are used to provide electrical power for all vehicle systems except the main propulsion engines, while AC turbine driven generators provide hydraulic and electric power for functions such as engine ignition, hydraulic and electrical system checkout (during prelaunch and re-entry), and vehicle steering during landing.

5.3.4.3 Technology Appraisal

The IES makes extensive use of equipment designs embodying technology used in spacecraft and aircraft programs with spaceflight development and flight experience. While the subsystems reflect the unique requirements of the shuttle program, they are also identifiable derivatives of existing designs. No major technical developmental problems are anticipated in the IES, but it may be expected that significant effort will be expended in the development of:

- systems engineering to satisfy mission requirements for:
- crew safety
- equipment refurbishment
- crew station displays and controls
- the DMS computer
- software

The development and verification of the software will require the extensive use of subsystem simulators. Flight simulators using realistic crew station mockups with operating controls and displays will be used for the dual purposes of the development of the interactive software and crew training.

5.4 New Expendable Launch Vehicle Family

The new expendable launch vehicle family (sometimes referred to as the new low cost expendables) represents a new group of related vehicles which, as stated in Reference [1] "through suitable arrangement of a set of stages and strap-ons, can efficiently and economically accomplish the spectrum of projected future low earth orbit and high altitude/high energy missions. The selected family is based primarily on existing Titan III components and their derivative and growth versions. Maximum use of common hardware and facilities results in a family of low cost launch vehicles with payload capabilities ranging from about 2000 kg to more than 50000 kg in low earth orbit."

When taken together with derivative Scout vehicles, the new expendable launch vehicle family consists of three classes; the Scout vehicles, a new family consisting of solid rocket first stages and existing liquid second stages, the Titan III vehicles described in Section 5.2.2 above and the new Titan III L vehicles. Characteristics of the Aerospace Corporation new expendable launch vehicle family are shown in Table 5.7.

The new expendable launch vehicle family as constituted by Aerospace Corporation was identified in its elements to consist of current low cost technology based primarily on Titan III which is the present launch vehicle family with the lowest cost per kilogram for placing payloads in earth orbit. While a family of launch vehicles that is so constituted may possibly represent an optimum short-term solution as an economically justifiable replacement for the current expendable launch vehicle family (either as an alternate or complement to the space shuttle in the 1980's), it should undoubtedly include recent and advanced technology in providing a higher performance, more flexibility and economic proposition. Such a family has not yet been proposed although some studies are understood to be in prospect. Any such proposed new family of expendable launch vehicles would have to prove cost effectiveness at a suitable social rate of interest when compared to this new expendable launch vehicle fleet.

5.4.1 Small Payload Class

The small payload class of the new expendable launch vehicles consists of improved versions of the four and five stage Scout. In appearance, the vehicle is very similar to the present Scout (Figure 5.1), however, several significant modifications are incorporated resulting in an increase of approximately 2.5 meters in overall length. These modifications include an improved guidance system incorporated into the fourth stage, steerable

Table 5.7

Characteristics of the New Expendable Launch Vehicle Family

N.B.: All data are approximate — References 5-1, 2, 3, 4

Launch Vehicle	GLOM(1) MT	Length,(2) m	Dia., m	Payload Vol,(3) m^3	Reliability
Improved Scout	32	25	1.5	1	(~ .95)
5 Seg/II/Agena	276	48.2	3	40	(> .95)
5 Seg/II/AKM	268	42.4	3	40	(> .95)
5 Seg/II/Centaur	288	52.5	3	50	(> .9)
Titan IIID	631	47	3	85-100	(> .95)
Titan IIID/BII	623	47	3	85-100	(> .95)
Titan IIID/Centaur	650	47	3	85	(> .95)
Titan IIID/Centaur/BII	638	47	3	85	(> .95)
Titan IIIF	821	50	3	85-100	(> .95)
Titan IIIF/AKM	821	50	3	85-100	(> .95)
Titan IIIF/BII	821	50	3	85-100	(> .95)
Titan IIIF/Centaur	841	50	3	85	(> .95)
Titan IIIF/Centaur/BII	843	50	3	85	(> .95)
Titan IIIM	723	55	3	85	(> .95)

(1) Gross Liftoff Mass, GLOM, 1 metric ton (MT) = 1000 kilograms (kg)
(2) Nominal overall with typical payload fairing
(3) Maximum with largest currently available fairing

Table 5.7 (cont'd-2)

Characteristics of the New Expendable Launch Vehicle Family

N. B.: All data are approximate — References 5-1, 2, 3, 4

Launch Vehicle	GLOM[1] MT	Length,[2] m	Dia., m	Payload Vol,[3] m^3	Reliability
Titan IIIL*4	1836	67	4.6	85-100	(> .95)
Titan IIIL*4/Centaur	1855	67	4.6	85	(> .95)

(1) Gross Liftoff Mass, GLOM. 1 metric ton (MT) = 1000 kilograms (kg)
(2) Nominal overall with typical payload fairing
(3) Maximum with largest currently available fairing

nozzles on the first and second stages, two strap-on castor solid rocket motors in order to allow increased launch mass. The payload volume will also be increased from its present 107 cm to 152 cm commensurate with its increased payload capability. The first stage is an ALGOL III and the second stage is a short ALGOL. The third stage is an X259 manufactured by Hercules and the fourth stage is a short 259 by UTC. The third and fourth stages incorporate bell nozzles. The performance of the improved Scout is shown in Figure 5.19 and its reliability should be approximately 0.95.

5.4.2 Medium Payload Class

The medium payload class of the new expendable launch vehicles consists of new combinations of existing elements of current launch vehicle systems. This solid (first stage)/liquid (second stage) class of vehicles utilizes the "building blocks" of the existing standard Titan III launch vehicles. For a first stage, these vehicles would use various lengths or segments of the 3 meter diameter solid rocket motor strap-ons (SRM's) employed on the current Titan III vehicles. The second stage would consist of Stage II on the current Titan III vehicles. Various existing upper stages such as an Agena or Centaur would be used for high energy missions. Representative vehicles are shown in Figure 5.20 and their performance is shown in Figure 5.21.

Various other modifications or additions would be required such as new adapter rings for the stages and payloads, and the addition of thrust vector control devices for pitch, roll, and yaw maneuvers. The projected reliabilities of these new solid/liquid launch vehicles is expected to be over 95 percent based on flight experience with similar technology.

Other vehicles in the medium payload class of the new expendable launch vehicles are two vehicles of the current expendable group that are compatible with the new low cost logic applied to the launch vehicles in this Section 5.4-1. The Titan III D is currently in development and the Titan III F needs little additional development. They have been discussed previously in Section 5.2.2.

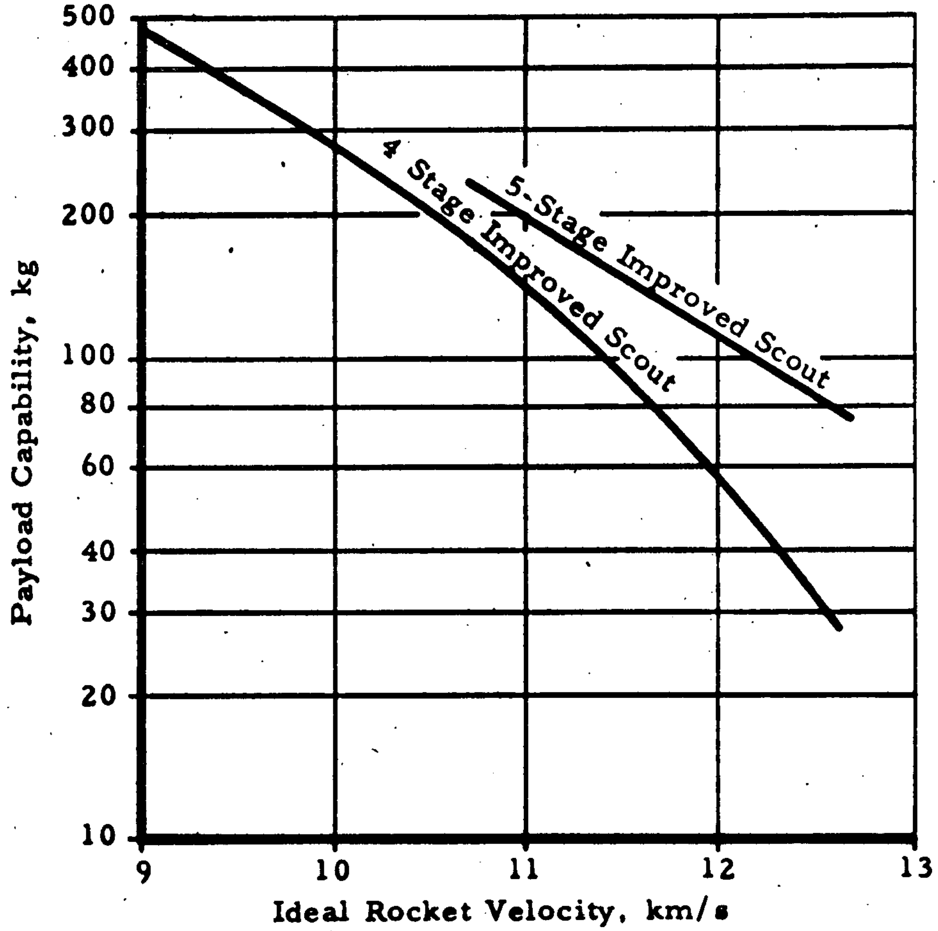


Figure 5. 19 Improved Scout Performance

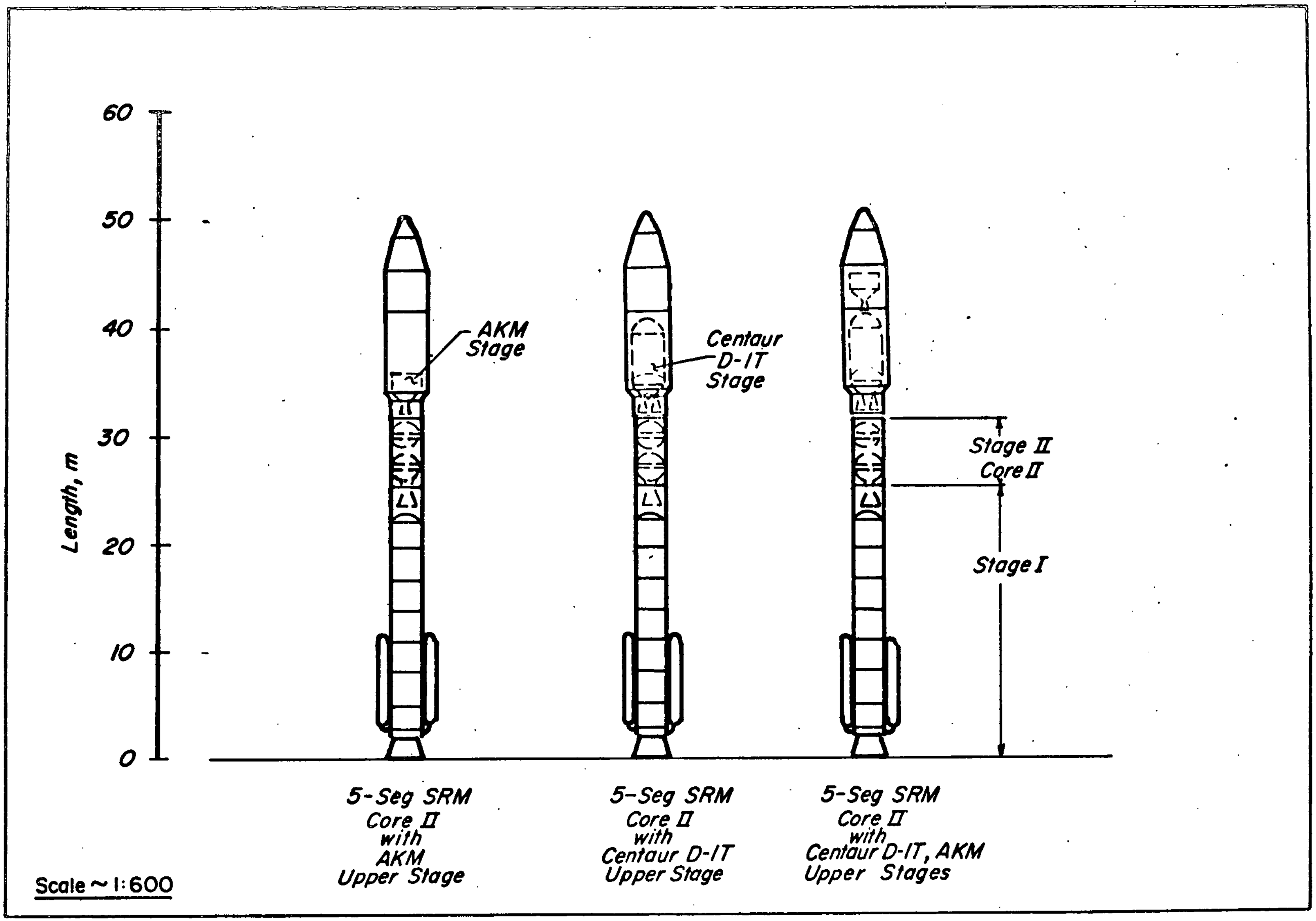


Figure 5.20 New Expendable Launch Vehicles—5 Segment SRM/Core II/with Centaur and AKM Upper Stages

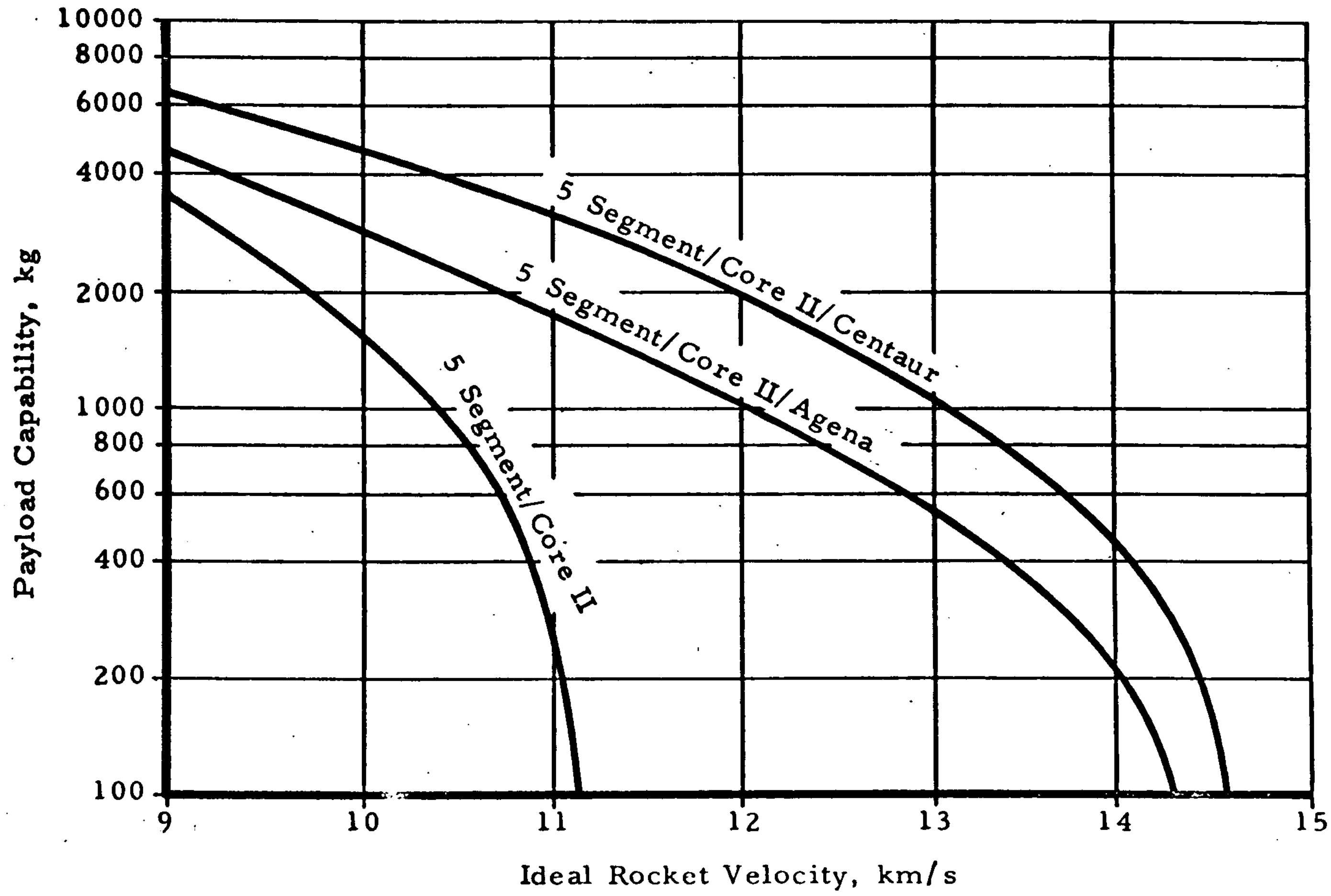


Figure 5.21 5 Segment SRM/Core II Launch Vehicle Family Performance

5.4.3 Large Payload Class

The large payload class of new expendable launch vehicles consists of the Titan III derivative/growth family of large diameter core (LDC) configurations, 4.58 meters diameter. These proposed Titan LDC configurations would use four LR-87 liquid rocket engines for Stage I. The LDC Stage II, 4.58 meters diameter would use one LR-87 engine. The SRM's for the Titan III L series of vehicles are the same as the seven-segment SRM's used with the stretched core vehicles mentioned previously in Section 5.2.2. The Titan III LDC vehicles can be used without an upper stage; however, optional upper stages are Transtage and Centaur. The proposed Titan III LDC vehicles can be used without an upper stage; however, optional upper stages are Transtage and Centaur. The proposed Titan III LDC configurations are called the L 2, L 4 and L 6 depending on the number of solid strap-ons used. However, only the L 4 version was assigned payloads in the 1979-1990 time period.

The Titan III L 4 is a three stage launch vehicle with a 4.58 meter diameter core (Stages I and II), and four seven segment SRM's (Stage 0). Typical configurations showing the L*4 without an upper stage and with the Centaur upper stage are given in Figure 5.22. The LDC Stage I uses a storable propellant consisting of fuel that is a 50-50 mixture of hydrazine and UDMH, and nitrogen tetroxide as the oxidizer. The propellant loading for Stage I is 476 metric tons. Stage II or LDC Core II uses the same propellant with a loading of 87,430 kilograms. A bulbous payload envelope measuring 6.7 meters diameter by 9.15 meters length is proposed for the L*4; however, later studies have shown that payload envelopes as large as 10.07 meters diameter by 11 meters length can be used [1]. The performance of the Titan III L*4 vehicles is shown in Figure 5.23.

5.5 Space Shuttle Concepts

5.5.1 Phase B Studies

The Space Shuttle was initially proposed as a new, reusable type, manned Space Transportation System designed to deliver and return various types of payloads and passengers between the earth's surface and low

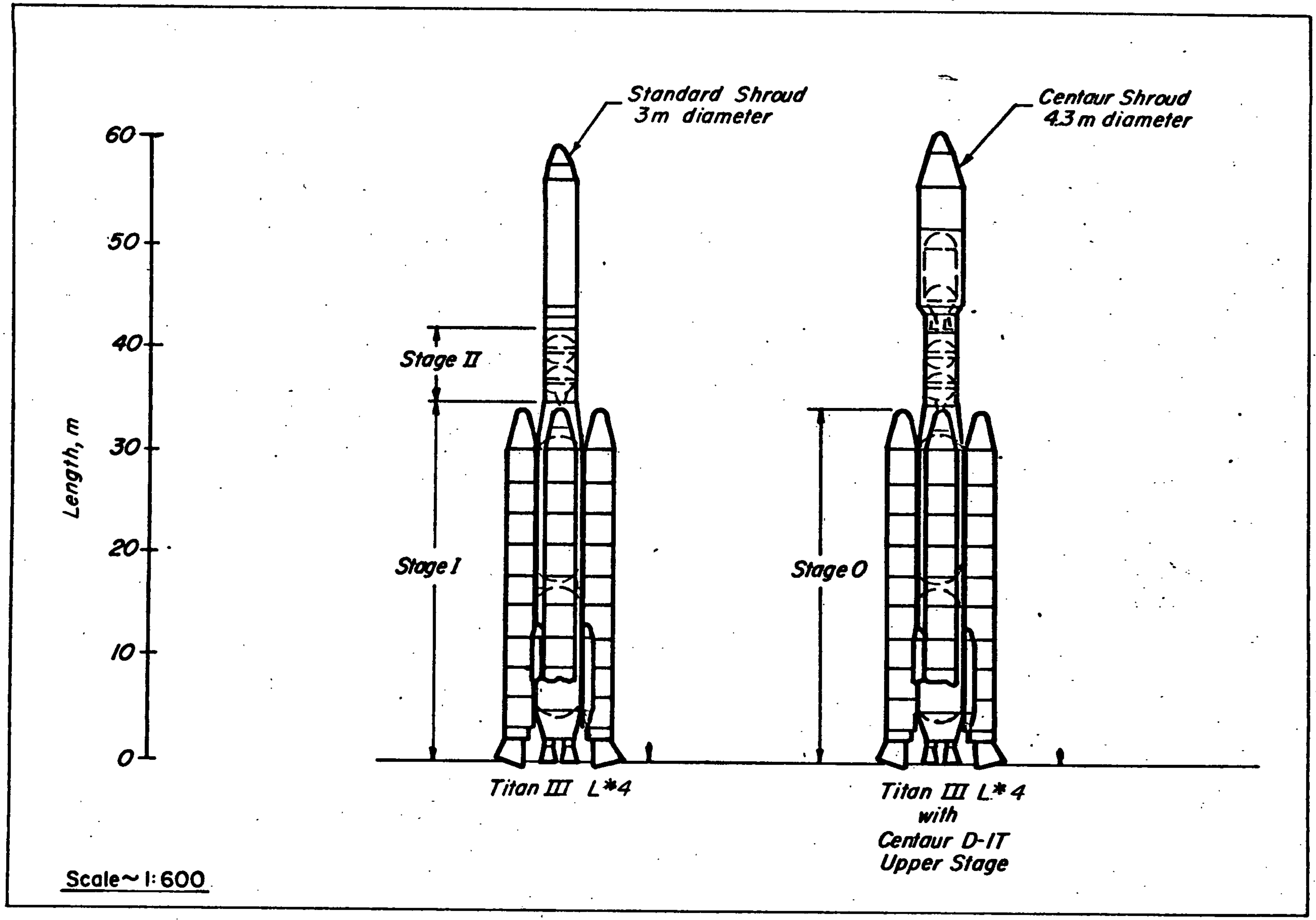


Figure 5.22 New Expendable Launch Vehicles—Titan III L*4 and Titan III L*4/Centaur

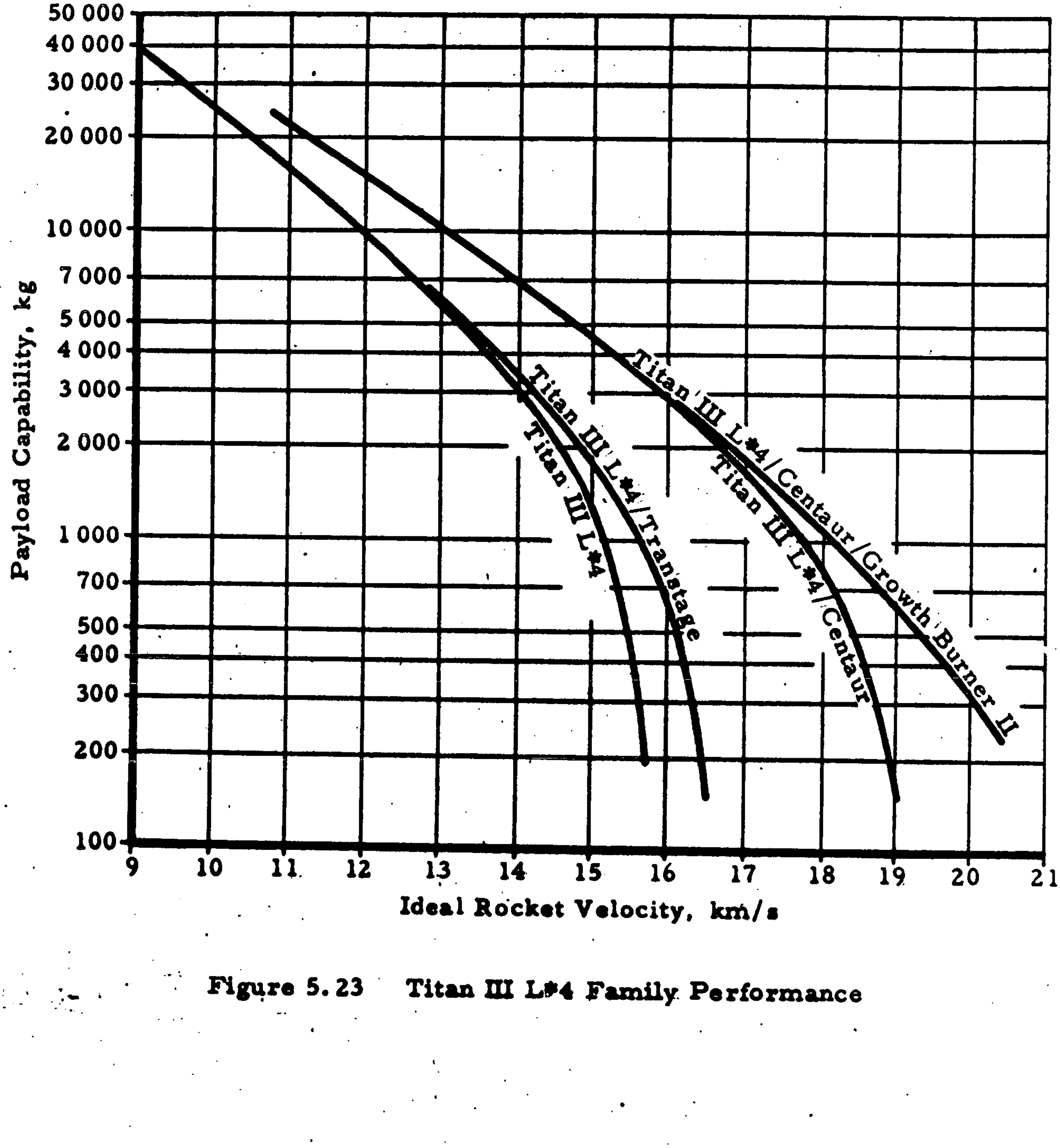


Figure 5.23 Titan III L*4 Family Performance

earth orbit. During the Phase B studies (ended June 30, 1971), the new Space Shuttle was envisioned as part of a Space Transportation System that would safely, efficiently and economically perform most future United States space missions. Under contract to NASA, various teams of aerospace companies conducted studies that resulted in the definition of the Space Shuttle (as dictated by NASA Directives and Requirements) as a fully reusable, two-stage Space Transportation System known as the original Phase B Baseline concept). The implications of the requirements for <u>full reusability</u> and <u>two stages</u> yielded launch configurations that, in a very general way, represented two large rocket powered airplanes (a manned orbiter vehicle and a manned booster vehicle) that were mated in a "piggy-back" fashion and launched vertically. These Phase B configurations, offered by various teams of aerospace companies such as McDonnell Douglas, North American Rockwell and Grumman/Boeing, are described in numerous publications [1, 57, 59 to 65] that define the evolution and changes that occurred to the original shuttle baseline design criteria as detailed by NASA and DoD Requirements and Technical Directives [37].

It is important to note that the emerging shuttle design consisted of a very large winged booster vehicle (essentially full of ascent propellants) and an orbiter vehicle (usually a delta or double-delta winged configuration) that contained large internal propellant tanks and a standard-sized internal payload bay. The large internal propellant tanks of the orbiter for oxygen and hydrogen) contained the additional propellant needed (after staging from the booster) to achieve low earth orbit. However, near the end of the Phase B studies it became apparent that an interesting innovation could be made in the Space Shuttle design by carrying the orbiter ascent fuel (hydrogen) in two, expendable, external tanks (mounted to the side of the orbiter's fuselage and above the wings) that could be ejected when empty. Some of the proposed advantages cited for this orbiter/booster concept design change were lower gross lift-off mass, lower cost, fewer engines, less TPS variety and a generally more flexible system [62].

During the last few months of the Phase B studies, the major contractor teams conducted studies of modified space shuttle designs (the later Phase B Baseline concept) incorporating expendable, external, hydrogen fuel tanks on the orbiter and utilizing a fully reusable heat sink type booster [5-65]. (Also, at that time, another company suggested an orbiter type design where the expendable externally mounted drop tanks contained all the orbiter's ascent propellants; both the oxygen and hydrogen [5-63].

The general impact (and potential advantages) of the various proposed alternatives incorporating the hybrid or partially-reusable orbiter/booster systems was sufficient to encourage further analysis of the "alternate" Space Shuttle concepts (Phase B-Extension Studies). The economic analysis shows that there exists a considerable latitude in trading off expected higher costs per launch in the 1980's for lower non-recurring costs in the 1970's (Reference 5-64, Chapters 2, 6 and 7).

New design guidelines permitted not only complete re-design of the orbiter but also redirected the booster design to include not only the fully reusable flyback booster but also, many new designs incorporating recoverable boosters that could be refurbished and reused, and completely expendable boosters. These alternate space shuttle-orbiter/booster designs are discussed in the next section.

5.5.2 Phase B Extension Studies

The Phase B Extension Studies were directed toward new analyses of Space Shuttle alternate designs and program options associated with a phased pattern of development. The direct development of the external hydrogen tank orbiter/heat sink booster, fully reusable Space Shuttle System (a Phase B Baseline option) revealed major areas of concern such as high peak annual and near term funding requirements, high total R&D investment, and high technical development risk. The primary emphasis of the Extension Studies was the determination of alternatives associated with decreased capability and more favorable expenditures and included evaluations of the merits of vehicle design/configuration changes and new program options. Among the considerations were the delay of the fully reusable booster development and the initiation of comparison studies of different booster systems for the shuttle orbiter. Also, there was a re-definition of the orbiter concept that dictated the use of external propellant tanks and reduced payload capability with emphasis on lower non-recurring costs. The initial options associated with low risk technology systems were studied in order to determine the possibilities of a continued process of upgrading to achieve the desired final capabilities later in the program. And finally, alternate development and program schedules were analyzed with regard to reductions in peak funding and near term expenditures (i.e., RDT&E and initial fleet investment).

The impact of the new guidelines on Space Shuttle concepts was enormous and many new alternative configurations were analyzed. Since a phased program could utilize an interim, expendable booster and reduced capability orbiter for the first years of operation, the extension studies centered on the selection of the most promising initial systems and the optimum paths toward the improved capability concepts.

Some of the alternative options and configurations resulting from these studies are discussed in the following sections.

5.5.2.1 Baseline Orbiter

As a result of the analyses during the final months of the Phase B studies and initial Extension studies, NASA detailed new system requirements that resulted in the definition of a "Baseline Orbiter" for the alternate Space Shuttle concepts. The Baseline Orbiter, designated as the 040A, is shown in a three view drawing in Figure 5.24 and an inboard profile is shown in Figure 5.25. [66, 67].

The Baseline Orbiter considered by November, 1971 is a delta wing vehicle with three main rocket engines for ascent propulsion. There are no internal ascent-propellant tanks since all orbiter ascent propellant (oxygen/hydrogen) is carried externally and fed to the orbiter main rocket engines through a propellant interface connection on the orbiter. The combination of the Baseline Orbiter together with its external hydrogen/oxygen propellant tanks was frequently referred to as the "HO orbiter".

In addition to the three main rocket engines located at the aft end of the orbiter, two orbital maneuvering system modules (OMS) are attached to the rear fuselage section. The OMS modules contain the OMS propellant

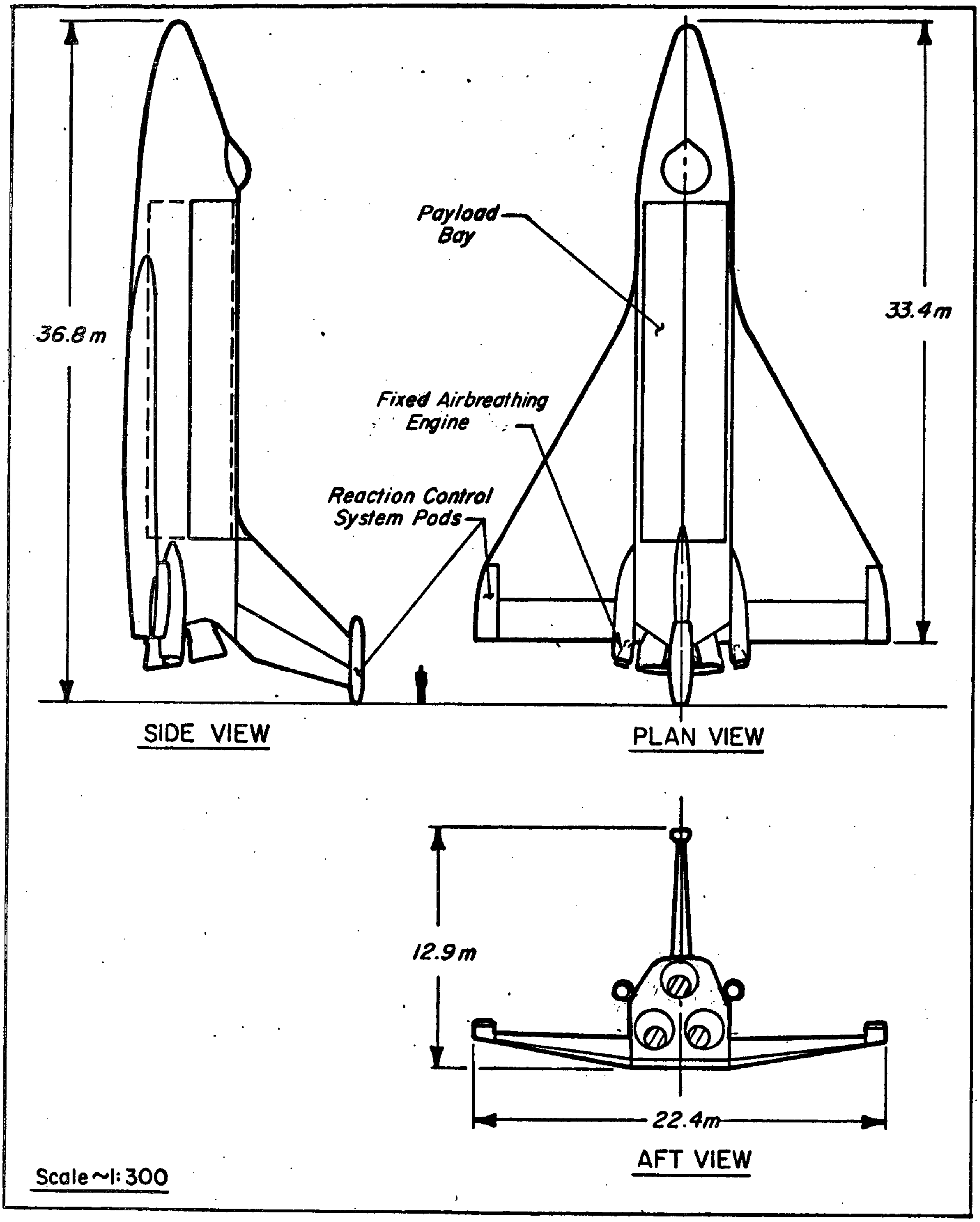


Figure 5.24 Current Baseline Orbiter-Three View

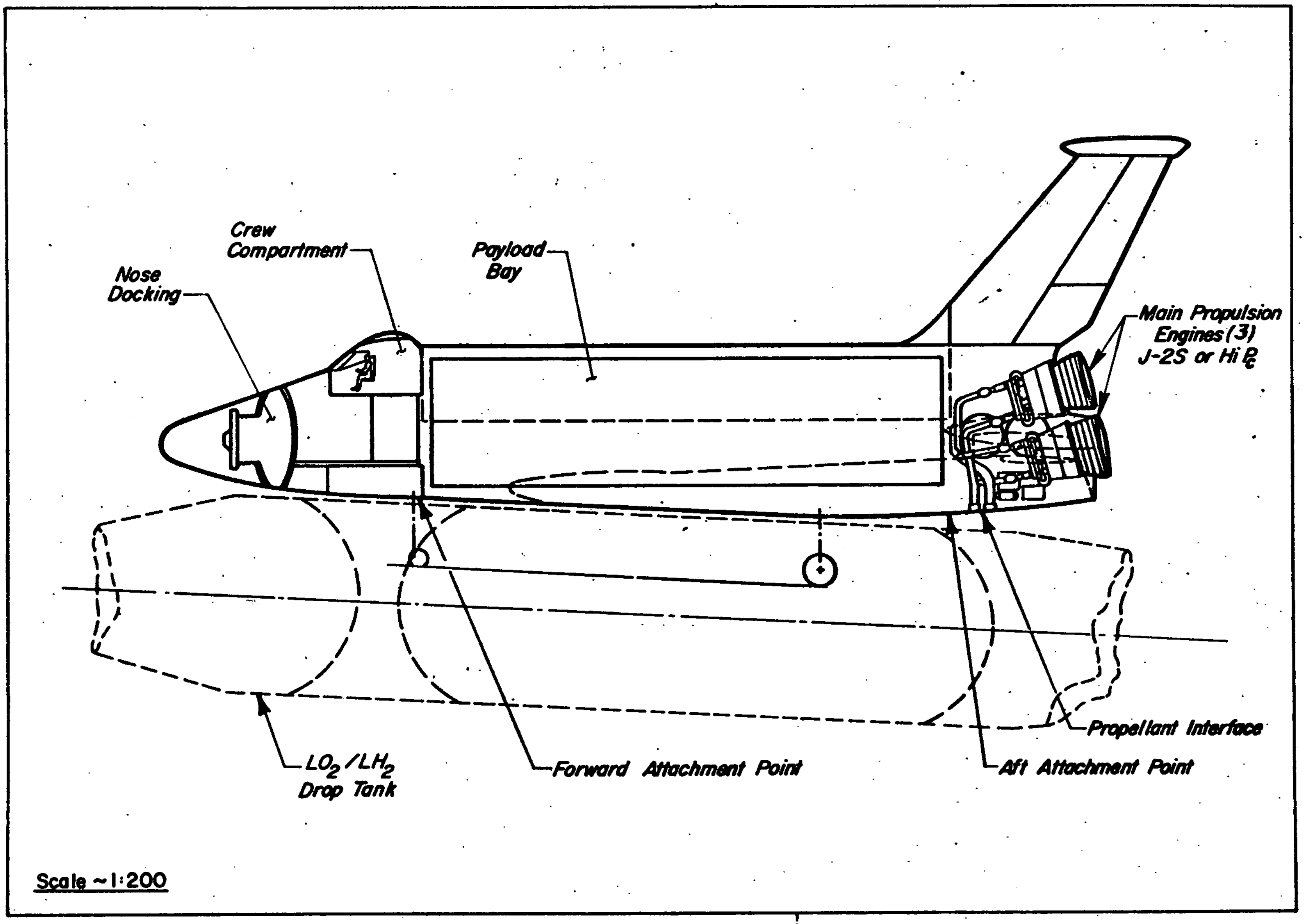


Figure 5.25 Current Baseline Orbiter -- Inboard Profile

tanks and rocket engine assembly designed to provide the necessary large maneuvering velocity increments required for rendezvous circularization, plane change, transfer and de-orbit thrust. One candidate for OMS engine selection is the Lunar Module Ascent engine using N_2O_4/A-50 propellants.

If the orbiter is required to cruise-fly a long distance cross range to a landing site after re-entry, an airbreathing engine system (ABES) would be included in the orbiter. Normally the ABES would utilize two gas turbine jet engines of the class JTF 22A or F101/F12A (JP type fuel) for air-breathing propulsion.

The orbiter also contains a type of reaction control system (RCS) or attitude control propulsion system (ACPS) to provide the propulsion needed for attitude control in space or to supplement control at the higher atmospheric altitudes. This type of control system typically utilizes a total of about 34 thrusters (with N_2H_4 propellant) arrayed in various pods or modules (located in the nose section, top of fins or wing tips) [68]. The RCS provides continuous vehicle attitude orientation and control in all six-degrees-of-freedom and facilitates precise control maneuvers such as terminal rendezvous, docking in space and re-entry attitude.

The Baseline Orbiter contains the standardized or nominal payload bay that will accommodate any candidate payload envelope measuring up to 4.6 meters in diameter and 18.3 meters in length.

One important consideration of the Baseline Orbiter is selection of the three or four main rocket engines. Several approaches were considered, e.g. an engine whereas, the later or upgraded orbiter versions will utilize the higher performance Hi-Pc (SSME) engines. This was consistent with the goals of the Phase B Extension Studies and the frequently used designators referred to the initial and upgraded Space Shuttle versions, respectively. For example, the Mark I or initial version of the orbiter would have utilized the J-2 type engines, a simpler thermal protection system (TPS) made up of expendable or ablative materials, existing aircraft and spacecraft type avionics, and have reduced capability in payload mass and flyback cross range. The Mark II or upgraded version of the orbiter would have utilized the better performance Hi-Pc main rocket engines, a more sophisticated reusable type TPS, an advanced integrated avionics system and have the full desired payload mass capability as well as the longer (2,037 kilometer) cross range fly-back capability. The (new) Baseline Orbiter has now been defined as using three high performance (SSME) engines.

The ascent propellant supply (LO_2/LH_2) for the orbiter's main rocket engines is always provided by external tanks. Among the various concepts, the HO drop tanks tended to vary in number (one to three), in size (both length and diameter), in mating arrangement, and in propellant supply or burn sequence (either parallel burn or series burn with respect to first stage or booster). Tank dry mass varies greatly (from about 20,000 kilograms to about 50,000 kilograms) depending on sizes, configurations and concepts. All orbiter concepts must execute a tank separation sequence and tank disposal maneuver after the main engines shut down and the powered ascent phase the trajectory is satisfactorily completed.

The typical orbiter characteristics are listed in Table 5.8 [66].

5.5.2.2 <u>Current Booster Concepts</u>

The Space Shuttle program allows the use of a variety of alternative booster concepts or interim expendable boosters that could be updated or even improved to include manned versions, new boosters that could be recovered and refurbished for reuse, and eventually new, flyback, manned boosters of the fully reusable type. The Phase B extension studies included all categories of existing boosters and their derivatives as well as new or used designs. Some of the vehicles or systems analyzed were Saturn S-IC or F-1 powered vehicles, Titans, SRM's (120, 156, 180, 260) and new pressure-fed systems.

Also, similar designators such as those used for Mark I and Mark II of the orbiter were applied to the booster concepts to distinguish between the initial designs and the updated versions. For example, the Mark II version of a booster, would have more sophisticated reusable TPS systems and more advanced integrated electronics systems than that of a Mark I type booster.

Although a very wide variety of different launch configurations were

Table 5.8

Typical Orbiter Characteristics
(Phase B Extension studies)

Gross Mass (Polar)	89,500 kg
Landing Mass (Polar)	79,000 kg
Dry Mass	59,000 kg
Main Engines:	
Number/Type	4/Hi-Pc
Thrust, vac./each Engine	1.36 mN
Subsonic L/D	6.3
Hypersonic L/D	1.8
Aspect Ratio	1.7
Leading Edge Sweep	60^{o}
Crossrange	2,130 km
Landing Speed	278 km/hour

studied, the most recent booster concepts considered in the past seven months tend to fall into about four or five classes of manned and unmanned booster configurations. These are discussed below.

Manned Boosters

The alternate, manned, Space Shuttle boosters suggested by the different concepts were generally winged versions of large SIC type or F-1 powered launch vehicles. A typical launch configuration is shown in Figure 5.26. A Baseline Orbiter with HO tank is tank-end-loaded atop a reusable flyback booster. This configuration designated as the " Reusable S-IC booster" (R-SIC) or "F-1 Flyback booster" [66, 68] utilizes five F-1 type rocket engines and SIC tankage with $L0_2$/RP propellants. It is a series-burn system since the HO tank orbiter ignites its main rocket engines after booster staging and continues its powered ascent trajectory to low earth orbit.

At this time, a review of certain available information on S-IC versions seems to indicate that other contractor concepts of this particular class and type are embodied in the general description given in the above discussion and figure. For example, another contractor's version of the reusable S-IC configuration is called the "HO/Flyback" concept and utilizes F-1 engines and S-IC components for a winged booster that is tank-end-loaded with an HO orbiter [66]. Generally speaking, the configurations tend to be similar.

During the Phase B Extension Studies regarding the reusable S-IC/F-1 powered booster concept, use was made of the designations "Block 1 program" and "Block II program." These designations were utilized to indicate initial and upgraded programs, respectively (in a similar manner as the designators Mark I and Mark II are used) and referred to progressive technology changes to systems and additional capability in number of launch complexes and number of operational vehicles [69]. For example, the reusable S-IC "Block I" program relied on existing F-1 engine technology (five for the booster) and existing J-2 engine technology (five for the orbiter) with two complete operational vehicle systems. Also, the Block I program R-SIC vehicles would use commercial based avionics with minimum on-board checkout.

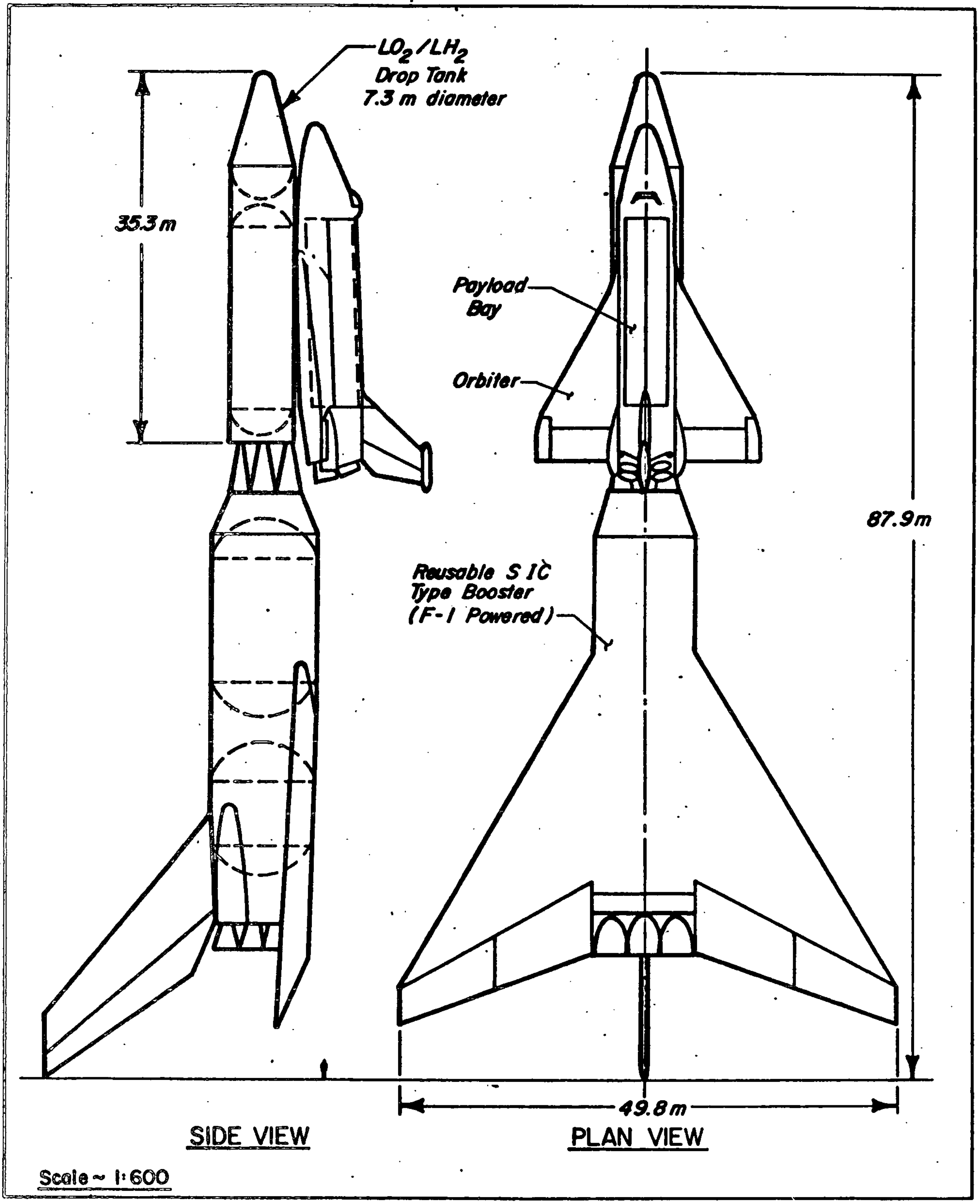


Figure 5.26. Reusable S-IC Booster (F-1 Flyback) Concept—Launch Configuration

The R-SIC Block II program would make use of advanced technology (F-1A/J-2S engines) and upgraded avionics in the form of automatic landing devices and advanced on-board checkout equipment together with the implementation of multiple launch complexes. Another reference discusses the R-SIC thermal protection systems, where the Block I TPS system would utilize ablative materials and the Block II TPS would use upgraded materials including HCF, highly compacted fibers [70].

It should be noted that several contractors have suggested similar unmanned and un-winged design utilizing the S-IC system as an expendable booster. These were proposed to be used for the earliest Shuttle launches with the HO Baseline Orbiter and are called the "S-IC Booster/HO Orbiter" concept [68].

Some of the characteristics of a typical reusable S-IC (F-1 powered) concept are listed in the following table Table 5.9 [66].

Unmanned Boosters

Parallel Burn Thrust Assisted Orbiter Shuttle (TAOS)

These alternate Space Shuttle concepts have been given the acronym, TAOS, by MATHEMATICA and include all parallel-burn, thrust assisted orbiter shuttle types. They embody all configurations using an external HO tank-Baseline Orbiter and some type of thrust assist either in the form of solid rocket motor boosters or liquid rocket motor boosters. Typical TAOS designs are shown in Figures 5.27 and 5.28 [66].

In Figure 5.27 the HO Baseline orbiter is shown mated, tank-side-loaded, with two 3.96 m 4 segment solid rocket motor (SRM) boosters containing 1,043,000 kg total propellant. The HO tank is center mounted underneath the Baseline Orbiter and contains 590,000 kg of usable ascent propellant (LO_2/LH_2) for the orbiter's main rocket engines.

This parallel-burn concept has been variously named as "rocket assisted takeoff" (RATO) concept and similar configurations and designations such as "rocket assisted orbiter" (RAO) concept and thrust augmented hydrogen oxygen (TAHO) concept have been offered by other contractors for these classes of orbiter/booster systems.

Table 5.9

Typical Characteristics of a Reusable S-IC Concept

Gross Lift Off Mass (GLOM)	2,540,000 kg
Staging Velocity	2,140 m/s
Maximum Dynamic Pressure	31,000 N/m^2
Booster Dry Mass	284,200 kg
Orbiter Dry Mass	59,000 kg
Booster Propellant Mass	1,765,000 kg
Orbiter Propellant Mass	318,000 kg
Orbiter Tank Dry Mass	16,000 kg
Main Engines:	
Booster, Number/Type	5/F-1
Orbiter, Number/Type	4/Hi-Pc
Engine Thrust	
Booster S.L. each	6.9 mN
Orbiter S.L. each	1.36 mN
Cruise Engines, Number/Type	9/GE 101-12-B3
Flyback Range	42.6 km

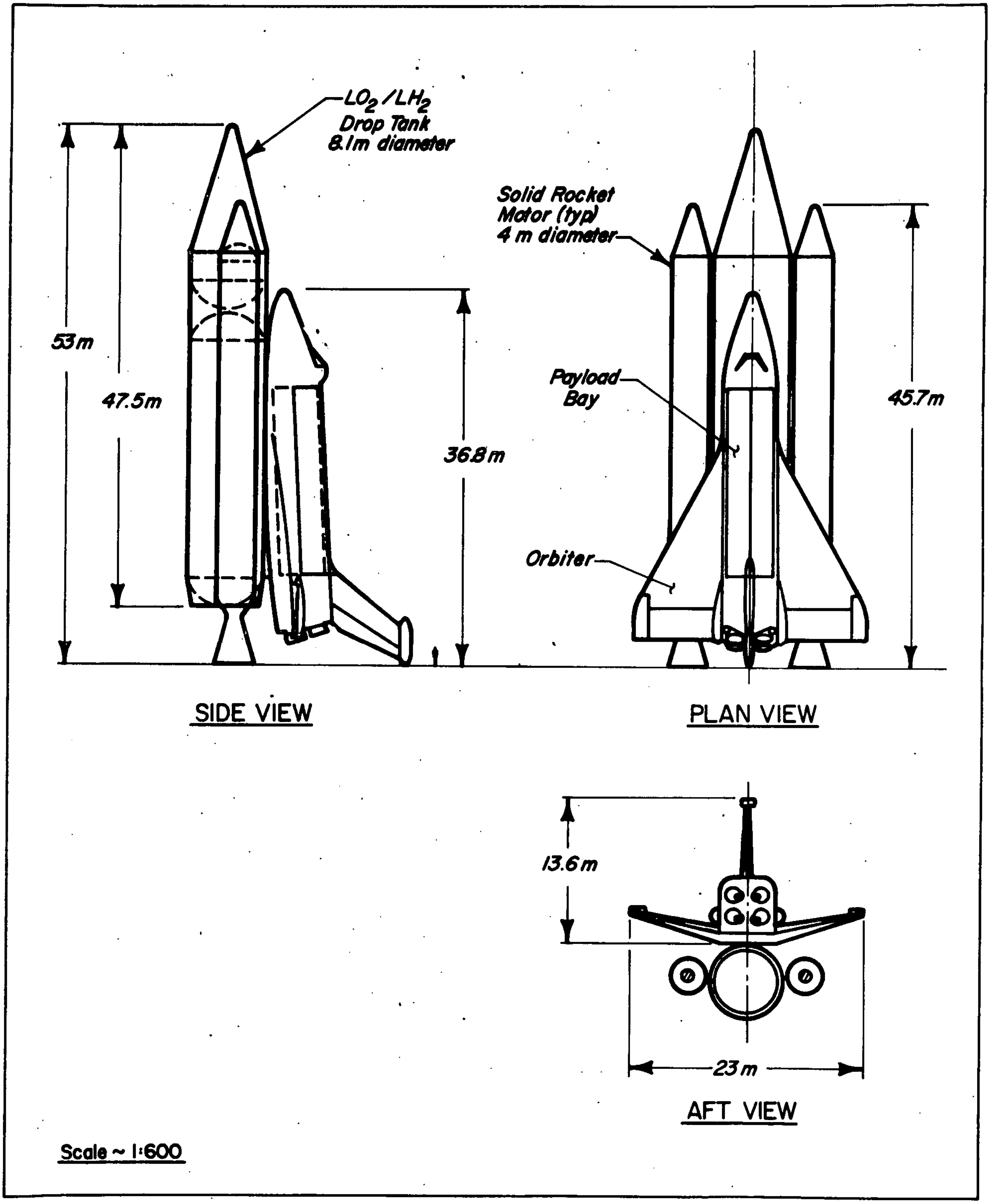


Figure 5.27 Baseline Orbiter with Twin Solid Rocket Motor Booster (TSRM) Concept, Parallel-Burn—Launch Configuration

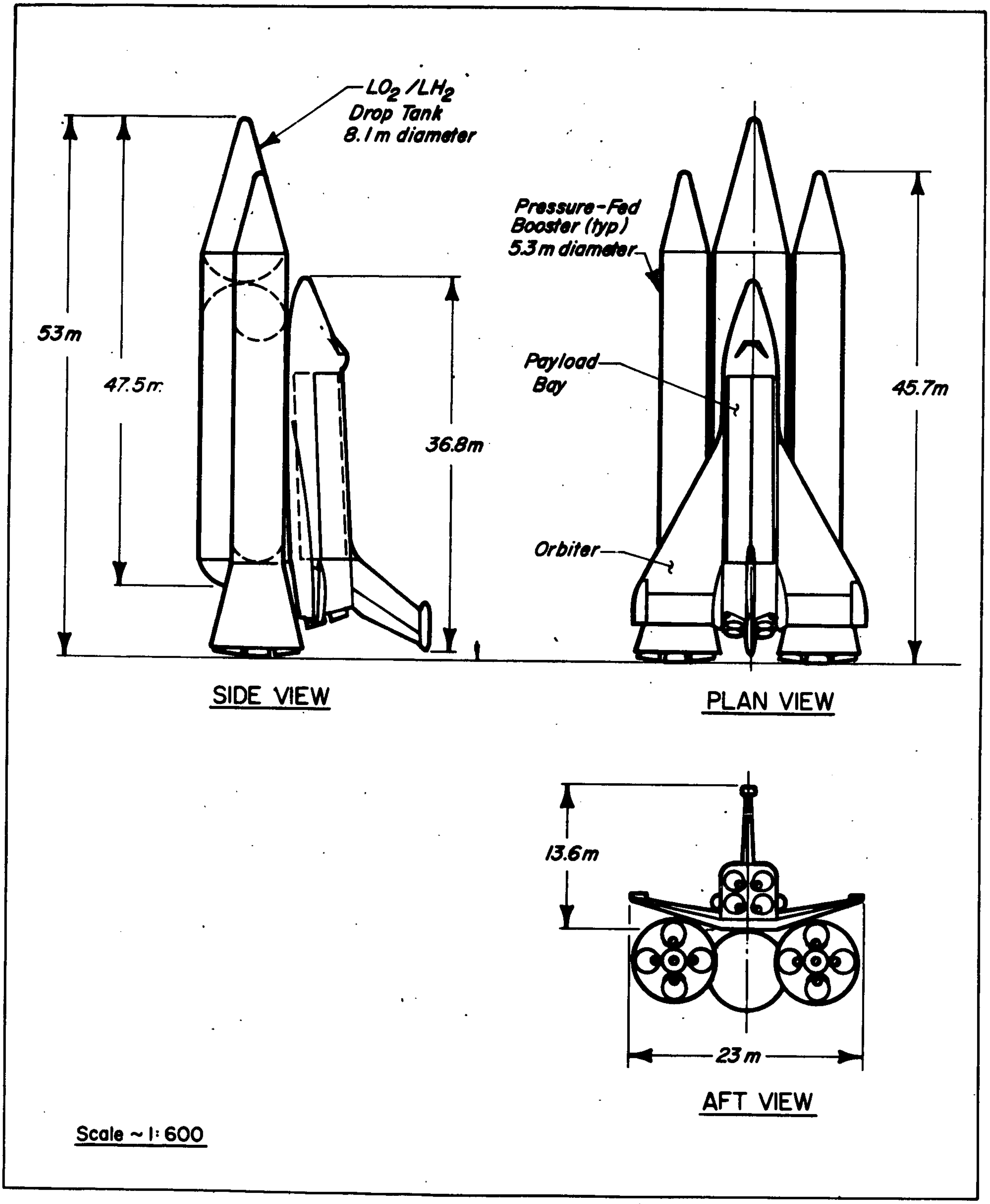


Figure 5.28 Baseline Orbiter with Twin Pressure Fed Booster (TPFB) Concept, Parallel-Burn—Launch Configuration

In Figure 5.28 the HO baseline orbiter is shown mated, tank-side-loaded, with two pressure-fed (LO_2RP) recoverable boosters. The propellant mass for the booster is 1,175,000 kg and for the orbiter 590,000 kg (LO_2/LH_2). It is presumed that procedures will be developed to recover the pressure-fed boosters (after staging and splashdown) for refurbishment and reuse. The oribter's HO tank will be expended due to re-entry.

It is proposed that both of these parallel-burn systems will rely on the thrust vector control and aerodynamic control of the orbiter for trajectory steering during ascent to orbit.

Some of the characteristics of typical parallel-burn systems are compared in Table 5.10 [66].

Some of the characteristics of typical parallel-burn systems are compared in Table 5.10 [66].

<u>Series Burn Boosters</u>

These new thrust assist booster concepts are identified not only by the burning sequence of stages (series-burn) but also by their size and/or the new technology they represent. In regard to burning sequence, series-burn generally means that the booster ignites its stage (s) first (to provide the total initial thrust required from the launching pad to the booster/orbiter staging point) and then the HO orbiter ignites its engines (after booster staging) to continue the powered ascent to low earth orbit. In regard to size and technology, these new series-burn thrust assist concepts represent an interesting variety of ideas and concepts. The boosters vary from single to multiple stage systems, and consist of either liquid rocket motors or solid rocket motors or a combination of the two. The propellants suggested for use vary from LO_2/RP, $N_2O_4/A\text{-}50$, LO_2/LH_2, LO_2/Propane and PBAN or HTPB (solids). The liquid rocket motors may represent existing technology (Titan III series <u>pump-fed</u> LR-87 systems) or the newer technology associated with very large <u>pressure-fed</u> systems. Similarly, the SRM's may represent existing technology (like the 120 SRM) or newer technology (like the 156, 180, and 260).

Table 5.10

Typical Characteristics of Parallel-Burn Shuttle Concepts

	Twin Pressure Fed	Twin SRM
Gross Lift Off Mass, kg	2,080,000	1,890,000
Staging Velocity, m/s	1,675	1,675
Maximum Dynamic Pressure, N/m^2	31,000	31,000
HO Tank Mass, kg	24,400	24,400
Main Engines:		
Booster (number & type)	2 X 5 pressure fed	2/156/4-seg
Orbiter (Mark II; No. & type)	4/Hi-Pc	4/Hi-Pc
Engine Thrust:		
Booster, S.L. each - mN	2.76	10.7
Orbiter, vac. each - mN	1.36	1.36

One of the more interesting aspects of the new booster concepts is that, through the use of the proper, special, design logic and the development of the appropriate operational techniques, the boosters can be recovered, refurbished, and re-used. This is being considered for the entire large pressure-fed boosters and for the solid rocket motor boosters also (or at least some components of them).

A typical large, series-burn, pressure-fed booster/HO orbiter launch configuration is shown in Figure 5.29 [66]. It is represented by an HO orbiter (the baseline orbiter complete with its underslung HO propellant drop tank) mounted atop (tank-end-loaded) the single (large, 10.4 meter diameter), pressure-fed booster.

Some of the baseline characteristics of systems typical of this pressure-fed booster concept are given in Table 5.11 [66].

Results of some preliminary studies have revealed that high tank pressure dominates the structural design, tank materials have been tentatively selected and propellants could either be LO_2/Propane or N_2O_4/UDMH. Intact ocean recovery, survival, and retrieval appear feasible and reasonable but some small percentage (5-10 percent) loss rate might be expected. However, certain key issues exist and are debated. Among them are [66]: lack of engine data, aerodynamic configuration required for re-entry, water entry loads, seaworthiness criteria, propulsion system selection, structural configuration and materials, refurbishment costs, and development of an operational recovery system. Each of these key issues increases in importance as one proceeds from Twin Parallel Burn Boosters to Series Burn Boosters.

Two ocean recovery system alternatives have been studied. One is a high angle of attack re-entry and the other is a low angle of attack re-entry with drag devices. Each of these permits two other alternatives for the final descent; one is rocket braking and the other is a drogue chute deploy with multiple main parachutes for splashdown or a combination.

Again, it should be noted that, with respect to the launch configuration designs, other contractors have offered similar class concepts. One of these uses the nomenclature "pressure-fed, ballistic recovered booster (BRB)" configuration [68] and, in a general way, is similar to the aforementioned

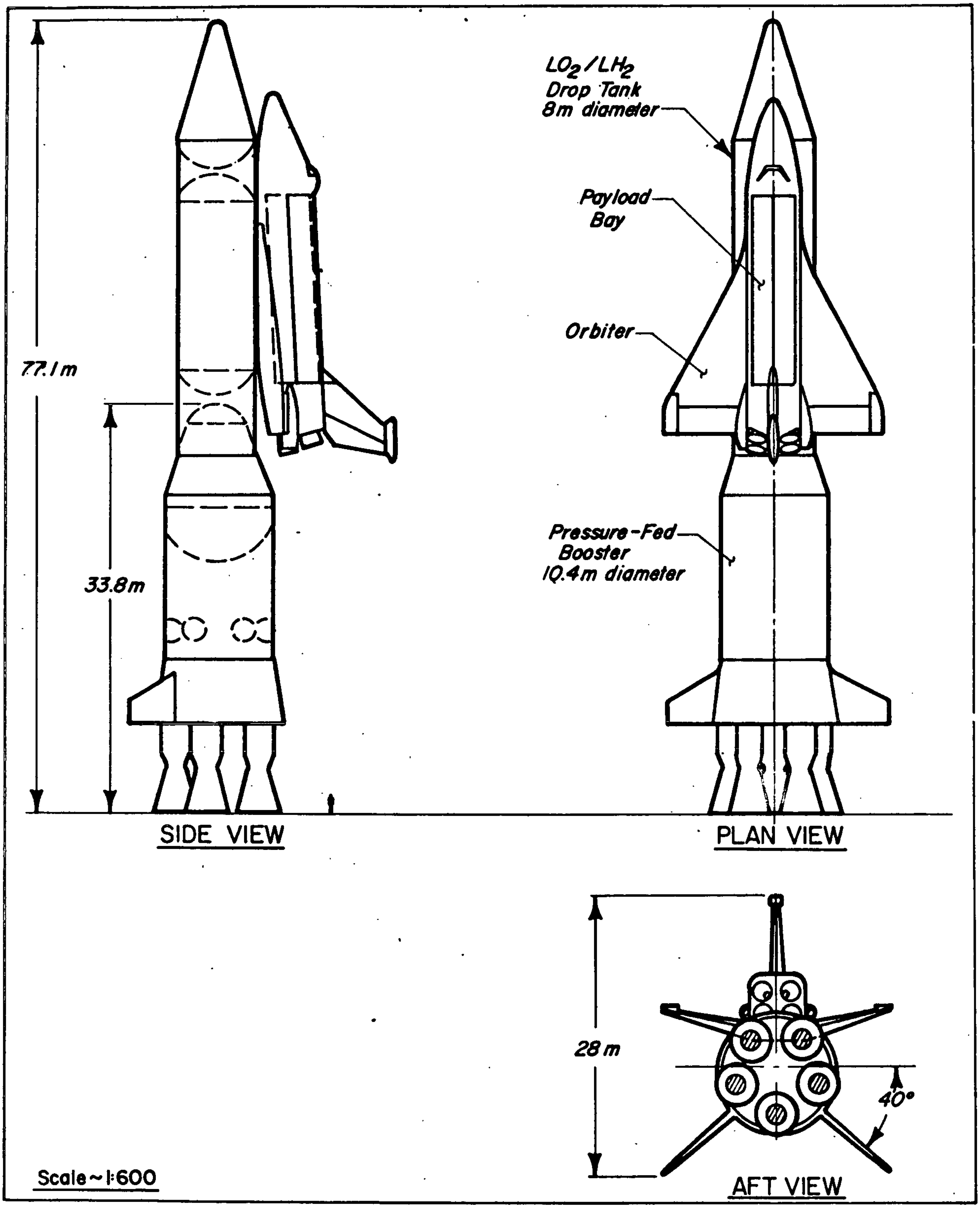


Figure 5.29 Baseline Orbiter with Large, Single Pressure Fed Booster (SPFB), Series-Burn—Launch Configuration

Table 5.11

Typical Characteristics of Single, Large Pressure-Fed Booster Concepts.

Gross Lift Off Mass, GLOM	2,350,000 kg
Staging Velocity	1,630 m/s
Maximum Dynamic Pressure	31,000 N/m^2
Dry Mass:	
Booster	220,000 kg
Orbiter	59,000 kg
Propellant Mass:	
Booster	1,588,000 kg
Orbiter	372,000 kg
Propellant type	LO_2/Propane
Orbiter HO Tank Dry Mass	19,000 kg
Main Engines:	
Booster: Number & type	5/pressure-fed
Orbiter: Number & type	4/Hi-Pc
Engine Thrust:	
Booster; each-S. L.	5,785,000 N
Orbiter; each-Vac.	1,360,000 N
Booster Pressurant; Volume/Type	35 $meters^3$/COLD He gas
Booster Engine Gimbal Angle	6° (square)
Booster Fin Area	71.5 m^2
Booster Splashdown Mass	282,000 kg
Booster Splashdown Velocity	45.7 m/s

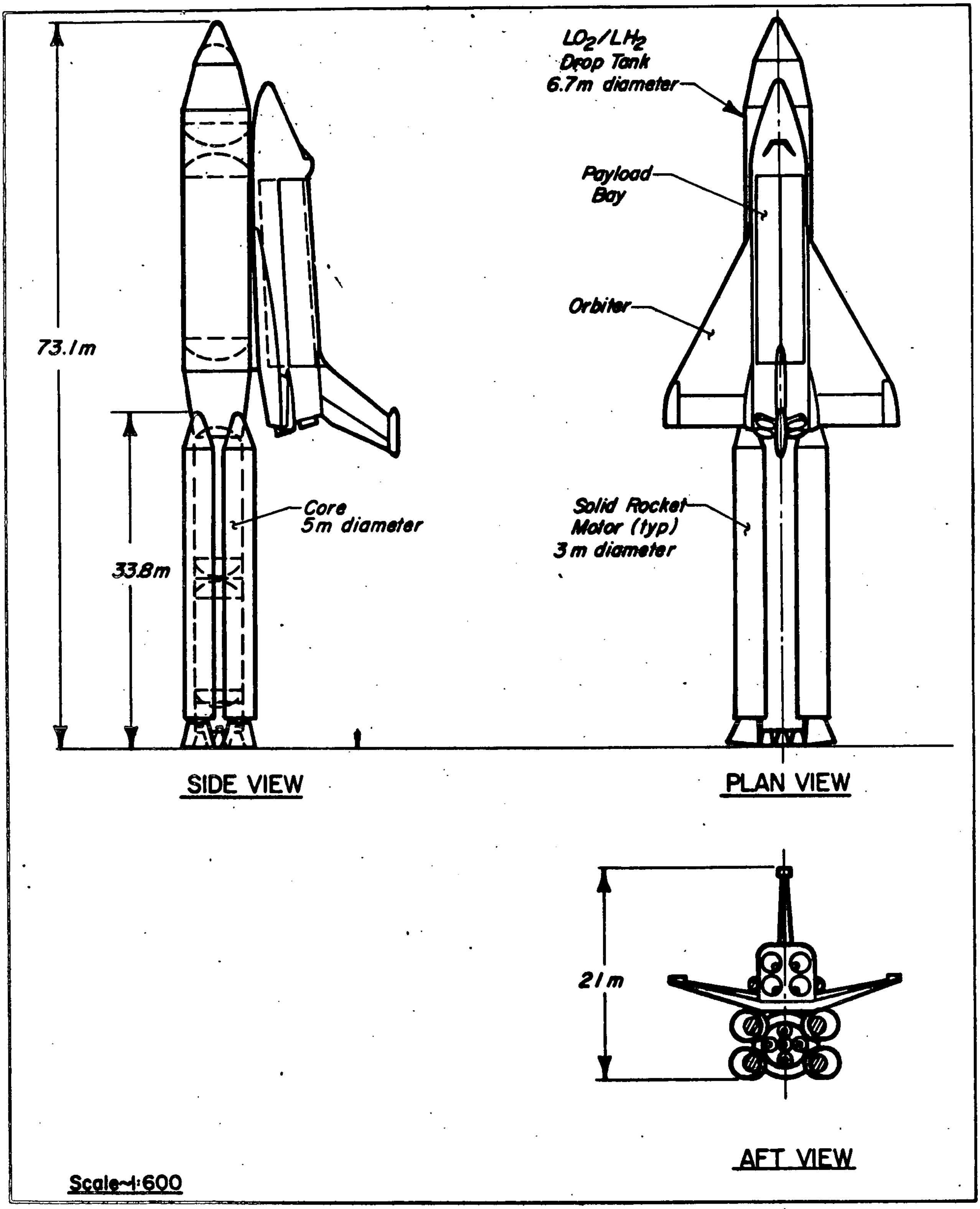


Figure 5.30 Martin Marietta Titan III L Booster Concept—Launch Configuration

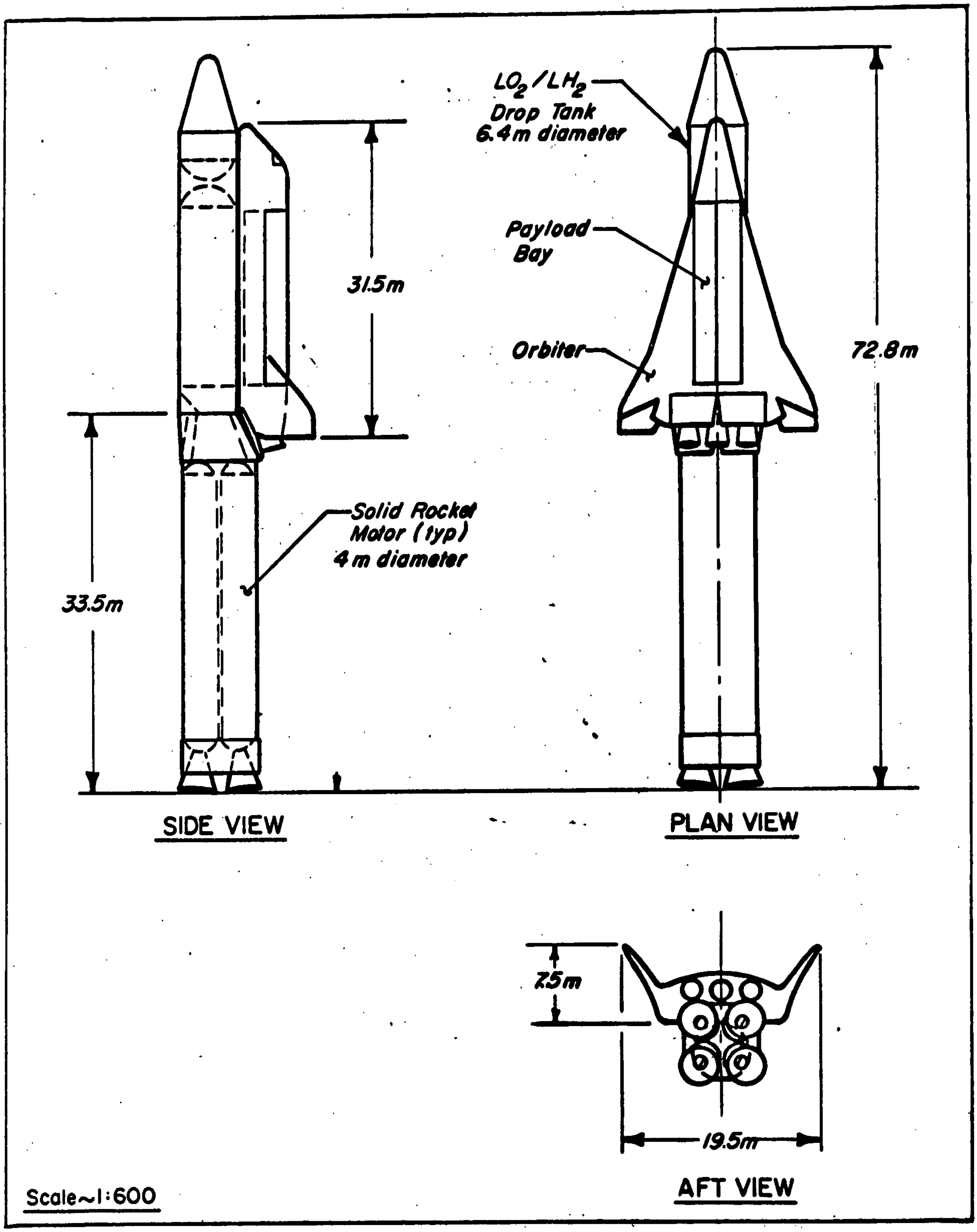


Figure 5.31 Lockheed Solid Rocket Motor Booster Concept—Launch Configuration

Table 5.12

Characteristics of a Titan III L 1207-4 Booster Concept

Gross Lift-off Mass	2 260 000 kg
Booster Lift-off Mass	1 810 000 kg
Propellant Type	Liquid and Solid
SRM, Quantity and Size	4 - 3 m
LRM, Quantity and Size	1 - 5 m

Table 5.13

Characteristics of the Lockheed 5B Series-Burn Solid Rocket Motor Booster Concept

Gross Lift-off Mass	2 345 000 kg
Booster Lift-off Mass	1 860 000 kg
Propellant Load	1 626 000 kg
Propellant Type	Solid Only
SRM, Quantity and Size	4 - 4 m

concept.

Another series-burn booster concept worthy of some note evolves from the role created for the Titan launch vehicle family by DoD and NASA, namely, the Titan III L series that utilizes both solid and liquid propellants. The booster system comprises a storable propellant core structure (containing N_2O_4/A-50 for five pump-fed LR-87 engines) and various arrays of multi-sized and multi-segmented solid rocket motor strap-ons. One such concept is given in Figure 5.30 and shows a baseline orbiter (with HO drop tank) mounted atop (tank-end-loaded) a Titan III L-1207-4.

A third series-burn booster concept has appeared in a report issued by the Lockheed Missiles and Space Company [5.17] which is comprised of a single stage cluster of solid rocket motors. The cluster of four 3.96 m SRM's is tandem mounted (tank-end-loaded) with the orbiter hydrogen/oxygen tank (Figure 5.31.).

Some typical characteristics of the Titan III L 1207-4 (solid and liquid system) and the Lockheed launch vehicle configuration 5B are given in Table 5.12 and 5.13 [17 and 7]. The economic tradeoff problem among all the alternative booster concepts for the baseline orbiter are highlighted elsewhere in this report.

5.6 Space Tugs

The Space Shuttle is capable of placing sizable payloads into low earth orbit and returning these payloads to earth. However, it is not capable of significant orbital maneuvering beyond low earth orbit. Thus, it is necessary to provide a reliable and usually, for economical purposes, reusable rocket vehicle for taking payloads to orbits which the shuttle cannot achieve. This vehicle is presently referred to as the Space Tug and it is important to recognize the fact that the Space Tug is an integral part of Space Shuttle Transportation System. In the early phase of Space Shuttle operations, kick stages can be used in lieu of a Space Tug to place payloads to the derived orbiters.

The Space Tug is generally thought of as a high performance stage using H_2/O_2 propellant. However, while this is desirable for performance-limited missions, some missions exist that are space-limited in the shuttle cargo bay. For these missions, it is possible that a more dense propellant might be used, for example, liquid propane/liquid oxygen.

5.6.1 United States Tug Concepts

The baseline Space Tug and alternate configuration are shown as presented by Lockheed Missiles and Space Company [72] in Figure 5.32. Characteristic parameters for a typical United States Space Tug [1] are given in Table 5.14. A considerable number of Space Tug concepts have been studied with various arrangements, propellant combinations, etc., and these studies are continuing. The optimum concept and its operational performance will vary depending on the payload and velocity requirements; however, it is hoped that a standardized Space Tug can be configured that will handle practically all missions with little change in design or operation.

Some proposals have included modifications of existing stages, for example, the Centaur [73], or development of new, multi-purpose stages, for example, the versatile upper stage (VUS) [74]. The Centaur was originally designed as a technology vehicle to show the feasibility of a high-performance H_2/O_2 stage and as such it was not designed for high reliability. Subsequently, significant design changes must be incorporated to achieve the very high desired tug reliability. However, General Dynamics is presently reviewing the Centaur systems in an effort to increase its reliability. The VUS is also being currently studied as a long-life (2000 days) propulsion system for interplanetary applications. However, while these designs provide useful technologies, the optimum tug design should probably be the result of an original effort as opposed to a redesign of preconceived stages designed originally for non-tug missions.

5.6.2 European Tug Concepts

Two European teams have studied the Space Tug on a preliminary basis with interesting results [75, 76, 77]. Both designs are similar at present and typical configurations of a reusable tug and an expendable tug are shown in Figure 5.33. The performance of the European tug system is shown in Figure 5.34. In addition to the preliminary design studies, Delft University of Technology [78] performed a mission analysis study for a European Space

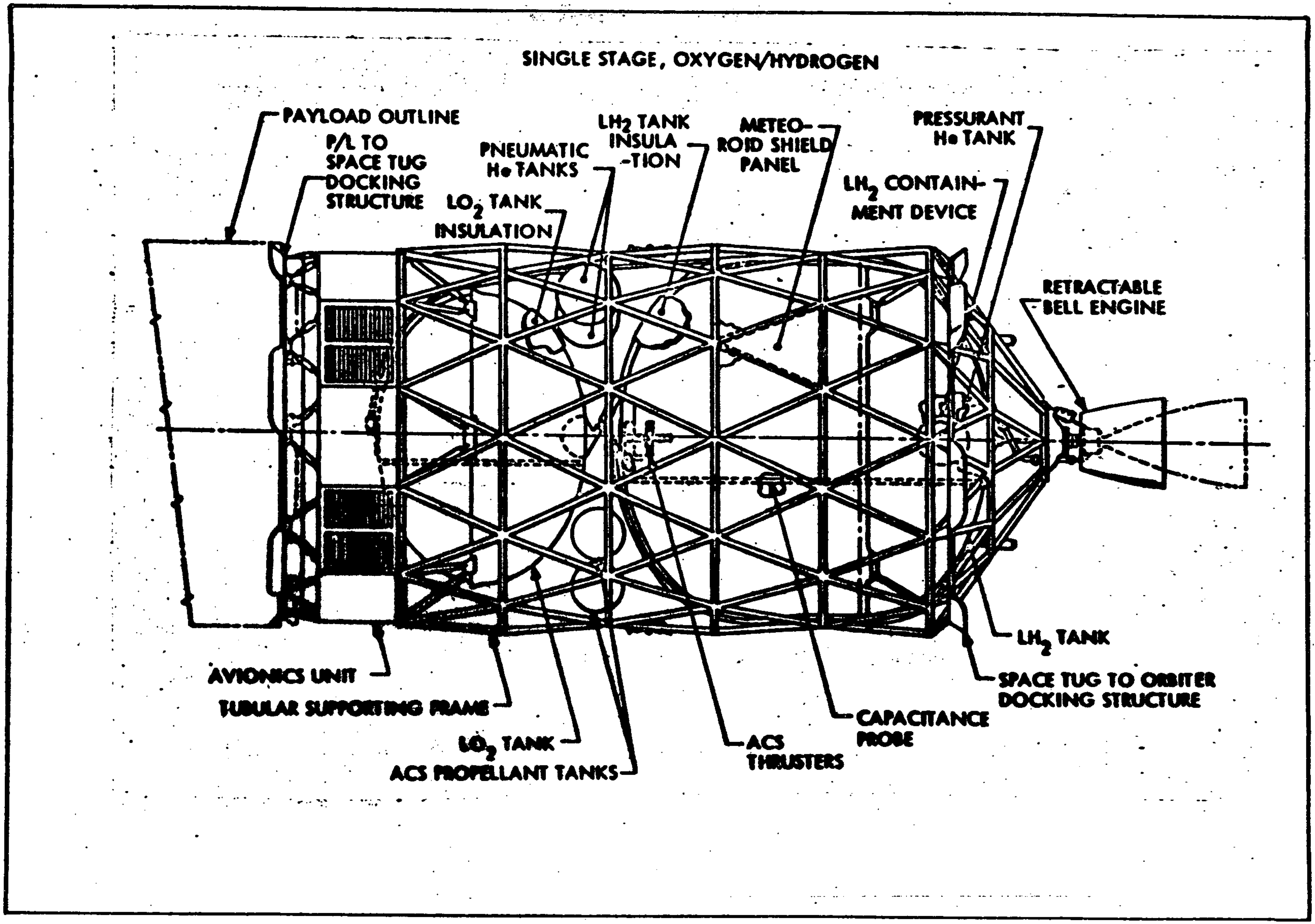


Figure 5.32 Typical U.S. Space Tug Configuration

Table 5.14

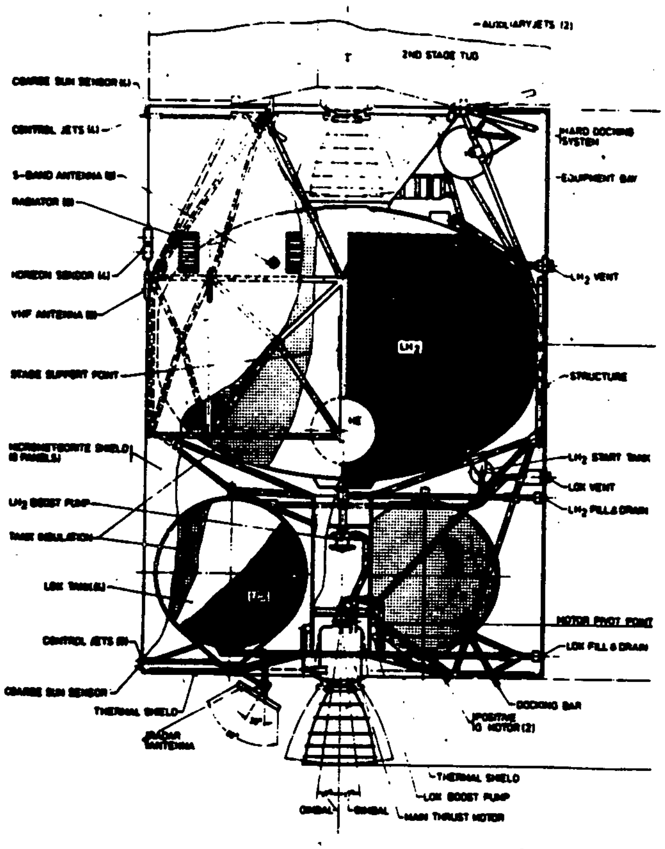
U.S. Space Tug Characteristic Parameters

Reference 1

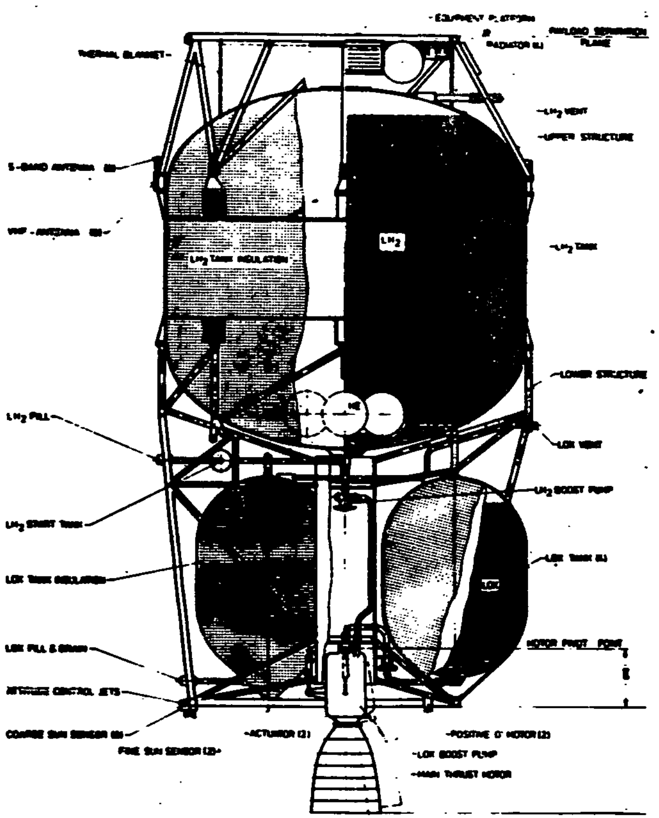
	Baseline Tug	Alternate Configuration
Propellant Mass, kg(1)	25 800	14 000
Burnout Mass, kg(2)	3 100	2 200
Total Mass, jg	28 900	16 200
Propellant Fraction(3)	.881	0.841
Number of Engines	One	One
Total Thrust, N	106 000	106 000
Effective Jet Velocity, m/s	4 508	4 508
Expansion Ratio	250	250
Propellants	LO_2/LH_2	LO_2/LH_2

(1) Includes Non-Usable and RCS Propellant
(2) Includes Residual Propellant

(3) Propellant Fraction = $\frac{\text{Impulsive Propellant Mass}}{\text{Total Mass}}$



General Arrangement 10.6 Metric Ton Reusable Space Tug Stage



General Arrangement 16.5 Metric Ton Expendable Space Tug Stage

Figure 5.33 Typical European Space Tug Configurations (Reference 5-76)

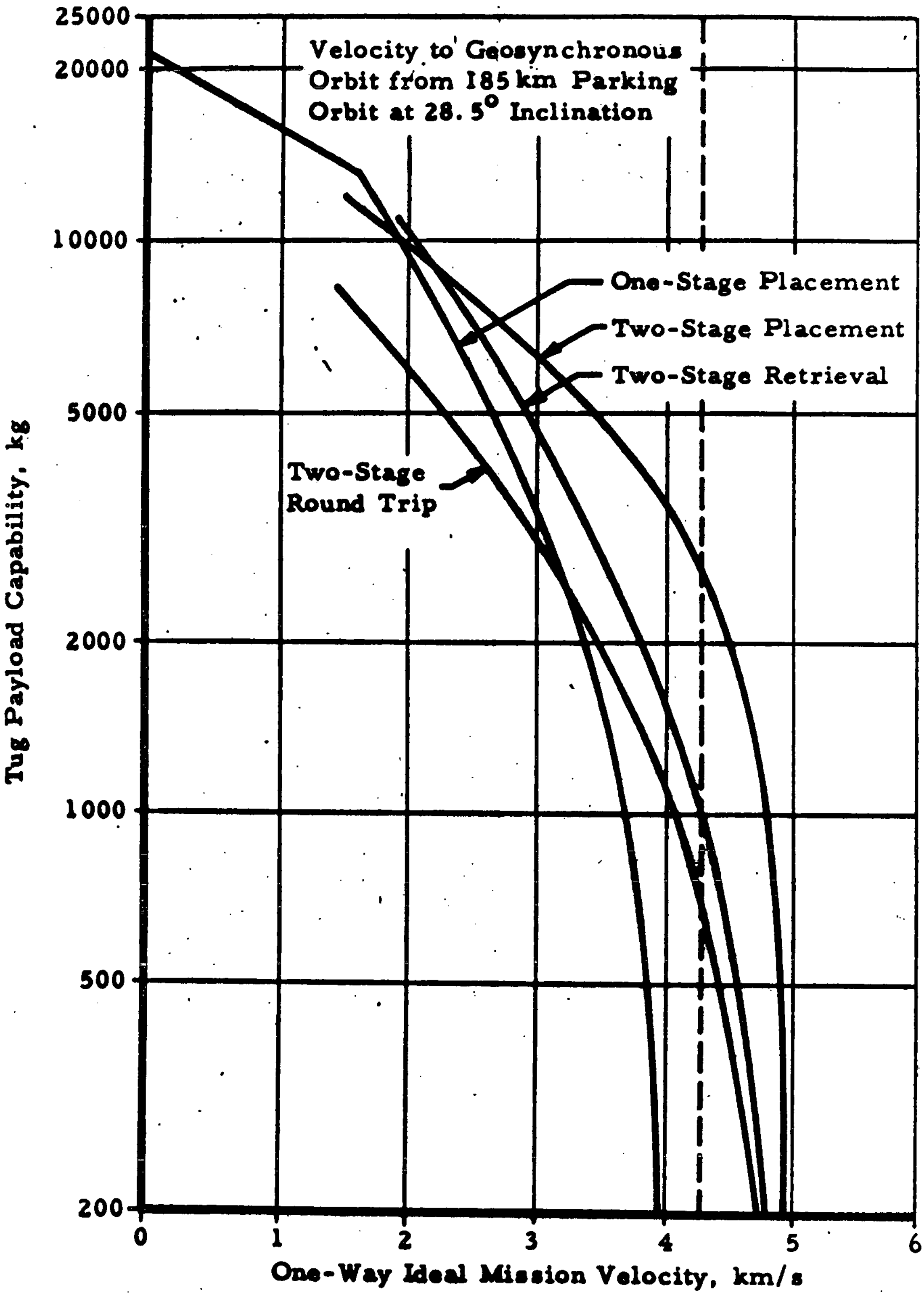


Figure 5.34 European Space Tug Performance

Tug. The conclusions of the Delft study are that the primary mission for the Space Tug is the geosychronous orbit satellite placement and retrieval and that over 97 percent of these satellites can be placed in orbit by the reusable tug. In addition, they expect that over half of the presently forecast interplanetary spacecraft could be injected into interplanetary space by the reusable tug. Whether this estimate remains correct with the possibility of more massive, high technology, planetary explorer satellites is yet to be seen. It appears that, to truly make use of advanced technology in exploring the solar system, satellite masses must increase significantly, thereby requiring advanced propulsion systems, for example, the nuclear rocket, for earth escape. A separate economic study of the Space Tug is presently in progress at LMSC and MATHEMATICA.

5.7 Extra-Vehicular Activity and Teleoperators

It has become increasingly apparent that the overall Space Transportation System must include consideration of the essential roles of extra-vehicular activity (EVA) by men and the use of teleoperators as appropriate to the various missions.

At the present time it seems that EVA will be preferred in the immediate vicinity of the Space Shuttle or Space Station while more remote assignments and those with unconstrained exposure requirements will be carried out by teleoperators. The exact roles to be assigned to each need further study on the earth and in space as they affect the STS and payload design concepts.

Based on the EVA experience to date a number of questions remain to be answered and some conflicting results need resolution. It is hoped that the remaining Apollo missions and the skylab can be used effectively to explore the efficacy of EVA in the STS.

Considerable interest has been generated recently by the role of teleoperators in the STS. Teleoperators are seen as necessary for certain essential functions relating to payload placements, servicing, maintenance, and retrieval. Teleoperators need to be closely coordinated with the Space Tug design and will be surely needed in the retrieval of spinning or disabled space craft and also in unexpected situations. The full range of functions and control that can be usefully performed by Teleoperators remain to be defined. One concept of a Teleoperator Slave Unit is shown in Figure 5.35.

5.8 Performance Analysis

The concept of ideal rocket velocity is introduced in Section 5.2.1. Ideal rocket velocity is a parameter that applies to expendable, recoverable, and reusable rocket vehicles, dependent only on the configuration of the vehicle, in terms of which payload capability can be readily determined. Curves of payload mass versus ideal rocket velocity are presented for several expendable vehicles in this chapter and similar curves could be constructed for Space Shuttle System. The ideal rocket velocity required to perform a given mission is equal to the ideal mission velocity for that mission plus the velocity loss. The purpose of this section is to present data for determining the ideal mission velocity for a wide variety of missions and also to estimate the velocity losses associated with these missions in order to illustrate sources of performance variability of various STS concepts. A second part of this section discusses range safety and other launch constraints that might apply to various STS's including the Space Shuttle and relates these constraints to their corresponding performance penalties via ideal mission velocity increments. The final part of this section discusses the Space Shuttle and Space Tug peculiar problems as they also relate to performance.

5.8.1 Mission Velocity Requirements

The mission velocity requirements are expressed as the sum of two quantities, the ideal mission velocity and the velocity loss. The ideal mission velocity is the minimum velocity change required to perform a mission assuming no drag loss and no gravitational loss, and furthermore assuming that the optimum (minimum velocity change) flight path is chosen, however, subject to certain mission-oriented constraints. Typical mission-oriented constraints are, for example, launch azimuth constraints (explained in Section 5.8.2.1) or the requirement for a temporary low-altitude parking orbit rather than a direct ascent to orbit. Because of the way in which the

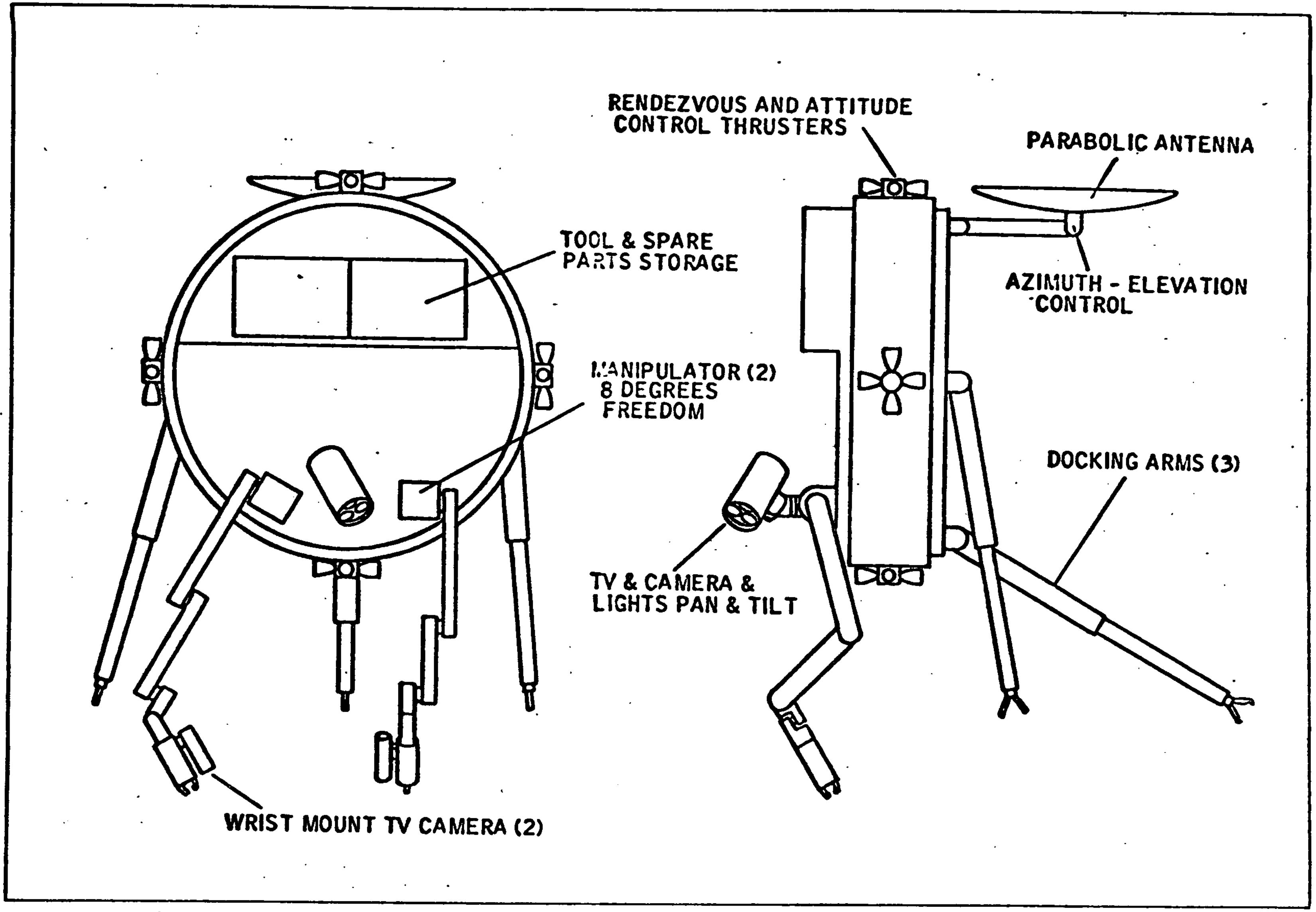


Figure 5.35 Teleoperator Slave Unit Configuration

ideal mission velocity is defined, it is a quantity that is dependent only on the mission to be performed and on the mission-oriented constraints imposed and it is not dependent on the vehicle used to perform the mission. The velocity loss results primarily from the effects of atmospheric drag and gravity. It is a more difficult quantity to determine accurately and depends on both the mission and the rocket vehicle.

5.8.1.1 Due East Launch

A rocket vehicle standing at any point on the earth (except at the poles) already has an eastward velocity corresponding to the earth rotational speed and latitude. A launch vehicle ascending in a due east direction takes maximum advantage of this velocity while a vehicle ascending in a due west direction receives the maximum penalty and must halt its eastward motion before it can begin to obtain a westward velocity. The eastward velocity of the earth's surface is greatest on the equator and zero at the poles. Thus, for eastward launches, launch sites at low latitudes are preferable, however, for westward launches additional factors enter into consideration and the best launch site latitude is not obvious. The compass direction on which a launch occurs is called the launch azimuth and for a due east launch is 90 degrees. The ideal mission velocity requirements for due east launch to orbit as a function of perigee altitude (closest approach to earth) and apogee altitude (greatest orbital distance from earth) are given in Figure 5.36 for launch from ETR. These velocities are computed assuming a Hohmann flight path from the launch site to orbit with no atmospheric drag. Thus, the ideal mission velocity for due east launch to 185 km (100 n.m.) circular orbit is 7.607 km/s from ETR. These numbers may be corrected for launch from WTR by adding 0.028 km/s so that, for example, the ideal mission velocity to 185 km circular orbit from WTR (due east launch) is 7.635 km/s. (An ideal mission velocity increment for launch azimuths other than 90 degrees is presented later.)

Two basic modes of attaining orbit are commonly used: the direct ascent to orbit mode and the parking orbit mode. The direct ascent to orbit typically requires two burn periods, the first, beginning at launch, to achieve the desired orbital apogee altitude and the second, occurring when the vehicle

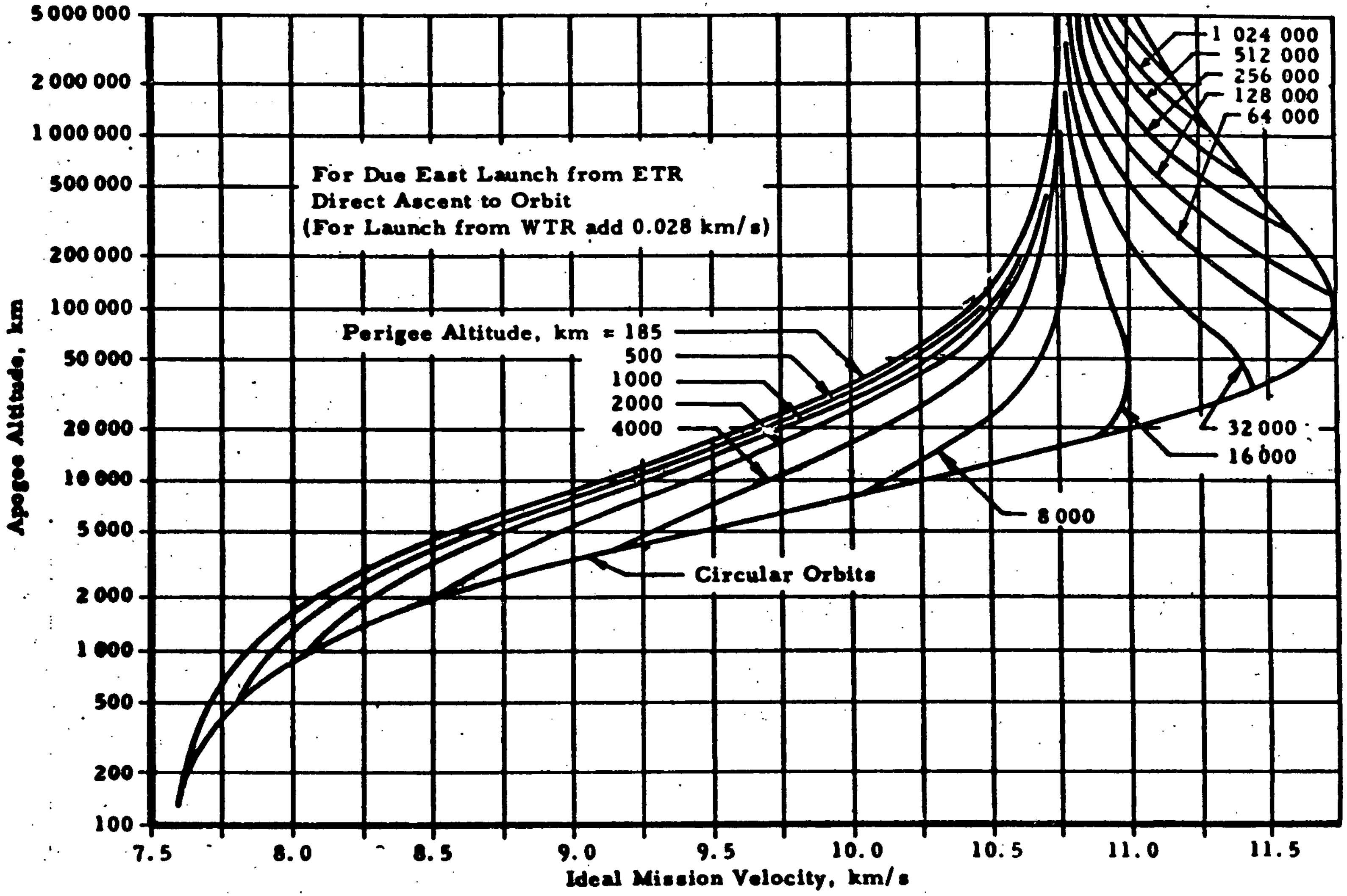


Figure 5- 36 Ideal Mission Velocity for Direct Ascent to Earth Orbit

reaches apogee, to provide the desired perigee altitude. The parking orbit mode typically requires three burn periods, the first, beginning at launch, to achieve a low circular orbit (normally 185 km) and, later a second burn to obtain the desired apogee altitude then, still later when the vehicle reaches apogee, a third burn to obtain the desired perigee altitude. The parking orbit mode must be used when it is desired to place the orbital perigee point over a different location than would be obtained from a direct launch to orbit, although the direct ascent mode requires a slightly lower ideal mission velocity. The ideal mission velocity increment required to transfer from a 185 km parking orbit to a higher orbit is shown in Figure 5.37. To compute the total ideal mission velocity for due east ascent to orbit via a 185 km parking orbit the ideal mission velocity increment from Figure 5.37 must be added to 7.607 km/s or 7.635 km/s for launches from ETR or WTR, respectively.

The ideal rocket velocity required for these missions is the sum of the total ideal mission velocity and the velocity loss. While the velocity loss is a difficult quantity to determine accurately, for most large launch vehicles a value of 1.409 km/s yields quite accurate results for the velocity loss to low earth orbit. To higher orbits the velocity loss is slightly higher and, similarly, there is a small velocity loss associated with any transfer from one orbit to another. For most high thrust-acceleration propulsion systems, such as the chemical rockets presently in use, only the velocity loss associated with the launch is of significance and reasonably accurate results can be obtained by neglecting all other velocity loss contributions. However, for more advanced propulsion systems with relatively low thrust-acceleration, it becomes necessary to account for all velocity loss contributions in order to accurately predict performance. The process for estimating the velocity loss accurately is quite complex and the results depend on both the rocket vehicle and the mission and their determination involves a branch of higher mathematics known as the calculus of variations.

5.8.1.2 Launch Azimuth and Orbit Inclination

When a vehicle is launched on a launch azimuth of 90 degrees, it goes into an orbit that is inclined to earth's equator by an angle, called the

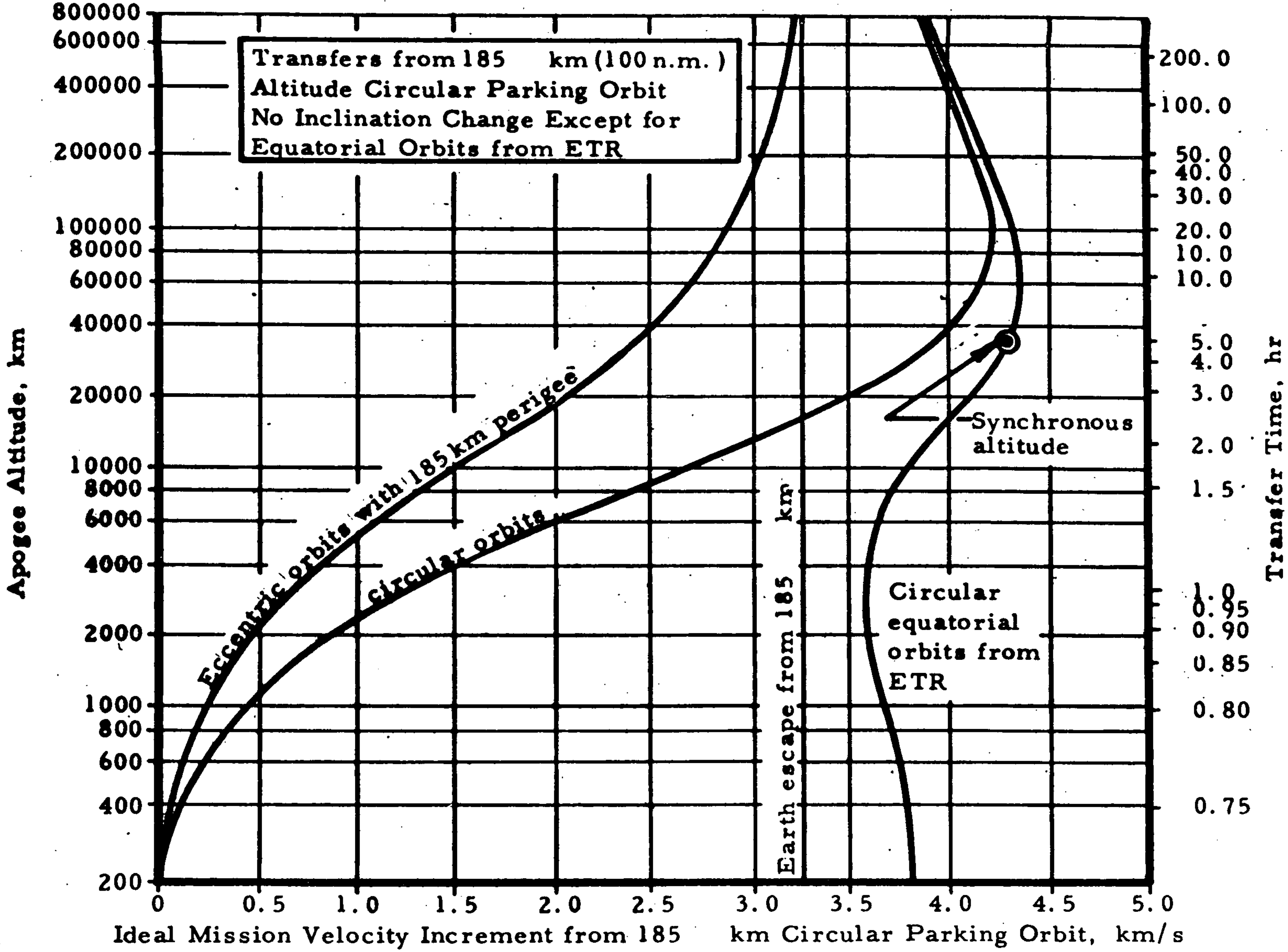


Figure 5.37 Ideal Mission Velocity Increment for Transfer from Parking Orbit (Reference 5-79)

orbit inclination, equal to the latitude of the launch site. Thus, a due east launch from ETR results in an orbit inclination of 28.5 degrees. An orbit that is inclined to the equator by 28.5 degrees passes over points on the earth's surface that oscillate between 28.5 degrees north latitude and 28.5 degrees south latitude. Thus, a satellite launched due east from ETR would never pass over much of North America, Europe or Asia. For many purposes, for example, to take weather pictures, it is desirable to have a satellite pass over more northerly (and more southerly) latitudes. This must be accomplished by increasing the orbit inclination. The orbit inclination can be increased by launching on an azimuth either greater than or less than 90 degrees. Figure 5.38 shows the effect of launch azimuth on orbit inclination. It is assumed here that the ascent to orbit occurs in a plane. The effects of changing orbit plane (dog-leg maneuvers) are discussed in Section 5.8.1.3 and Figure 5.38 is further discussed in Section 5.8.2.1. Note that a satellite in an orbit with an inclination of less than 90 degrees moves around the earth's poles in the same direction as the earth rotates, thus its orbit is referred to as a posigrade orbit. A satellite in an orbit with inclination greater than 90 degrees moves around the earth's poles in a direction opposite to the rotation of the earth. These orbits are referred to as retrograde orbits.

When a vehicle is launched on an azimuth other than 90 degrees, it is not taking maximum benefit of the earth's rotational speed. Thus, for launches on azimuths other than 90 degrees, the vehicle must supply some additional velocity to achieve orbit. This ideal mission velocity increment is shown in Figure 5.39. To obtain the total ideal mission velocity, the ideal mission velocity increment from Figure 5.39 must be added to the other components of ideal mission velocity. Thus, a due west launch from WTR to a 185 km circular orbit requires an ideal mission velocity of 7.635 km/s plus 0.760 km/s or a total ideal mission velocity of 8.395 km/s. The ideal rocket velocity required is obtained by adding the velocity loss to the total ideal mission velocity; 8.395 km/s plus 1.409 km/s gives 9.804 km/s. Comparing this number to the ideal rocket velocity required to achieve a 185 km circular orbit by due east launch from ETR which is 9.016 km/s shows the significant variability of ideal rocket velocity required to achieve similar, but not identical, orbits.

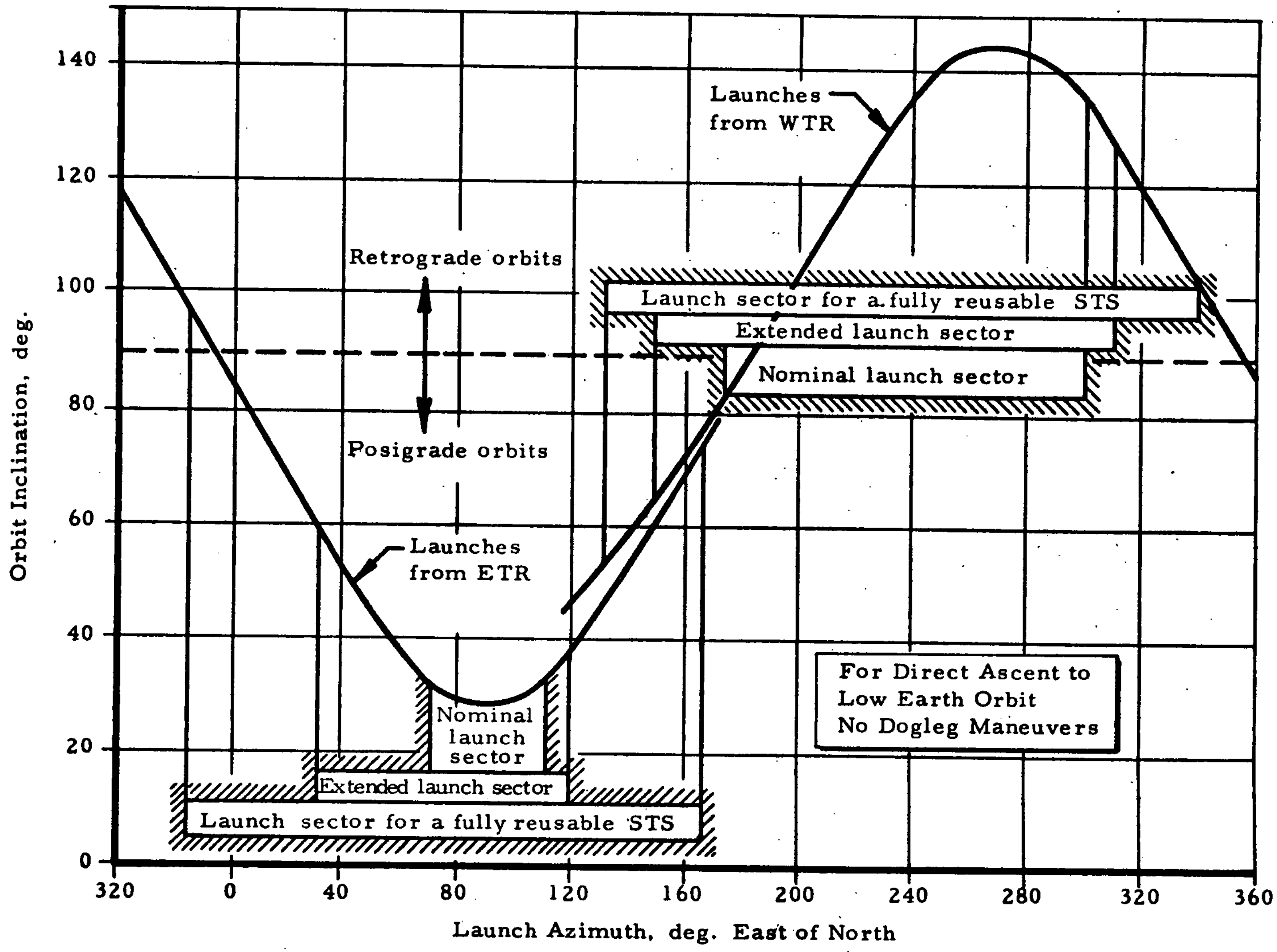


Figure 5.38 Orbit Inclination Versus Launch Azimuth for Direct Ascent to Low Earth Orbit (References 5-79, 80)

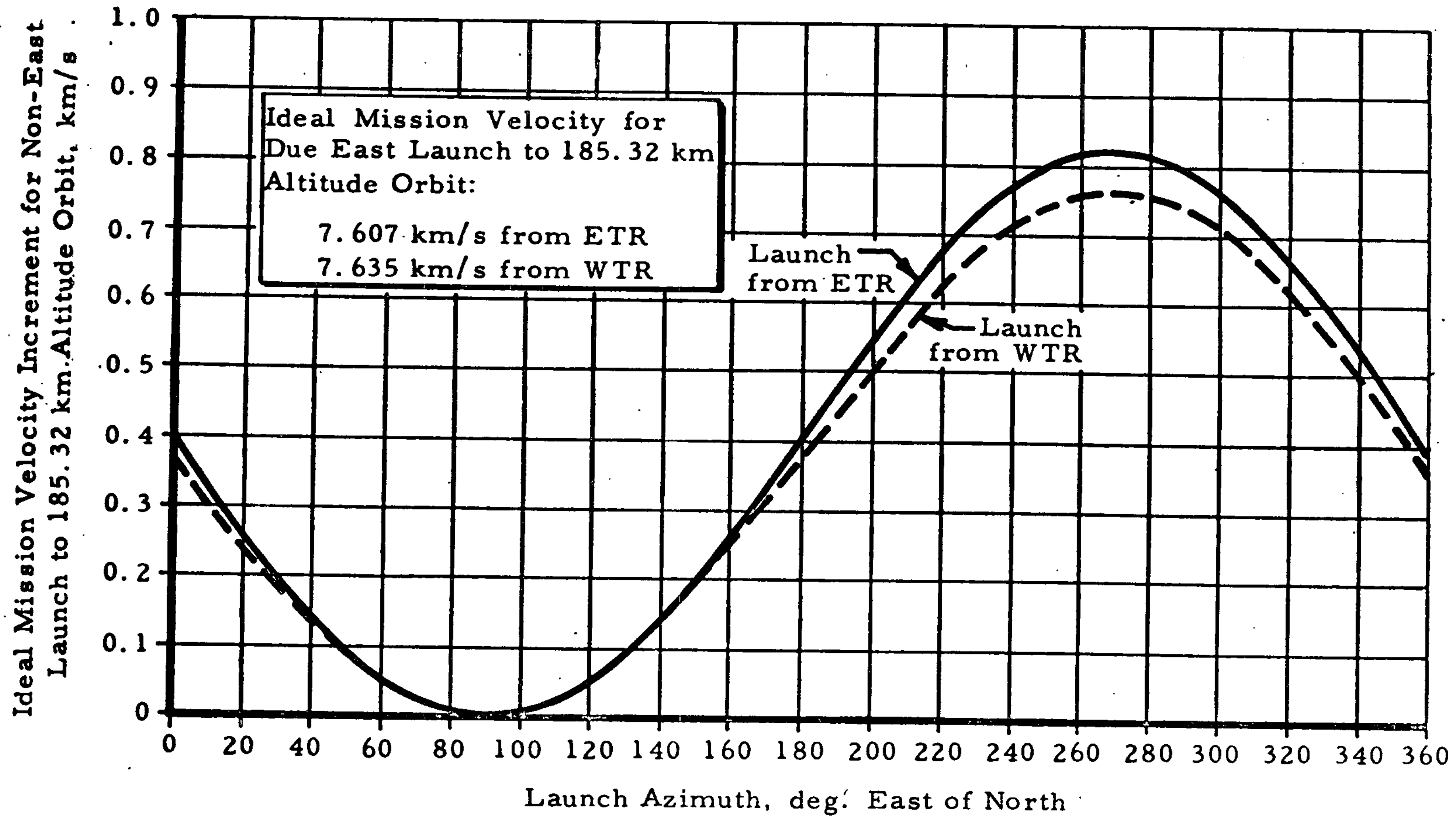


Figure 5.39 Ideal Mission Velocity Increment for Non-East Launch (Reference 5-81)

5.8.1.3 Orbit Plane Change

Two reasons exist which can make it impossible to launch a payload directly into an orbit with the desired inclination without performing a plane change maneuver. The first reason is that the minimum orbit inclination obtainable by a planar launch occurs for a due east (or due west) launch and is equal to the launch site latitude, for example, 28.5 degrees from ETR. It is often desirable to place a payload into an orbit with a lower inclination, for example, zero degrees for a synchronous equatorial communications satellite. This can be accomplished only by a plane change maneuver either during ascent to orbit (if the vehicle reaches a sufficiently low latitude during ascent) or after initial orbit insertion. The second reason involves a range safety constraints (discussed in Section 5.8.2.1) which might prohibit launching on a launch azimuth required to obtain the desired inclination. In this case, plane change maneuvers to increase the orbital inclination are preferably performed at the lowest possible velocity, generally as soon as the restricted areas are cleared.

The ideal mission velocity increment required for plane change depends on the vehicle velocity at the time the plane change maneuver is initiated, the vehicle velocity immediately after the plane change is completed and the amount of plane change. Figure 5.40 shows the ideal mission velocity increment for plane change as a function of these parameters. Notice that the ideal mission velocity increment is greatest when the vehicle velocity is the same at the end of the plane change as it is at the beginning, that is, when the maneuver is performed solely to change the orbit plane. It is also important to recognize that the plane change maneuver must occur at the intersection of the initial and final orbit planes (referred to as the line of nodes). Thus, the plane change maneuver to place a payload into an equatorial orbit must occur as the vehicle is passing through the equatorial plane. For a payload ascending from a low earth orbit to synchronous equatorial orbit, some plane change is performed on departure from the low earth orbit as the vehicle crosses the equatorial plane and the remaining plane change is performed as the vehicle reaches synchronous altitude. However, since the velocity is lowest at synchronous altitude, most of the plane change is generally performed at that altitude.

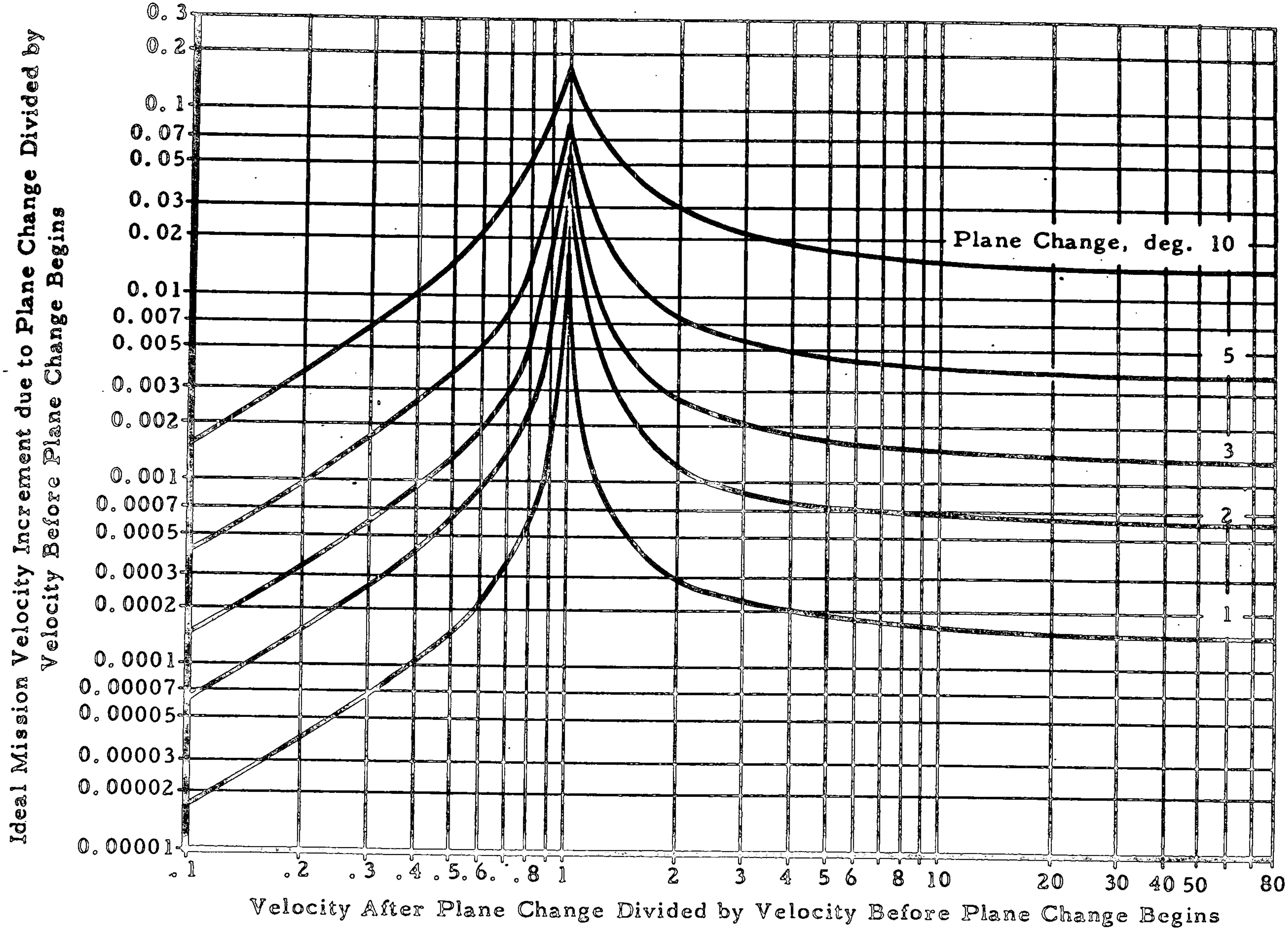


Figure 5.40 Ideal Mission Velocity Increment for Plane Change

5.8.1.4 Earth Escape Missions

For certain missions it is necessary to escape from the earth's gravitational field. These missions, comprised largely of the NASA interplanetary program, are called earth escape missions. Earth escape is achieved when the vehicle attains sufficient velocity that, without the influence of the gravitational fields of other bodies, it will continue to recede from the earth indefinitely. Initially, as a vehicle that has exceeded the escape velocity coasts away from the earth, the earth's gravitational field will cause the vehicle to slow somewhat. However, after some period of time the vehicle will have receded to a distance from the earth at which the earth's gravitational field has only a very small effect on the vehicle. Thereafter, the vehicle will continue its motion away from the earth indefinitely with a nearly constant velocity. This velocity is referred to as the hyperbolic velocity.

The magnitude and direction of the hyperbolic velocity relative to the earth determine the ultimate destination of the payload. If the hyperbolic velocity is directed in the direction of the earth's orbital velocity, the payload will go to destinations outside of earth's orbit whereas if the hyperbolic velocity opposes the orbital velocity of earth, the payload will go to destinations inside earth's orbit. The ideal mission velocity increment required to achieve a particular hyperbolic velocity is a function of the hyperbolic velocity and the orbit from which the vehicle departs. Figure 5.41 shows the ideal mission velocity increment to obtain various hyperbolic velocities for departure from two different circular orbits. Figures 5.42 and 5.43 relate both hyperbolic velocity and ideal mission velocity increment to the heliocentric (sun-centered destination of the payload.

5.8.2 Mission Mode Restrictions

It is not always possible to fly a rocket vehicle on a flight path that is best from a performance standpoint. Reasons why this may be the case fall into two general categories: safety factors and operational constraints. Safety factors, for example, influence permissible launch azimuths while operational constraints may involve maximum allowable pitch or yaw rates due to structural limits, limits on thrust controllability or requirements for

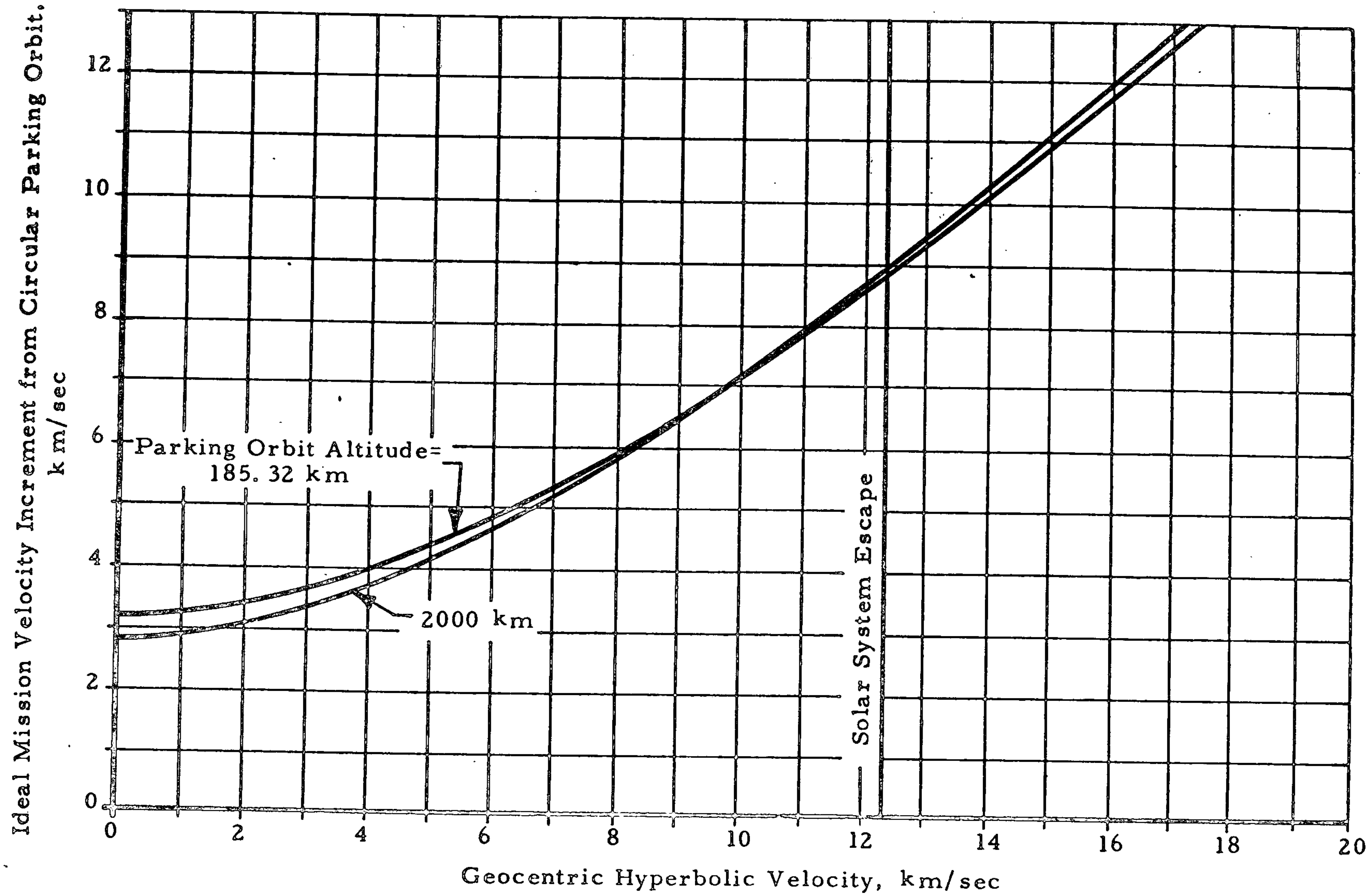


Figure 5.41 Ideal Mission Velocity Increment to Achieve Various Hyperbolic Velocities

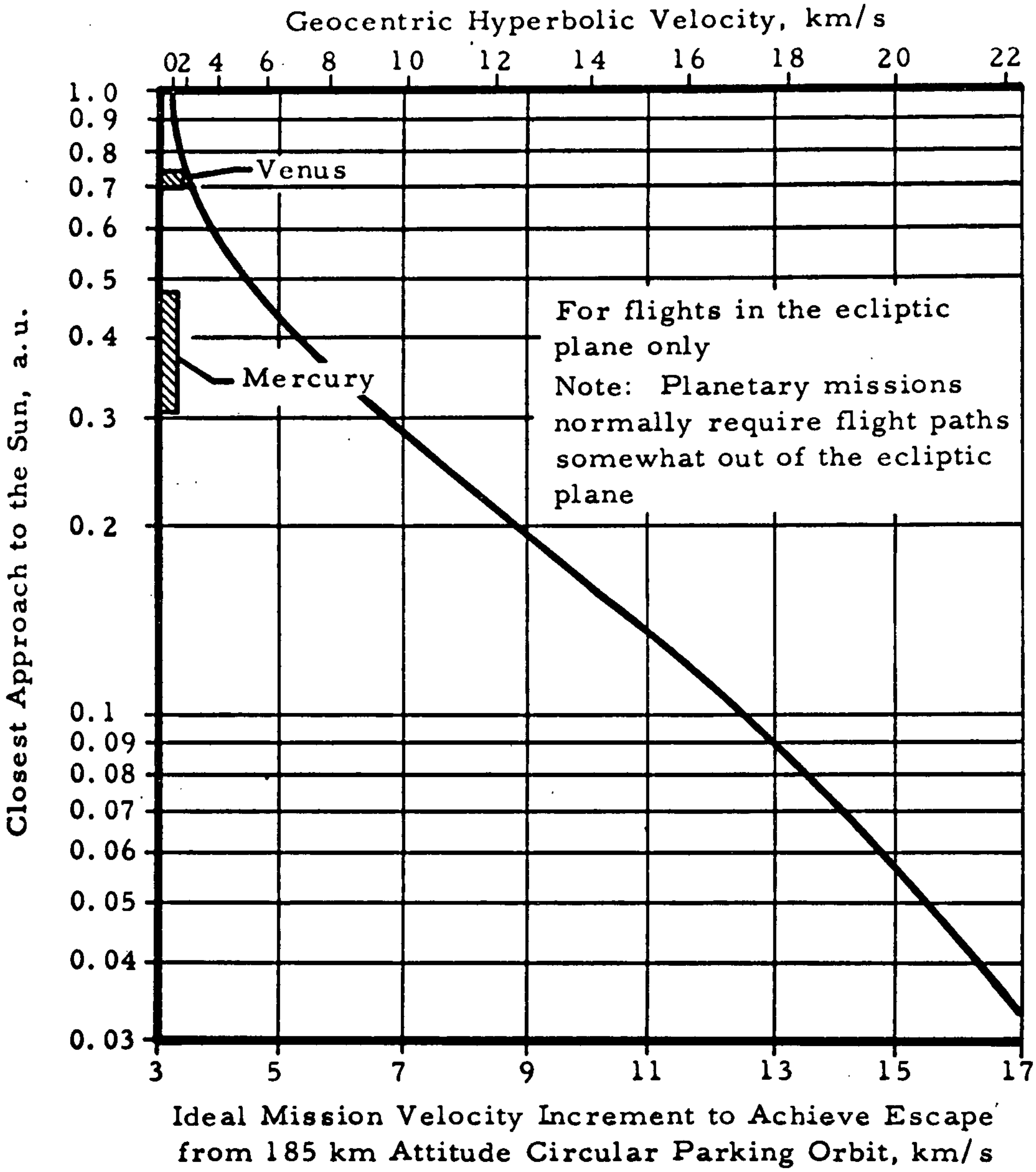


Figure 5.42 Ideal Mission Velocity Increment for Heliocentric Missions Inside Earth Orbit (Reference 5-79)

* a.u.: astronomical units - 1 a.u. = mean distance from the sun to to the earth = 149,600,000 km

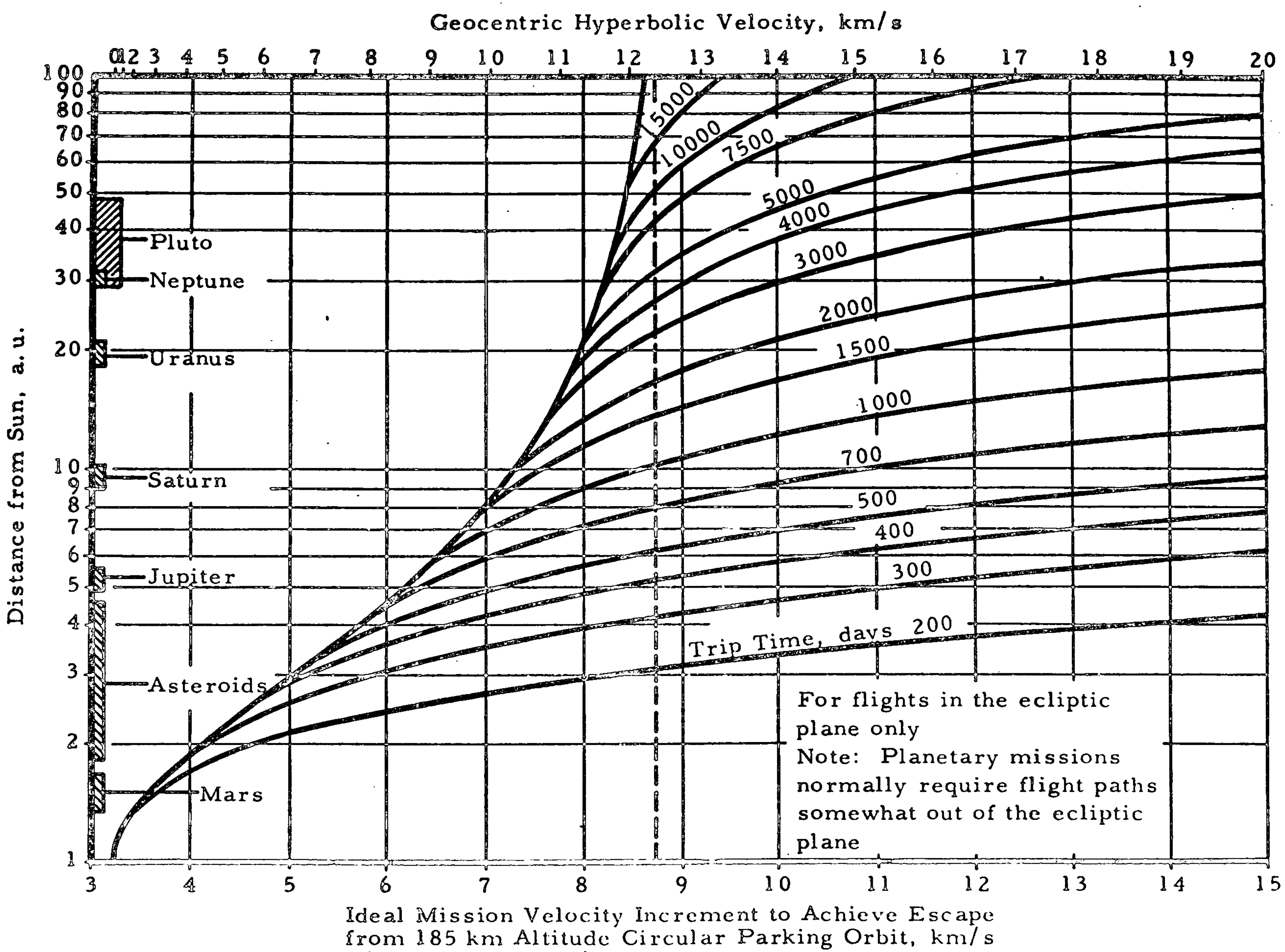


Figure 5.43 Ideal Mission Velocity Increment for Heliocentric Missions Outside Earth Orbit (Reference 5-79)

recovery of components of the total rocket vehicle, for example, recoverable boosters or, for that matter, return of the Space Shuttle to the launch site. This section discusses the concept of hazard as it influences the allowable launch sectors and requirements of recoverable vehicles and components as examples of mission mode restrictions and their influence on performance.

5.8.2.1 Launch Sectors and Hazard

Launch and recovery or return operations of any rocket vehicle can be potentially dangerous to the civilian population that is overflown. All persons, however, are exposed to certain hazards in their daily lives and if the hazard due to rocket vehicle operations is small compared to their other daily hazards it will probably be considered acceptable. Hazards associated with rocket vehicle operations can be categorized as planned events or failure modes. The nature of these hazards can be significantly different for different launch vehicles and Space Shuttle concepts. For example, Figure 5.44 shows a Titan/Centaur launch profile which is typical of expendable launch vehicles. It can be seen from the figure that at various points during the ascent of the vehicle, spent stages, insulation panels, payload shroud and a multitude of miscellaneous parts fall off the vehicle. In fact, to some considerable extent, the good performance obtainable from the expendable launch vehicles derives from their ability to drop off parts as they become no longer necessary. On the other hand, this conglomeration of hardware falling back to earth at velocities too low for burnup can pose a considerable hazard. To reduce this hazard, the vehicle must use a flight path that does not permit the hardware to fall on populated areas. Unfortunately, much of the hardware is released at quite high velocity and does not drop straight to earth. Rather it follows, more or less, a ballistic path and can impact several thousand miles downrange. The expected impact point is a function of the vehicle's velocity and altitude at each instant in time and is referred to as the instantaneous impact point (IIP). Since it is possible for debris to fall anywhere in the vicinity of the entire IIP trace, care must be taken to prevent the IIP trace from passing over heavily populated areas. However, assuming that the vehicle operates as planned, the impact points of the various pieces of hardware can be predicted reasonably well in advance of the flight. Thus, it may be acceptable

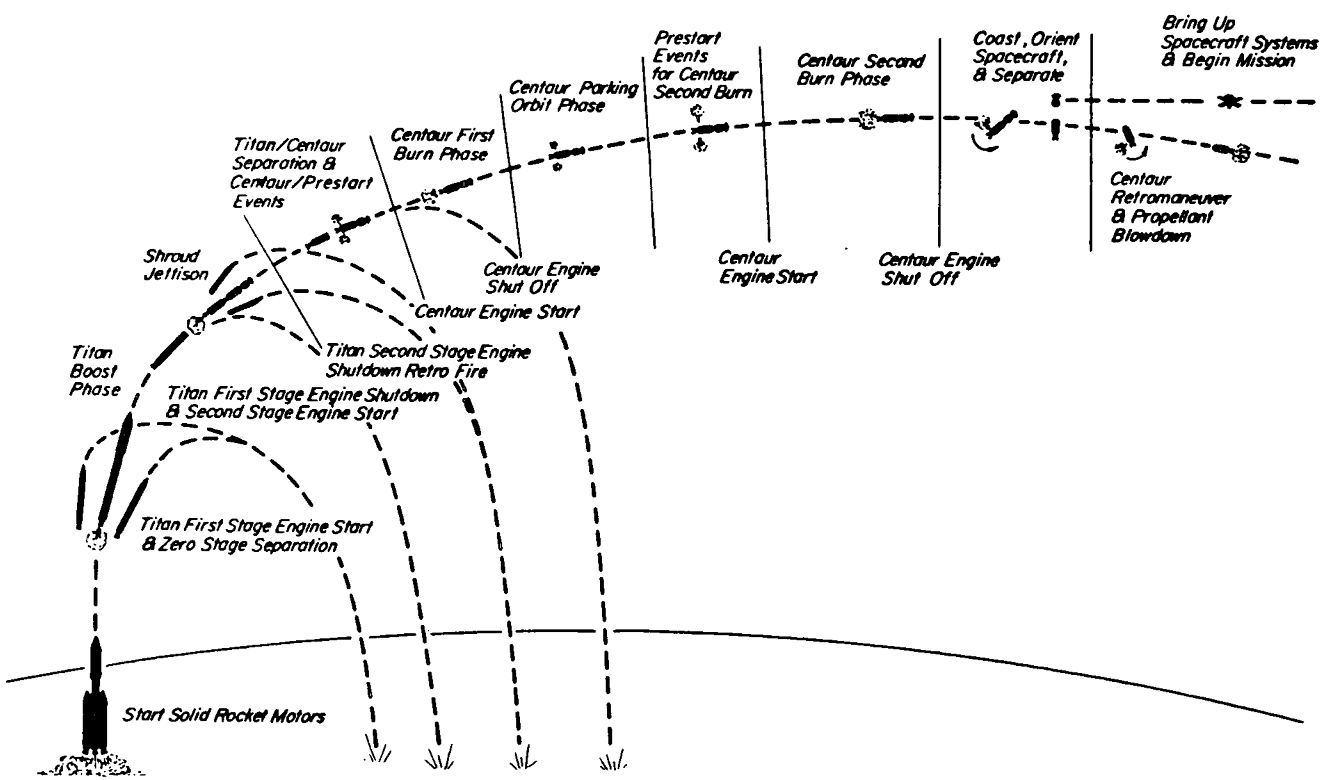


Figure 5. 44 Typical Expendable Launch Vehicle Flight Profile

to allow the IIP to pass over certain populated areas if care is taken to prevent individual pieces of hardware from falling on these areas.

The failure modes of the launch vehicle comprise the second part of the launch hazard. It is always possible, though highly unlikely, for a wide variety of catastrophies to occur. Thus, current expendable launch vehicles are generally equipped with range safety destruct mechanisms capable of destroying a vehicle at any time that the IIP poses a significant unplanned threat to human safety. Of course, it is still possible that the destruct mechanisms will fail and everything else will go wrong all at once, but the probability of this happening is so low that it does not pose a significant hazard to civilian life. Thus, this threat is acceptable. However, this conclusion results from the assumption of a highly reliable range safety destruct device.

The hazard associated with a launch is a function of the population density under the planned IIP trace, the expected variability of this trace, the planned crash-down points of the jettisoned hardware and their relative size, and of the possible magnitude of any catastrophic failures tempered by the probability that such a failure could occur (i.e., the reliability of the launch vehicle) and where it might occur [82]. The acceptable hazard is also a variable depending on the importance of the mission to the nation and/or the world. A reconnaisance satellite launched to survey the Gaza Strip during a military build-up that could result in a world crisis, for example, would probably be approved for a higher hazard than an orbiting solar observatory. The range of all flight paths that result in hazards below the maximum acceptable hazard for the mission determine the acceptable range of launch azimuths from each launch site. Figure 5.38 shows the generally acceptable launch azimuths from ETR and WTR for the current expendable system, a projected extended launch sector for partially reusable vehicles or new launch vehicles of high reliability and a launch sector for the fully reusable Space Shuttle.

The fully reusable Space Shuttle profile (Figure 5.45) is significantly different from that of an expendable system. Because the system is fully reusable, the various components cannot be jettisoned as they complete their respective tasks. Furthermore, the reliability and controllability of a

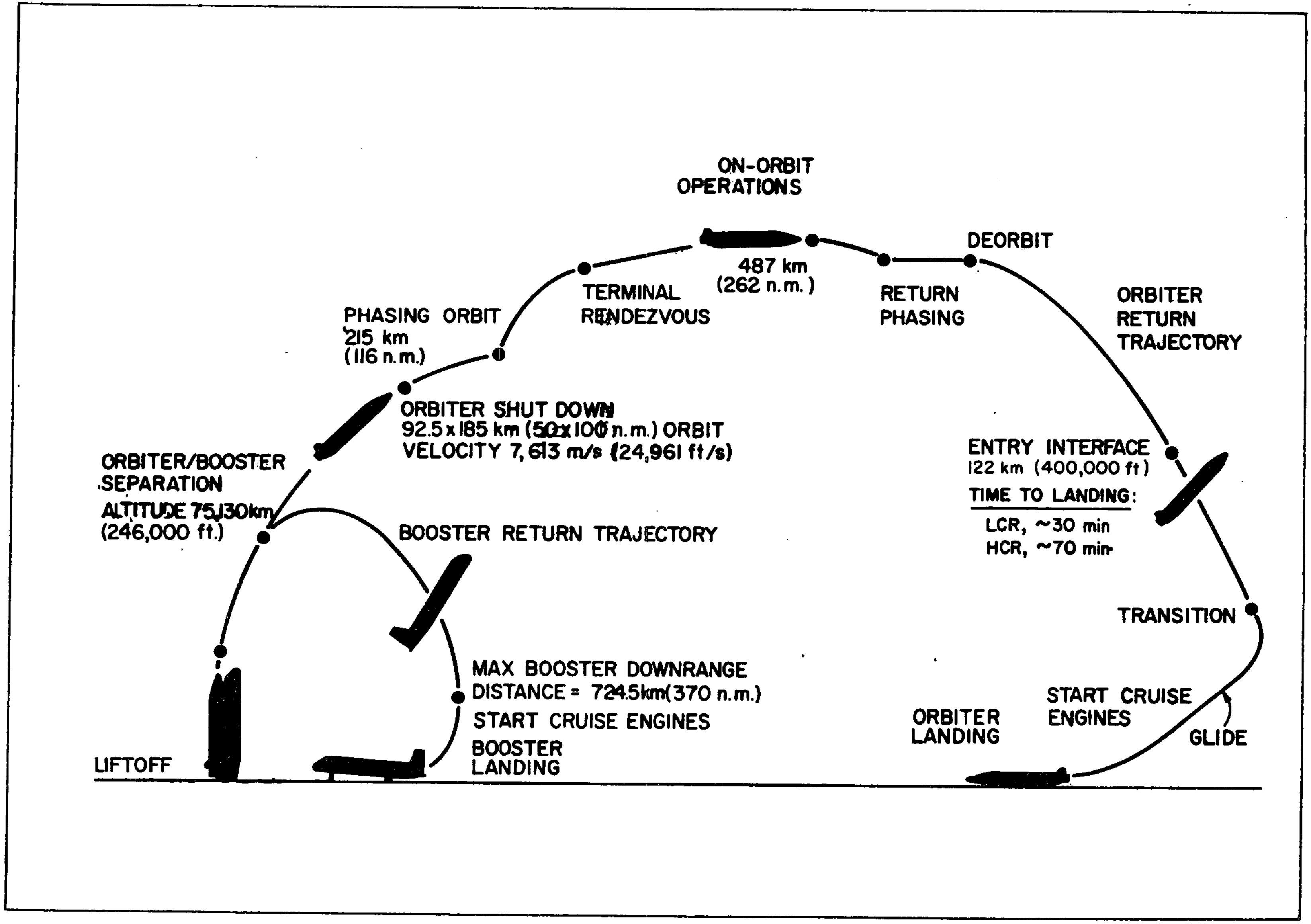


Figure 5.45 Typical Fully Reusable Space Shuttle Flight Profile

manned system presents a totally different picture of launch hazard. Recall from Section 5.8.1.3 that if a system cannot, due to launch azimuth constraints, perform a planar launch to the desired orbit, subsequent plane change maneuvers are required with a very high reduction in payload mass in orbit. Thus, because of the launch hazard, reliability and reusability can have profound effects on performance.

5.8.2.2. Recoverable Vehicles and Components

Certain vehicle configurations include the concept of recovering some components that would otherwise be expended, for example, drop tanks, boosters, et cetera [83]. In addition to the launch hazard considerations for these vehicle configurations, one must now also be concerned with the recovery aspects of the recoverable hardware. Definite drop zones, probably outside shipping lanes, must be established and provision for returning recovered hardware must be made. It is possible that these aspects might place additional launch azimuth constraints on particular vehicles and that these constraints could adversely affect performance.

It is also possible to treat the downrange distance that recoverable components travel as a variable in cost analyses as the cost of recovery is clearly a function of the recovery distance. It may be desirable to constrain the operations of recoverable components to minimize recovery costs.

5.8.3 Shuttle and Tug Peculiar Mission Aspects

Because the shuttle and tug are very expensive but highly reliable and reusable, there are certain peculiar mission aspects regarding their use. This section touches briefly on some of these aspects.

5.8.3.1 Mission Abort Requirements

To provide the desired probability of recovering the space shuttle on any particular flight it is necessary to provide for single engine out operation. The requirement imposed on the shuttle design is that the orbiter is to have the capability of abort to a once-around flight with a single engine out condition at and/or after booster/orbiter separation for the design and reference missions [57]. The mode of operation for a single engine out condition is to burn the OMS engines in parallel with the operating main engine. However, in this mode the thrust acceleration is substantially reduced and a significant additional velocity loss occurs. Thus, particularly if an engine failure occurs shortly after booster/orbiter staging, it becomes impossible for the orbiter to continue to fly into orbit. However, sufficient propellant must be provided to assure abort capability. For higher orbit missions, the mission velocity (ideal plus loss) requirements are higher than the abort mode velocity requirements and the requirement for abort capability poses no constraint. On the other hand, for low orbit missions, the mission velocity requirements are lower than the abort mission velocity requirements and additional propellant must be provided to achieve the abort capability. Since this propellant is not used on successful flights, the additional propellant mass that must be carried is reflected in a one-for-one reduction in payload mass. Thus, it can be seen that the mission requirements do not always limit the shuttle performance.

5.8.3.2 Branched Trajectories and Round Trip Missions

The optimization of flight paths (minimization of various vehicle requirements and/or maximization of payload mass) for the Space Shuttle and the Space Tug poses a problem which, to date, has received only a limited amount of attention [84, 85]. This problem, referred to as a branched trajectory optimization problem, occurs when the state (position and velocity time history) of the vehicle (s) becomes a multi-valued function. For example, consider the placement of a synchronous equatorial satellite into orbit. The Space Shuttle orbiter, with Tug and satellite onboard, flies into earth orbit. Then the tug and satellite separate from the orbiter and fly to synchronous orbit. During this time the state of the system is dual-valued. At synchronous orbit the tug and satellite separate and the tug returns to the Space Shuttle. The state during this phase is triple-valued. Finally the Tug and Shuttle return to the launch site. Because of the round trip nature of this problem, to be properly considered it must be treated as a branched trajectory.

A second shuttle-peculiar problem lies in the facts that the effects of atmospheric lift and drag are very significant to the operation of the

Shuttle and thermal flux and structural loading of the vehicle impose significant constraints on the optimal flight path for a given mission. Many Shuttle study groups are studying these problems and company-proprietary computer programs have been written. However, in addition, the Martin Marietta Corporation is presently under contract to NASA Langley Research Center to provide a computer program to optimize Shuttle trajectories. This program, called POST [86], is now in the final documentation stage.

5.9 Operations, Maintenance, Refurbishment and other Ground Based Considerations

This section is an attempt to delineate the Space Transportation ground operation complex. Because of the lack of detailed and complete information only a broad overview is possible at this time. Since ground operations can significantly affect the cost of the Space Transportation System, the lack of detailed plans and assessments at this time should be viewed as a major uncertainty in system costs.

The following sections discuss briefly the various concepts of the ground operation complex.

5.9.1 Requirements and Constraints

The ground operation complex will be designed to perform the functions of:

- prelaunch assembly and checkout for the booster, orbiter, and payloads
- launch support for the integrated vehicle systems
- orbiter landing
- booster landing and drop tank recovery, or booster (complete or partial) and drop tank recovery, depending upon the selected booster configuration
- maintenance and refurbishment of the orbiter, booster, and drop tanks in preparation for subsequent flights.

5.9.1.1 NASA Requirements

The basic guidelines for the design of the ground operations complex have been established by NASA [57]. The NASA requirements may be sub-divided into the two general catagories of site and operations related. The site related requirements specify that the launch sites may be located at the Kennedy Space Center (KSC), the Western Test Range (WTR), or at an inland site; but that the launch facilities, landing site, and servicing facilities must be located in the same general area.
The two significant operations related requirements are:

(1) minimal assembly and checkout at the launch pad,

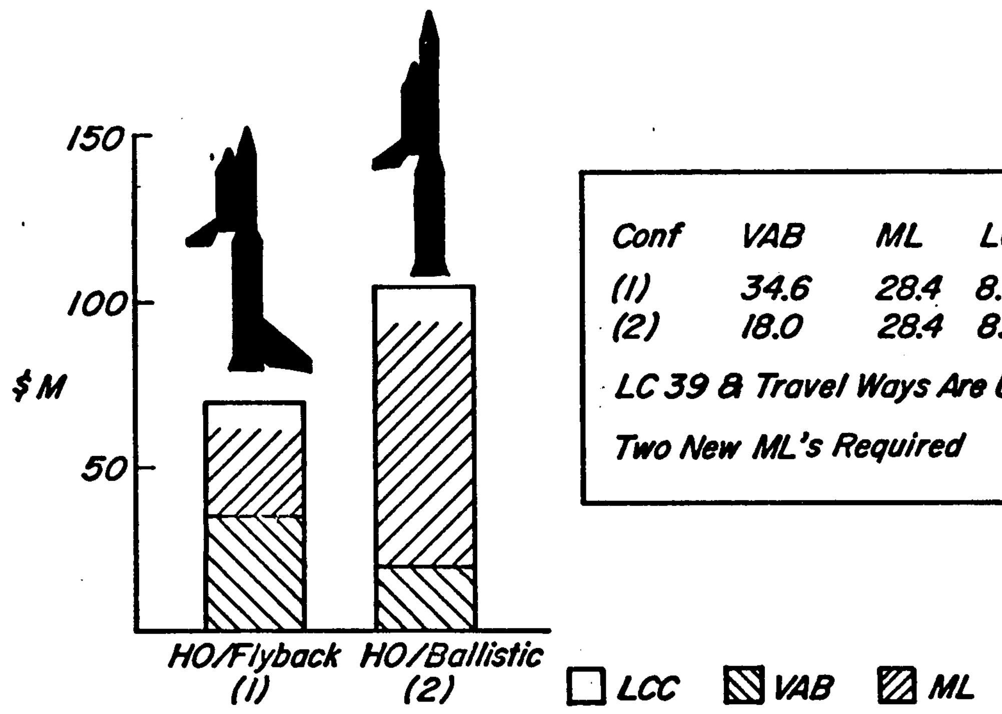
(2) use of specialized facilities such as clean rooms and environmental test facilities in preparation for launch shall be minimized.

The importance of requirement (1), above, is emphasized by a further requirement for minimal service lines at the launch pad with the objective of supplying only the main propulsion system propellants at the pad.

The objectives of these requirements is to shape the ground operations of the booster and orbiter toward present day concepts of maintenance, preflight checkout, and take-off operations of large military or commercial aircraft, as opposed to present day space vehicle launch techniques.

5.9.1.2 Dependency of the Ground Operations Complex on the Selected Flight Vehicle Configuration

A review of candidate booster and orbiter configurations indicates a large degree of dependency of the design of the ground operations complex on the selected booster configuration, but only a small dependency on the orbiter and payloads. Although the candidate orbiter configurations differed in detail, all of the configurations past and the current baseline orbiter are manned, recoverable, aerospacecraft. On the other hand, the candidate boosters considered over the past seven months differ markedly in design concept and operational philosophy, ranging from manned, recoverable, aerospacecraft to unmanned expendable boosters. Other candidate booster configurations are wholly or partially reusable, but are unmanned. This wide range of candidate booster concepts cannot be accommodated by a single ground operation complex design. At present, the choice of boosters has narrowed down to four (unmanned) thrust assist booster concepts. Figure 5.46 illustrates this dependency of the ground complex costs for the two selected



Conf	VAB	ML	LCC	Total
(1)	34.6	28.4	8.0	71.0M
(2)	18.0	28.4	8.0	103.2M

LC 39 & Travel Ways Are Used As Is

Two New ML's Required

Figure 5. 46 KSC Facility Modification Costs

booster configurations [68]. As shown in the study, additional KSC facility modification costs of approximately $32 million are incurred with the HO/ Ballistic configuration over the costs associated with the HO/Recoverable configuration.

5.9.2 Space Shuttle Ground Operation

One important aspect of a successful Space Shuttle System will be the ability to incorporate an efficient, airline type ground service and operations activity for the booster, orbiter and payload system. Airline-airfreight type maintenance, refurbishment and service are regarded as vital to the success of the Space Shuttle and necessary ground checkout and launch preparations must be reduced to their simplest terms to achieve the rapid turnaround times and economic, efficient ground operations. Between landing and relaunch, four major phases of ground operations may occur for a typical Space Shuttle activity. They are Post Landing Operations, Maintenance and Refurbishment Operations, Pre-launch Operations and Launch Operations. A two week turnaround cycle (between landing and relaunch is required [57]. It is presumed that the turnaround would occur only at a centralized operation of the booster and orbiter with the launch, recovery and ground operations being performed at the same site.

It should be noted that significant investments in both facilities and human resources for ground operations exist at three locations. These resources could be used singularly, or in combination, to support Space Shuttle ground operations. The Kennedy Space Center has both the facilities and a proven capability to support the launch of manned space vehicles, while the WTR has supported unmanned vehicle launches. The flight test facility at Edwards Air Force Base has been used to support the assembly, flight and landing of the X-series of experimental aircraft.

Figure 5.47 depects the flow of system elements to the launch pad for a Titan booster/orbiter configuration for launch from launch complex 39 at the Kennedy Space Center. While the specific operations and facility usage will vary with the selected booster, this flow is indicative of the operations to be followed in the case of an unmanned expendable (or partially reusable)

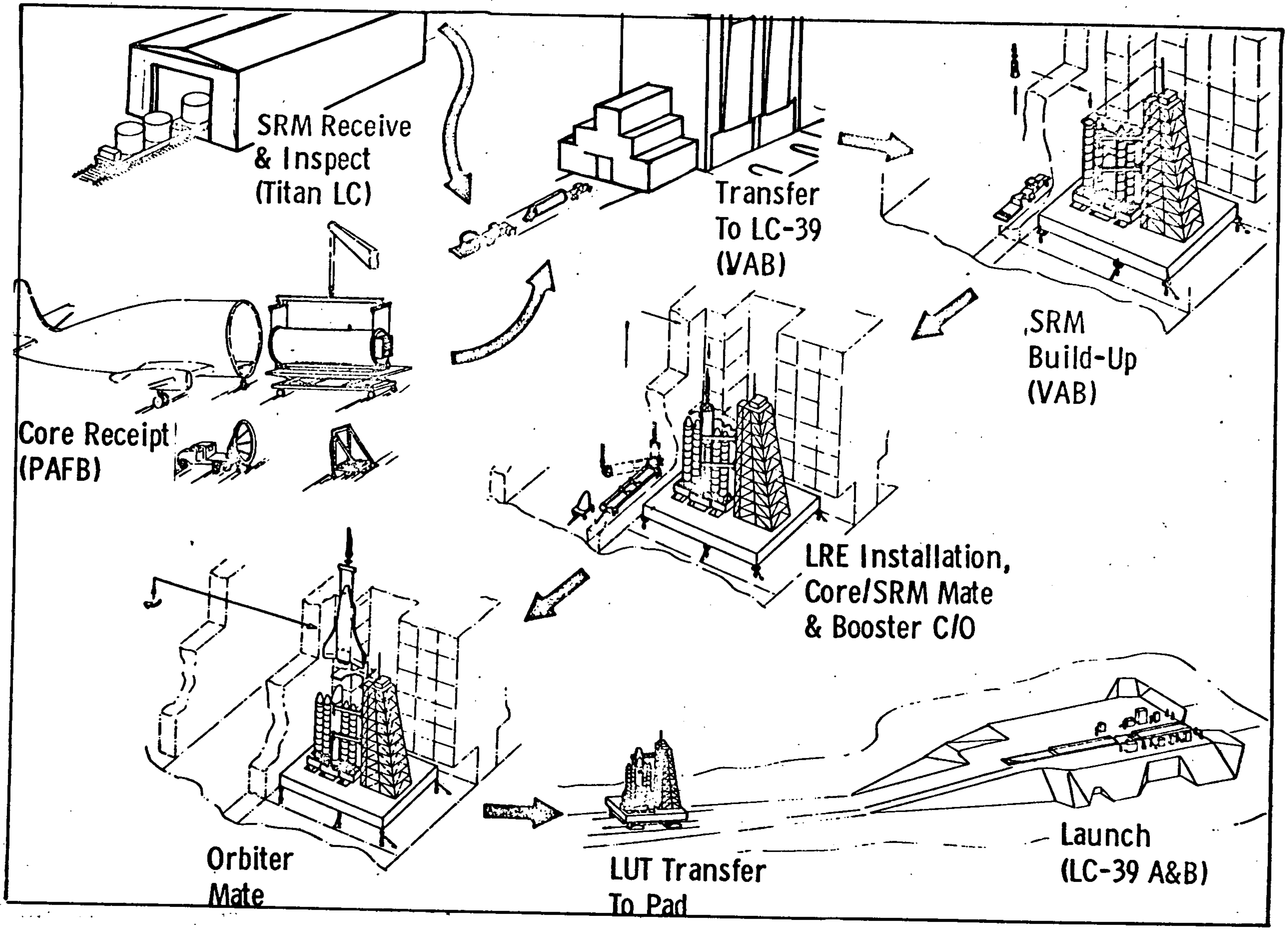


Figure 5.47 Assembly Flow— Titan III L Booster

booster consisting of a liquid core augmented with solid rockets [71].

A time-line diagram for the Mark II shuttle turn-around is shown in Figure 5.48 [87]. The turn-around is accomplished in 214 working hours, and is within the 14 day turn-around requirement [57]. The booster portion of this time-line is constructed for a manned recoverable booster, and will be simplified for an unmanned vehicle; however, the major operations of booster, drop tank, and orbiter readiness will be inherent in any plan. The operations performed are described in the following sections.

5.9.2.1 Post Landing Operations

The post landing operations begin as soon as the orbiter has accomplished its horizontal landing and is taxied or towed to a post landing facility. The orbiter, will be subjected to a relatively long heat pulse duration of about 2,000 seconds and after re-entry and cruise flight back to the landing site, this will be critical as far as post landing procedures and the avoidance of highly heated surfaces are concerned. The crew and/or payload of the vehicle will have to be removed with care and sub-system management must be tended to, to promptly secure the vehicle safely. Some typical required post launch activities would be removal of data packs, vehicle "cool-down", open access doors and attach de-fueling and de-servicing ground equipment, safe ordnance, drain and purge cryo-tanks, remove any cargoes and cargo modules, remove ground service equipment, close access doors and finally, tow the vehicle to maintenance and refurbishment area.

Booster post landing operation will be a function of the selected booster configuration. In the case of the unmanned reusable booster, the liquid rocket motors and the reusable drop tanks will make a parachute controlled descent to the ocean surface and will be located either by an underwater sound net or by integral radio locater beacons. As shown in Figure 5.49 these units will then either be lifted aboard a recovery ship or towed to the recovery site for refurbishment [68]. If the fully reusable booster configuration is employed, the booster will return to the site after separation from the orbiter at about 75,130 m altitude and a re-entry heat pulse

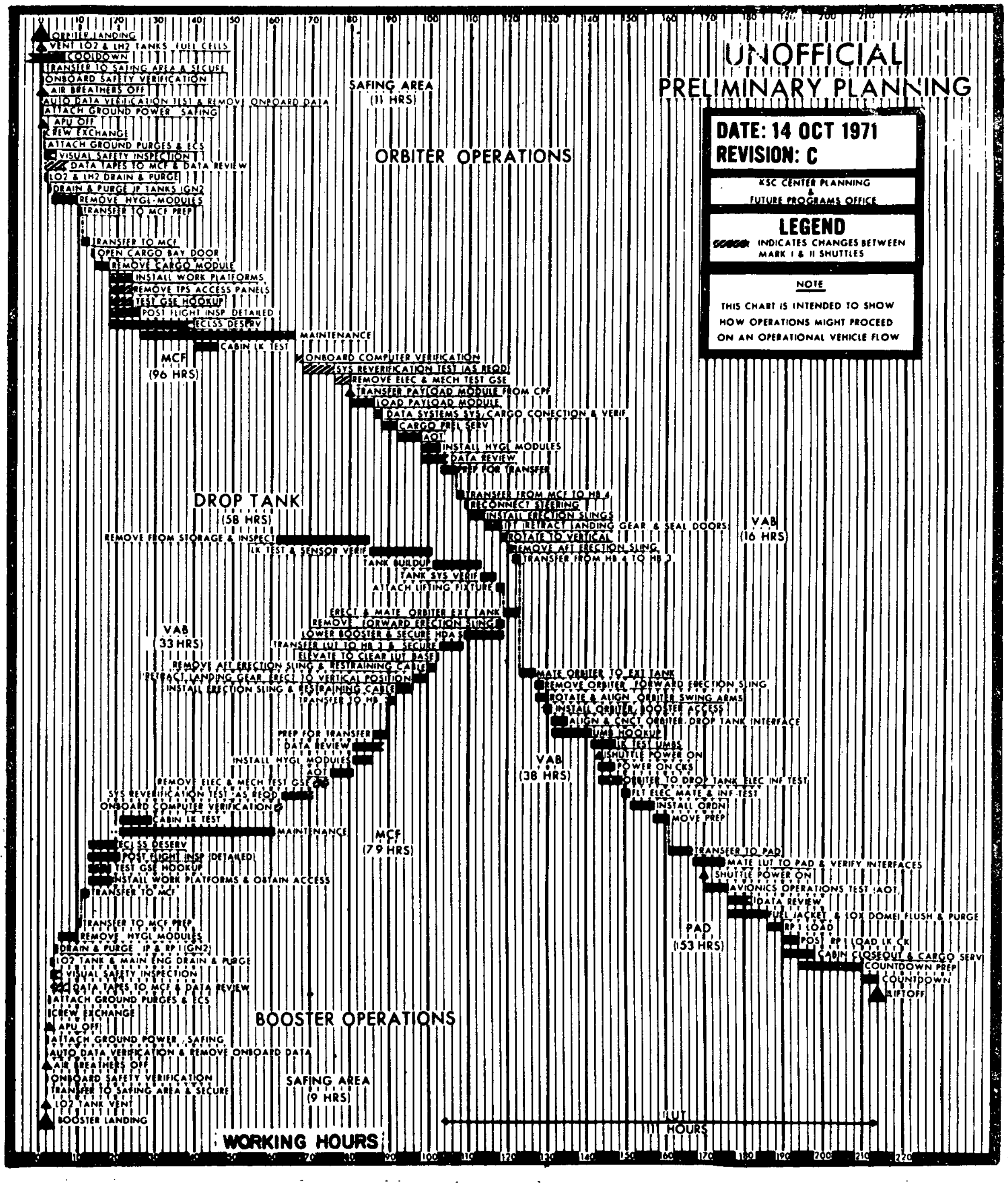


Figure 5.48 Mark II Shuttle Flow-Plan

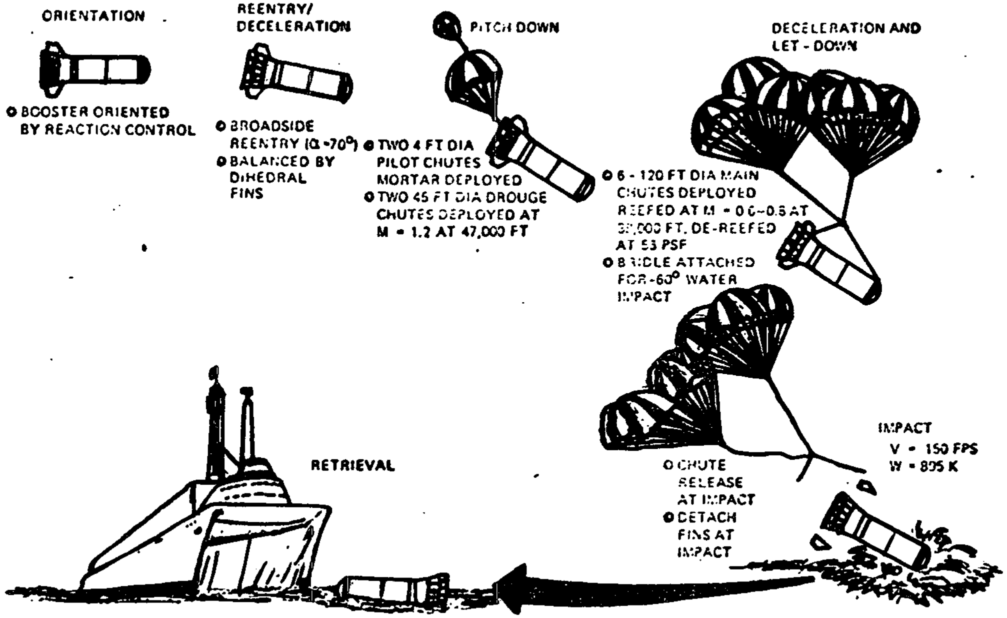


Figure 5. 49 Booster Recovery Operation

of 400 seconds. In the case of the recoverable booster, the entire booster and orbiter post landing operation will be completed within 12 hours after landing. This booster is, however, no longer considered under the present Space Shuttle program.

5.9.2.2 Maintenance and Refurbishment Operations

The maintenance and refurbishment operations activities begin as soon as the orbiter is hangared in the facility, Figure 5.50. Typical duties would include opening and removal of all access doors and servicing plates and immediate implementation of airline type preventative and corrective maintenance. After all the required maintenance and refurbishment of the orbiter is accomplished, a post maintenance checkout would be completed and the cargo module would be installed. A detailed flow plan for the maintenance and refurbishment operations is shown in Figure 5.48. Contractor estimates indicate that the entire maintenance, refurbishment, and payload installation may be completed in five days.

5.9.2.3 Pre-Launch Operations

The pre-launch operations phase of ground operations includes all the important procedures devoted to the determination that the thrust assist motors and orbiter are made ready to be transported to the launch pad. These procedures would be completed prior to moving the vehicle and include vehicle and ground service equipment power-up and propulsion subsystem checkout. Also, a complete checkout of the integrated electronics, avionics and all mechanical subsystems would be performed.

Up to this point, all the ground operations activities on the orbiter have been accomplished with the vehicle in the horizontal position with the obvious advantages of ease of service. The next step in the pre-launch operations requires a decision on the best approaches concerning the transporting and erection of the vehicle. A possible approach for mating and erecting the orbiter, solid rocket motors, and HO tanks is shown in Figures 5.50 and 5.51. The solid rocket motor segments receive incoming inspection in a separate facility and are integrated in the solid rocket motor assembly building at the VAB. An external HO tank assembly area is provided at the VAB.

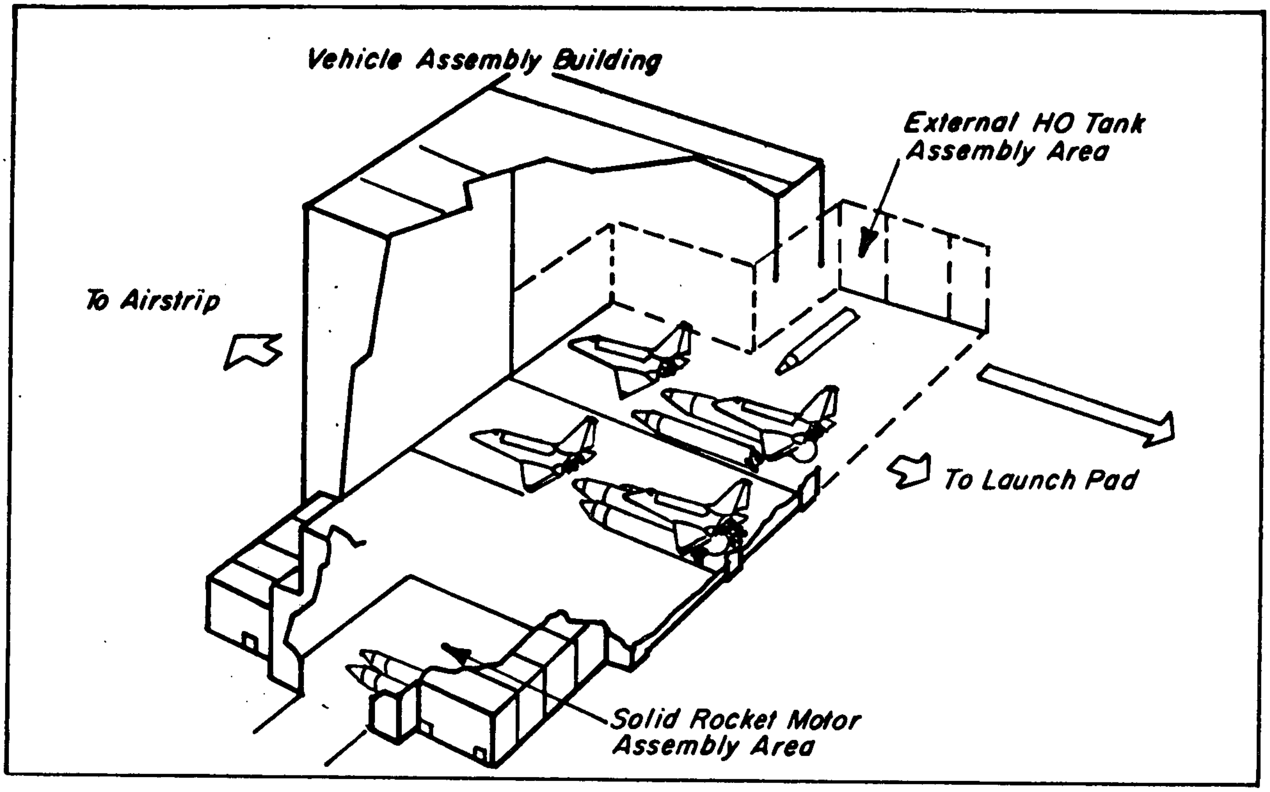


Figure 5. 50 Typical Ground Operations for TAOS

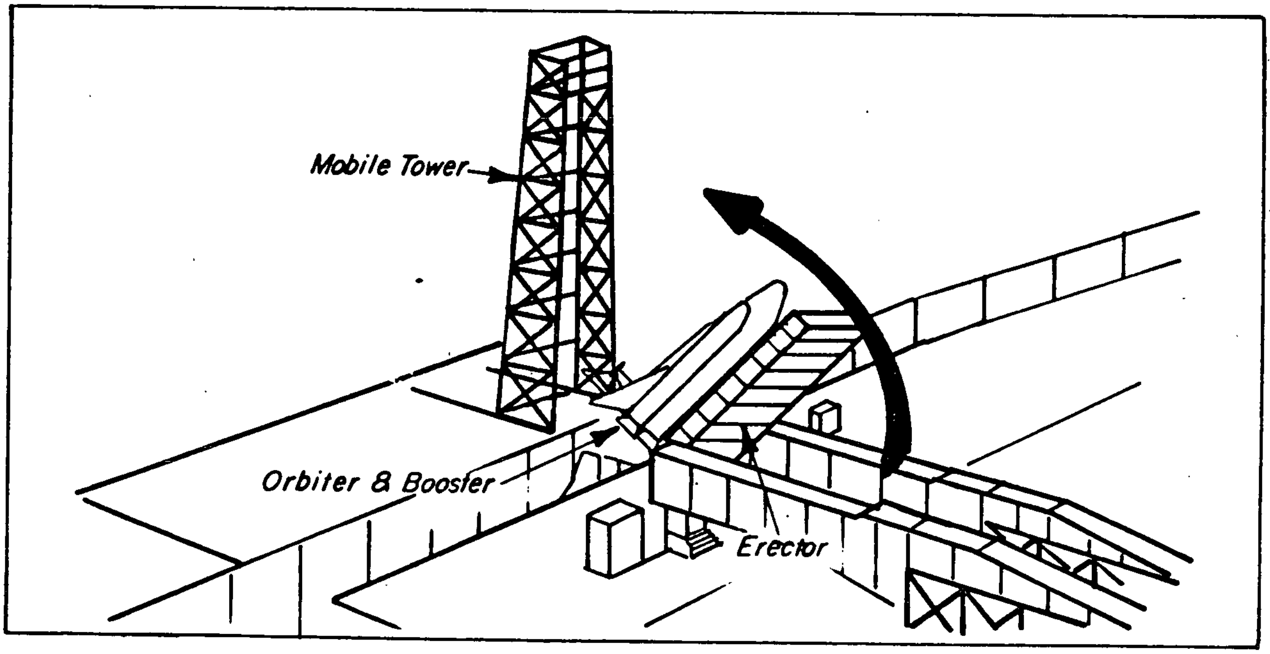


Figure 5. 51 Typical Erection and Launch Operations for TAOS

The sequence of operations in the VAB consists of:

- mate external HO tanks to orbiter
- mate solid rocket motors to orbiter

All operations in the VAB are performed with the orbiter in a horizontal attitude to facilitate access to the vehicle during assembly operations. The assembled vehicle is then installed in a horizontal attitude on an erector (which could be an integral part of the LVT) and transferred to the pad. At the pad the vehicle is moved to a vertical position and fueled with cryogenics.

Partial checkout of the orbiter takes place in the VAB prior to installation on the mobile erector. Final checkout takes place at the launch pad when the vehicle is erected and fueling completed.

5.9.2.4 Launch Operations

The launch operations must assure a safe flight for the Space Shuttle crew and passengers as well as safety for all persons on earth. Adequate procedures must be implemented to preclude any unusual occurrence or accident.

The program goal of the launch phase of ground operations as described for the typical case detailed in Reference [88] is to schedule the lift-off within 24 hours from the time of leaving the vehicle assembly building. With the vehicle in the launch attitude and two hours before lift-off, the Space Shuttle is powered up and the final mission trajectory data is loaded into the guidance systems. The launch pad will be cleared of most personnel and hazardous operations such as loading and servicing the cryogenic propellants will commence. After the servicing is complete the crews and passengers will board the vehicles. Facilities and procedures will need to be provided so that rapid egress (to a safe area) of all crews and passengers may be made in the event of any emergency requiring abandonment of the launch site. Upon successful completion of the launch, the pad area is inspected and prepared for the next launch.

5.9.2.5 Facility Safety Requirements for Pre-Launch Abort

Since the orbiter may carry a flight complement of 12 personnel (two crew and ten passengers), as opposed to a total of three crew members in the Apollo mission, it is felt that facility provisions for crew and passenger safety in the event of prelaunch abort must be enhanced. Facility requirements will include [58]:

- gas purge system for orbiter and booster
- leak detection and warning
- semi-free fall elevators at crew and passenger emergency egress exist levels
- access arms which will rapidly rotate from a stowed position on the launch tower to the emergency egress exists to provide an enclosed passageway to the elevator for fire protection.

5.9.3 Spacecraft (Payload) Ground Operations

The Shuttle payloads will be delivered to the VAB in hermetically sealed shipping containers following the successful completion of flight acceptance tests at the payload contractor's facility. Upon designation for flight the payload will be removed from the shipping container and will be erected in a spacecraft test area. With the expected standardization of payload electronics for Shuttle operations, it is expected that the payload test equipment will consist principally of interface electronics, and that maximum use will be made of a central computer facility in the VAB for payload testing. Upon successful completion of electrical testing in the spacecraft test area the payload will be moved to the Shuttle for integration. It is not anticipated that environmental testing of the payload will be performed at the VAB.

Upon return of a payload from orbit for maintenance or refurbishment, the payload will be demated from the orbiter and returned to the spacecraft test area for post-flight electrical test. First level maintenance consisting of module replacement will be performed in the spacecraft test area. Payloads requiring significant rework may be returned to the contractors facility.

5.9.4 Environmental Considerations

5.9.4.1 Combustion Pollution

It should be evident that, in the present climate, every large public or private system will receive close scrutiny regarding any possible environmental issues that may be noted during the operation of the system. One environmental aspect of the Space Shuttle or any Space Transportation System will certainly be centered on the products of combustion resulting from the use of the large masses of propellants. If there are any resulting contaminants, they will become noticeable if a large traffic launch rate occurs. The most optimistic launch rate, however, would still be essentially insignificant conquered to commercial aviation even considering the greater masses of fuels. Some of the combustion pollutants of the most utilized or projected propellants are shown in Figure 5.52. Concepts such as the proposed fully reusable, two stage Space Shuttle with LO_2/LH_2 propellants contain quantities of compounds that affect the respiratory tract and adjacent areas of the body. Still others contain compounds that are asphyziant or depressant in character. Further attention should be given to assess the importance of this particular problem area.

5.9.4.2 Noise

Local and near-distance noise could be a problem of some consequence as shown in Figure 5.53, where noise from the Shuttle will exceed that from the Saturn V significantly in power level across the entire frequency range. Frequence of flight and launch site location will also be factors in determining the basic acceptability of the noise resulting from the launch of these massive vehicles. Although this problem is recognized and being dealt with, only a full and candid revelation of the extent of the problem will lead to a satisfactory solution.

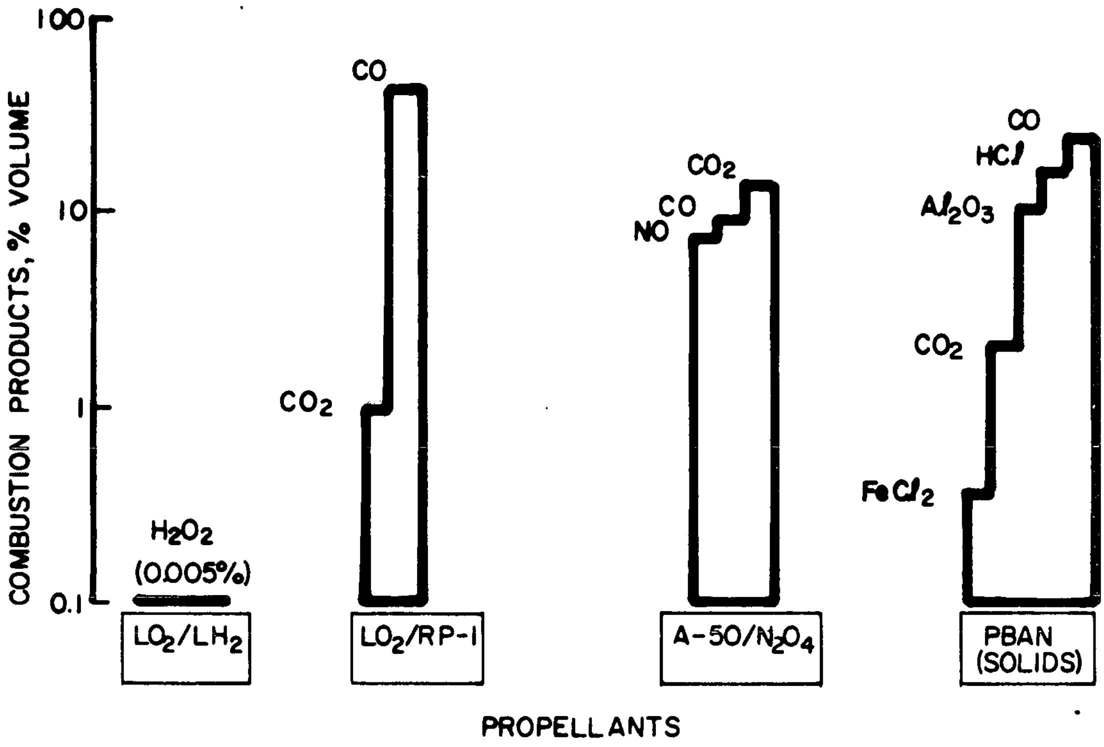


Figure 5.52 Combustion Pollution (Reference 5-89)

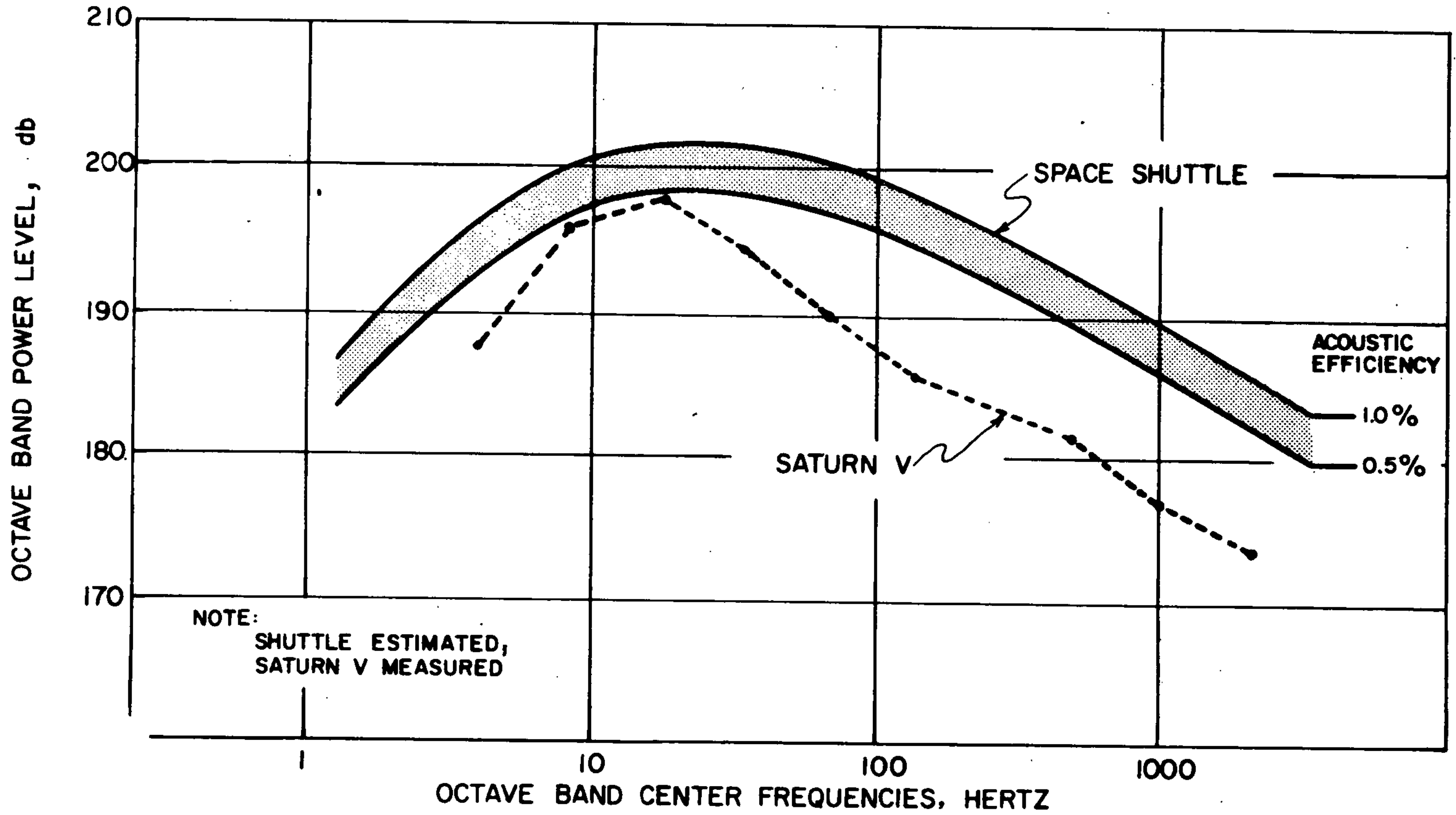


Figure 5.53 Space Shuttle Noise Levels

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Appendix 5.1: Table of Metric to English Conversions

Length

$1\ cm = 0.394\ in$

$1\ m = 3.281\ ft$

$1\ km = 0.621$ statute miles $= 0.539$ nautical miles

Area

$1\ cm^2 = 0.155\ in^2$

$1\ m^2 = 10.76\ ft^2$

Volume

$1\ cm^3 = 0.0610\ in^3$

$1\ m^3 = 35.31\ ft^3$

Mass

1 kg = 0.0685 slugs*

1 metric ton** = 68.5 slugs

Force

1 N*** = .2248 pounds

Pressure

$1\ n/cm^2 = 0.0987$ atm = 1.450 psi

Speed

1 m/s = 2.237 mph = 1.944 knots

---

* 1 slug (1 pound-sec^2/ft) weighs 32.1579 pounds at Princeton, N.J., hence, 1 kg weighs 2.2035 pounds.

** 1 metric ton is defined as 1000 kg and weighs 1.102 tons (adv.).

*** 1 N (Newton) is defined as 1 kg-m/s^2.

CHAPTER 6.0

SPACE TRANSPORTATION SYSTEM COSTS AND UNCERTAINTIES

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CHAPTER 6.0

SPACE TRANSPORTATION SYSTEM COSTS AND UNCERTAINTIES

6.1 Introduction

In this chapter we discuss the measurement and classification of launch vehicle and payload costs for the various Space Transportation Systems and present their costs. Undiscounted costs are reported for current and new expendable launch systems and reusable and partially reusable Space Shuttle concepts. Much of the following, especially the treatment of current and new expendables and payloads, follows the format of MATHEMATICA's May 31st report since these have undergone only evolutionary changes since that time. However, this is not the case for the discussion of Space Shuttle costs. Funding constraints have led to drastic changes in the conception of the Space Shuttle concept which has rapidly evolved into several partially reusable configurations.

Section 6.2 describes the data as to primary and secondary sources, and discusses the evolution and present status of Space Shuttle costing efforts. Launch vehicle and payload costs are reported for each Space Transportation System. The NASA-DoD baseline mission model (Scenario 1) and the other traffic models are articulated and cost savings from the design, reuse, updating and refurbishment of payloads are discussed. Qualitative assessments of data reliability are also made.

Section 6.3 is devoted to the problem of uncertainty in cost estimates. The use of scenarios to simulate the uncertain future environment of the space program is discussed as is the problem of cost growth in large, high technology development programs. A sample risk analysis is also performed for Space Shuttle recurring costs.

6.2 Data Description

6.2.1 Space Shuttle Configurations

Aerospace Corporation was the prime source of cost and configuration data for the expendable launch systems but not for the current partially reusable systems. Payload effects as studied by Lockheed were reported by them, but also obtained more directly from LMSC. Costs for the Space Shuttle configuration came from the competing contractors and display more of a scatter than was the case with the fully reusable Phase B Baseline. This is because there are currently many variations of at least three different shuttle concepts, whereas in May we were dealing with a single design -- the McDonnell Douglas high cross range shuttle as costed by Aerospace.

The rapid evolution in Space Shuttle designs over the past few months was induced by funding constraints. Non-recurring costs of the Phase B Baseline of $13.6 billion with peak annual funding of $2.4 billion [2] were simply too high. Attempts to extend the program development so as to stay below a maximum peak year funding of $1 billion were unsuccessful since shuttle benefits were deferred too far into the future.

The current approach calls for a "minimum technology" design and development effort that will trade off lower non-recurring costs for higher operations costs. Note that this higher direct cost per launch is not of overriding importance so long as the Shuttle is still capable of capturing payload benefits. The majority of shuttle benefits come from the ability to update, reuse and refurbish payloads, not from lower launch costs. In fact, a Space Shuttle does not appear to be an economic investment when only launch costs are considered. Launch costs, though important, are generally on the order of a third of payload costs which would tend to imply that, from a purely economic standpoint, a Space Transportation System should be designed more to maximize payload benefits than to minimize operations costs.

The design philosophy adopted was to develop the Space Shuttle in two phases, Mark I and Mark II. In Mark I a lower risk, less sophisticated vehicle would be developed so that benefits might be obtained earlier. Meanwhile, a more sophisticated Mark II vehicle with significantly lower operations cost would be in development for deployment somewhat later.

Once the Mark II configurations were operational, the Mark I vehicles would be retrofitted to the more sophisticated design. The situation in which a Mark II capability was developed with no interim program was also considered.

Lower cost options considered have included orbiters with either external liquid hydrogen tanks, external liquid hydrogen and liquid oxygen tanks, or a single tank holding both. The orbiters have either been end loaded for series burn, mounted side by side with the booster for parallel burn, or mounted in tandem with another orbiter, no booster, and three lox-hydrogen tanks sandwiched between. Booster options considered have included a modification of existing expendable liquid propellent boosters by adding wings, modifying for reusability, and giving a flyback capability. Also evaluated were reusable pressure fed boosters, either a large series burn concept or two smaller boosters burning in parallel with the orbiter. Finally, expendable solid rocket motor strap-ons were suggested although there is some indication that these could eventually be made partially reusable [10].

There are currently three shuttle concepts being considered, but two of the options imply the possibility of using either pressure fed or solid boosters so there are really five configurations of interest. All the orbiter designs are similar in that they include a single expendable hydrogen/oxygen drop tank.

Total Space Transportation Systems costs are summarized in Table 6.1. Both launch vehicle and payload costs are included for the Current Expendable, New Expendable and reusable systems. Totals have been rounded to the nearest billion dollars because of the inherent uncertainty in these costs.

Non-recurring costs for the various shuttle configurations are reported by contractor in Table 6.2 along with cost per flight. These non-recurring costs include both RDT&E and investment for the Space Shuttle and Tug. The cost of the Space Tug and additional investment for Western Test Range (WTR) have raised these figures above contractor

TABLE 6.1

SPACE TRANSPORTATION SYSTEMS COST SUMMARY[1]

(Millions of Undiscounted 1970 Dollars)

	Current Expendable	New Expendable	Space Shuttle and Tug
EXPECTED LAUNCH VEHICLE COSTS			
Non-recurring costs (FY 1972-87)	1,620	2,000	7,450
Recurring Costs (FY 1977-1990)	10,600	8,760	4,800
TOTAL LAUNCH COSTS	12,000	11,000	12,000
EXPECTED PAYLOAD COSTS (Satellites)			
RDT&E (FY 1975-1990)	11,000	10,600	9,880
Recurring Costs (FY 1976 - 1990)	18,800	18,400	12,700
TOTAL PAYLOAD COSTS	30,000	29,000	23,000
EXPECTED TOTAL SPACE PROGRAM COSTS	42,000	40,000	35,000

(1) Source: September Contractor Data

TABLE 6.2

NON-RECURRING COSTS (INCLUDING TUG & WTR) VERSUS COST PER FLIGHT[1]

(Millions of 1970 Dollars)

	Non-recurring Cost	Cost per Flight [2]
McDonnell Douglas		
Flyback	10,600	6.2
Single RAO	8,100	6.5
Twin RAO	7,800	6.9
Twin RAO (SRM)	6,900	10.0
North American Rockwell		
Flyback	9,000	5.6
Single RAO	7,800	7.9
Twin RAO	7,400	10.2
Grumman		
Flyback	8,700	6.7
Single RAO	8,300	8.1
Twin RAO (SRM)	7,900	8.1

Note: Single RAO (Rocket Assisted Orbiter) is series burn, Twin RAO is parallel burn.

(1) Source : Contractor data, all estimates rounded

(2) Tug not included in cost per flight

shuttle estimates by about $1.6 billion.

Space Tug cost estimates have changed only slightly from those reported in our May 31st report. The Tug design was assumed not to differ because of changes in the shuttle design. Also, only one Tug concept was considered. Costs for the Tug were obtained from Aerospace's August report [2]. It can be argued that these estimates are too low since the Tug design in question was configured for the fully reusable baseline system. The Space Shuttle, as originally conceived, was to have more energy available to stage the Tug. The clear implication is that any degradation in the first stage (Shuttle) performance will have to be made up by the second stage (Tug). From which one can conclude that the Tug estimates involve a lower technology and lower cost system than will eventually be needed.

This is perhaps true. However, note that not all payloads require maximum Tug performance. Not all Tug oriented missions involve payload delivery to synchronous equatorial orbit. Furthermore, the direct cost per flight of the Space Tug is significantly less than the cost to fly the Shuttle. Large percentage increases in Tug operating costs manifest themselves as much smaller percentage increases in total operations costs.

Finally, it is not at this time clear that Tug development would be incurred by the United States since the Europeans have expressed an interest in the project. However, this does not mean that the Tug would be obtained "free". The argument that shuttle users be charged only the incremental cost per launch in order to maximize the benefits to society as a whole does not apply to users from other countries.* The Europeans would tend to charge the United States more than marginal costs for the use of the Tug and the United States would charge more than marginal costs for the use of the Space Shuttle. Note that fees would not necessarily include amortized investment and RDT&E but would be "whatever the market will bear." It is to one's advantage to sell a trip if one makes more money on the sale than if one does not. Note that there have been no launches for

---

*We are speaking strictly economically, here, of course. Politically it might even be considered advantageous to charge a foreign government <u>less</u> than the marginal cost.

foreign countries included in MATHEMATICA's analysis. Although the European space program has not in the past been very large, it is certainly quite probable that there will be an increase in the next two decades. Each mission flown by the United States for the Europeans would imply an additional benefit of the Shuttle and one larger than would be the case for a mission flown for some intra-American group. It is thus quite likely that any optimism in Tug estimates is more than compensated by other considerations.

Space Tug cost estimates were:

Space Tug Costs

RDT&E	$600M
Investment (non-recurring)	$180M
Total	$780M
Direct Cost per flight	$.49M

Boosters

Let us consider some of the factors involved in choosing a first stage, this being the major area of difference among the present configurations. Figure 6.1 illustrates some of the relevant issues involved in choosing the Shuttle first stage. If one opts for expendable boosters, the choice would probably be a solid. A liquid propellent booster would then be too costly and a pressure fed would have to be developed as there are no pressure fed engines of sufficient size currently in operation. A cost-effective solid would also need some development, but non-recurring costs would be much lower since the technology is better understood.

For reusable first stages, there exist two options -- ballistic return and manned flyback. Pump fed liquid propellent engines are needed for the flyback case since the wing structure weight requires the most efficient engine possible. For ballistic re-entry, the pump fed would probably be excluded since its complexity would make it difficult to refurbish after it is pulled out of the ocean. Both the solid and the pressure fed may be optimal for this last case, there being a trade off between non-recurring

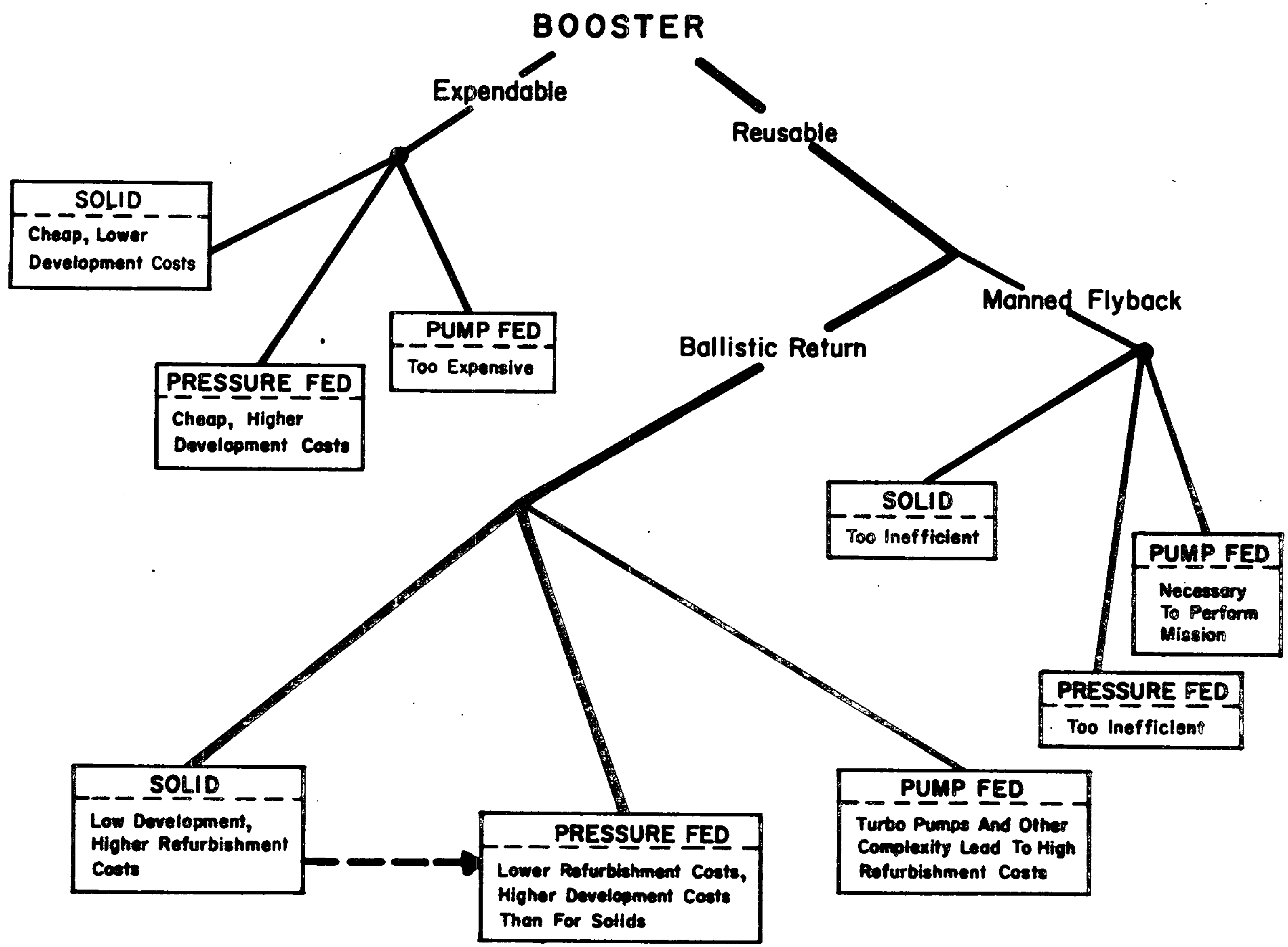

Figure 6.1: THE BOOSTER DECISION

and recurring costs. Costs for the solid are lower for the former and higher for the latter. It is also quite possible that one might opt for a solid booster initially and later phase into a pressure fed.

Table 6.3 gives total program costs, RDT&E costs and Mark II costs per flight for the five configurations as reported by Grumman, McDonnell Douglas, North American Rockwell and Lockheed in December. Great care should be taken in interpreting it. It would be unwise to attach too great a credence to differences between contractors for the same configurations. Differences are in part due to costs being classified differently, different assumptions as to procurement, amortization, Shuttle flights and so on. What each contractor means by a particular configuration is not necessarily the same. It is, however, useful to compare a single contractor's estimates for the different Shuttle designs.

Consider the more detailed cost breakdown in Tables 6.4 and 6.5. These costs, are based on October contractor data which will account for any differences from Table 6.3. Note that the non-recurring costs reported in Table 6.4 comprise both RDT&E and investment and cannot be directly compared with the RDT&E costs in Table 6.3.

The Twin RAO (SRM) figures were calculated by assuming that the use of solid rockets would affect only booster related costs. This appears to be a fairly safe assumption since the solid rocket options are generally costed on the basis of freezing the design and sizing the booster accordingly. Non-recurring costs were estimated by taking the difference between the pressure fed and SRM Twin RAO options in Table 6.3 and charging it as a reduction in both booster and flight test costs.

For estimating cost per flight using solid rocket motors, the booster related costs were subtracted. These included booster (with engine) amortization and booster related operations costs consisting of manpower and materiel expenditures. To this was added $6.4 million, the cost of procurring two 156 inch solid rocket motors. Grumman estimates of $8 million for two SRM's [5] were not used.

TABLE 6.3: REPRESENTATIVE CONTRACTOR COST ESTIMATES[1][2]

(Millions of 1970 Dollars)

	Flyback Booster	Single RAO		Twin RAO	
		Pressure Fed	SRM (3 x 156)	Pressure Fed	SRM (2 x 156)
Grumman					
Total Program	9,300	10,800	14,500	10,500	11,500
RDT&E	4,900	5,000	4,100	4,600	3,900
Cost per Flight (Mark II)	6.7	8.1	22.0	8.1	15.3
McDonnell Douglas					
Total Program	12,900	10,900	13,500	10,500	10,600
RDT&E	7,400	5,700	4,400	5,200	4,200
Cost per Flight (Mark II)	6.2	6.5	15.7	6.9	10.0
North American Rockwell					
Total Program	10,100	9,800	---	10,700	---
RDT&E	6,200	5,200	---	4,300	---
Cost per Flight (Mark II)	5.6	7.9	---	10.2	---
LMSC					
Total Program	8,100	8,400	---	7,600	8,100
Cost to FMOF	4,000	4,000	---	3,600	3,500
Cost per Flight (Mark II)	7.6	8.9	---	8.9	9.9

(1) Source: December contractor data. All estimates rounded.

(2) All Space Tug related costs excluded.

TABLE 6.4: NON-RECURRING COST BREAKDOWN OF FOUR REPRESENTATIVE CONFIGURATIONS[1]

(Millions of 1970 Dollars)

	Reusable Flyback Booster	Single Pressure Fed RAO (Series Burn)	Twin Pressure Fed RAO (Parallel Burn)	Twin RAO SRM [2] (Parallel Burn)
Orbiter	(4340)	(4050)	(3990)	[[3990]]
Mark I	2510	2510	2830	[2830]
Mark II	1830	1540	1160	[1160]
Drop Tanks	290	350	340	[340]
Booster	(1350)	(940)	(660)	[630]
Mark I	1310	890	630	
Mark II	38	41	27	
Flight Test	(640)	(680)	(680)	
Mark I	540	580	580	
Mark II	96	93	93	
Management & Integration	430	490	490	[490]
TOTAL	7000	6500	6200	[5500]

(1) Source: October Grumman data. Numbers do not add to totals shown because of rounding.

(2) MATHEMATICA estimates derived from December Grumman data.

An estimate of $4 million for each strap-on is not consistent with other contractor estimates; therefore, the smaller figure is reported in order to make Table 6.5 more general. The estimate was derived from plots of unit cost versus propellent weight. Solid booster costs are pretty much a function of propellent weight.

Using a propellent weight of 1.2 million pounds [5], Ref. [14] yielded an estimate of $2.8 million per booster based on a propellent production rate of forty million pounds per year. Note that this rate is a bit low for Shuttle use and could only accommodate about eighteen flights per year. Shuttle use of SRM's would imply higher propellent production rates and thus somewhat lower costs. McDonnell Douglas [6] predicts costs of $2.7 million for a similar size booster. Reference 16 estimates $3.2 million for motors with 1.37 million pounds of propellent. This last number was used in Table 6.5 simply because it was the largest.

Non-recurring costs for solid rocket motors are lower than for any other booster development. The estimates for non-recurring costs are the order of $145 million [10] to $183 million [5]. The 156 inch diameter vehicles and estimates for the 120 inch strap-ons are even lower. Development times are estimated to be about three and one-half years [16].

For the series burn configuration, development costs of the large reusable pressure fed booster as estimated by Grumman [5] are $890 million. Engine development would be an additional $180 million for a total of $1170 million. This is for a phased orbiter engine program development with a J2S in Mark I and an SSME in Mark II. Going directly to a Space Shuttle Main Engine (SSME) would reduce these figures to $820 million and $170 million for a total cost of about $1 billion.

Similarly, the Twin RAO program costs are estimated at $600 million for booster development plus $130 million for engine RDT&E in the Mark I/Mark II program. Again, these figures are reduced when one opts to go directly to advanced orbiter engines. In this case, costs would be $520 million for the booster plus $120 million for the engines to give a total development cost of $640 million.

TABLE 6.5: COST PER FLIGHT BREAKDOWN OF FOUR REPRESENTATIVE CONFIGURATIONS[1] (Millions of 1970 Dollars)

	Reusable Fly-back Booster[2]		Single Pressure Fed RAO[3] (Series Burn)		Twin Pressure Fed RAO[3] (Parallel Burn)		Twin RAO-SRM (Parallel Burn) (3), (4)	
	Mark I	Mark II	Mark I	Mark II	Mark I	Mark II	Mark I	Mark II
Drop Tank	2.1	2.1	2.4	2.4	2.4	2.4	[2.4]	[2.4]
Booster	3.0		7.6	3.6	6.9	3.1	[6.4]	[6.4]
Booster Engine	.7	.7	1.8	.4	1.1	.3	---	---
Orbiter Engine	1.1	1.1	.4	.3	.7	.2	[.7]	[.2]
Operations	6.1	3.7	15.0	4.5	15.0	4.5	[13.0]	[2.9]
TOTAL	13.0	7.5	27.1	11.1	26.0	10.4	[22.5]	[11.9]

(1) Source: October Grumman data. Numbers do not add to totals shown because of rounding.

(2) Based on 123 flights in Mark I, 322 flights in Mark II.

(3) Based on 49 flights in Mark I, 411 flights in Mark II.

(4) MATHEMATICA estimates obtained by subtracting pressure fed booster related operations cost; booster cost from Reference [2] . See text for details.

More conservative (higher) estimates for Twin RAO costs are reported by Chrysler [17] which estimates $1260 million for RDT&E, $1430 million for investment and $1410 million for operations to give total undiscounted program costs of $4.1 billion. Unamortized cost per flight is $3.2 million or $6.4 million with booster amortization. Chrysler assumes a 445 flight program and a development time of some 7 years. The study also seems to indicate a more conservative design than those of the other contractors -- weights are higher, load factors are less.

Lockheed Stage-and-One-Half Concept

Not explicitly treated in the economic analysis was the Lockheed stage-and-one-half concept. The costs for this configuration were significantly lower than those for any other configuration reported by any contractor. Cost for RDT&E and investment was given as $4.1 billion [33]. This included the procurement of five orbiters. Cost per flight was given as $5.65 million of which $2.8 million is for the purchase of two drop tanks. If these values are realistic, this concept is economically superior to all others considered. This can be seen if one notes the location of this configuration on the tradeoff graphs in Chapter 1. The point for $5.7 billion in non-recurring costs (includes Tug and Western Test Range) and operation costs of $5.6 million is farthest from the tradeoff line.

We note, however, that these costs seem somewhat more optimistic than those reported by other contractors. Development costs for the configuration are lower than those we have seen for orbiters that are being staged by boosters. First unit cost for the Lockheed orbiter is significantly below other contractor estimates despite the fact that it has over twice as many engines and a higher inert weight. Nevertheless, it appears that the concept warrants study. All booster related costs are eliminated in the stage-and-one-half and it might be that even pessimistic assumptions of costs may still lead to it being the most economic system.

Drop Tanks

The viability of partially reusable Space Shuttle configurations will in large part be dependent on drop tank costs. It was therefore deemed useful to pursue this question in some depth.

In 1969 an Ad Hoc Committee for drop tanks, established as part of the DoD Space Transportation System Working Group, concluded that costs of less than $30 per pound might be possible for tanks with dry weight fractions of .05 to .06. Aerospace Corporation went into this question in detail and developed cost estimating relationships based on tank dry weight for drop tank unit and development costs [18] [19].

As there exists no comparable drop tank data base, those relationships were based on body-tank structure cost experience from the Saturn SIC, SII, SIVB and Titan III programs. It was further assumed that drop tank development costs would be 70 percent of what the above experience would predict for body-tank structure. Similarly, a likeness factor of 50 percent was used for predicting first unit cost. Aerospace also suggested that a learning curve factor of 88 percent be used.

These cost estimating relationships were multiplied by a factor chosen to account for design and manufacturing complexities. For example, the data indicate that a tank containing liquid hydrogen or liquid oxygen will be about twice as expensive as one containing a non-cryogenic propellent.

We have applied these relationships to two sample hydrogen/oxygen drop tanks with structural weights of 60 and 100,000 pounds. These weights are representative of current designs. A complexity factor of two was used since the designs must carry both liquid hydrogen and liquid oxygen. It was further assumed in the calculation of unit cost that 445 would be built with a learning of 88 percent. The use of an 88 percent learning curve means that each time the number of units doubles, the <u>average</u> unit cost goes down by 12 percent. Thus, after 445 drop tanks the average unit cost has gone down nearly 70 percent.

	60,000 lbs.	100,000 lbs.
Development Cost	$280M	$320M
First Unit Cost	$ 11M	$ 16M
Unit Cost	$ 3.5M	$ 5.2M
Per Pound Unit Cost	$ 58	$ 51

Contractor cost estimates have generally been lower than the above would indicate.

Time Phased Costs

Appendix 6A contains two plots of time phased contractor estimates for a Twin RAO configuration. These are broken down into categories to illustrate the impact of the various components. Also plotted is the same configuration but with certain assumptions of slippage and cost growth. The large scatter in the cost data makes these sensitivity studies even more needful than usual. However, approaches such as that depicted in Figure 6A.3 do not eliminate the need for resolving the anomalies in the data. This brings us to the discussion of uncertainty which is treated in Section 6.4.

Infinite Horizon

Also included in Appendix 6A is Table 6A.1 which presents time phased costs for a particular scenario using the New Expendable System. It is reported there as an illustration of the use of the infinite horizon. All numbers in the box become recurring costs when this is considered. The boxed costs are repeated into the indefinite future; new orbiters are purchased, new payloads are developed and procurred and so on. These discounted values are then totaled to get the programs net present value using an infinite horizon. The subject is discussed in more detail in our May 31st report.

6.2.2 Expendable Configurations

The source of all cost data presented in this section on expendable systems is the Aerospace Final Report, Volume III [2].

6.2.2.1 Current Expendable System

The Current Expendable (CE) fleet articulated by Aerospace includes configurations from the Scout, Thor, Atlas, Titan III, and Saturn vehicles. Aerospace carried out cost estimates in the standard life cycle format (RDT&E, Investment, and Operations), but a redivision of costs for both Current Expendable and New Expendable fleets into recurring and non-recurring costs was necessary in utilizing the SAMSO/Aerospace expendable vehicle costing program.

The Current Expendable costing effort by Aerospace was directed toward "off the shelf" vehicles, and thus was characterized by the comparatively simple task of determining where each vehicle is in its "learning curve" production process, and how future launch rates will affect future unit costs. The (wholly non-recurring) RDT&E costs were generated by making point estimates of cost for each specific vehicle configuration, and prior studies were used where possible. Non-recurring investment costs (chiefly for extra launch facilities) were largely based on similar costs for existing facilities.

These costs are summarized in Table 6.6. The non-recurring costs, however, are dominated by the high Recurring Investment and Operations costs of expendable launch vehicle programs; consequently, Aerospace Corporation devoted its major effort to recurring-cost estimation. It utilized a cost model requiring two categories of input data:

1. Quantities of vehicle configurations launched by year and by launch site (Eastern Test Range, Western Test Range).

2. Cost data as a function of quantity of vehicle hardware elements and operations elements, for each vehicle configuration in the fleet.

The model develops (average) cost versus use-rate curves, from which the average or unit-recurring vehicle cost can be subsequently extracted based on the launch rate required by the traffic model (c.f. [1] for a discussion of how these use rates are determined). Table 6.7 presents these costs based on use of the baseline traffic model's vehicle use rates; Table 6.8 presents total recurring launch costs (again, based on the baseline traffic model), broken down further into Investment, Operations, and Range (WTR and ETR) cost components.

TABLE 6.6: TOTAL NON-RECURRING COSTS

CURRENT EXPENDABLE FLEET*, FOR 1978-1990

U. S. SPACE PROGRAM (SCENARIO 1)(1)

(Millions of Undiscounted 1970 Dollars)

COST CATEGORY	TOTAL
RDT&E	
TITAN IIIB/CENTAUR	20
TITAN IIID/CENTAUR	10
TITAN IIID (7)/CENTAUR	5
TITAN IIIM	75
OTHER (PAYLOAD/VEHICLE INTEGRATION)	50
TOTAL	160
INVESTMENT (NON-RECURRING)	
ADDITIONAL TITAN III CAPABILITY, ETR	24
CENTAUR CAPABILITY, ETR	26
AGENA CAPABILITY, ETR	8
SUBTOTAL, ETR	(58)
TITAN III (7) CAPABILITY, SLC-4E/W, WTR	23
CENTAUR CAPABILITY, WTR	26
SUBTOTAL, WTR	(49)
TOTAL	107

* Excluding Big G costs.

(1) Source: Table 2.3-3, reference [2]. No Non-recurring costs are expected beyond 1982: the above costs are incurred in the 1975-1982 period.

TABLE 6.7: UNIT RECURRING COSTS

CURRENT EXPENDABLE FLEET[1]

VEHICLE	LAUNCH SITE	TOTAL NUMBER OF FLIGHTS	COST PER FLIGHT (DIRECT OPERATING COST) $ M, 1970
SCOUT	WTR	4	3.4
TAT (3C)/DELTA	ETR	22	6.2
	WTR	80	5.8
TAT (3C)/DELTA/TE-364	ETR	12	7.0
TAT (9C)/DELTA/TE-364	ETR	8	7.0
	WTR	40	6.4
TITAN IIIB/AGENA	ETR	71	8.9
	WTR	12	10.7
TITAN IIIB/CENTAUR	ETR	77	10.5
	WTR	3	12.6
TITAN IIIC	ETR	88	12.6
	WTR	20	13.4
TITAN IIID	WTR	66	9.8
TITAN IIID/CENTAUR	ETR	48	16.1
TITAN IIID (7 SEG)	WTR	60	10.6
TITAN IIID (7 SEG)/CENTAUR	ETR	14	16.6
	WTR	9	18.9
TITAN IIID (7 SEG)/CENTAUR/ BURNER II	ETR	6	17.6
TITAN IIIM	ETR	65	15.1*
INTERMEDIATE 21	ETR	1	240.0

* Plus 30.0 for BIG G

(1) Source: Table 2.3-5, reference [2].

TABLE 6.8: TOTAL RECURRING COSTS
CURRENT EXPENDABLE FLEET, 1978-1990(1)

(Millions of Undiscounted 1970 Dollars)

VEHICLE	TOTAL
INTER *21	240.0
INVESTMENT	160.0
OPERATIONS (ETR)	80.0
SCOUT	13.5
INVESTMENT	11.0
OPERATIONS (WTR)	2.6
TAT 3 DEL	601.1
INVESTMENT	392.3
OPERATIONS	208.8
ETR	52.5
WTR	156.3
TAT 3 DEL 64	83.5
INVESTMENT	48.3
OPERATIONS (ETR)	35.2
T3D	647.5
INVESTMENT	569.6
OPERATIONS (WTR)	77.9
T3D/CENT	772.6
INVESTMENT	658.2
OPERATIONS (ETR)	114.4
T3D/7SEG	633.7
INVESTMENT	553.8
OPERATIONS (WTR)	79.9
T3D/7 CENT	402.3
INVESTMENT	326.6
OPERATIONS	75.7
ETR	33.2
WTR	42.5
T3D7CENTB2	105.6
INVESTMENT	90.3
OPERATIONS (ETR)	15.3

TABLE 6.8: (continued)

VEHICLE	TOTAL
TAT9DEL364	310.0
INVESTMENT	211.1
OPERATIONS	98.9
ETR	20.7
WTR	78.2
T3B/AGENA	758.2
INVESTMENT	608.5
OPERATIONS	149.6
ETR	109.2
WTR	40.4
T3B/CENTAUR	847.6
INVESTMENT	701.9
OPERATIONS	145.7
ETR	134.2
WTR	11.5
T3C	1378.9
INVESTMENT	1207.9
OPERATIONS	171.0
ETR	127.6
WTR	43.3
T3M	979.6
INVESTMENT	887.4
OPERATIONS (ETR)	92.2
TOTAL	7773.9
INVESTMENT	6427.0
OPERATIONS	1347.0
ETR	814.4
WTR	532.6

(1) Source: Table 2.3.4, Reference [2]. Includes some investment in 1977. Figures may not add to the totals shown because of rounding.

In support of manned space flights, a modified Gemini vehicle (Big G) was costed as the required re-entry/logistics vehicle. It consists of two modules -- a twelve-men re-entry vehicle module and a cargo/propulsion trailer module. The costs of the Big G were developed based on a 1969 study by MDAC, and comprise about one-fifth of the total Current Expendable launch costs. These costs are summarized in Table 6.9.

Also, for both Current and New Expendable fleets, additional operating costs are incurred for operating the launch sites. Annual range costs of $80 million/year at ETR and $40 million/year at WTR are utilized as being typical of the costs now attributed to space launch vehicles, excluding missile and ICBM programs, in budgeting forecasts for the late 1970's.

Reliability of Current Expendable Cost Data

The unit recurring launch costs utilized by Aerospace for Current Expendable vehicles are in general considerably lower than current launch costs, the predicted reduction in unit costs being due to procurement rates generally higher than current rates. It should be kept in mind that any reduction in total usage of a particular vehicle would result in increased unit costs: this is an important effect if traffic volume is reduced in the traffic model, and also accounts for higher unit costs when CE vehicles are used during the Space Shuttle phase-in period. Given these considerations, the Aerospace estimates for the Current Expendable recurring costs should be quite reliable, and their results should be comparatively easy to verify by NASA and industrial parties.

We note that the SAMSO/Aerospace Corporation model does not actually incorporate any "learning effects" in the costing of expendable launch vehicles; thus unit costs are in general not reduced as the program progresses. Consequently, even though unit costs may vary from year to year due to different procurement rates, the use of an average unit cost for the entire program is reasonable and does not cause any significant bias when these costs are discounted in our present value calculations.

TABLE 6.9: BIG GEMINI PROGRAM COSTS*

(Millions of Undiscounted 1970 Dollars)

			INVESTMENT COST				
			NON-RECURRING		RECURRING		
FISCAL YEAR	NO. OF LAUNCHES (1)	RDT&E COST	FACILITIES	RVM (2)	CPM (3)	OPERATIONS COST (4)	TOTAL COST
1973	0	0	0	0	0	0	0
1974	0	0	0	0	0	0	0
1975	0	50	0	0	0	0	50
1976	0	150	0	0	0	0	150
1977	0	250	0	0	0	0	250
1978	0	200	20	50	0	0	270
1979	0	100	20	125	0	0	245
1980	0	50	10	125	12	0	197
1981	1	0	0	125	72	38	235
1982	6	0	0	75	72	128	275
1983	6	0	0	0	72	128	200
1984	6	0	0	0	72	128	200
1985	6	0	0	0	96	128	224
1986	8	0	0	0	96	164	260
1987	8	0	0	0	96	164	260
1988	8	0	0	0	96	164	260
1989	8	0	0	0	96	164	260
1990	8	0	0	0	0	164	164
TOTALS:	65	800	50	500	780	1,370	3,500

* Source: Table 2.3-6, reference [2].

(1) Launches occur in calendar year

(2) RVM - Reentry vehicle module (12 new, 1 from RDT&E)

(3) CPM - Cargo/Propulsion Module

(4) Includes $20 million/year for indirect support

The total Current Expendable program costs are summarized in Table 6.10 below.

TABLE 6.10 TOTAL LAUNCH VEHICLE COSTS CURRENT EXPENDABLE FLEET(1) FOR 1978 - 1990 U. S. SPACE PROGRAM (SCENARIO 1)

(Millions of Undiscounted 1970 Dollars)

	LAUNCH VEHICLES	BIG G	TOTAL
RDT&E	160	800	960
INVESTMENT	6,534	(1,330)	7,864
NON-RECURRING	107	550	657
RECURRING	6,427	780	7,207
OPERATIONS	2,787	(1,370)	(4,157)
DIRECT	1,347	1,170	2,517
INDIRECT	1,440	200	1,640
TOTAL PROGRAM	9,481	3,500	12,981

(1) Source: page 2-20, reference [2].

6.2.2.2 New Expendable System

The objective of defining a new expendable vehicle fleet is to show the full potential of the expendable vehicle concept in comparing it with reusable launch vehicle concepts. The Aerospace Corporation approach to the definition effort was to employ a vehicle family concept based on the current Titan III family. This use of a family concept was felt to minimize non-recurring costs and to provide a high degree of commonality to gain the desirable effects associated with increased production.

The total Non-recurring costs (RDT&E and Non-recurring Investment) for the New Expendable program are shown in Table 6.11. RDT&E costs include all engineering, hardware, and test activities required to make each configuration operational; completion of the 7 segment SRM development is charged to the Titan III M, which would use that SRM as a strap on. No flight tests are required for the 5 segment SRM/Core II vehicle, but one Titan III M flight test and two Titan III L2, L4 flight tests are included in their costs. The Non-recurring Investment costs are due to additional facilities and equipment needed to launch the vehicles at rates required by the New Expendable baseline traffic model.

Unit recurring costs were predicted using the same cost model as for the Current Expendable fleet and are shown in Table 6.12. The input cost data for the new vehicle elements were estimated on a judgmental and past experience basis by Aerospace.

Total program Recurring costs for the New Expendable STS, based upon the predicted unit recurring costs and upon the New Expendable baseline traffic model, are shown in Table 6.13. The (recurring) Investment costs include all hardware costs and supporting in-plant functions; the Operations costs include all launch operation, propellants, transportation and other on-site support functions. The costing effort assumed single government agency procurement of all hardware of each vehicle family, with hardware costs based on total annual concurrent production of common elements.

TABLE 6.11: TOTAL NON-RECURRING COSTS
NEW EXPENDABLE FLEET*,
FOR 1978-1990 U. S. SPACE PROGRAM (SCENARIO 1)[(1)]

(Millions of Undiscounted 1970 Dollars)

COST CATEGORY	TOTAL
RDT&E	
3 SEG SRM/CORE II	25
5 SEG SRM/CORE II	10
7 SEG SRM/CORE II	10
UPPER STAGE INTEGRATION 3, 5, 7	25
TITAN IIID/CENTAUR	10
TITAN IIID (7)/CENTAUR	5
TITAN IIIM	75
TITAN IIIM L2, L4	175
OTHER (PAYLOAD/VEHICLE INTEGRATION)	50
TOTAL	385
INVESTMENT (NON-RECURRING)	
5 SEG SRM/CORE II AT ETR 36, INCLUDING CENTAUR	28
ADDITIONAL T-III AT ETR 40 41, INCLUDING UPPER STAGES	50
T-III L2, L4 AT ETR 37, INCLUDING CENTAUR	81
SUBTOTAL, ETR	(159)
5 SEG SRM/CORE II AT WTR SLC-4W, INCLUDING CENTAUR	26
CENTAUR CÅPABILITY WTR SLC-4E	26
T-IIID (7 SEG) AT WTR SLC-6, INCLUDING CENTAUR	51
SUBTOTAL, WTR	(103)
TOTAL	262

* Excluding Big G costs.

(1) Source: Table 2.4-4, reference [2].

TABLE 6.12: UNIT RECURRING COSTS, NEW EXPENDABLE FLEET[1]

VEHICLE	LAUNCH SITE	TOTAL NUMBER OF FLIGHTS	COST PER FLIGHT (DIRECT OPERATING COST) $ M, 1970
SCOUT	WTR	4	3.4
5 SEG SRM/CORE II/TE-364	ETR WTR	16 48	4.6 4.5
5 SEG SRM/CORE II/CENTAUR	ETR WTR	56 35	10.0 10.4
5 SEG SRM/CORE II/CENTAUR/TE	ETR	41	10.3
TITAN IIID	ETR WTR	0 60	0 9.8
TITAN IIID/BURNER II	ETR WTR	17 22	10.4 10.3
TITAN IIID/CENTAUR	ETR WTR	38 12	15.5 15.9
TITAN IIID (7 SEG)	ETR WTR	0 60	0 10.2
TITAN IIID (7 SEG)/BURNER II	ETR WTR	7 6	11.0 10.9
TITAN IIID (7 SEG)/CENTAUR	ETR WTR	33 9	16.0 16.3
TITAN IIID (7 SEG)/CENTAUR/B-II	ETR	6	16.8
TITAN IIIM	ETR	65	15.0*
TITAN III L2/CENTAUR	ETR	0	0
TITAN III L4	ETR	2	35.8
TITAN III L4/CENTAUR	ETR	1	40.8

* Plus 30.0 for Big G

(1) Source: Table 2.4-6, reference [2]. "B2" denotes Burner-2, "SEG" denotes segments of a Solid Rocket Motor (SRM). The AKM used is the TE-364.

TABLE 6.13: TOTAL RECURRING COSTS, NEW EXPENDABLE FLEET FOR 1978-1990 U. S. SPACE PROGRAM (SCENARIO 1)[1]

(Millions of Undiscounted 1970 Dollars)

VEHICLE	TOTAL
SCOUT	13.5
INVESTMENT	11.0
OPERATIONS (WTR)	2.6
CORE 2*364	290.2
INVESTMENT	251.5
OPERATIONS	38.7
ETR	10.2
WTR	28.5
CORE2*CENT	923.9
INVESTMENT	765.0
OPERATIONS	159.0
ETR	89.4
WTR	69.6
CORE2*C*TE	422.4
INVESTMENT	353.0
OPERATIONS (ETR)	69.5
T3D	586.8
INVESTMENT	520.9
OPERATIONS (WTR)	65.8
T3D BURN II	401.5
INVESTMENT	354.7
OPERATIONS	46.8
ETR	21.4
WTR	25.5
T3D/CENT	779.7
INVESTMENT	662.9
OPERATIONS	116.8
ETR	84.9
WTR	31.8

TABLE 6.13: (continued)

VEHICLE	TOTAL
T3D/7 SEG	613.2
INVESTMENT	547.4
OPERATIONS (WTR)	65.8
T3D/7/B2	142.1
INVESTMENT	126.2
OPERATIONS	15.9
ETR	8.7
WTR	7.2
T3D/7/CENT	675.2
INVESTMENT	575.9
OPERATIONS	99.3
ETR	76.0
WTR	23.3
T3D7 CENT B2	100.7
INVESTMENT	86.4
OPERATIONS (ETR)	14.3
TCM	975.2
INVESTMENT	873.8
OPERATIONS (ETR)	101.4
T3L4	71.6
INVESTMENT	56.7
OPERATIONS (ETR)	14.9
T3L4/CENT	40.8
INVESTMENT	32.2
OPERATIONS (ETR)	8.6
TOTALS	6036.8
INVESTMENT	5217.3
OPERATIONS	819.5
ETR	499.4
WTR	320.1

(1) Source: Table 2.4-5, reference [2]. Includes some investment in 1977.

Big G costs (for manned reentry) and range costs were assumed to be identical to those presented for the Current Expendable fleet. The total New Expendable program costs are summarized below.

TABLE 6.14 TOTAL LAUNCH VEHICLE PROGRAM COSTS FOR NEW EXPENDABLE [1] FLEET FOR 1978-1990 U. S. SPACE PROGRAM (SCENARIO 1)

(Millions of Undiscounted 1970 Dollars)

(COSTS IN MILLIONS OF 1970 DOLLARS)

	LAUNCH VEHICLE	BIG G	TOTAL
RDT&E	385	800	1,185
INVESTMENT	(5,479)	(1,330)	(6,809)
NON-RECURRING	262	550	812
RECURRING	5,217	780	5,997
OPERATIONS	(2,259)	(1,370)	(3,629)
DIRECT	819	1,170	1,989
INDIRECT	1,440	200	1,640
TOTAL PROGRAM	8,123	3,500	11,623

(1) Source: Page 2-39, reference [2].

6.3 Payloads

An extensive study of payload effects was conducted by Lockheed Missiles and Space Corporation (LMSC) from September, 1970 to June, 1971 and the results used by MATHEMATICA in the present analysis.* Although we have been continuously informed of the study's progress, Lockheed was not the direct source of data. Aerospace Corporation's payload cost data was the direct source, although its direct source for the payload information was LMSC.

This section briefly summarizes some of the results of the payload effects studies and is primarily an updating of Section 6.2.3.2 of MATHEMATICA's May 31st Report. For an in-depth treatment of the subject and a more complete data description, the reader is directed to Aerospace's Integrated Operations/Payloads/Fleet Analysis Final Report, especially Volumes II, III, IIIA and VI. Lockheed's reports are, of course, also important as the original source of payload savings from low cost design and reuse.

One of Aerospace's basic tasks in its payload costing effort was the description of the baseline (current design principles) payloads to be flown in the NASA-DoD baseline mission model. This effort, reported in [2], describes the NASA-OSSA, NASA-OMSF, non-NASA application and DoD payloads by their preferred orbit, IOC date, lifetime, dimensions, weight of mission equipment, and total weight. These parameters were utilized in a spacecraft cost-estimating model (developed by Aerospace for SAMSO) to determine baseline payload costs for the entire set of missions.

The LMSC parametric and "bottoms up" analyses were used for all payloads in the model that Aerospace believed was applicable to provide the low-cost expendable and (low-cost) reusable payload weight and volume data for the capture analysis; this subsequent capture analysis assigned payloads to launch vehicles for each of the current expendable, new expendable

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* LMSC's effort is continuing with emphasis being given to the economic benefits associated with standardized spacecraft.

and reusable Space Transportation Systems, and was necessary in costing the launch operations and determining whether low-cost designs resulted in a net cost reduction (in many cases they do not, since payload weight increase results in higher launch costs).

Aggregate payload costs as estimated for their baseline space programs in Aerospace's August Report [2] are shown in Table 6.15. Note that reuse and refurbishment of payloads causes the greatest benefits. The major part of the savings due to the relaxation of mass and volume constraints can be realized with the new expendable system as well as a Space Shuttle. These cost reductions are evidenced by the differences between the current and new expendable payload costs. The larger savings from reuse and refurbishment are partially implied by the differences between new expendable and new Shuttle payload costs, although this does not include payload refurbishment (contained in operations costs) which is, on the average, 39 percent of new unit costs.

LMSC provided Aerospace Corporation with the subsystem costs reported in Tables 6.16 and 6.17. These estimates were used to calculate the cost factors in Tables 6.18 and 6.19. These factors were calculated from the following relationship:

$$\text{Cost Factor} = \frac{\text{"Low Cost" Payload Cost Estimate}}{\text{"Baseline" Historical Payload Cost}}$$

The low cost payload cost estimate comes from the Lockheed payload effects study. The baseline historical payload cost in the denominator, though essentially the same as historical cost figures, is not exactly the same. The baseline costs were estimated by recosting the baseline payloads using the estimating techniques developed and following NASA ground rules. This was done to improve the compatibility between the two figures.

Aerospace reviewed the factors reported in Tables 6.18 and 6.19 and found the OAO communication and stability subsystem factors were not well suited for use with the Mission Model. These factors as well as the total OAO satellite factors were therefore not used by them. Propulsion factors were established by Aerospace on a judgemental basis since the payloads investigated did not contain propulsion systems. These were:

Table 6.15

Payload System Cost Summary For
Baseline Traffic Models [(1)] Scenario 1

(Billions of Undiscounted 1970 Dollars)

	Current Expendable	New Expendable	Shuttle
NASA			
RDT&E	9.00	8.60	7.58
Investment	8.49	8.28	3.11
Operations	1.01	.96	3.93
Total	(18.50)	(17.83)	(14.62)
Non NASA			
RDT&E	.37	.34	.35
Investment	1.80	1.66	.55
Operations	.32	.29	.75
Total	(2.49)	(2.29)	(1.65)
DoD			
RDT&E	1.64	1.59	1.43
Investment	4.47	4.42	2.01
Operations	.97	.94	1.95
Total	(7.08)	(6.96)	(5.38)
Total			
RDT&E	11.01	10.53	9.36
Investment	14.76	14.36	5.67
Operations	2.30	2.19	6.63
Total	28.07	27.08	21.65

Note: Payload refurbishment and maintenance is carried as an investment cost.

(1) Source: Reference [2], Table 3.32

TABLE 6.16

LMSC Cost Estimates, Recosted Baseline and Low Cost SEO (2 yr) Payload (1)
(Thousands of 1970 Dollars)

	Recosted Baseline			Low Cost-New Expendable			Low Cost-Shuttle		
	RDT&E	Unit Invest-ment	Unit Ops	RDT&E	Unit Invest-ment	Unit Ops	RDT&E	Unit Invest-ment	Unit Ops
Structure (2)	12,979	1,627		8,509	819		6,917	682	
Electrical	14,121	2,070		11,844	1,971		10,256	1,753	
Communication	25,450	3,358		21,534	3,401		17,846	2,855	
Stability (3)	21,296	3,061		15,112	2,761		12,965	2,555	
Propulsion	--	--		--	--		--	--	
Spacecraft	73,846	10,116		56,999	8,952		47,984	7,845	
Mission Equipment	46,979	3,036		40,990	2,217		37,716	1,985	
Satellite (4)	120,825	13,152	4,013	97,989	11,169	3,390	85,700	9,830	2,988

(1) Source = Reference [2], Table 3.2.1

(2) Includes Adapter, Structures and Mechanisms, and Environmental Control

(3) Includes G & N, Stabilization, and Attitude Control

(4) Nonallocated Costs Distributed Over Other Elements

TABLE 6.17

LMSC Cost Estimates, Recosted Baseline and Low Cost OAO-B (1 yr) Payload (1)

	Recosted Baseline			Low Cost-New Expendable			Low Cost-Shuttle		
	RDT&E	Unit Investment	Unit Ops	RDT&E	Unit Investment	Unit Ops	RDT&E	Unit Investment	Unit Ops
Structure (2)	17,270	2,311		11,142	1,666		9,717	1,369	
Electrical	17,532	3,607		11,579	3,461		11,390	2,867	
Communication	39,209	6,969		22,204	3,741		21,178	2,805	
Stability (3)	77,773	15,481		34,041	7,062		31,707	5,801	
Propulsion	--	--		--	--		--	--	
Spacecraft	151,784	28,368		78,966	15,930		73,992	12,842	
Mission Equipment	15,880	3,597		10,444	3,205		10,040	2,972	
Satellite (4)	167,664	31,965	10,992	89,410	19,135	6,669	84,032	15,814	5,349

(1) Source: Reference [2], Table 3.2.2

(2) Includes Adapter, Structures and Mechanisms, and Environmental Control

(3) Includes G & N, Stabilization, and Attitude Control

(4) Nonallocated Costs Distributed Over Other Elements

TABLE 6.18

Cost Factors (1)(2) Low Cost SEO Payloads

	Low Cost-New Expendable			Low Cost-Shuttle			Comments
	RDT&E	Unit Invest-ment	Unit Ops	RDT&E	Unit Invest-ment	Unit Ops	
Structure (3)	0.71	0.50		0.53	0.42		
Electrical	0.84	0.95		0.73	0.85		
Communication	0.85	1.01		0.70	0.85		
Stability (4)	0.71	0.90		0.61	0.83		
Propulsion	--	--		--	--		Low Cost SEO Did Not Include Separate Propulsion System Used Primarily For Propulsion Purposes
Spacecraft	0.77	0.88		0.65	0.74		
Mission Equipment	0.87	0.73		0.80	0.65		
Satellite	0.81	0.85	0.84	0.71	0.75	0.74	

(1) Reference [2], Table 3.2.3

(2) Cost Factor = $\dfrac{\text{LMSC Low Cost SEO Cost Estimate}}{\text{LMSC Recosted SEO Baseline Cost}}$

(3) Includes Adapter, Structure and Mechanisms, and Environmental Control

(4) Includes G & N, Stabilization and Attitude Control

TABLE 6.19

Cost Factors (1)(2) Low Cost OAO - B Payloads

	Low Cost-New Expendable			Low Cost-Shuttle			Comments
	RDT&E	Unit Invest-ment	Unit Ops	RDT&E	Unit Invest-ment	Unit Ops	
Structure (3)	0.65	0.72		0.56	0.59		
Electrical	0.66	0.96		0.65	0.79		
Communication	0.57	0.54		0.54	0.40		Cost Factors Not Applicable to Mission Model
Stability (4)	0.44	0.46		0.41	0.37		SEO Factors Used for These Subsystems
Propulsion	--	--		--	--		Low Cost OAO Did Not Include Separate Propulsion System
Spacecraft	0.52	0.56		0.49	0.45		Not Applicable To Mission Model, SEO Factors Used
Mission Equipment	0.66	0.89		0.63	0.83		-------------
Satellite	0.53	0.60	0.61	0.50	0.49	0.49	Not Applicable to Mission Model, SEO Factors Used

(1) Reference [2], Table 3.2.4

(2) Cost Factor = $\frac{\text{LMSC Low Cost OAO Cost Estimate}}{\text{LMSC Recosted OAO Baseline Cost}}$

(3) Includes Adapter, Structure and Mechanisms, and Environmental Control

(4) Includes G & N, Stabilization, and Attitude Control

	Low Cost Expendable
RDT&E	.95
Unit	.85
	Low Cost Space Shuttle
RDT&E	.85
Unit	.75

The formulation of non-dimensional cost factors enables one to extrapolate the cost reductions estimated for the Orbiting Astonomical Observatory (OAO) and the Synchronous Equatorial Orbiter (SEO) to all the satellites in the mission model.

A refurbishment factor was also estimated for those payloads brought back from orbit and later reused. These factors were developed by LMSC on a subsystem level, but the overall average was judged sufficiently accurate since the variation from it was small. Two average rates were produced, 39 percent for SEO type satellites and 32.5 percent for payloads similar to the OAO. This means that a satellite can be refurbished for that percent of its unit cost.

6.4 Uncertainty

A decision on a system as technologically complex, long in development, and expensive as the Space Shuttle cannot ignore the effects of uncertainties. These uncertainties will include those in cost, technology, schedule, the number and types of missions flown, and strategic and tactical considerations. The concern of this section is with the effect of uncertainties upon eventual costs and benefits and thus their implications of the economic portion of the Space Shuttle decision. A tabulation of some relevant uncertainties is reported in Table 6.19 of Reference [1].

Any cost estimate must be considered as representing a range of possible values, the distribution of which will vary as a function of such parameters as technological complexity, program length, or even managerial skill. The choice among any set of systems thus cannot be made solely on the basis of predicted costs since each prediction may have a different probability associated with it. This is illustrated by Figure 6.2 which might represent the choice between a more expensive but lower technology expendable solid rocket booster and the reusable but riskier pressure-fed system.

Some economists have suggested that risks inherent in particular programs should be taken into account either by adding a risk premium to the discount rate, or by adding to future costs and/or subtracting from future benefits. Such adjustments do not seem appropriate in our evaluation of alternative STS's. It is most unlikely that any simple index (such as the net present value) adequately assesses relative performanace of one project over another; rather, whichever indices we use must implicitly be qualified by our knowledge of the program's future environment and of its various uncertainties.

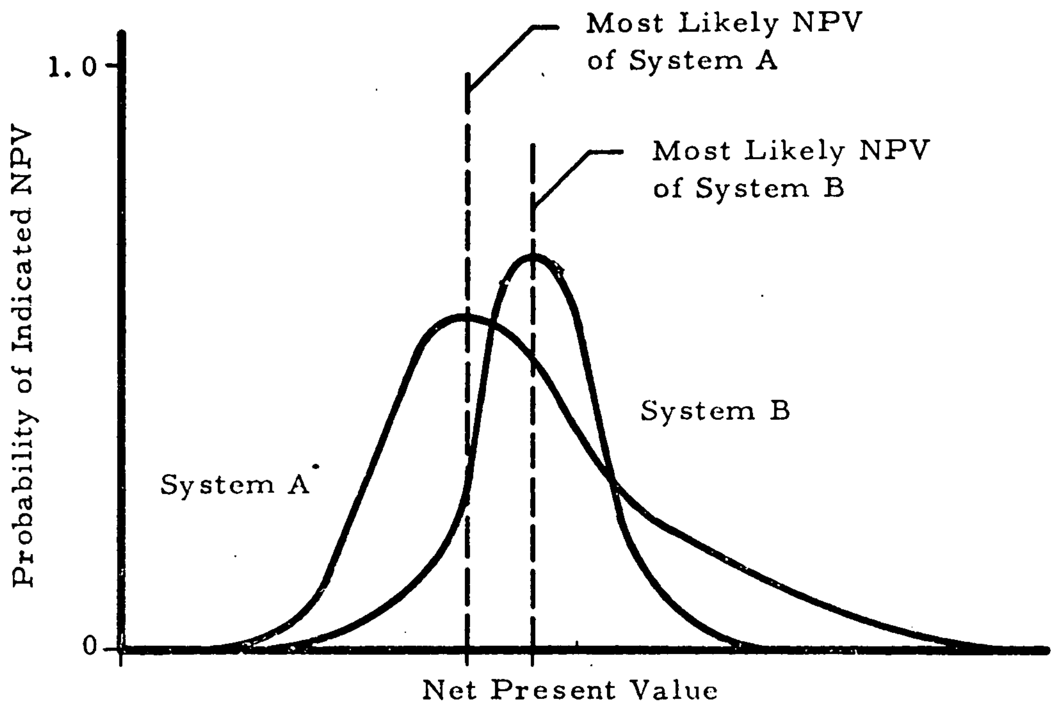

Figure 6.2

The Choice Between Two Alternatives with Different Associated Risks and Expected Values

If we rely on an NPV measure plus some strictly numerical "uncertainty" information such as the standard deviation, skewness, confidence limits, etc., we still have not accounted for the environment in which the program selection decision must be made. However, this, combined with the extra information from scenario analyses, can come close to providing all of the economic information which we can use concisely in advising on what we think is the best choice. The different scenarios are discussed in detail in the next section.

There are different questions involved in the uncertainties in non-recurring and recurring cost estimates. In the former, RDT&E phase uncertainties in cost, scheduling, technological readiness, etc., have high impact on risk. In the latter case, future cost streams (particularly for pay load and launch operation costs) are somewhat repetitive, correlated from year to year and the risk is of exceeding the costs of equally effective, competing systems.

It is desirable for several reasons to separate the uncertainty analysis of recurring costs from that on non-recurring costs:

(a) Non-recurring costs are severaly affected and constrained by budgetary, scheduling and development-phasing alternatives; recurring costs are less affected by these factors.

(b) Due to the above constraints, non-recurring costs are correlated in a fashion that is very difficult to simulate. For example, higher than expected cost for a particular development item early in the hardware program may mean that the development is in trouble and further annual and total cost overruns are likely or, it may mean that the program manager is accelerating development of this item and hence costs in later years may be lower than budgeted.

(c) Development time is an essential output factor and cost driver in the Space Shuttle development program; time is not a critical factor in examining cost streams which begin some ten years from now.

The rest of this chapter will be divided into three sections. First, there will be a discussion of the uncertainties in the environment of the eighties and scenario analysis. This will be followed by a discussion of the uncertainties implicit in a research and development effort. Finally, the question of Space Shuttle recurring costs will be considered by taking a 'snapshot' of a one year interval starting with about the fiftieth launching.

6.4.1 Uncertainty in Demand: Scenario Analysis

In analyzing the benefits and the cost-effectiveness of a new Space Transportation System, a very fundamental question is how much space activity can one expect the United States to perform in the 1980's and beyond. Advocates of a Space Shuttle System intuitively believe and claim that, once a reusable Space Transportation System with a low cost per launch is developed, the demand for space transportation will increase beyond anything done in the past or at present, since completely new uses of space can and will be found, for the direct benefit of the United States in commercial, civilian and defense applications, as well as for the benefit of other nations. These intuitive feelings may well prove correct, as history has shown so often in the past when a new field of technology was opened up.

Opponents and critics of a Space Shuttle System feel that many of the expenditures that went into space exploration and particularly manned space flight programs, were and are a "waste" of money, and the development of a Space Shuttle System amounts to "throwing good money after bad money." Yet, even the most severe critics of the U. S. space program grant that the unmanned space program of the United States, and other countries, is of great value and benefit.

The crucial question regarding a decision on a Space Shuttle System development, therefore, is what level of space activity justifies the development of a reusable Space Transportation System and are the space activities that would justify -- economically -- such a development inordinately high and how do these activity levels compare with historical activities of the United States and other countries. One has to keep in mind, thereby, that payloads have to be delivered, over time, to very different orbits, requiring at times additional stages or a Space Tug, and the mere fact of a payload bay of 60 x 15 feet and a 40K pound capability to polar orbit of 100 miles altitude does not mean that the actual, or even optional use of the Space Shuttle System will ever approach these payload weights in each and every flight.

A very detailed analysis of the loading and scheduling of Space Shuttle flights in the 1978-1990 and, again, the 1979 to 1990 period was performed by Aerospace Corporation in support of this study effort. The expendable, NASA and DoD baseline mission model was taken as the basis, and it was this mission model that the Space Shuttle System --of whatever configuration -- had to meet. As it turns ;out, the actual loading of the Space Shuttle System in terms of satellite payload weight comes to about 8,000 pounds on the average (not the 40,000 pounds theoretically available). This reflects some of the conservative operating assumptions imposed on the Space Shuttle System, and these are fully reflected in the life cycle cost streams of the economic analysis. Certainly, in the 1980's the operating knowledge gained on the new Space Transportation System will lead to a better and more efficient use of the Space Shuttle. But such improvements would then only help the economic analysis in favor of a Space Shuttle System. Since operating difficulties will certainly arise, particularly in the early period of Space Shuttle operations, such improvements are not allowed for in this analysis. It is also wrong, however, to simply infer, from the number of Space Shuttle flights, a very large amount of satellite payload weight in Earth orbit, the Space Shuttle. Some very enormous and misleading statements have been and can be made in this connection.

What matters are the actual space missions performed in the 1980's and used in the economic analysis when comparing Space Shuttle Systems to expendable modes of operations. What matters most, in the economic analysis, is the cost of payloads and of space missions in the 1979 - 1990 period. Since no certainty exists as to these requirements, we created the scenario approach to determine the limits within which a Space Shuttle System makes economic sense.

In order to subject the candidate Space Transportation System to economic analysis across a broad spectrum of possible future events and levels of activity, a number of scenarios, i. e., alternative models of future space activity, were constructed and presented in MATHEMATICA's May 1971 report, incorporating variations in the mix, rate of traffic build-up, phasing-in of the Space Shuttle, as well as the actual level of future space activity, a working framework for a broadened evaluation of the Space Shuttle decision. Nineteen scenarios were analyzed in the May, 1971 report, of which Scenario 1 was the NASA-DoD baseline mission model provided to MATHEMATICA. The remaining were formulated by MATHEMATICA and were essentially parametric variations from the baseline model. Each Space Program for the 1980's can be understood to consist of a set of space missions that will be performed -- (e.g., those described in Chapter 4 of this report for NASA) and are independent of the question of whether or not we will have a Space Shuttle System or not. These space programs are taken as the baseline requirements that any Space Transportation System has to meet on an equal capability basis. Table 6.20 describes such a space program in summary form, by year, in the case of the Space Shuttle System. In Table 6.20 we use the Space Shuttle System to meet the NASA-DoD baseline mission model of 736 Space Shuttle flights for 1978 to 1990, the baseline mission model of the May 31, 1971 report.

Table 6.20

NASA and Non-NASA Applications Shuttle Flights 1978 - 1990 (Not including DoD)

FY	New Satellites Deployed	Refurbished Satellites Deployed	On-Orbit Maintenance	On-Orbit Refurbishment	Flights to Space Station
1978	39	--			5
1979	36	--			8
1980	26	17	1		5
1981	13	22	3	1	5
1982	19	23	2	3	5
1983	6	26	4	2	4
1984	26	15	2	4	4
1985	25	14	5	2	5
1986	17	26	2	5	4
1987	10	30	5	2	5
1988	10	37	2	5	4
1989	6	33	5	2	4
1990	9	31	1	4	4
Total NASA	242	274	32	30	62
Total DoD	103	302	Some	Some	n.a.

A similar mission model, now, however, reduced to 624 Space Shuttle flights was taken as a baseline for the economic analysis since then.

What matters most, however, to the economic analysis of alternative Space Transportation Systems, is the costs associated with these payloads, over time, and their breakdown mission by mission. Table 6.20 gives such a breakdown, again for the old NASA-DoD baseline mission model of 736 Space Shuttle flights. In Table 6.20 the payload costs are shown for the Space Shuttle System and the exactly equivalent costs for the Current Expendable System for the OSSA (unmanned) part of the NASA space program. Similarly detailed breakdowns exist for the non-NASA applications, the DoD (unmanned) space program, as well as for OMSF. Also shown in Table 6.21 are the relative (percentage) cost distribution Space Transportation Systems. In similar fashion, we could also show the expendable payload costs. Figure 6.3 shows the results of Table 6.21 in diagrammatic form. The overall effects of a Space Shuttle System are a relative reduction of expected payload unit costs (as shown here), as well as payload RDT&E costs, and, through refurbishment and updating, of the costs of space missions over an extended time period. These alternative cost streams are all described, in detail, in Chapter 8.

The important aspect of the scenario approach is that, by reducing the number of space missions, and satellites to be deployed, one reduces also the overall costs of the portions affected (NASA, DoD, commercial applications), and not just the number of Space Shuttle flights. Furthermore, since the composition of new satellites deployed, refurbished satellites, etc., between agencies is very different, one can also assess the effects of these different contributions to the economics of the Space Shuttle System by substantially changing the NASA, DoD or commercial component in these space programs. The purpose of the scenario approach was therefore twofold: to measure the economic effects of substantially reducing or expanding the overall level of space program activity in the 1980's and to

Table 6.21

Distribution of OSSA Payload Costs, (Scenario 1)

Range of Payload Unit Costs (Millions of 1970 Dollars)	Space Shuttle		Current Expendable	
	# Payloads	% of Total	# Payloads	% of Total
0 - 19.9	162	36	17	4
20 - 39.9	181	40	242	61
40 - 59.9	62	13	46	12
60 - 79.9	13	3	17	4
80 - 99.9	22	5	1	<1
100 - 119.9			22	6
120 - 139.9	12	2	22	6
140 - 159.9			11	3
160 and over			20	5
Totals	452	100	398[1]	100

DISTRIBUTION OF OSSA PAYLOAD COSTS

SCENARIO I

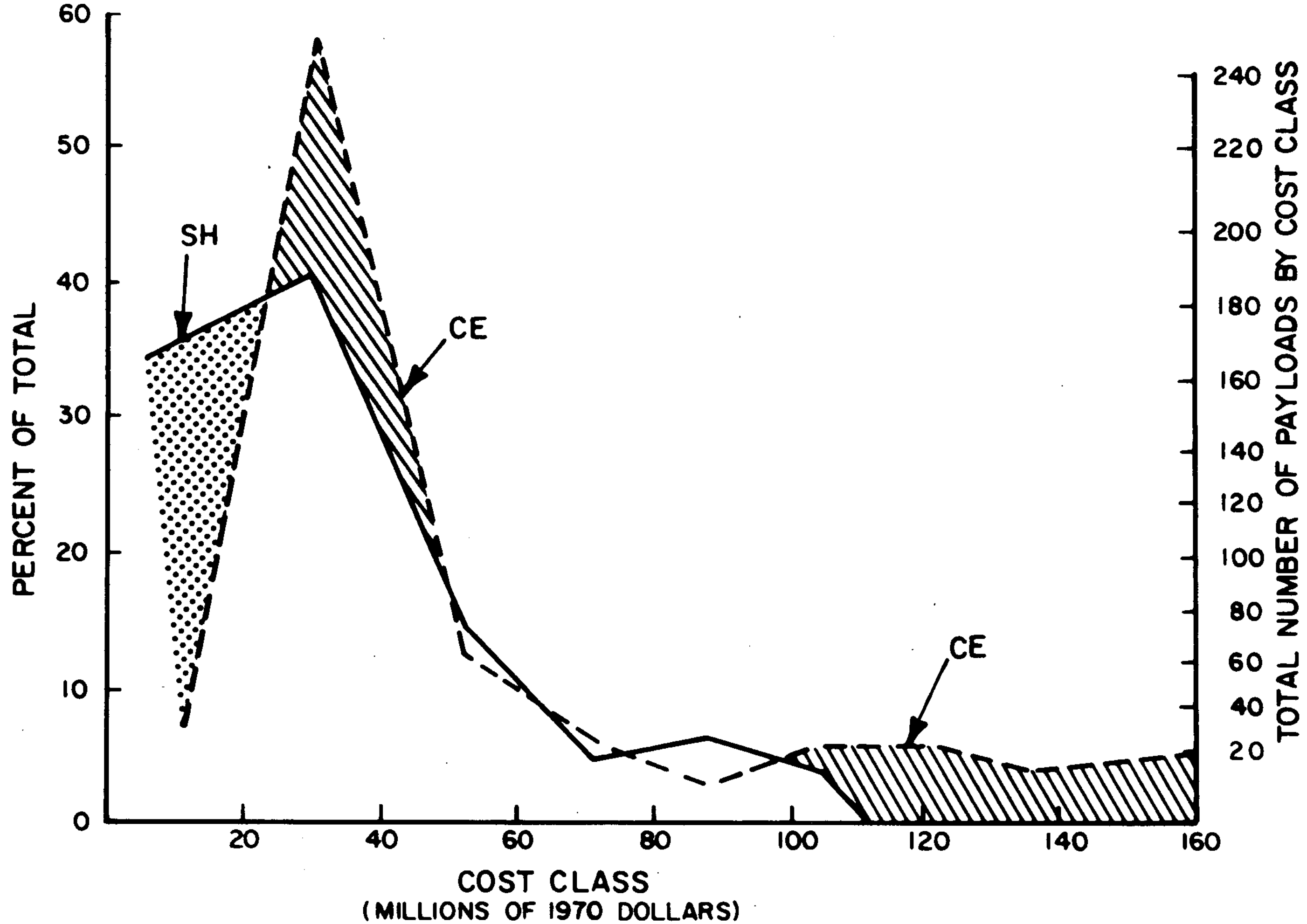

Figure 6.3

TABLE 6.22

DISTRIBUTION OF OSSA SPACE SHUTTLE PAYLOAD COSTS, SCENARIOS 1 AND 3[1]

Range of Payload Unit Costs Millions of 1970 Dollars	Scenario 1 # Payloads	Scenario 1 % of Total	Scenario 3 # Payloads	Scenario 3 % of Total
0 - 19.9	162	36	81	36
20 - 39.9	181	40	90	40
40 - 59.9	62	13	31	13
60 - 79.9	13	3	7	3
80 - 99.9	22	5	11	5
100 - 119.9				
120 - 139.9	12	2	6	2
140 - 159.9				
160 and over				
Totals	452	100	226	100

1 Based upon the 736 flight mission model

assess the effects of a substantially different mix of space programs between NASA, DoD and commercial space activities.

In doing so, however, the relative cost distribution of satellite payloads within each agency was held constant. That is, we still use 13 percent of satellites in the $40 million to $60 million cost class, 36 percent of satellites in the cost class of up to $20 million, etc. What changes is the absolute budget of each agency, but not the relative cost distribution of its space program by satellite cost class. The changes in the number of satellites, and their distribution by cost class, is shown in Table 6.23 for the first 3 scenarios (for the OSSA component.)

Had the economic analysis been biased in the adjustment of space activities either toward expensive (but fewer) or inexpensive (but more) satellite payloads, then the economic results would have been degraded or improved respectively. This is shown in Figure 6.4 in a general way. At this point no clear statement as to the actual situation in the 1980's can be made, with assurance, by anybody. The important economic parameter remains, however, the overall budget level of space activities actually implied in the 1980's and this component was changed dramatically over the range of interest in the scenario approach. (see Chapter 8).

Scenario 3, referred to in the May 1971 report as the MATHEMATICA baseline, was derived from the NASA-DoD baseline by reducing all identified OSSA costs by 50 percent. The reason for choosing this new baseline was that the average annual budget requirement of $1,750 million resulting for the OSSA under the NASA-DoD baseline for current expendable costs is four times the OSSA's average 1963-1971 budget ($450M), and two and one-half times the guideline of $750M set by the Bureau of the Budget for the First Interim Report.

For Scenario 3 and all other scenarios, MATHEMATICA did not request that the Aerospace Corporation attempt to perform new mission and traffic capture analyses based on the deletion of specific flights.

RELATIVE PAYLOAD COST DISTRIBUTION SHIFTS OVER TIME WITHIN AGENCIES

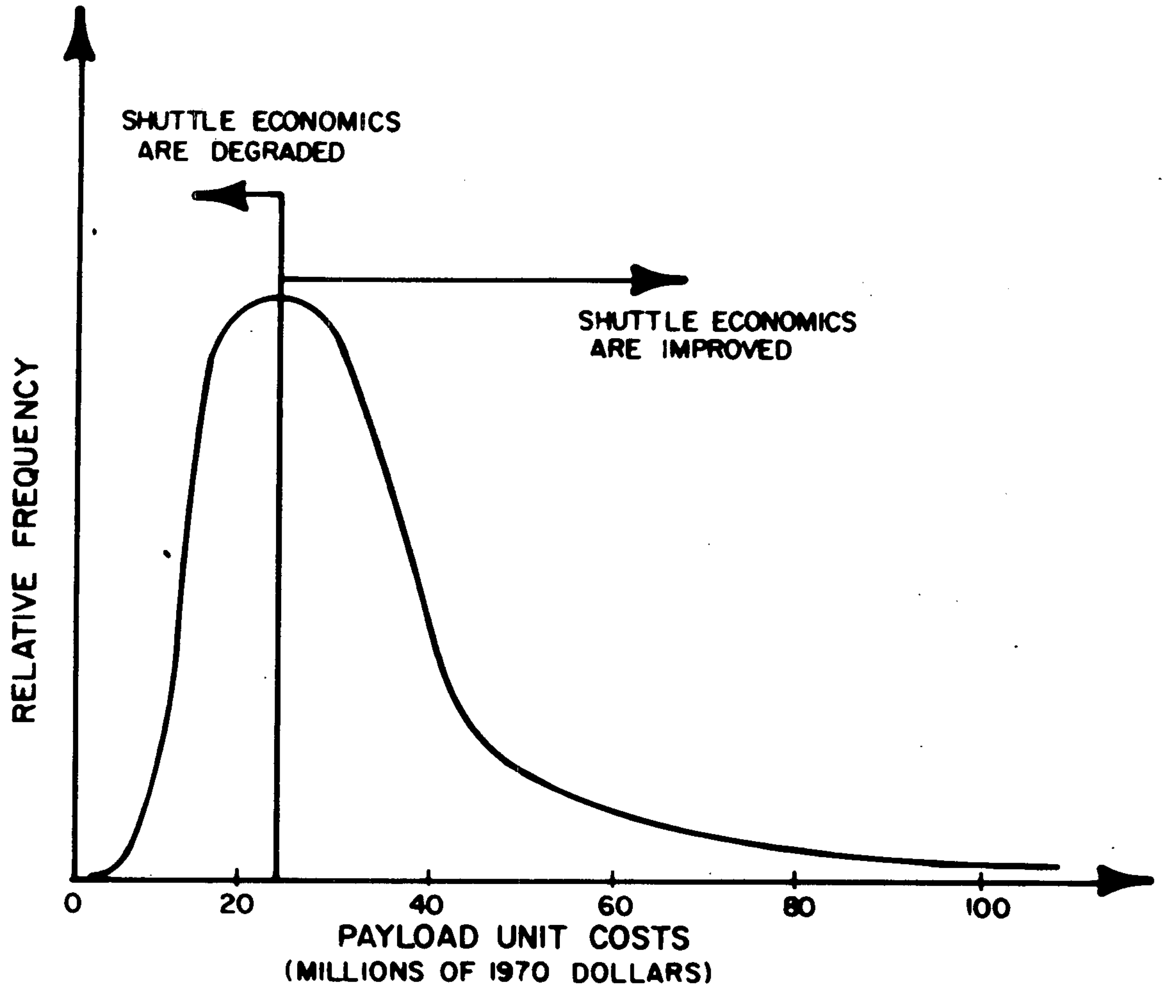

Figure 6.4

Table 6.23

Construction of Scenarios From NASA-DoD Baseline Sample Shuttle Payloads OSSA

Range of Payload Unit Costs (Millions of 1970 Dollars)	NASA-DoD Baseline		Scenario 2		Scenario 3		Scenario 4	
	# Payloads	% of Total	# Payloads	% of Total	# Payloads	% of Total		
0 - 19.9	162	36	120	36	81	36	--	--
20 - 39.9	181	40	135	40	70	40	--	--
40 - 59.9	62	13	47	13	31	13	--	--
60 - 79.9	13	3	9	3	7	3	--	--
80 - 99.9	22	5	17	5	11	5	--	--
100 - 119.9 120 - 139.9 140 - 159.9 160 and over	12	2	9	2	6	2	--	--

Rather, the 50 percent decrease in OSSA costs for Scenario 3, for example, was obtained for the STS by halving the activity level dependent (incremental) costs for OSSA, with the simplifying assumption that the number of OSSA flights for each year was also halved. This is illustrated by Table 6.24. Additionally, the RDT&E costs for OSSA payload development were also halved.

As is demonstrated in Table 6.24, the basic assumption made in developing each scenario is the following: a percentage variation away from the "baseline" costs for any mission category (NASA-OSSA, DoD, non-NASA applications) entails the same percentage variation in the number of flights for that category, and the relative frequency of mission costs within the category remains unchanged.

The nature of this underlying assumption for cost mission activity variations can be illustrated adequately by considering Scenario 3, in which only OSSA flights have been varied from the baseline, Scenario 1. For Scenario 1, the payload costs of OSSA flights for the Space Shuttle and Current Expendable systems have the distributions shown in Table 6.24 below.

Scenario 3 is derived by assuming in effect that half as many OSSA payloads are flown from each cost group indicated in Table 6.24 for the Space Shuttle and the NE and CE systems; the frequency of payloads from each cost interval, indicated by the "percentage of total" column remains unchanged. Thus, for Scenario 3 of the May 1971 Report 81 OSSA payloads in the $0-19.9 million range are to be flown by the Space Shuttle. The reduced numbers are of course approximate, representing the expected number of payloads in each cost interval if the "constant frequency" rule were imposed literally.

A summary of the nineteen scenarios performed for the May 1971 Report is presented in Table 6.24. The scenarios performed for this current report are presented in Chapter 8. Essentially, they include the

TABLE 6.24: SCENARIOS OF FUTURE SPACE ACTIVITY BASED UPON THE 736 SHUTTLE FLIGHT MISSION MODEL

Scenario	Description	Total Shuttle Flights 1978 - 1990
1	NASA-DoD Baseline Model	736
2	OSSA flights reduced to 75%	666
3	OSSA flights reduced to 50% -- MATHEMATICA BASELINE	600
4	Mathematica Baseline with DoD flights increased 50%	753
5	Mathematica Baseline with DoD flights doubled	912
6	Mathematica Baseline with DoD flights reduced to 75%	516
7	Mathematica Baseline with non-NASA applications increased 50%	645
8	Mathematica Baseline with non-NASA applications doubled	697
9	Mathematica Baseline with non-NASA applications tripled	794
10	Mathematica Baseline with the Shuttle phased in over the period 1978-1979	585
11	Mathematica Baseline with the Shuttle phased in over the period 1978-1981	547
12	Mathematica Baseline with the Shuttle phased in over the period 1978-1983	494
13	The Mathematica Baseline flight rate is reached in 1980	584
14	The Bathematica Baseline flight rate is reached in 1982	561
15	The Mathematica Baseline flight rate is reached in 1984	537
23	Flights based on average FY1963-FY1971 payload and launch vehicle funding of NASA (unmanned), DoD, and non-NASA applications flights	678
24	Similar to 23, but with flights based on FY1970-FY1971 average funding levels	562
25	NASA-DoD Baseline Model plus Lunar Option 1	1221
26	Mathematica Baseline plus Lunar Option	1085

baseline case, now 624 Shuttle flights, and scenarios 2 through 9 of the May 1971 report. As shown in the above-mentioned table, the activity level of the NASA-DoD mission model is 736 Shuttle flights over the period 1978-1990. The activity level of the MATHEMATICA Baseline was 598 Shuttle flights, and over the seventeen scenarios, the variation around this activity ranged from 494 to 910 flights. While in Table 6.24 only the number of Shuttle launches for each scenario has been indicated, the variation in the number of Current and New Expendable STS flights was incorporated into the economic models.

In the case of Scenarios 2 through 9, the number of Space Shuttle flights and the corresponding budget requirements for each agency were determined by adjusting the NASA-DoD Baseline mission and budget models by the factors given in Table 6.24. These factors have been applied uniformly to all activity level dependent (incremental) launch vehicle costs associated with each agency and to all payload costs, i.e., RDT&E, Investment and Operations. Estimated launch vehicle activity level independent operations costs were not changed.

Based upon the new MATHEMATICA Baseline (Scenario 3), Scenarios 10, 11 and 12 were formulated by adjusting the activity level dependent costs for the Space Shuttle to simulate its phasing in over two years (1978-1979), four years (1978-1981), and six years (1978-1983). The assumption was made that total space activity in each year is unchanged, and only the mix between Space Shuttle and Current Expendable flights varies with the Shuttle capturing an increasing proportion of the traffic over the course of the phase-in period.

Scenarios 13 through 15 simulated alternate patterns of the rate of space traffic build-up for NASA and non-NASA Applications. The mission and traffic model for the DoD, however, is unaltered.

Of several subsequent analyses, Scenarios 23 and 24 were of particular interest. For Scenario 23, the historical yearly average funding for payloads and launch vehicles in the four categories (NASA-OSSA, NASA-OMSF, DoD, and non-NASA Applications) for the Fiscal Year 1963-1971 period were used as the projected yearly funding level of the Current Expendable system; under the equal capability analysis, total Space Shuttle funding levels are somewhat lower. For the OSSA and non-NASA categories, RDT&E costs were uniformly reduced from the corresponding (Current Expendable) NASA-DoD baseline costs (Scenario 1) by a factor approximately equal to the total percentage reduction for all costs in each category; Activity Level Dependent (ALD) costs were then computed for each year to give total costs by year equal to the desired average 1963-1971 cost being imposed. For OMSF and the DoD, RDT&E costs were not changed; but for DoD, a rather large increase in ALD costs at the end of the program was required to offset the truncation of the RDT&E costs in the NASA-DoD baseline. Having used the Current Expendable costs to thus adjust program costs in Scenario 1 to agree with the historic average funding, identical adjustments were made in each category to determine New Expendable and Space Shuttle activity levels and costs.

Scenario 24 was developed in identical fashion, but based on the lower Fiscal Year 1970-1971 average funding levels for each category.

Scenario 25 is the NASA-DoD baseline (Scenario 1) plus Lunar Option 1. For Lunar Option 1, only launch cost data were available; however, the lack of payload data did not affect the economic analysis since these payloads are assumed to be identical for each Space Transportation System. Scenario 26 is the MATHEMATICA baseline (Scenario 3) plus Lunar Option 1.

The major result of the scenario analyses performed in the May 1971 Report is that the exact composition of the mission model is of secondary importance to the scale of the mission model; i. e., the activity level. As was shown in the Summary (Chapter O) of the Report, 99 percent of the variation in the Allowable Non-Recurring Costs evaluated at the 10 percent discount rate is associated with changes in the level of activity over the 1978-1990 period. It is on the strength of this result that for the analyses of the two-stage fully reusable and alternative shuttle configurations within this report a reduced number of scenarios is used. A broad range of activity has been covered and results appear within Chapter 8.

6.4.2 Non-Recurring Costs

Although the problem of non-recurring cost uncertainties was addressed by MATHEMATICA, it proved impossible to reach explicit quantitative conclusions. Two parametric models for predicting cost growth were considered, but for reasons to be discussed, were not useful. Qualitative observations of some significance can, however, be made.

Consider the evolution of a large, technologically difficult, design and development program. Initially, conception of the end product is by necessity incomplete and ill-defined. This is followed by rapid evolution of the concept normally accompanied by large changes in cost estimates as the designers become more aware of the problems and, perhaps more important, better able to define the objective. The uncertainties inherent in this phase of a program are thus very large and probably should not be subject to too much quantitative scrutiny. With time, the rate of increase in knowledge slows down and development becomes primarily the refinement of a conception. Most knowns have been quantified and educated guesses about known unknowns have been made.

Somewhat later in the development process, slippages may occur. Any large development is comprised of many smaller interdependent development efforts, so that a delay in one can mean a delay in many and even in the entire project. Deviations from schedule will lead to a higher probability of increasing time or costs than of decreasing them. This will hold even if there is as much chance of finishing a subprogram ahead of schedule as behind schedule. Since communication is imperfect, the manager of any part of the development will not always know the status of all the other subprograms that are important to him. If someone else finishes ahead of schedule, he may or may not be able to capitalize on it. However, if there is a delay in one of these other subprograms, it will always affect him since some part of his development may be delayed or some corrective action necessary.

The above would seem to indicate that if one were to plot cost estimates versus time, errors in the early stages would be due to flaws in the estimating technique; errors in the later stages would be due to poor management. Note that it will be difficult to distinguish the effect of good or bad estimating techniques from those of good or bad management.

Initially, it was hoped that some parametric means of predicting cost growth might be found. Although this proved impossible on the program level, it is commonly employed on the subsystem level where there is a clearer functional relationship between dollars and some independent variable like weight.

In references [21] and [22], examples of attempts to parametrically quantify the evolution of program cost estimates, were considered as to their applicability to the Space Shuttle program. Reference [22] employs an empirical least-squares fit of an exponential form to estimate cost data, in which the time to initial operational capability was used as the independent variable; Reference [21], in a somewhat more complicated approach, uses length of the development program, fraction of the program so-far elapsed, technological complexity, and calendar year of the estimate. The results of applying these methodologies to the Space Shuttle program are contradictory, with the result that one can prove whatever one wants by the adroit choice of very plausible assumptions.

For example, in Reference [21], Summers postulates that cost estimating techniques have been improving with time. (Note that this would be functionally equivalent to assuming management techniques have been improving with time). Direct application of his model to the Phase B Baseline, fully reusable Space Shuttle, will predict a significant cost underrun. However, Summers' assumption of an exponential improvement in cost estimating implies that estimates made today will be twice as good as those made when Summers first published his work in 1962. It is possible to modify this assumption without violating the fit of the data by assuming some different functional form for later years. But this will lead to predicting overruns whose magnitude is a function of the assumption.

Reference [22] suggests that cost histories of the 1960's show an improvement over those of the 1950's. Although it is quite probably true that costs are better understood and controlled today than in the past, it is not clear that Reference [22] supports this thesis. Examination of the data suggests that the difference between the fifties and sixties data sets may not be time. It would appear that the difference could also be the degree of technical difficulty relative to the state of the art of the day.

Consideration of price indices for various fabrication techniques, materials and subsystems shows that cost changes have not been the same for all categories. As the programs used in the data bases for References [21] and [22] will involve different mixes of these components, the use of the models defined to predict cost growth becomes more questionable.

The problem would appear to be one of finding a clear functional (not correlative) relationship between cost and some independent variable(s). Thus, it is difficult to predict the overall costs of a program by analogy to the aggregate costs of historical systems since with a new development they will probably be more dissimilar than similar. One can expect to be wrong and, if right, it will probably be for the wrong reasons.

Let us consider what can be said about non-recurring costs on a more specific level. There are in general use several methods for predicting subsystem costs, the two most common being the so-called "bottoms up" approach and the use of parametric cost estimating relationships (CER's). The first consists of adding up the costs of every component and every hour spent. Its primary flaw is that some items tend to be left out, especially when a system is only partially designed. The second also suffers from uncertainty. Cost estimating relationships are developed by plotting historical costs versus some independent variable like weight or thrust. The data used to develop these relationships will include some errors and will not perfectly reflect the costs incurred. There is a second, more subtle, area of uncertainty that involves the degree of analog between historical data and the system in question. In a development program there will be much that differs from past experience. The degree of comparability between the shuttle and past programs will be good in some areas (e. g. engines) and poor in other (e. g. thermal protection systems). The degree of confidence in an estimate will depend significantly on this comparability.

Inflation has not yet been discussed. Since it is fairly difficult to predict it, most estimates are made in constant dollars which will generally be satisfactory on a program level, but can lead to difficulties on the subsystem level. For example, in developing its cost estimating relationships, Aerospace Corporation used price index data for various component parts. Historical data was related to 1969 dollars on a more specific level than the inflation rate of the entire economy. Component costs have been inflating (or deflating as in the case of titanium fabrication) at different rates, the effects of which were included. It then becomes clear that today's constant dollars cannot necessarily be related to future buying power by a single factor, but rather by a spectrum of factors whose aggregate impact may be larger or smaller than the inflation rate of the entire economy.

What is the likelihood that development of the Space Shuttle will cost significantly more than anticipated? This question cannot be satisfactorily answered by going to historical data for precedents that can be parametrized and applied. However, consider that where a program maintains flexibility and where concurrency has been minimized, there will be less chance of cost growth since slippages in one part will be less catastrophic to the whole. Where the design and state of the art are not far apart, uncertainties will be less.

The current conception of a Space Transportation System is significantly "safer" than the original fully reuseable Phase B Baseline. Performance slippages in the engines can, in part, be made up by increasing the size of the fuel tanks. Whereas this could lead to very costly orbiter redesign were the tanks internal, the external drop tank can be modified with relative ease. Risks in the development of reuseable pressure fed boosters are not critical to the overall program since solid rocket motors are satisfactory. In fact it is not presently clear that solids would be worse than pressure feds. This is the old trade-off between high operations and low development cost and low operations and high development cost. Finally, consider that NASA's development of the Space Shuttle is a different "game" than that of a military development project.

One of the many reasons for cost overruns has been changes in the engineering design once development is well underway. Such design changes have been responsible for some of the more spectacular cost increases. Although one might argue that this cost growth is not an overrun as such, the fact remains that costs have increased and we, in attempting to evaluate the Space Shuttle, cannot ignore this situation. As non-recurring cost growth of this sort has generally been positively correlated with higher operations cost, this becomes especially significant. (Note that this is to be distinguished from the situation portrayed by the trade off graphs in Chapters 2 and 8. There, higher non-recurring costs imply lower recurring costs and vice versa. This, however, compares <u>different</u> configurations, not different evolutions of the same conception.)

By far the great majority of high technology projects conceived in this country have been in the province of the military. Military planners have a more difficult task than we do in defining a "pay-off" function. The problem of quantifying the "value" of a life saved or an additional enemy killed is more difficult than that faced by the designer of a transportation system where benefits may be measured by dollars saved. Thus there is a heavy bias towards increased capability which in turn implies higher complexity and costs. Because of this we would argue that a Space Transportation System, if developed with an eye to economic considerations, will be less prone to incorporate new technology into its design.

A Space Shuttle has a clearly defined function: to deliver a payload into earth orbit. There is no "opponent" set to degrade its capabilities. Certainly there will be an evolution from today's conception, but the drivers should be towards performance of a specific mission at minimum cost. In areas of rapid technological evolution one can expect component costs to go down since one will be able to do the same thing cheaper. In a military development there is a strong bias towards the use of any new technology for increased capability (usually at higher cost), but the shuttle design (presuming rationality!) should evolve towards lower cost for the same mission. We again stress that this hypothesis assumes that the project is managed with an eye to maximizing dollar quantifiable benefits -- as would be the case for a private company competing for profits.

6.4.3 Uncertainty in Recurring Costs

To a large extent uncertainties in technology, schedule, and missions flown can be reflected as cost uncertainties. Since the Space Transportation Systems under evaluation utilize different technologies and are in various stages of planning, research, design and development, different levels of uncertainty and hence risk exist. The comparison and selection of the alternative Space Transportation System should consider the different levels of risk which can be described as the possible variability of net present value. Because of the uncertainties which exist, net present value will not be single valued but probablistic in nature and must therefore be represented by a probability distribution characterized possibly by an expected value and standard deviation. Even if development and operating costs could be predicted with certainty, net present value would still be probabilistic. This results from the fact that it is not possible to achieve systems having perfect (unity) reliability. Boosters, orbiters, payloads, et cetera, will undoubtedly achieve reliabilities near, but less than, unity. A reliability less than unity introduces uncertainty and hence risk. Reliability considerations will affect fleet size (boosters and orbiters), number of payloads, number of launch attempts, number of refurbishments, et cetera.

Reference [1] describes the various areas of uncertainty which affect costs and which make it impossible to consider costs as being well defined, single valued functions. Costs must be described as illustrated in Figure 6.5 where annual costs are shown as ranges of possible values with different probabilities of falling into different parts of the range. The probabilistic recurring costs are the result of (a) uncertainties associated with predicting mission requirements, booster cost, orbiter cost, payload cost, et cetera, (b) less than unity reliability of launch success, orbiter payload injection, orbiter recovery, et cetera, and (c) payload failure (Mean-Time-Before-Failure, MTBF) characteristics. Since net present value is the result of considering a time dependent stream of probabilistic costs, it must also be categorized by a probability distribution. The probability distribution represents the chance of achieving each of the possible levels of net present value. The probability of net present value exceeding a specified level may be determined by obtaining the area under the probability distribution curve for all values greater than the specified level. This is normally referred to as the cumulative probability distribution (henceforth referred to as a <u>risk profile</u>). A typical risk profile of net present value is shown in Figure 6.6 where the vertical scale represents the probability or chance, p, of exceeding the various levels of net present value, NPV, indicated by the horizontal scale. In general, the steeper the curve the lower the risk (or variability). When comparing alternatives it is important to compare the expected or most likely net present values. It is equally or perhaps more important to also compare risk levels. In the certainty situation, it is generally desirable to select the alternative which yields the minimum net present value of costs when all alternatives are evaluated on an equal capability basis. The selection process becomes more difficult when uncertainties are considered; tradeoffs must be made between alternatives possessing different expected net present values

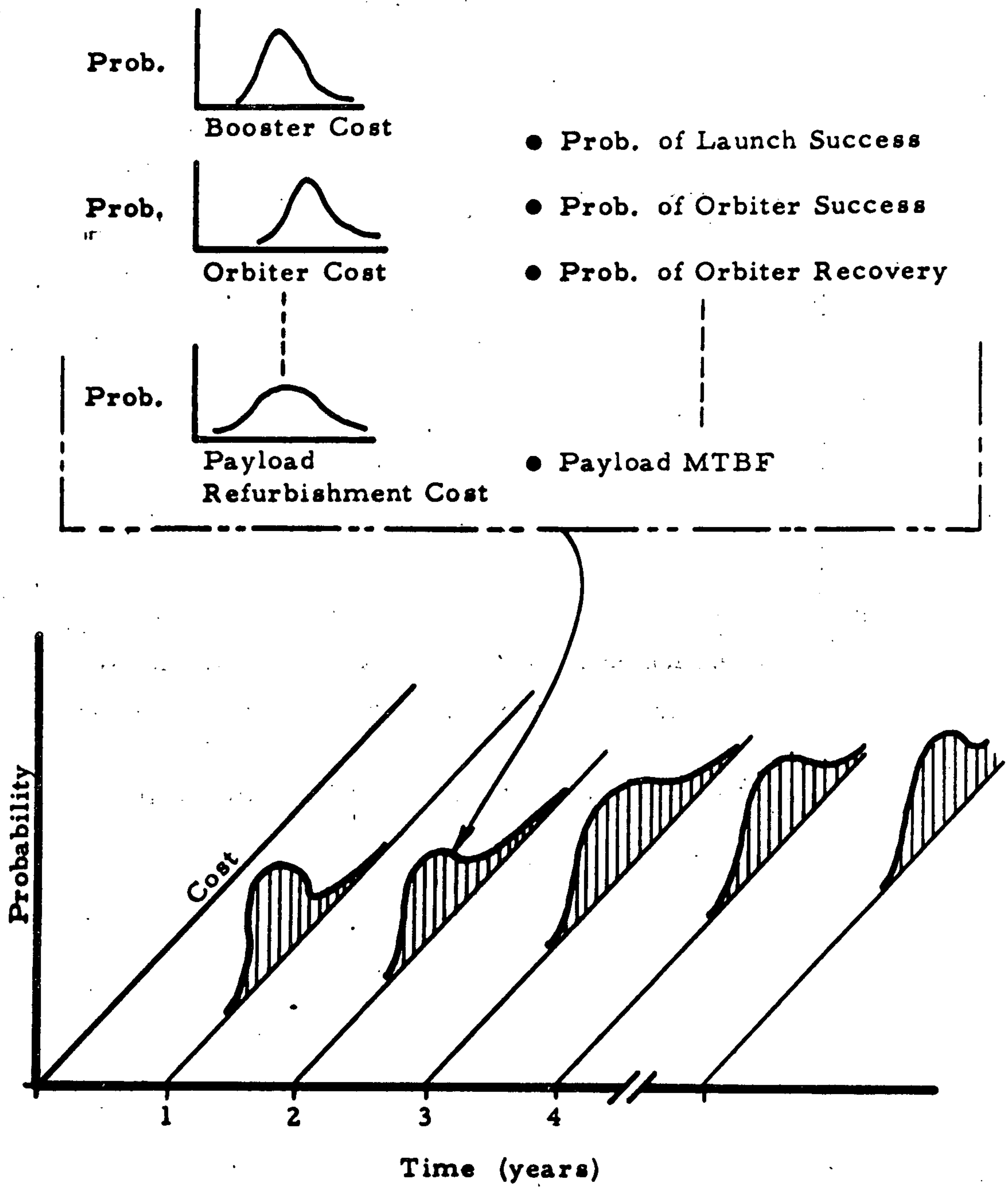

Figure 6.5: Probability Density Functions for a Stream of Uncertain Costs

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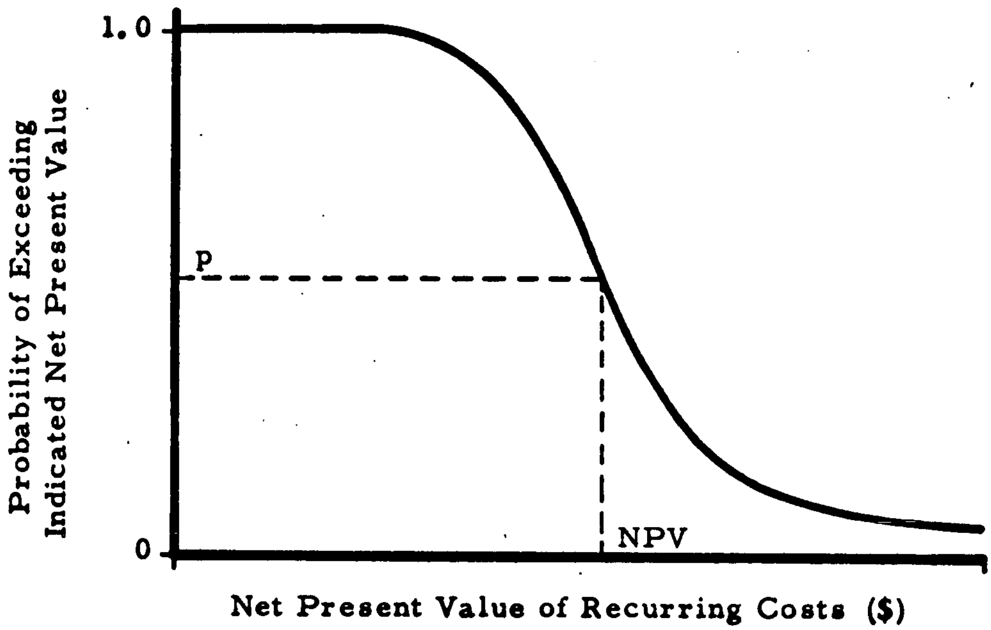

Figure 6. 6: Risk Profile of Net Present Value

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and associated levels of risk. This is discussed in more detail in Section 6.4.3.7.

To date, primary concern has been with trying to establish reasonable cost levels [2, 19, 24]. For example, the Aerospace Corporation [19] has developed a cost estimating methodology based on the concept that the cost of the shuttle and its parts can be shown to be a function of one or more design, performance, or program paramenters in suitable equation form (the CER's--cost estimating relationships). The credibility and the accuracy of the overall system estimate is a function of the number and type of individual estimates, the quality of the data on which the individual estimates are based, and the correlations of cost to the independent variables. The cost estimating relationships are single valued cost functions which provide little or no indication of the variability or uncertainty involved in establishing the relationships. Thus, the use of CER's leads to a single valued STS cost estimate with no insight into the associated risk.

Since reliability and cost uncertainties can significantly affect risk, a methodology has been developed for explicitly considering and evaluating their effects. In order to demonstrate the importance of considering reliability and cost uncertainties a typical mission was postulated and resultant recurring costs evaluated for both a typical expendable launch vehicle system and a typical space shuttle system consisting of an expendable booster and recoverable orbiter. The following paragraphs describe the methodology. Typical results are presented and should be considered only as a demonstration of the type of analysis which should be part of future evaluation efforts. The analysis is motivated by the shortcoming of any formal consideration of uncertainties and reliability effects up until now, by the expectation that Space Shuttle operation costs will vary much more widely than has been allowed for over the 1978-1990 period and by the ready applicability of simulation and risk analysis techniques to this situation.

6.4.3.1 General Procedure for Evaluating Effects of Reliability and Cost Uncertainties

The basic starting point of the procedure is to face up to the fact, from the very beginning, that reliabilities will not be equal to 1.0 and that significant cost uncertainties exist. With this in mind the general evaluation procedure shown in Figure 6.7 was established. The procedure consists of establishing a mission simulation model and a mission cost model. The mission simulation model represents the many possible sequences of events which may take place in the process of performing the desired space mission. The space mission may be a single space flight or an interrelated group of space flights. The mission simulation model is concerned with establishing the number of events (for example, launch attempts) in terms of the various pertinent reliability factors.

The mission cost model establishes the mission recurring costs including replacement and refurbishment costs for boosters, orbiters and payloads. The mission cost model combines the results of the mission events with the appropriate cost per event. The costs are considered as uncertainty variables where ranges of possible values are specified as well as subjective estimates of the form of the uncertainty (the probability density function--pdf-(henceforth referred to as the uncertainty profile).

The simulation uses Monte-Carlo* techniques to establish the probability distributions (risk profiles) of the different events, their associated costs, and total mission cost. In order to demonstrate the technique a typical mission was postulated, i.e., the establishment and maintenance of a low altitude satellite system.

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*Monte-Carlo implies the repetition of a modeled experiment, sequence of events, physical process, etc. whose component outcomes are probabilistic, a sufficient number of times to generate a "smooth" profile or histogram of all possible outcomes. This resulting profile of predicted outcomes for the model is then normalized to a relative frequency profile which represents the probability density function for the experiment's outcome.

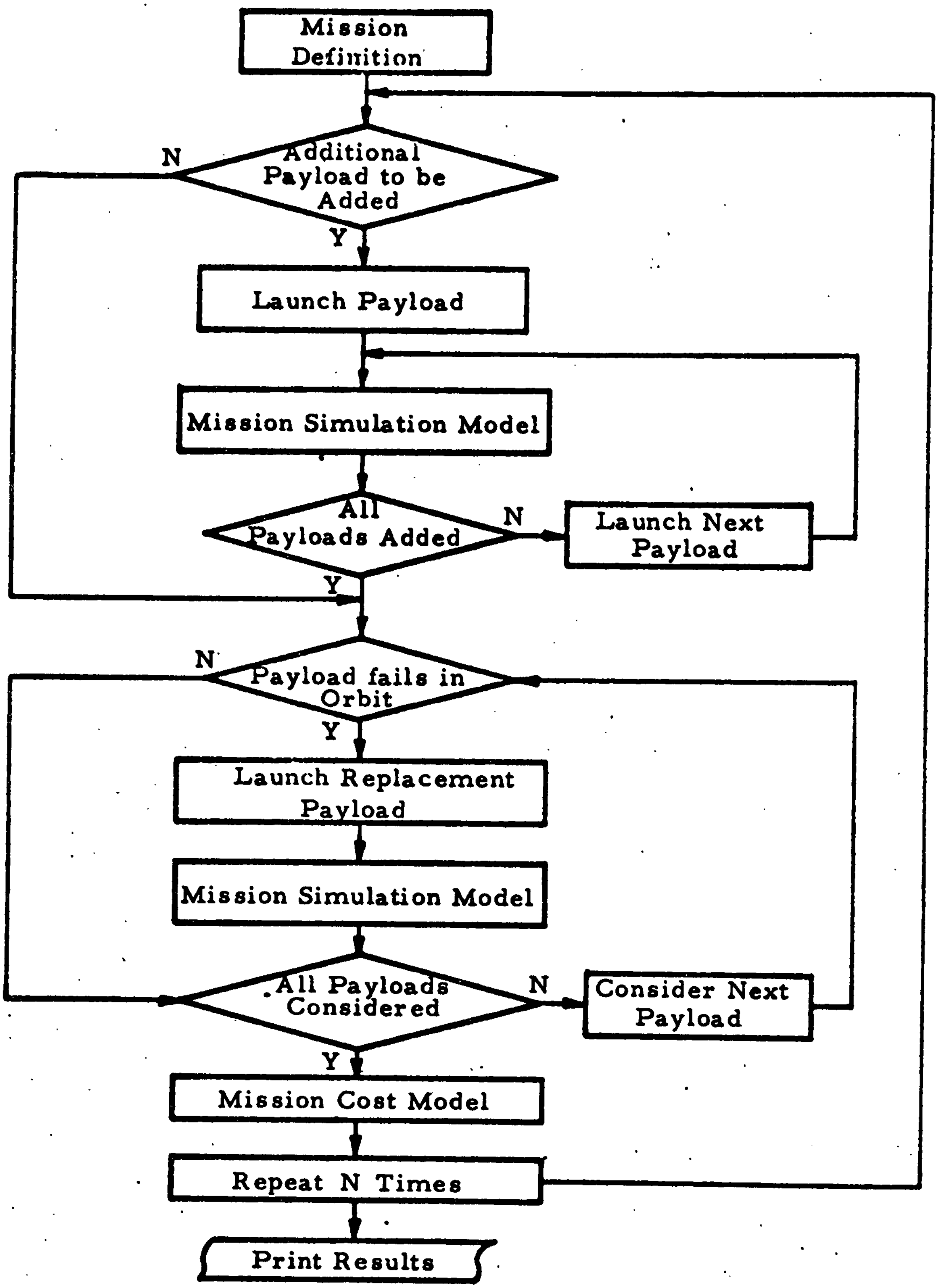

Figure 6. 7: General Procedure for Evaluating Effect of Reliability and Cost Uncertainties

Mission Simulation Model

The mission simulation model considers a wide range of possible situations within the framework of the establishment and maintenance mission. In general, a two-stage launch vehicle is considered. Each stage may be reusable, recoverable, or expendable or consist of a reusable and expendable portion. Table 6.25 summarizes the various reliability aspects considered. It should be noted that the representative shuttle system is based upon a non-recoverable booster. Figure 6.8 depicts the various possible sequences of events which might occur in the process of performing the establishment and maintenance mission. Each of the diamond shaped boxes represents one of the reliability aspects considered. The particular path through the network is probabilistic and depends on the various reliability components. The simulation (experiment) is repeated a large number of times for each mission in order to establish a histogram of outcomes. The particular path taken through the network represents or simulates a particular sequence of events which might occur. For example, a booster might be launched successfully and thence recovered successfully. This necessitates a refurbishment of the booster. The orbiter might abort and not be recovered necessitating the acquisition of another orbiter and payload and requiring another launch attempt. Thus the path through the network determines the number of launch attempts, the number of additional boosters, orbiters and payloads required in order to satisfy the mission requirements. It also determines the number of booster, orbiter and payload refurbishments and the number of payloads which fail in orbit and must be replaced. The probability of a payload failing in orbit depends upon the payload mean-time-before-failure (MTBF).

The number of times through the simulation model depends on the number of payloads required by the mission definition and the number of payloads which fail and must be replaced. Upon placing the desired number of payloads into orbit the mission cost model is entered. At this point the computations represent a set of events and their associated costs which result from satisfying the mission requirements. This process is then

Table 6. 25 Typical Aspects Considered

Assumed Probability of:	Representative Shuttle System	Representative Expendable Launch System*
• Booster Launch Success	.99	.98
• Booster Recovery Given a Launch Success	.00 +	.00 ++
• Booster Recovery Given a Launch Failure	.00 +	.00 ++
• No Orbiter Abort	.99	.99
• Orbiter Recovery Given a Launch Failure	.95	.00 ++
• Orbiter Recovery Given an Orbiter Abort	.95	.00 ++
• Orbiter Injecting Payload into Desired Orbit Given No Orbiter Abort	.99	.98
• Orbiter Recovery Given an Otherwise Successful Mission	.99	.00 ++
• Orbiter Recovery Given That Payload Placed in Incorrect Orbit	.99	.00 ++
• Payload Operating Successfully when Initially Placed in Desired Orbit	.99	.97

* For Expendable Launch System, Orbiter Refers to Upper Stage

\+ Representative Shuttle System based upon a non-recoverable booster

++ Due to the expendable nature of the upper and lower stages

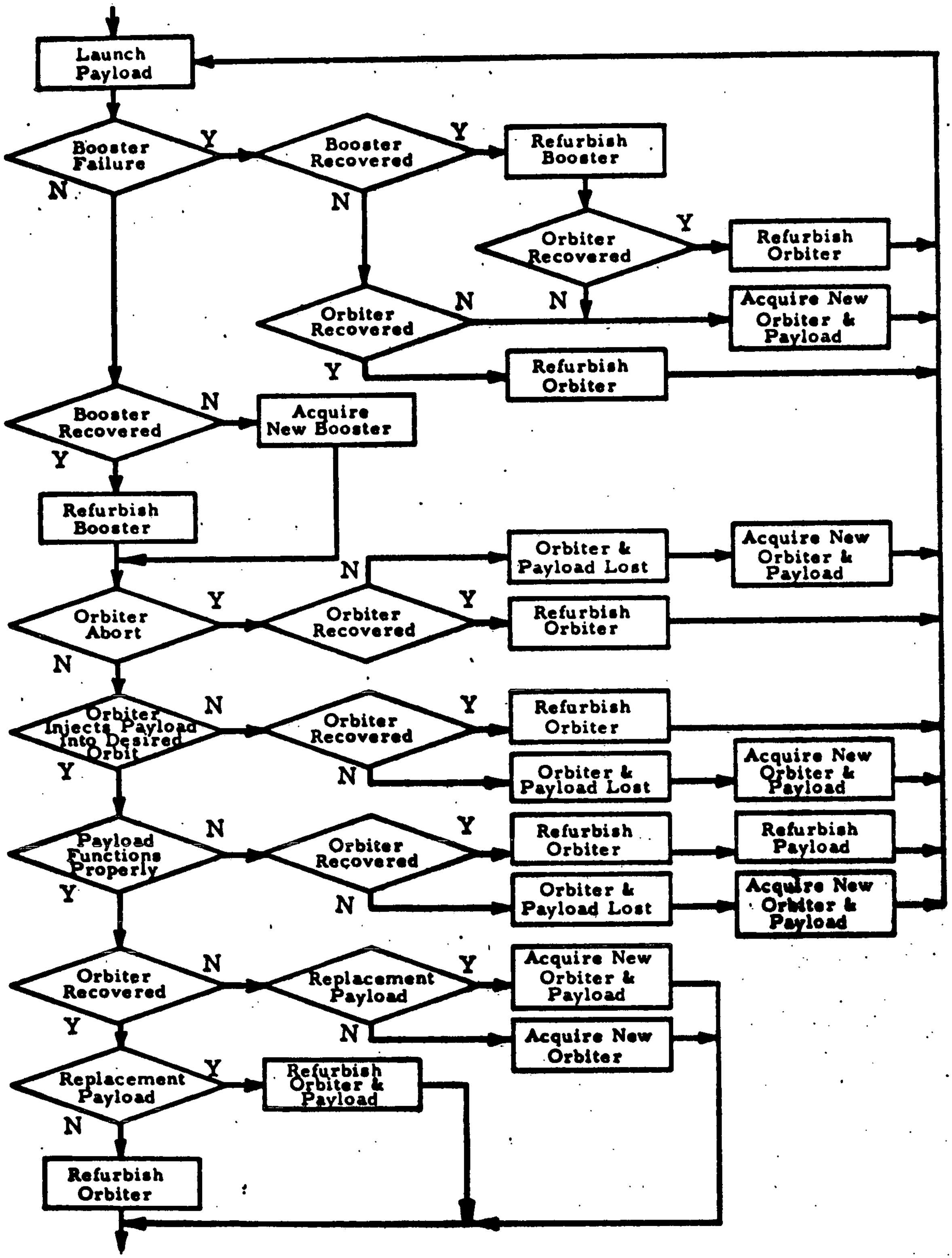

Figure 6.8 Mission Simulation Model—Establishment and Maintenance of a Low Altitude Satellite System

repeated a large number of times (1000 or more) so that a histogram of all possible outcomes can be established.

Mission Cost Model

The concept of the mission cost model is shown in Figure 6.9. The mission cost model utilizes the computed number of events and the uncertainty profiles of the appropriate costs to establish the total mission cost and the associated component costs. The cost model treats the following costs as uncertainty variables:

- Booster cost (reusable portion)
- Booster cost (expendable portion)
- Orbiter cost (reusable portion)
- Orbiter cost (expendable portion)
- Booster refurbishment cost
- Orbiter refurbishment cost
- Payload cost
- Payload refurbishment cost
- Operations cost per launch

Each of the uncertainty variables is characterized by a range of uncertainty and the probability distribution of cost within the range. A method for establishing the uncertainty profiles is discussed in Section 6.4.3.4. Monte-Carlo sampling techniques are used to establish values of the cost elements used in each of the 1000 or more sets of cost computations which result in the histograms of possible costs.

6.4.3.2 Effects of Reliability

The various reliability aspects considered are summarized in Table 6.25. In order to demonstrate the type of results to be expected when reliability is explicitly considered, reliability estimates have been made for "typical" expendable launch vehicle and Space Shuttle systems. These reliability estimates should be considered as being only representative. At the time of this writing little data is available to substantiate these or other estimates. The criticality of reliability will become read-

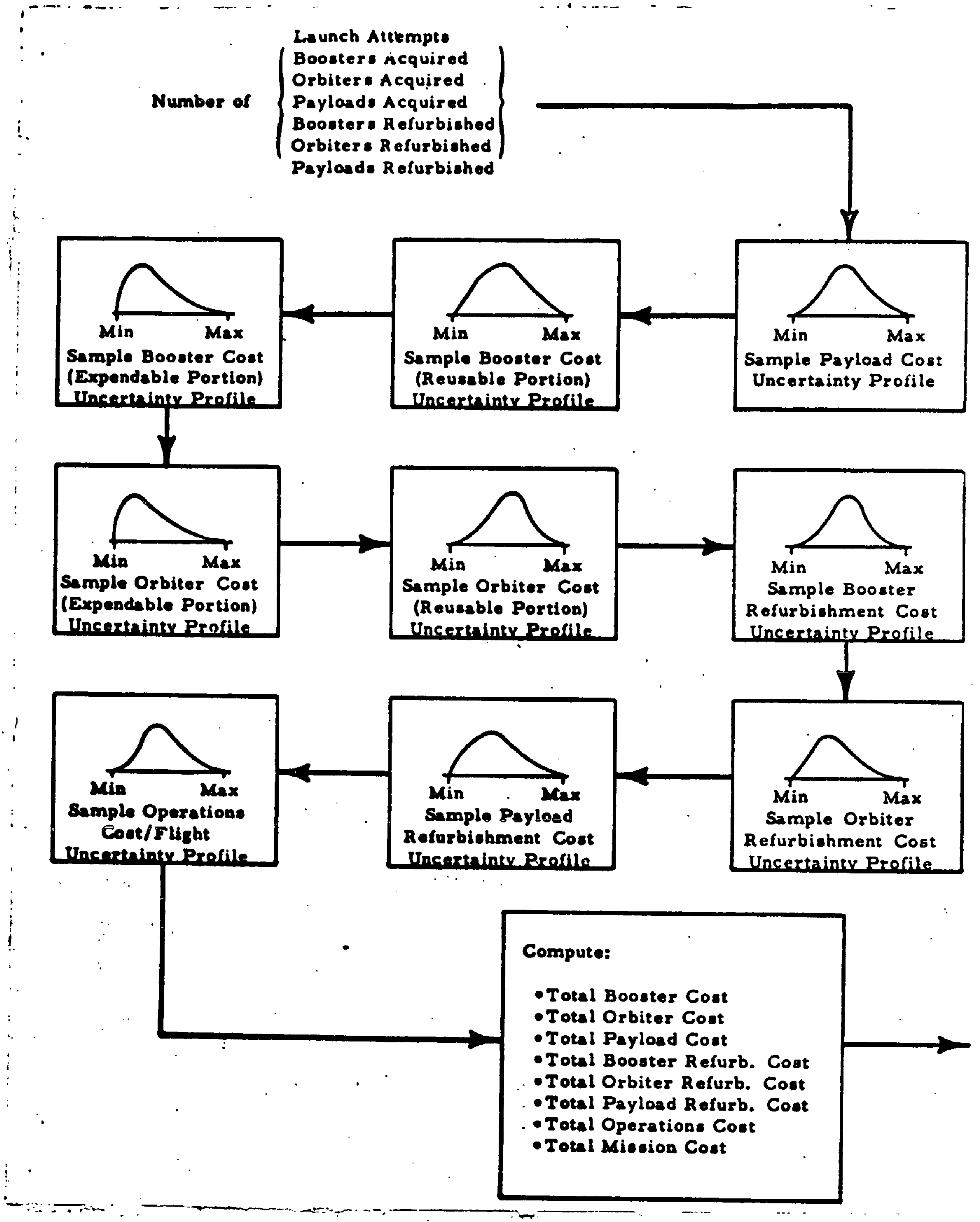

Figure 6.9 Mission Cost Model

ily apparent in the following paragraphs. The Space Shuttle reliability estimates are based on a system having an expendable booster and reusable orbiter (for example, Twin SRM Boosters, Parallel Burn Orbiter with External Single HO Tank) and the expendable launch vehicle system is characterized by the Titan III M.

The specific parameters assumed for the establishment and replacement mission are illustrated in Table 6.26. A one-year snapshot in time is considered during which new payloads are added to the system and payloads which fail are replaced. The snapshot in time is considered to start with approximately the fiftieth launching.

Figure 6.10 illustrates the effect of reliability on the number of launch attempts for a typical expendable launch vehicle system and the effect of payload mean-time-before failure on the number of payloads which fail and have to be replaced. For example, there is a 25 percent chance that more than three payloads will fail and about a three percent chance that more than five payloads will fail. The number of launch attempts (a minimum of five for the assumed mission) is a function of launch vehicle reliability and number of payload failures. There is approximately a 40 percent chance that more than eight launch attempts and a ten percent chance that more than ten launch attempts would be required to establish and maintain the postulated system of payloads.

Similar data have been established for the typical Space Shuttle system. Figure 6.11 illustrates the probability density functions (pdf) of a number of important events. For example, referring to the first column, there is a small chance (four percent) of having only five launch attempts and a large chance (24 percent) of having eight launch attempts. On the other hand, because of the ability of the shuttle system to reclaim payloads which fail, there is a 97 percent chance that only six new payloads (five required by the mission specification plus one standby spare) will be required and a three percent chance that seven payloads will be required. These data are illustrated in the risk profile format in Figures 6.12 and 6.13. The effect of reliability on risk is clearly evident.

Table 6.26	Typical Mission Parameters
o Time Period Considered	1 year
o Number of Satellites Operating at Start of Time Period	10
o Number of Additional Satellites Added During Time Period	5
o Replacement of Satellites Which Fail is Required	As req'd
o Satellite Life (MTBF)	5 years

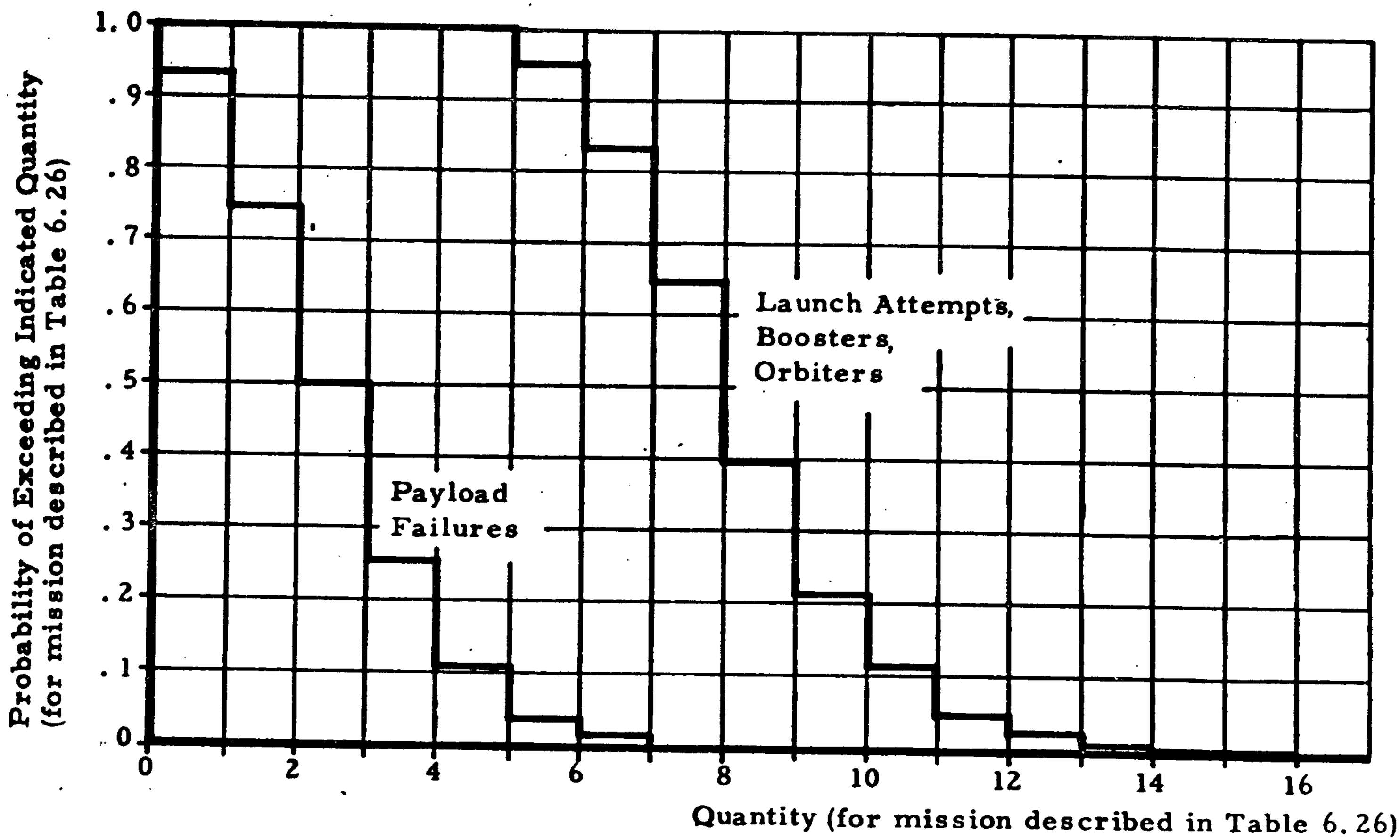

Figure 6.10: Probability of Exceeding Indicated Number of Launch Attempts, Boosters, Orbiters and Payload Failures for a Typical Expendable Launch Vehicle System to Meet Mission Requirements Specified in Table 6.26

ESTABLISHMENT & MAINTENANCE
OF A SYSTEM OF LOW ALTITUDE SATELLITES

PROBABILITY OF INDICATED QUANTITY

QUANTITY	LAUNCH ATTEMPTS	ADDIT. BOOSTERS REQUIRED	ADDIT. ORBITERS REQUIRED	ADDIT. PAYLOADS REQUIRED	BOOSTERS REFURB.	ORBITERS REFURB.	PAYLOADS REFURB.	PAYLOAD FAILURES
14	0.0020	0.0020	0.0	0.0	0.0	0.0010	0.0	0.0
13	0.0060	0.0060	0.0	0.0	0.0	0.0070	0.0	0.0
12	0.0170	0.0170	0.0	0.0	0.0	0.0170	0.0	0.0
11	0.0390	0.0390	0.0	0.0	0.0	0.0380	0.0	0.0
10	0.0980	0.0980	0.0	0.0	0.0	0.0830	0.0	0.0
9	0.1760	0.1760	0.0	0.0	0.0	0.1770	0.0	0.0
8	0.2430	0.2430	0.0	0.0	0.0	0.2370	0.0010	0.0010
7	0.2220	0.2220	0.0	0.0280	0.0	0.2270	0.0090	0.0060
6	0.1530	0.1530	0.0	0.9720	0.0	0.1610	0.0270	0.0230
5	0.0440	0.0440	0.0	0.0	0.0	0.0500	0.0800	0.0700
4	0.0	0.0	0.0	0.0	0.0	0.0020	0.1650	0.1630
3	0.0	0.0	0.0	0.0	0.0	0.0	0.2400	0.2500
2	0.0	0.0	0.0010	0.0	0.0	0.0	0.2510	0.2610
1	0.0	0.0	0.0780	0.0	0.0	0.0	0.1750	0.1710
0	0.0	0.0	0.9210	0.0	1.0000	0.0	0.0520	0.0550
EXPECTED NUMBER	7.94	7.94	0.08	6.03	0.0	7.86	2.69	2.63
STD. DEV.	1.62	1.62	0.27	0.16	0.0	1.63	1.49	1.44

Figure 6.11 Probability Distributions for Typical Shuttle System for a Low Altitude Satellite Mission as Described in Table 6.26

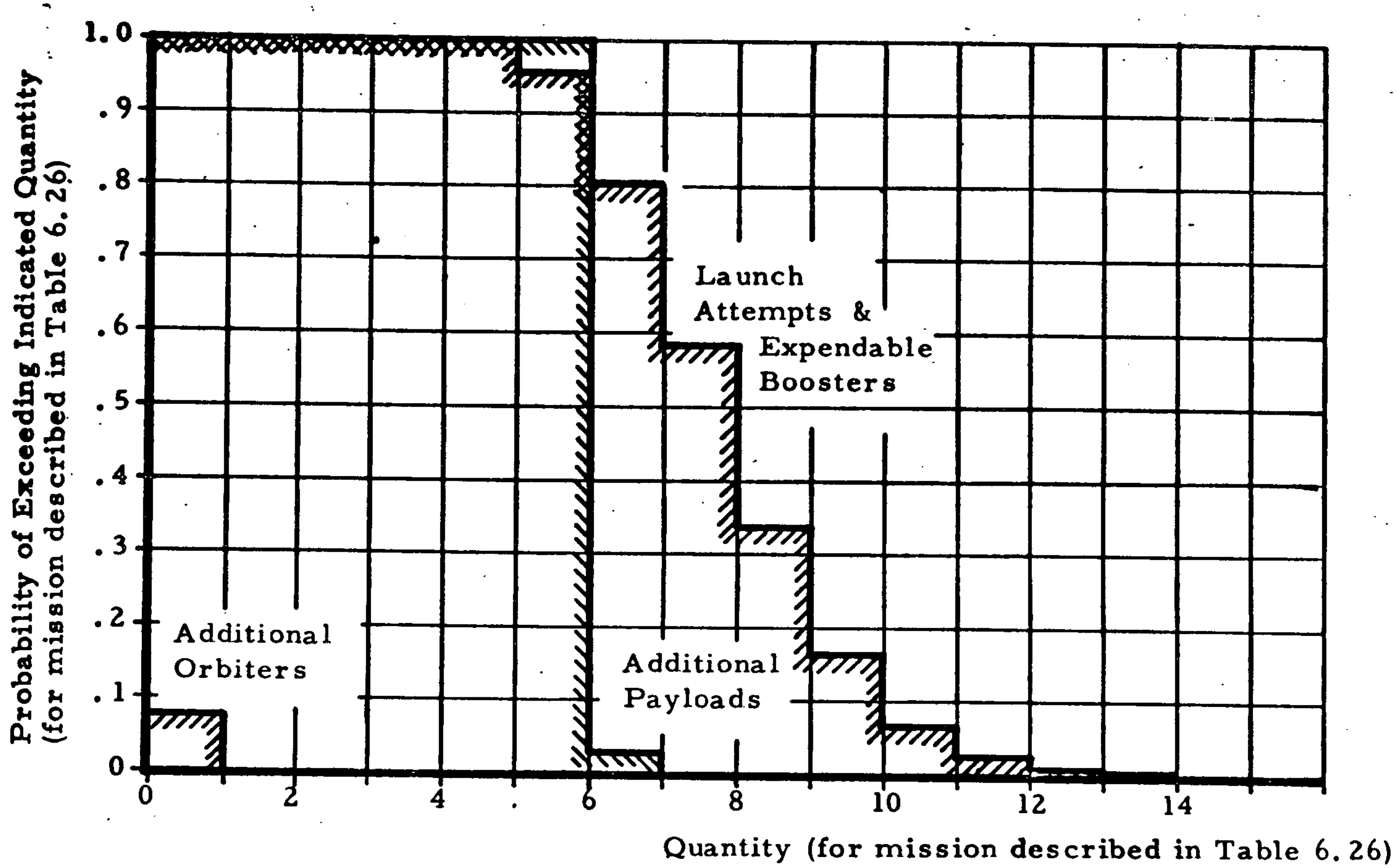

Figure 6.12 Probability of Exceeding Indicated Number of Launch Attempts, Boosters, Orbiters and Payloads for a Typical Shuttle System to Meet Mission Requirements Specified in Table 6.26

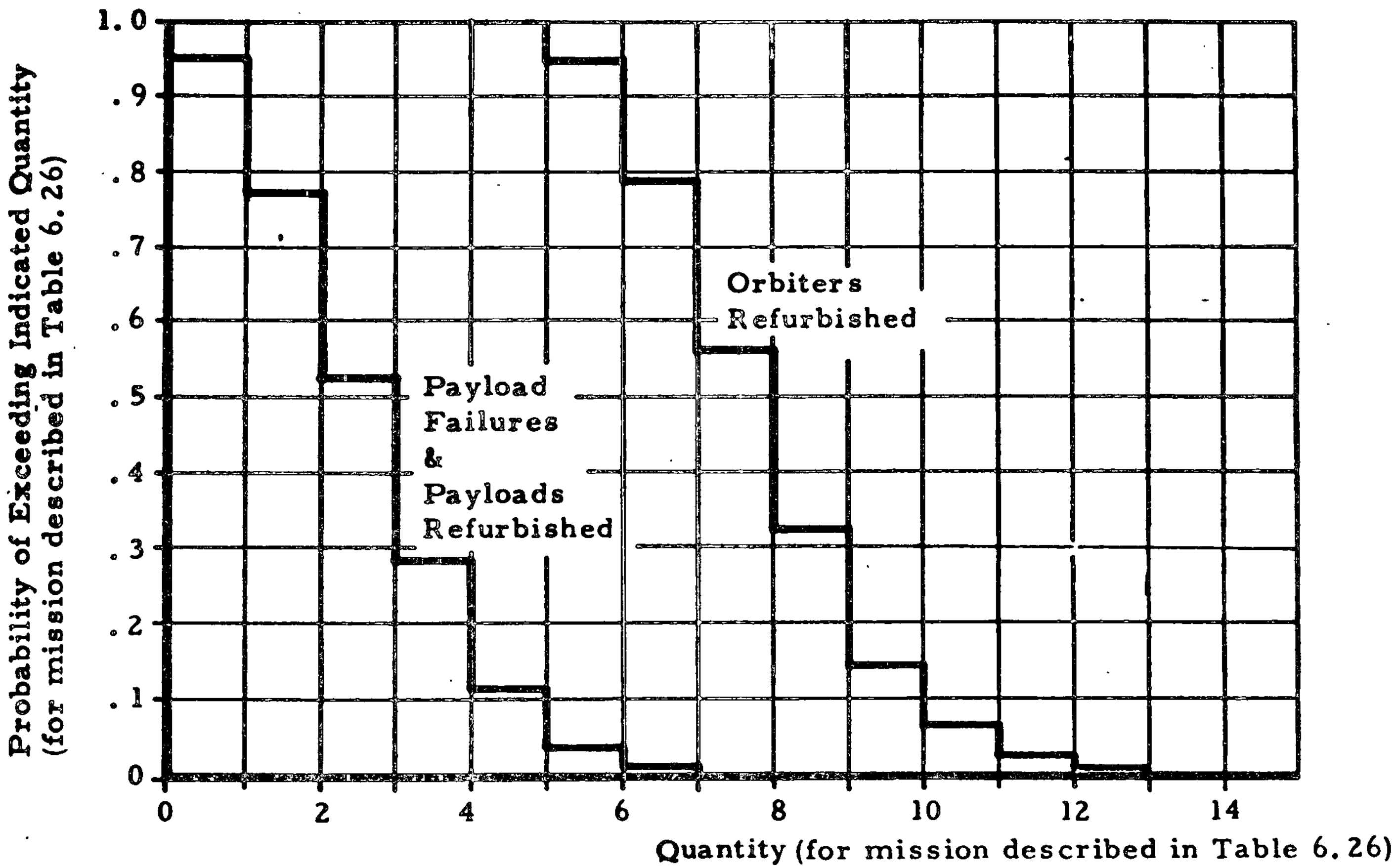

Figure 6.13 Probability of Exceeding Indicated Number of Payload Failures and Orbiter and Payload Refurbishments for a Typical Shuttle System to Meet Mission Requirements Specified in Table 6.26

The results of a sensitivity analysis are shown in Figure 6.14 and 6.15 where the sensitivity of launch attempts to reliability and payload MTBF are illustrated. Specifically, the sensitivity of the expected number and standard deviation of launch attempts is shown about two nominal values indicated as points "A" and "B". Point A (Figure 6.14) is based on all reliabilities being 1.0 and a payload MTBF of five years. Point B (Figure 6.15) is based on all reliabilities being .95 and a payload MTBF of five years. The thick bands represent the range of variability of the expected number and standard deviation of launch attempts as each of the reliabilities is varied (one at a time) from .9 to 1.0. This establishes the sensitivity of launch attempts to changes in probability of booster success, orbiter success and the payload operating successfully when initially placed in orbit. The sensitivity of launch attempts is approximately the same for each of these parameters. The dashed curves represent the sensitivity of launch attempts to variations in payload MTBF.

Point "A", representing a perfect launch system and a payload with an MTBF of 5 years indicates, for the mission described in Table 6.26, approximately 7.5 launch attempts are to be expected with a standard deviation of 1.3 launch attempts. Since Point "A" represents a perfect launch system the significance of payload MTBF becomes clear. It can be seen that if an MTBF of 3 years is achieved, approximately 9.3 launch attempts should be expected. On the other hand, extremely large MTBF's will reduce the number of expected launch attempts to 5 (the number of new satellites to be added to the system) as indicated by the dotted line in Figure 6.14. Also from Figure 6.14, it can be seen that if <u>one</u> of the reliability components is reduced from 1.0 to .95, the expected number of launch attempts increases from 7.5 to 8.0 and the standard deviation of launch attempts increases from 1.3 to 1.6. Point "B" (Figure 6.15) illustrates the consequences if all three reliability components are reduced simultaneously from 1.0 to .95. It can be seen that the expected number of launch attempts increases from 7.5 to 9.2 and the standard deviation of launch attempts increases from 1.3 to 2.2.

The magnitude of the effects of achieving less than unity reliability and less than infinite MTBF have been illustrated. The effects of reliability

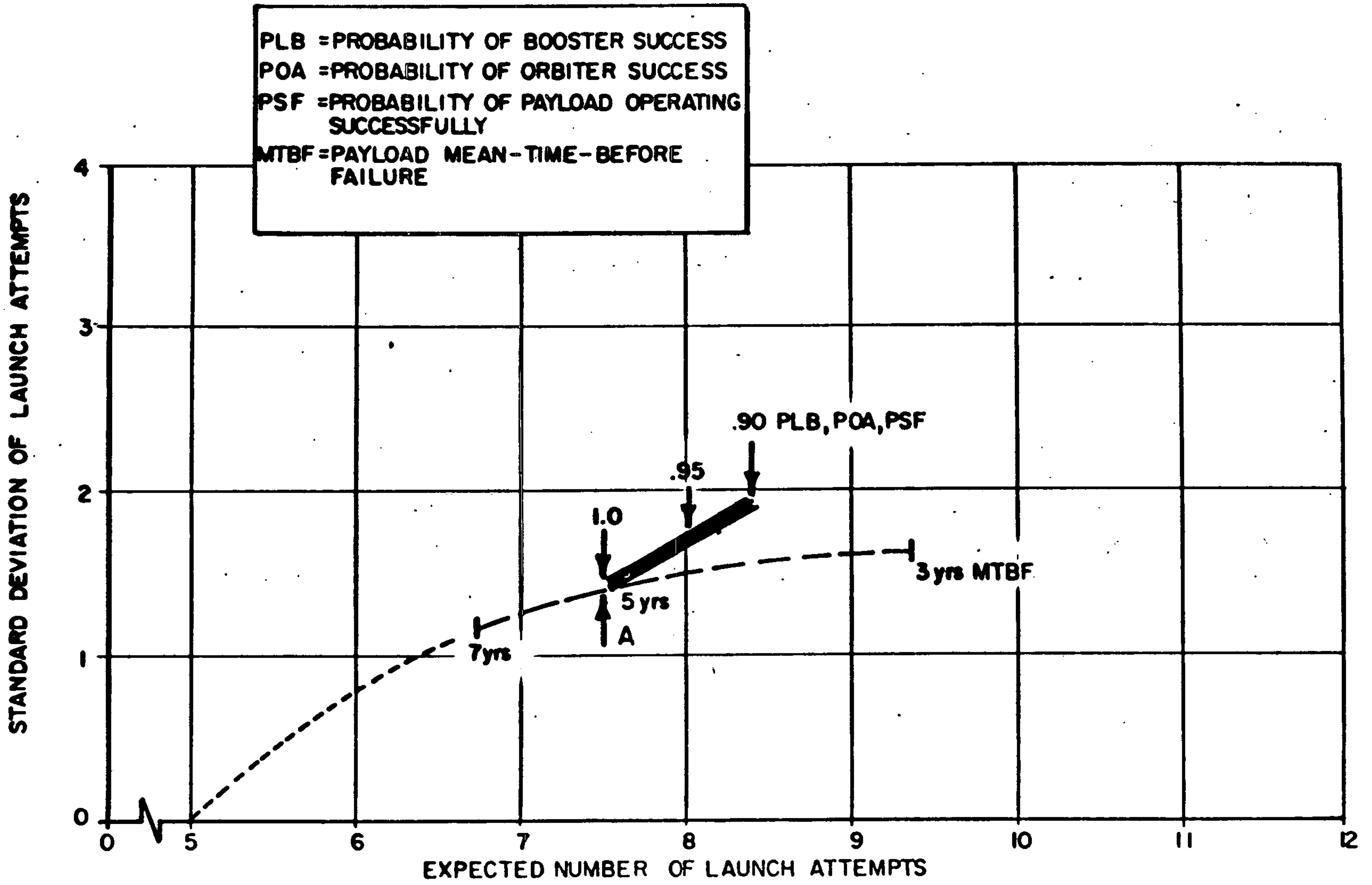

Figure 6.14 Sensitivity of Launch Attempts to Launch System Reliability and Payload MTBF for a Typical Shuttle System to Meet Mission Requirements Specified in Table 6.26

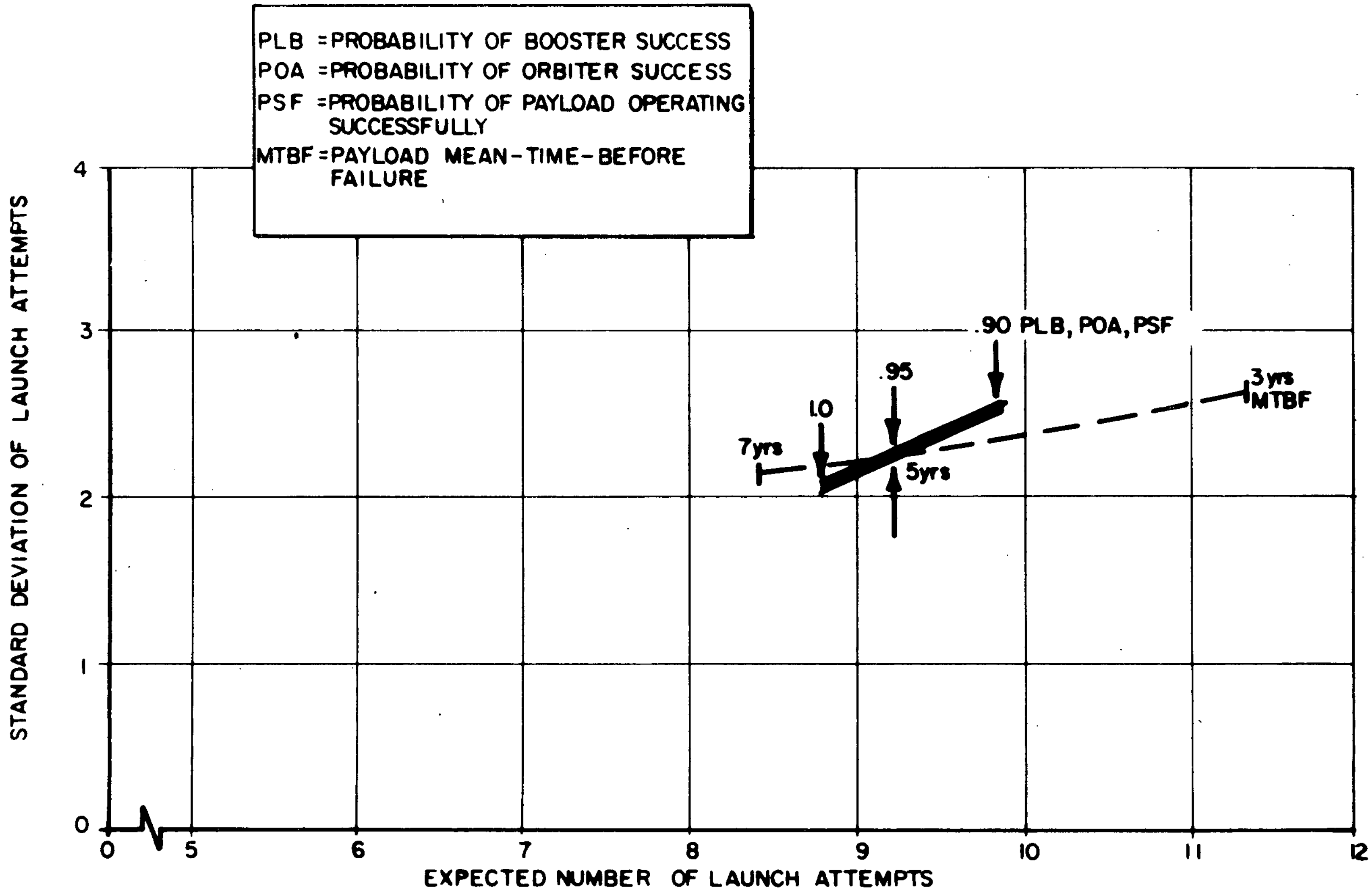

Figure 6.15 Sensitivity of Launch Attempts to Launch System Reliability and Payload MTBF for a Typical Shuttle System to Meet Mission Requirements Specified in Table 6.26

on mission cost will be discussed in Section 6.4.3.5.

6.4.3.3 Cost Uncertainties and the Development Cycle

The magnitude of cost uncertainties is related to the time in the development cycle that the estimates are made -- early in the cycle implies large uncertainties, late in the cycle implies small uncertainties. This is particularly true when new technologies and/or concepts are utilized. Thus it is anticipated that cost uncertainties associated with employing the current expendable fleet will be less than those associated with a new expendable fleet which will in turn be less than those associated with a new reusable Space Shuttle System.

Cost uncertainties are the result of uncertainties in the basic cost estimating relationships, lack of detailed understanding or appreciation of problems encountered in achieving desired technical solutions, variability of design goals, mission traffic estimates, et cetera. Uncertainties exist and should be considered as part of the evaluation process. Consideration of only the most likely or expected costs implies that the future is known with certainty.* Cost uncertainties are admittedly difficult to quantify. However, it might be inferred that the more difficult it is to quantify cost uncertainties the greater is the uncertainty.

6.4.3.4 Estimation of Cost Uncertainties
General Problem of Quantifying Uncertainties

The basic problem is how can uncertainty be quantified. The quantification of uncertainty requires that informed estimates be made of ranges of uncertainty of key cost variables and their probability distri-

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*This is analogous to the following: A man standing on a street corner is faced with a decision as to whether or not he should cross the street. He decides not to make a decision and remains standing on the corner. He has in fact made the decision not to cross the street. Thus the lack of explicit consideration of uncertainty would imply a condition of certainty.

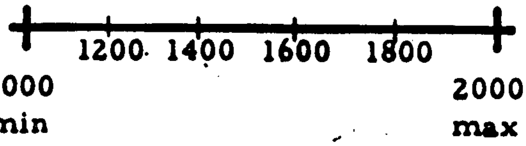
butions within the range. The estimates of uncertainty might be made, for example, at the CER level or they might be made at the unit cost (payload, orbiter, etc.) level. The uncertainty assessments should be made by an experienced group of individuals using Delphi type techniques. The estimates are very subjective in nature and quantitatively express the consensus of a group of well informed persons. The estimates reflect past experience with similar efforts, problems which have been encountered in the past, insights into problem areas which might develop, et cetera. They are the best estimates possible at any given time. An example might serve to illustrate how subjective estimates of uncertainty might be arrived at. In a recent discussion a propulsion system manufacturer outlined a development schedule for his proposed new propulsion system. It was stated that there was little chance of reducing the schedule and that based upon past experience delays of up to two years might be encountered. This implies at least a two-year range of uncertainty with a very large chance of the schedule being exceeded (in fact, it might be inferred that the uncertainty profile is exponential in nature).

Cost uncertainties can be quantified. Most large corporations use risk analysis techniques employing uncertainty assessments as standard procedure in the evaluation and comparison of new business alternatives [25, 26, 27, 28, 29, 30].

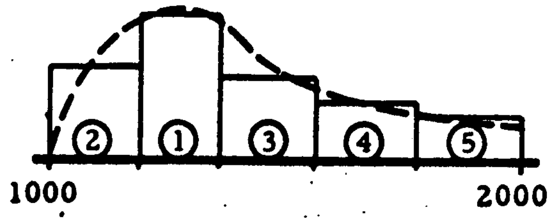
A Methodology for Quantifying Uncertainties

A methodology is now described for establishing the shape of the cost uncertainty profiles. This methodology has been employed in risk analyses performed for numerous industrial corporations. The methodology is illustrated in Figure 6.16.

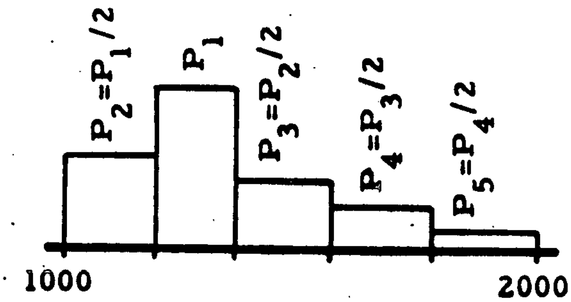
The first step is to establish the range of uncertainty. The range is based upon knowledgeable persons assessing what can go right and what can go wrong. The range is thence divided into five equal intervals (it has been found that it is difficult to "think" in terms of more than five or six intervals). The second step is to perform a relative ranking of the liklihood of the cost variable falling into each of the intervals. Once this has been



(A) **Specify Range of Uncertainty**



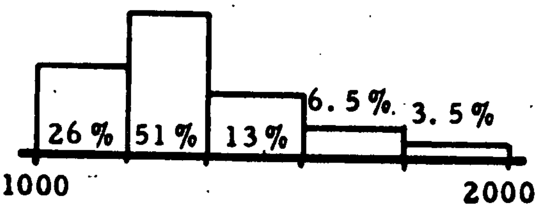
(B) **Perform Ranking (Qualitative)**



(C) **Establish Relative Values**

$$P_1 + P_2 + P_3 + P_4 + P_5 = 1$$

By Substituting from (C) Solve for P Values



(D) **Establish Quantitative Values**

Figure 6.16 **Methodology for Establishing Shape of Cost Uncertainty Profile (pdf)**

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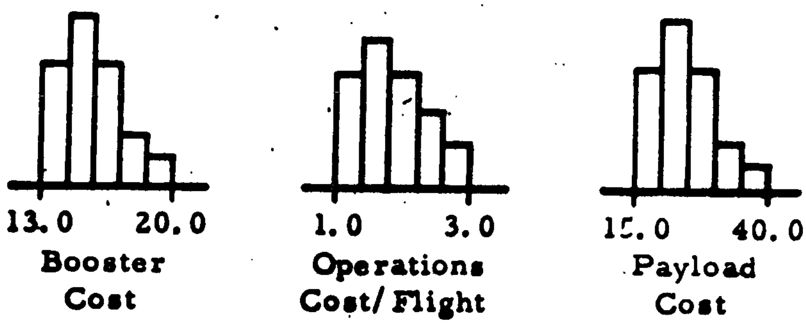
accomplished, the general shape (skewed left, skewed right, central, etc.) of the uncertainty profile has been established. The third step is to establish relative values of the chance of falling into each of the intervals (for example, in the illustration, the chance of falling into the first interval is estimated to be half as likely as falling into the second interval). The last step is to solve the illustrated equation for the quantitative values by substituting the data of the previous step.

6.4.3.5 Effects of Cost Uncertainties

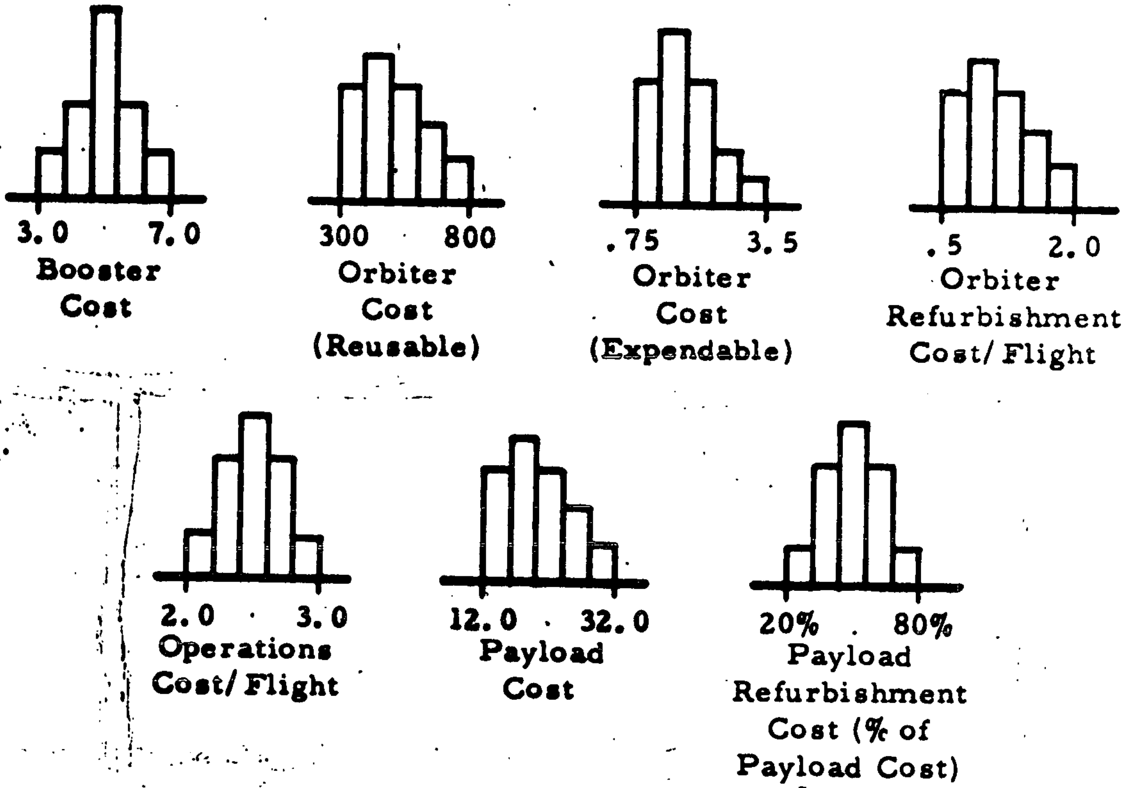
In order to demonstrate the effect that cost uncertainties coupled with reliability effects might have upon the comparison and evaluation of alternatives, the typical cost uncertainty profiles illustrated in Figure 6.17 have been used. Only recurring costs are considered. Replacement of elements of the fleet due to failures are considered as recurring costs. It should be noted that the uncertainty variables are at a relatively "gross" level. In the future it would be more meaningful to establish uncertainty profiles at the CER level.

Different Space Transportation System configurations will result in different uncertainty profiles. It should be noted that the uncertainty profiles immediately make apparent the degree of optimism or pessimism associated with the cost estimate.

As mentioned previously the cost model combines the uncertainty assessments with the outputs of the mission simulation model and establishes the probability distributions of booster cost, orbiter cost, payload cost, booster and orbiter and payload refurbishment cost, and total mission cost. Also as mentioned previously, both reliability and cost uncertainties affect the risk associated with total mission cost. Figure 6.18 illustrates the risk profiles (probability of exceeding indicated cost) associated with a typical expendable launch vehicle system and a typical Shuttle system for performing the previously defined (Table 6.26) mission. It should be noted that the most likely cost of the Shuttle System is less than the most likely cost of the expendable launch vehicle system. It should also be noted that the risk (variability) or standard deviation of the Shuttle System is greater



(A) Typical New Expendable System



(B) Typical Shuttle System

Figure 6.17 Typical Cost Uncertainty Profiles

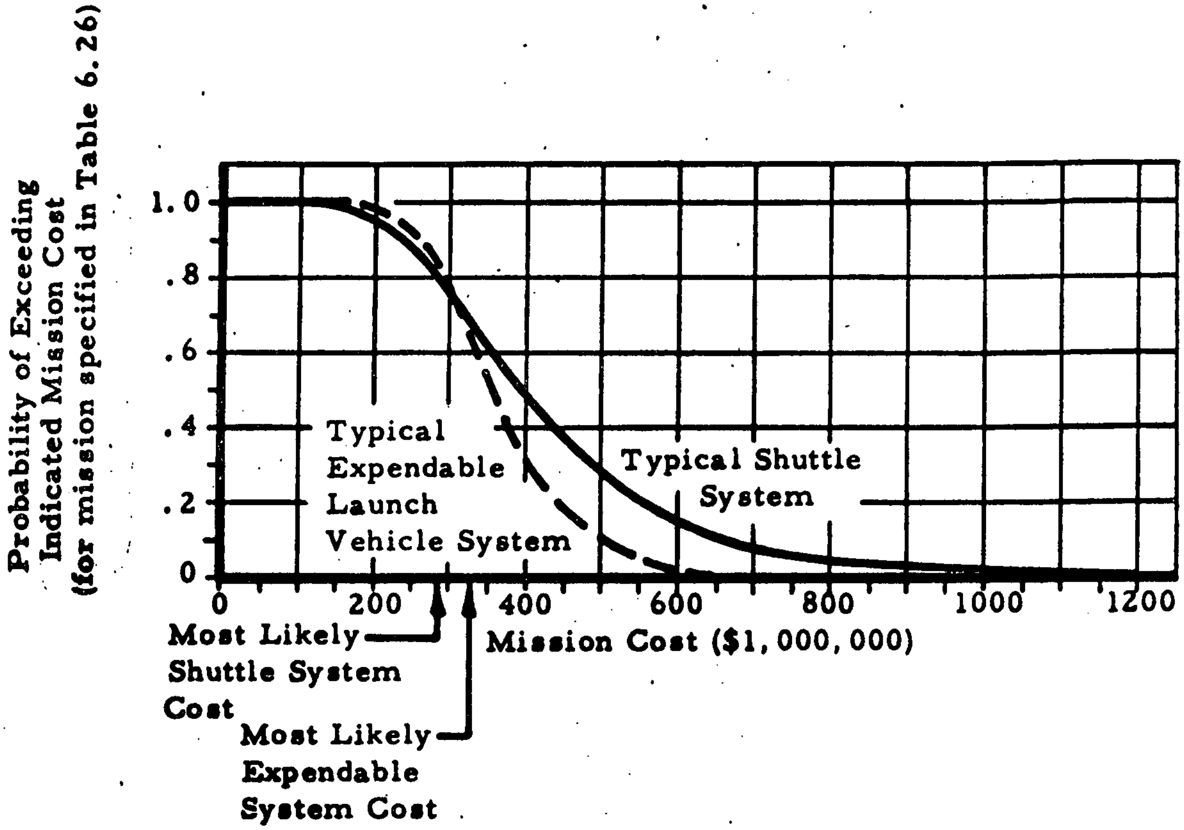


Figure 6.18 Typical Total Mission Cost Risk Profiles for Mission Requirements specified in Table 6.26

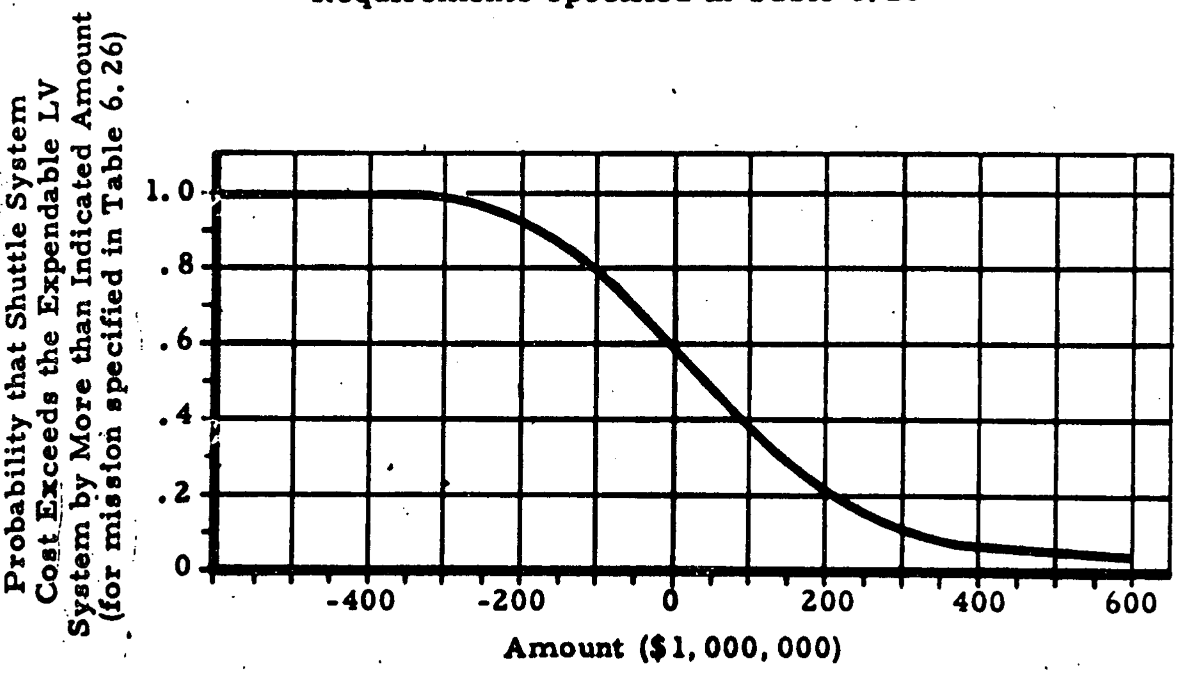


Figure 6.19 Comparison of Risk for Mission Requirements Specified in Table 6.26

than that of the expendable launch vehicle system.

In order to bring this into perspective, the probability that the Shuttle System cost exceeds the expendable launch vehicle system costs by various amounts* was computed and is illustrated in Figure 6.19. It can be seen, that for the case considered, there is a 60 percent chance that the Shuttle System cost will exceed the expendable system cost even though the most likely Shuttle System cost is less than the most likely expendable system cost. There is a 20% chance that the Shuttle System cost will exceed the expendable system cost by more than $200 million.

6.4.3.6 Extensions of Techniques

A method of explicitly considering the effects of reliability and cost uncertainties has been demonstrated by considering a single time period of a single mission. The consideration of a single mission and typical reliability and cost uncertainties has demonstrated the importance of risk analysis. Before conclusions can be drawn regarding the comparison and selection of Space Transportation Systems the simulation and risk analysis techniques should be extended to consider and include the following:

- multiple missions and payload types
- multiple time periods
- effect of multiple launch sites upon fleet size
- effect of refurbishment and checkout time upon fleet size in terms of mission requirements
- multiple payloads per flight
- space tug
- et cetera

Reliability and cost uncertainties must be considered when evaluating and comparing alternatives. They should also be considered when performing a capture analysis.

Risk analyses leads to variations in costs year by year. In the above analysis, no attention was paid to a fixed budget constraint. This

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*$p(A > B + Z)$ where Z is the amount

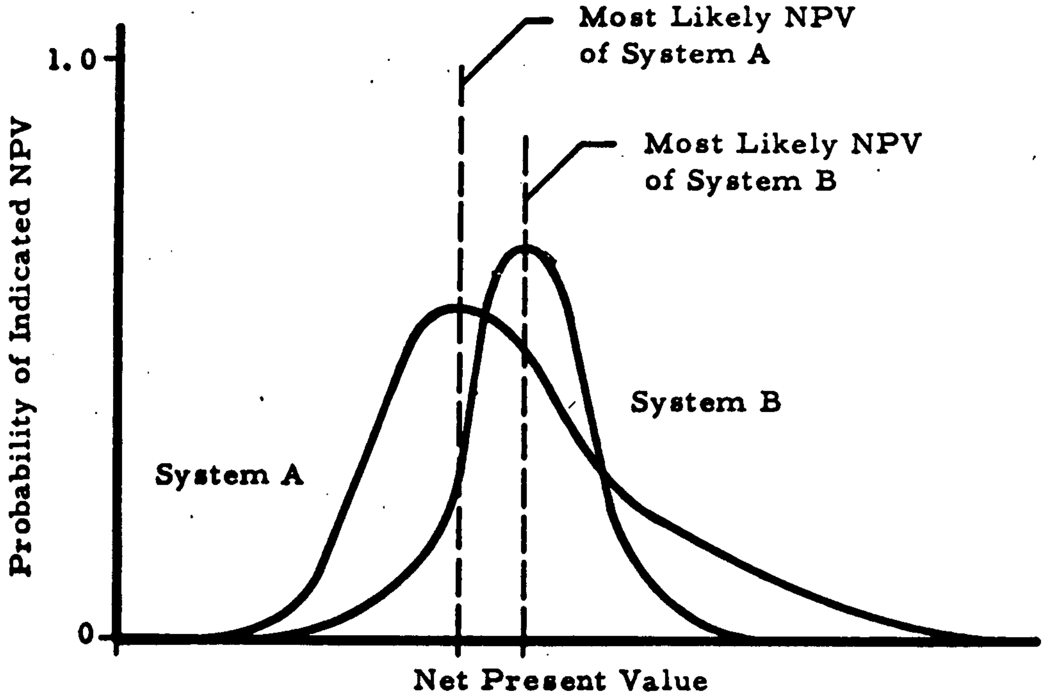
situation should be considered in extensions of the risk techniques. If the budget is fixed, then uncertainties and risk can be reflected back into the number of allowable flights. Thus, risk analysis comparison with a fixed budget constraint might be more meaningful if performed on a payload or number of allowed flight basis. If this is done, risk profiles of payloads per year and their "value" become important for comparison purposes.

The illustrated analysis considered an expendable STS and a typical Space Shuttle configuration. The same techniques should be used to evaluate and compare the many possible Space Shuttle configuration.

6.4.3.7 General Problem of Decision Making Under Uncertainty

Using Monte-Carlo techniques the probability distribution of net present value associated with alternate Space Transportation Systems can be determined in much the same manner as the mission cost distributions. This requires an expansion of the concept to include multiple missions, multiple time periods and non-recurring costs. When choosing from amongst several alternatives where costs are known with certainty, the alternative having the minimum present value of cost should be selected. When uncertainties exist, net present value will be a probabilistic quantity. Typical probability density functions of net present value for two alternative systems are illustrated at the top of Figure 6.20. The most likely NPV of System A is less than that of System B. The risk, as measured by the standard deviation, associated with System A is greater than that of System B. The choice of alternative depends upon the decision maker's aversion to risk. That is, if the possibility of high costs occurring is not tolerable, System B is preferable. If the possibility of high cost is tolerable when considering the low cost potential, then System A is preferable.

Another way of illustrating this is shown in the bottom illustration of Figure 6.20. The vertical scale represents the risk as measured by the standard deviation of net present value and the horizontal scale represents the expected (mean) net present value. Each alternative (three are illustrated) is represented as a point. A conservative decision maker will select alternative B rather than C since the risk associated with B is less



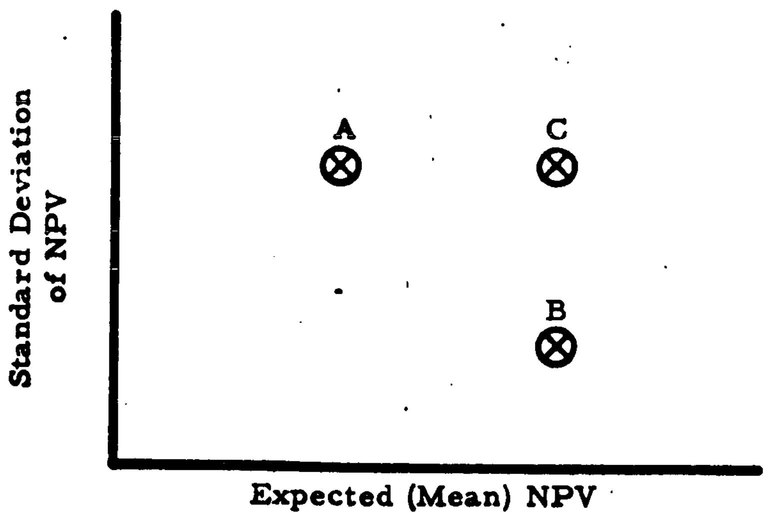


Figure 6.20 General Problem of Decision Making Under Uncertainty

than that of C and both alternatives have the same expected net present value. Similarly, alternative A will be selected rather than C since A and C have the same level of risk but the expected net present value of cost of A is less than that of alternative C. Thus the problem boils down to a selection between Alternatives A and B which have different levels of risk and different expected NPV's. At this point the analyst has provided all possible information, including quantitative assessments of risk, to the decision maker. The decision maker now must make choice considering his attitudes toward risk [31, 32].

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APPENDIX 6A

PHASED COSTS: TWIN RAO AND NEW EXPENDABLE SYSTEMS

TABLE 6 A.1

LIFE CYCLE COST SUMMARY DATA
SCENARIO 32 - NEW EXPENDABLE SYSTEM

(Millions of Undiscounted 1970 Dollars)

FISCAL YEAR	NON-RECURRING COSTS LAUNCH RDT&E	NON-RECURRING COSTS VEHICLE INVEST.	NON-RECURRING COSTS PAYLOAD RDT&E	RECURRING COSTS LAUNCH	RECURRING COSTS PAYLOAD	TOTAL
1971	0	0	0	0	0	0
1972	0	0	0	0	0	0
1973	0	0	0	0	0	0
1974	0	0	0	0	0	0
1975	60	0	38	0	0	98
1976	190	10	114	0	1	315
1977	355	80	441	81	308	1265
1978	315	194	978	240	883	2610
1979	195	193	1121	460	1163	3132
1980	60	135	1133	605	1311	3244
1981	5	125	1285	812	1423	3650
1982	5	75	1212	758	1639	3689
1983	0	0	1241	751	1649	3641
1984	0	0	905	806	1750	3461
1985	0	0	606	799	1580	2985
1986	0	0	455	837	1656	2948
1987	0	0	409	856	1684	2949
1988	0	0	362	834	1739	2935
1989	0	0	242	653	1261	2156
1990	0	0	64	269	328	661
TOTAL	1185	812	10606	8761	18375	39739

Figure 6A.1 Phased Non-Recurring Cost Breakdown for Twin RAO Operational Capability in 1979 (Engine costs not included in subtotals)

Source: McDonnell Douglas October Data

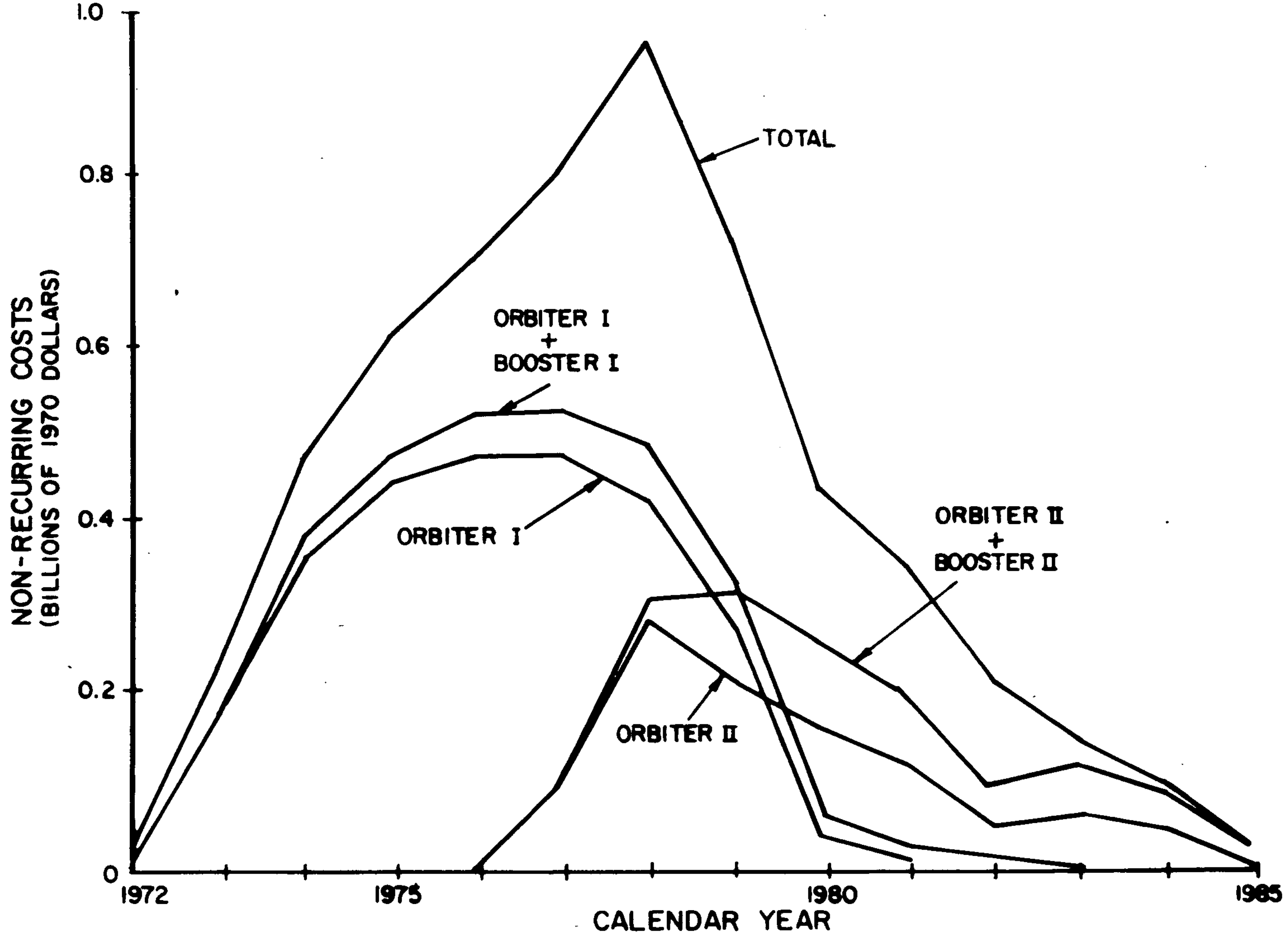


Figure 6A.2 Phased Non-Recurring Cost Breakdown for Twin RAO, Operational Capability in 1979

Source: McDonnell Douglas October Data

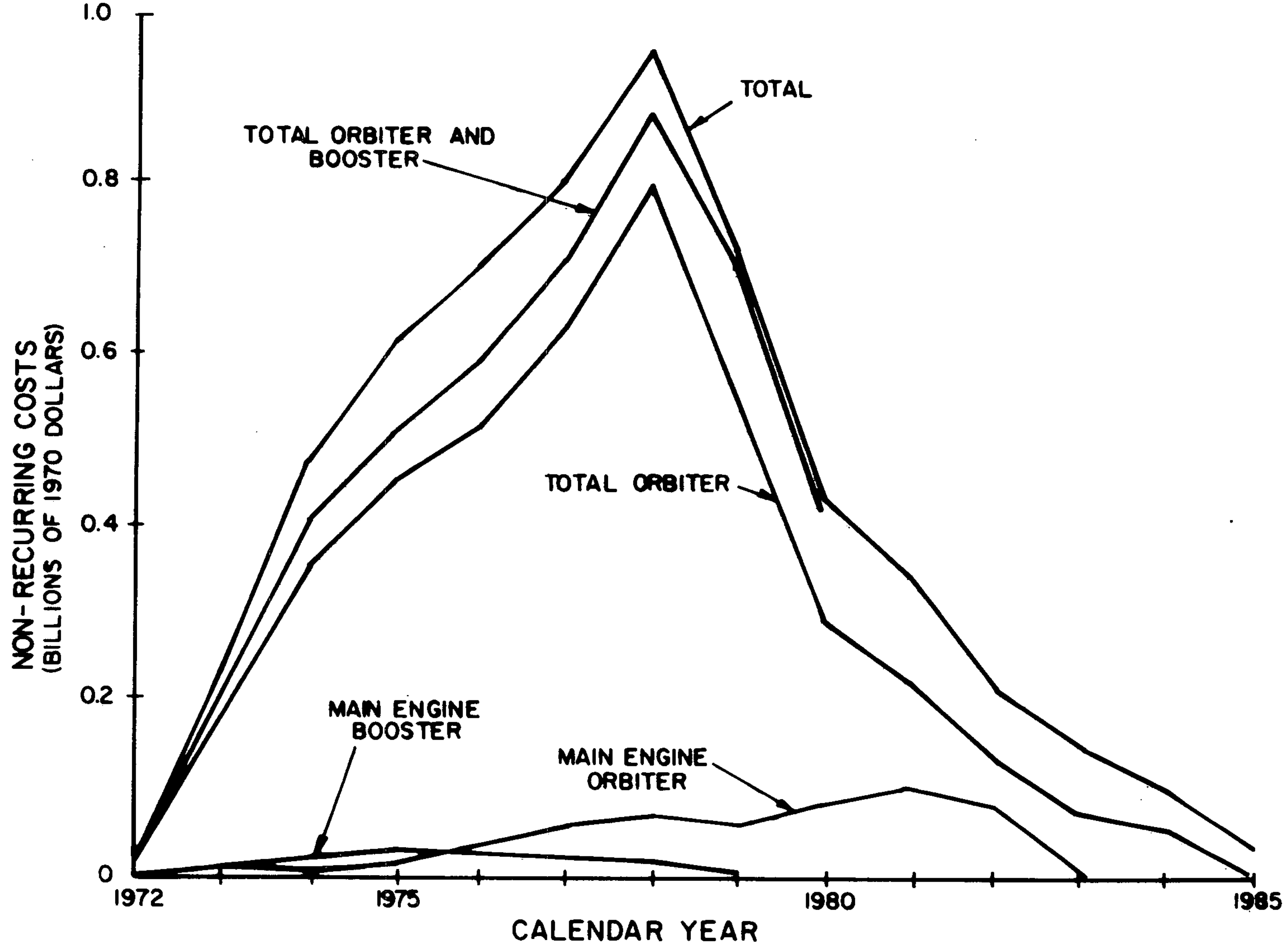
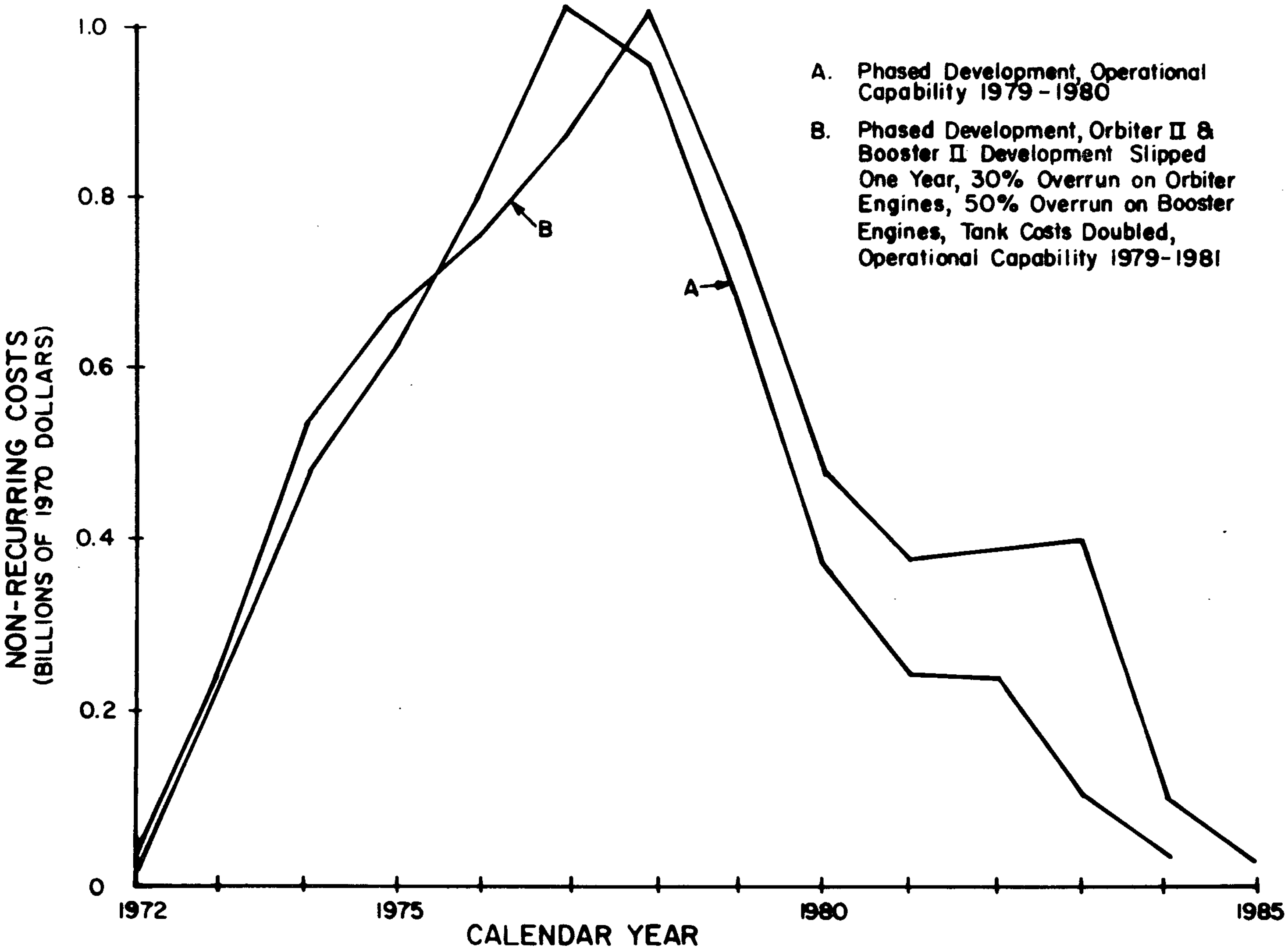


Figure 6A.3 Sensitivity to Slippages, Twin RAO

Source: McDonnell Douglas October Data



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