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TOR-92(2464)-1
Reissue A

Space Transportation Analysis and Design

Prepared by

Space Launch Operations

17 February 1993
(Supersedes and replaces TOR-92(2464)-1
dated 15 March 1992)

Prepared for

SPACE AND MISSILE SYSTEMS CENTER
AIR FORCE MATERIEL COMMAND
Los Angeles Air Force Base
P. O. Box 92960
Los Angeles, CA 90009-2960

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Programs Group

THE AEROSPACE CORPORATION
El Segundo, California

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SPACE TRANSPORTATION ANALYSIS AND DESIGN

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FOREWORD

This TOR is intended to be a living document that will be updated at regular intervals and will be linked to an ever-increasing body of easily accessed space transportation corporate knowledge in the form of both paper and magnetic media. This body of knowledge will include reports, computer-based analytical tools, technical databases, and human skills/knowledge databases. It is expected that its growth will be very much influenced by customer demand and the evolving future role of The Aerospace Corporation in the global space community.

Users of the TOR are invited to submit suggestions for future growth.

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I. INTRODUCTION

The intent of this TOR is to provide a guide to the processes and technical considerations involved in the selection of a launch vehicle for a given satellite mission, the integration of the satellite with the launch vehicle, and the many elements required to derive life-cycle costs of viable launch system candidates. As such, the document is expected to be of interest to satellite mission planners for preliminary assessments of launch vehicle candidates and of general use to experienced specialists and new engineers in both the satellite and launch vehicle sectors. Sufficient quantitative data, orbital and vehicle performance formulae, environmental data, and references have been provided to allow preliminary, first-order assessments which should suffice at the very early planning stages. As the planning process matures, the satellite planners are urged to get the much higher fidelity data on both technical and cost information required for programmatic decisions from the appropriate launch vehicle system program offices (SPOs).

The remaining six sections of this report are organized as follows: Section II summarizes the range of orbits/trajectories most frequently employed by satellites and the velocity requirements to achieve them. Section III lists and describes the capabilities of most domestic and foreign launch vehicles currently in service. Also discussed are upper stages and launch site locations. The many airborne mechanical/electrical interfaces and interactions between the spacecraft and launch vehicle that must be carefully engineered are covered in Section IV. Integration requirements at the launch site and orbital mission control centers are described, and typical time lines for the spacecraft/launch vehicle integration process are provided. Rocket propulsion, which has significant impact on the design of launch vehicles, is discussed in Section V. Included is a review of the basic principles of propulsion; types of engines, both current and envisioned; and their application to the different needs of launch vehicles, orbital transfer stages, and satellites. Trajectory-approximation computer routines, basic principles, and equations governing launch vehicle performance are presented in Section VI to enable mission planners to make first-order estimates of the vehicle size required for a given spacecraft weight to be placed in a desired orbit. Finally, in Section VII, a selection of future launch vehicles and their stage of development as of this writing are described. Included are the National Launch System, the Heavy-Lift Launch Vehicle, the Nuclear Transfer Stage, the Single Stage to Orbit, and the National Aerospace Plane.

II. MISSION REQUIREMENTS ANALYSIS

Joseph Statsinger

This section summarizes briefly the range of orbits/trajectories most frequently employed by satellites and the velocity requirements to achieve these orbits. The velocity requirements, combined with the spacecraft weight and physical dimensions, determine the minimum requirements placed on the launch vehicle. Various other requirements/constraints that the mission places on the launch system are discussed briefly. A detailed assessment of the impacts of these requirements on the launch system and a broader discussion of all important interactions and interfaces between the spacecraft and the launch system are part of the launch system integration process covered in Section IV.

A. ORBIT/TRAJECTORY CHARACTERISTICS

The characteristics of some typical trajectories and orbits frequently employed by satellites are summarized in Table 2-1. The "characteristic velocity" (third column) represents the velocity (exclusive of launch losses) that all the launch vehicle stages together must provide to achieve the desired orbit. Depending on the mission orbit, there is generally more than one option available for a launch vehicle/satellite system to achieve orbit. For some low-Earth orbit (LEO) missions, the booster delivers the payload directly into the operational orbit. For missions requiring high altitude orbits or Earth-escape trajectories, the payload is mated to an upper stage (orbit transfer vehicle) that provides the energy to transfer the payload from an initial low-Earth parking orbit to the desired mission orbit. The determination of the upper stage requirements is an important element of mission design. Based on performance and cost tradeoffs, the mission designer may choose a separate upper stage or one that is integral to the spacecraft. Intermediate choices are also available, involving partial transfer with a separate upper stage followed by final injection using spacecraft propulsion.

B. LAUNCH VEHICLE/SPACECRAFT CONSTRAINTS

A major launch constraint that must be evaluated and planned for each mission relates to the time period or interval during which the launch opportunity exists. This so-called "launch window" usually ranges from a few minutes to several hours in duration. Launch windows may reoccur periodically at frequent intervals or in the case of planetary missions, for instance, be available only infrequently. Some of the factors involved include:

- Event timing. For suborbital missions, the timing of the event to be observed may be the major constraint.
- Sun–Earth–spacecraft geometry during ascent and initial orbit. This affects solar heating of the spacecraft and the availability of solar energy for generating spacecraft power.

Table 2-1. Typical Orbits and Trajectories

Orbit or Trajectory Type	Description	Characteristic Velocity kft/sec	Typical Mission
Suborbital	Altitude up to several hundred miles. Return to Earth	Various	Experiments-measurements
Low-Earth orbit (LEO)	Altitude up to 1000 nmi. Period up to 3 hr. Various inclinations including equator, polar, sun synchronous	100 nmi 25.6 1000 nmi 28.3	Earth observation, scientific payloads, weather, manned orbital missions
Inclined elliptical orbits (Molniya)	Eccentricity: 0.75 Inclination: 63.4 or 116.6 Apogee altitude: ≈21,700 nmi Period: 12 hr	~33.5	Communications
Geosynchronous Earth orbits (GEO)	Altitude: 19,323 nmi Period: 23 hr, 56 min, 4 sec	38.5–39.8	Weather, communications
Escape trajectories	Hyperbolic	≥36.1	Lunar missions, deep-space missions
Geosynchronous transfer orbit (GTO)	Hohmann transfer from 100 nmi LEO to geosynchronous	8.08*	Missions employing geo-synchronous orbits

*Delta velocity at perigee

- *Location at the* Earth's equator of the orbital trace (right ascension of ascending node). This determines the relation of the orbit plane to other orbits and to the desired ground trace of the spacecraft.
- Constellation establishment. When the spacecraft is one of a constellation, the launch window is selected to provide proper phasing among the spacecraft with minimum orbit-adjust fuel expenditure.
- Astronomical configurations. For deep-space probes, the location of planets, comets, or other astronomical objects may determine launch timing requirements.
- Random events. Weather and other phenomena for which long-term predictions are not available may produce unexpected restrictions on launch window availability at any particular time.

Other examples of spacecraft-imposed constraints on the launch vehicle are:

- Dynamic loads on the spacecraft at stage shutdown. Loads are lower for a command versus a propellant depletion shutdown; performance is sacrificed if command shutdown is required.
- Trajectory shaping to satisfy location and time of allowable telemetry transmission, thermal, maximum allowable dynamic pressures, and other constraints.

- Collision avoidance maneuvers of the final propulsive stage at separation of the spacecraft.
- Orbital accuracy at spacecraft orbit injection.

III. CURRENT LAUNCH SYSTEMS

Richard A. Hartunian, Malcolm G. Wolfe, Carole S. Tanner,
Bruce H. Wendler, James S. Whittier, Edmund Blond

This section provides essential information required to select a launch vehicle for a given space mission. Five topics are covered:

- Description and capabilities of current domestic and foreign launch vehicles
- Launch sites
- Payload compartment environments
- Reliability assessment
- Launch system costs

These influence launch vehicle selection and are considered in early system tradeoff studies where more than one launch vehicle is a viable candidate.

A. EXPENDABLE LAUNCH VEHICLES

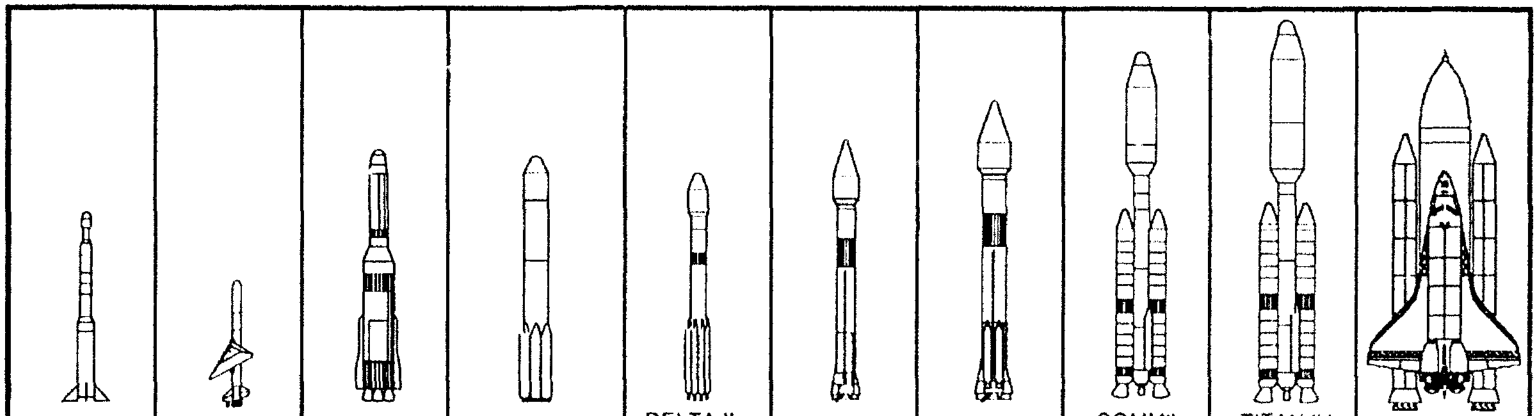
Basic data are provided on launch vehicles currently in use, as well as those programmed to be available in the near future. Both domestic and foreign vehicles are included. A brief description of each of the vehicles is provided. Data are provided in tabular form on the lift capability to frequently employed generic orbits, dimensions of the payload compartment available for the spacecraft, launch vehicle reliability, approximate cost per launch, and dates of initial launch capability (ILC). The selected generic orbits are:

- Low-Earth Orbit (LEO) - 100 nmi circular polar or due east
- Geostationary Transfer Orbit (GTO) - Elliptic with 100 nmi perigee, 19,323 nmi apogee
- Geosynchronous Orbit (GEO) - 19,323 nmi circular

Launch vehicle performance to other orbits may be obtained from the prime contractors (Reference 3-1) or, for preliminary estimation purposes, by application of the algorithms provided in Section VI. The costs per flight should be considered as planning estimates only, since they vary with number of flights per year and many other factors.

1. Domestic Vehicles

The current and programmed U.S. stable of launch vehicles is depicted in Figure 3-1 along with the data described above for each vehicle. A brief description of each vehicle follows.



	SCOUT	PEGASUS	ATLAS E	TITAN II	DELTA II 7920/7925	ATLAS II	ATLAS IIAS	COMM'L TITAN III	TITAN IV (SRMU)	STS
RESPONSIBLE AGENCY	NASA	DARPA	USAF	USAF	USAF	USAF	COMM'L	COMM'L	USAF	NASA
PRIME CONTRACTOR	LTV	OSC	GD	MMC	MDC	GD	GD	MMC	MMC	RI
PERFORMANCE LEO POLAR LEO DUE EAST [1] GTO GEO	 460 600 (2.9°) 120 —	 800 1,000 275 —	 1,800 — — —	 4,200 — — —	 8,420 11,100 4,010 —	 11,900 14,100 5,900 1,270	 15,000 18,500 7,700 2,310	 — 32,000 9,500[2] 3,000[3]	 41,000 47,700 19,000[3] 12,700[4]	 16,300 53,700 13,000[5] 5,200[5]
RELIABILITY SUCCESS RATE ~% [7]	95.0	50.0	92.0	93.0	97.0	100.0	NOT FLOWN	93.0	93.0	97.4
PAYLOAD ACCOMMODATION PAYLOAD DIA ~ft CYLINDER LENGTH ~ft CONE LENGTH ~ft	 2.5; 3.2 2.8; 4.0 1.3; 2.5	 3.8 3.0 3.3	 7.0 16.9 3.8	 9.3 12.0; 17.0; 22.0 5.1	 9.2 12.0/6.3 7.0	 9.6; 12.0 12.8; 13.7 12.6; 18.2	 9.6; 12.0 12.8; 13.7 12.6; 18.2	 12.0 26.4 14.1	 15.0 40.0 19.8	 15.0 60.0 —
GLOW ~Klbs	48	42	267	340	506	413	516	1,500	1,900	4,500
COST/FLT ~$91M	10–12	7–12	49	43	47-52	74-84	116–126	130–150	154, 227[4]	275[6]
INITIAL LAUNCH CAPAB	1960	1990	1974	1988	1990	1991	1993	1989	1989	1981

1) ~28.5°
2) WITH TRANSTAGE
3) WITH KICK MOTOR
4) WITH CENTAUR
5) WITH IUS
6) SHUTTLE FULL COST = 275M (248 IN $88), 130M ($88) FOR COMMERCIAL USERS AND 109M ($88) FOR DOD
7) LAST 88 FLIGHTS (or less); EARLY FLIGHTS OMITTED FOR MATURE SYSTEMS

Figure 3-1. U.S. launch vehicles.

a. Scout

The NASA Scout vehicle became operational in 1960; as of July 1991, only four vehicles remain. It is a four-stage, solid-propellant, series-burn rocket. The Scout can deliver 460 lb to LEO polar [launched from Western Test Range (WTR)] and 570 lb to an easterly (37.7 deg) orbit launched from Wallops Island, Virginia. An upgraded version (Scout II) has been studied that could double the payload capability. Strap-on solids and an apogee kick motor added to the existing core vehicle yield the enhanced performance.

b. Pegasus

The Pegasus air-launch space booster is a new launch vehicle developed as a privately funded joint venture by Orbital Sciences Corporation (OSC) and Hercules Aerospace Company. Pegasus is a three-stage, solid-propellant, inertially guided, all composite, winged launch vehicle. The vehicle is approximately 50 ft long and 50 in. in diameter, and has a gross weight (excluding payload) of approximately 41,000 lb. It is carried aloft by a conventional transport or bomber-class aircraft (such as a Boeing B-52) to level-flight conditions at approximately 40,000 ft and a speed of Mach 0.8. After release from the aircraft and ignition of the Stage I motor, the autonomous flight control system provides all the guidance necessary to produce a wide range of suborbital and orbital trajectories. The development program was begun in early 1987, and the Pegasus vehicle has been available for launch services to both government and commercial users beginning with its first orbital flight in April 1990. The second flight of the Pegasus featured a new operational propulsion unit and a guidance upgrade. After burnout of the third solid-fuel stage, a small liquid propulsion unit, dubbed HAPS (Hydrazine Auxiliary Propulsion System), functions as a precision orbital injection kit. HAPS has three 50-lb thrusters and holds 160 lb of hydrazine. The Inertial Measurement Unit (IMU) serves as the primary source of navigation information, while the Global Positioning System (GPS) receiver serves as a redundant unit. The current configuration, when air-launched from a B-52 bomber, delivers 750 to 1000 lb of payload to a due east orbit and 560 to 750 lb to a polar LEO orbit. Improvements have been identified to increase the payload capability to 2500 to 3100 lb to a due east orbit and 1880 to 2300 lb to polar LEO.

c. Atlas Family

The Atlas is a former Air Force Intercontinental Ballistic Missile (ICBM) weapon system converted for use as a space launch vehicle. The Atlas D was man-rated and flew four successful missions in the Mercury program, including the John Glenn first U.S. orbital flight. The system concept is now over 30 years old, and has gone through a series of modernizations and upgrades which have enhanced its payload performance capability. There are several Atlas versions, existing and planned, but common to all is the use of one and a half liquid propellant stages. Both stages (booster and sustainer) are ignited on the ground and burn in parallel. After the booster engines are jettisoned, the sustainer engine continues burning to orbit. The Atlas family of vehicles is designed and manufactured by the Space Division of General Dynamics Corporation.

1. Atlas E
The Atlas E is a Department of Defense (DOD) launch vehicle presently used to launch smaller payloads to polar orbit from Space Launch Complex 3 (SLC-3) at Vandenberg Air Force Base (VAFB). The Atlas Es are the former Atlas ICBMs that have been refurbished, modified, tested, and certified for space flight. The Atlas E can place up to 1800 lb in low polar orbit, and it is primarily intended for these missions. It is compatible with a number of upper stages, including the single and dual solid motor spin-stabilized SGS I and II and Burner II stages. The remaining four vehicles are expected to be flown by 1994.

2. Atlas I
The Atlas I is a commercial launch vehicle used to launch medium-size payloads to GEO from Cape Canaveral Air Force Station (CCAFS), Launch Complex 36 (LC-36). It is an upgraded version of the Atlas G/Centaur. The vehicle can place about 5100 lb into geostationary transfer orbit (GTO) and 1000 lb to GEO using a kick stage, with an 11-ft diameter payload fairing. It is compatible with the Centaur I upper stage.

3. Atlas II
The Atlas II was selected as the Air Force Medium Launch Vehicle (MLV) II and is designed to perform LEO and GTO missions. The Atlas II was procured primarily to support the Defense Satellite Communication System (DSCS) and other medium-class payload requirements. It can lift payload weights of approximately 14,000 lb into LEO due east, 6100 lb into GTO, or 3100 lb into GEO using a kick stage. Initial launch of the Atlas II occurred in February 1992 when it boosted a DSCS III satellite into GTO from the Eastern Test Range's refurbished Complex 36A at Cape Canaveral. The Atlas contractor is offering another version of the Atlas II, the Atlas IIA, for launching NASA commercial payloads. The Atlas IIA is an improved version of the Atlas II in which the Pratt and Whitney RL-10 engine in the Centaur stage will be increased in thrust from 16,500 lb to 20,000 lb. The Atlas IIA payload to GTO will be 6400 lb.

4. Atlas II Derivatives
Versions of the Atlas with more capability than the Atlas II are being developed and planned. An Atlas IIAS configuration, in development for Intelsat, adds four Castor IVA solid motor strap-ons to the IIA and increases the LEO due east payload to 18,000 lb, the GTO payload to 7700 lb, and the GEO payload to 4100 lb, using a kick stage. The proposed versions, the JII and JIIS, are similar to Atlas II and IIAS, respectively, but without the Centaur upper stage. The JIIS has a LEO polar capability of approximately 7600 lb.

   Another proposed configuration, Atlas IIe, stretches the Centaur stage by 10 ft and uses larger (length and diameter) Graphite-Epoxy Motor (GEM) strap-on solid rockets.

d. Delta Family

The NASA Delta vehicle was derived from the Thor IRBM by adding several small, solid rocket motors (SRMs). It is used to launch payloads into GEO and LEO from CCAFS LC-17A and LC-17B and to polar orbit from VAFB SLC-2 (see Figure 3-2). The capabilities have grown through a series of upgrades and are now approximately 1450 lb to GEO, 750 lb to LEO, and 500 lb to polar for the Delta 3925 model. The 3925 model is now out of production. The Delta family of vehicles is manufactured by the McDonnell Douglas Company.

1. Delta II
   The development of the DOD Delta II was begun following the Shuttle Challenger accident in 1986 as a means to off-load the Global Positioning System (GPS) satellites from the shuttle manifest and speed up the operational availability of the GPS. The interim version, Delta 6920, is capable of placing 8800 lb into the due east LEO. An upgraded version, Delta 7920, supports the heavier weight of current GPS satellites. This version is capable of placing 11,100 lb into LEO due east. The main difference between the Delta II vehicles (6920 vs 7920) lies in the use of Castor Solid Rocket Motor (SRM) strap-ons (steel cases) versus the use of GEM strap-ons (graphite epoxy cases). The Delta II is launched from LC-17A and LC-17B at CCAFS. In addition to launching GPS satellites, the Delta contractor provides Delta II launch services for NASA and commercial payloads. The 7925, which uses a third stage, provides performance to GTO.

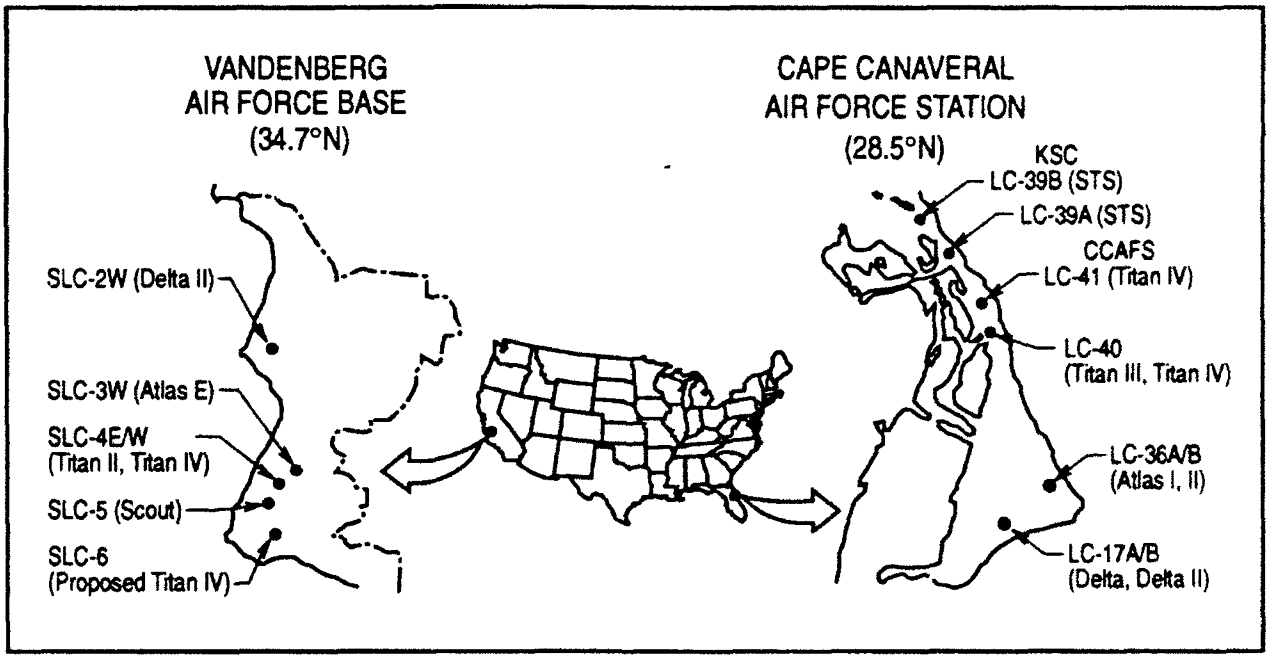


Figure 3-2. Launch complex locations.

This third or upper stage is a Payload Assist Module (PAM-D), with a Star 48B solid rocket motor mounted on a spin table to spin-up the stage/payload combination before deployment. Incorporation of this third stage provides the capability to deploy 4010 lb of payload into GTO (without the third stage, only 2800 lb can be deployed). An additional spacecraft kick motor is required for geosynchronous orbits.

2. Delta II Upgrades
Upgraded versions of Delta II are planned, including the current 7920 with cost improvements and a higher performance derivative capable of handling multiple or larger payloads. Cost improvements will be achieved by improving manufacturing methods, reducing on-pad time, and reducing subcontracting costs.

e. Titan Family

The Titan Space Launch System is a series of Air Force vehicles that has evolved from the Titan ICBM system over the past 35 years. The Titan ICBM weapon systems have been decommissioned. Three main launch vehicle configurations presently exist, with variations of each: Titan II Space Launch Vehicle (SLV), Commercial Titan III, and Titan IV. All have two core stages using liquid propellants. The last two use segmented SRMs for the initial stage to enhance performance. Composite-case SRMs (designated SRMU) will soon replace the steel-cased solids to provide increased performance. Additional proposals for increased performance include the use of liquid cryogenic propellant boosters, more and longer SRMs, and more liquid engines with stretched and/or larger diameter core stage tankage. The Titan family of vehicles is manufactured by the Martin Marietta Corporation.

1. Titan II
The Titan II Space Launch Vehicle was developed by DOD to replace the Atlas E launch vehicles for launching smaller payloads to polar orbit. The Titan II vehicles are former ICBMs that have been removed from their silos, refurbished, configured for space launch, tested, and certified for space flight. There were originally 55 Titan ICBMs in inventory, and the Air Force has a continuing program to modify and launch these vehicles, as required. The initial launch capability (ILC) of the Titan II was achieved in September 1988. The Titan II can place up to 4200 lb in polar LEO from Space Launch Complex-4 (SLC-4) West at VAFB. Proposals for upgrading Titan II performance include developing coast capability and long-duration circularization burns with added propellants and SRM strap-ons; the addition of eight Castor IVs, for example, would launch 8900 lb to LEO. Configuration studies include the addition of up to 10 GEM (Graphite Epoxy Motor) SRM strap-ons.

2. Commercial Titan III
The Titan III is derived from the Titan 34D with a stretched second stage and a hammerhead shroud (larger diameter) for dual or dedicated payloads. The first commercial Titan III was launched on 31 December 1989 with two communication satellites, one British and one Japanese. The Titan III can launch 32,000 lb into LEO. It is compatible with the McDonnell Douglas PAM-DII, the Martin Marietta Transtage, and the Orbital

Sciences Corporation Transfer Orbit Stage (TOS) upper stages to provide GTO capability of 4080 lb, 9500 lb, and 11,000 lb, respectively. Performance to GEO orbits is approximately 5500 lb using a spacecraft kick motor.

3. Titan IV
The DOD's Titan IV development was begun as a complement to the space shuttle. However, following the Challenger accident in 1986, the program was expanded to accommodate critical DOD payloads. The program was further expanded in 1988 as the full impact of the shuttle delays and cancellation of the Shuttle-Centaur program became clear. When NASA cancelled the Shuttle-Centaur program, the Centaur development was transferred to the Titan IV program. Initial launch capability of the Titan IV/Centaur is planned for the latter part of 1993; the vehicle is capable of delivering 10,000 lb to GEO with the current SRMs. When the SMRU boosters (now under development) are available, the Titan IV/Centaur will be able to launch a 12,700 lb payload to GEO. (It is important to note, however, that the current Centaur is structurally limited to 11,500 lb).

The Titan IV/IUS (Inertial Upper Stage) is currently operational and is capable of delivering 5250 lb to geosynchronous orbit. A Titan IV with no upper stage is used for launching into polar and high inclination orbits. This configuration can deliver 31,000 lb to a 100-nmi polar orbit.

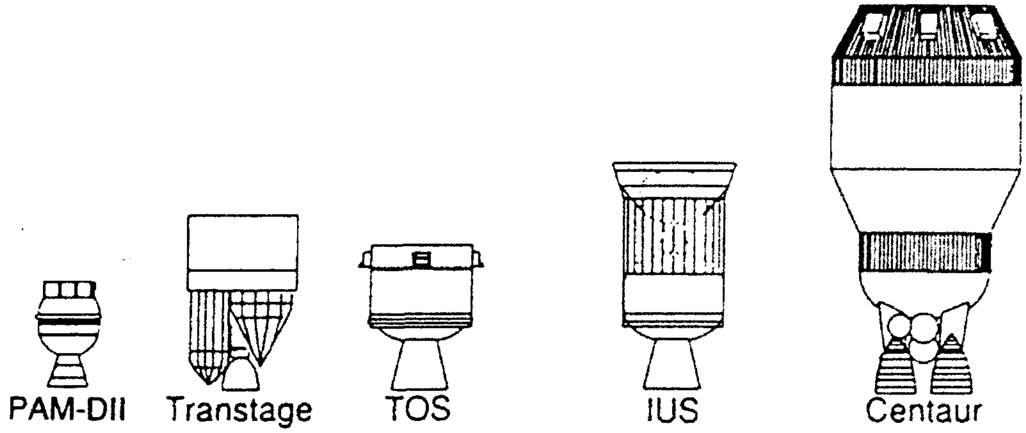
The Titan IV vehicles are launched from LC-40 and LC-41 at CCAFS and from SLC-4 East at VAFB.

2. Upper Stages

For high-energy orbits such as geosynchronous, launch vehicle performance can be augmented by an additional stage, either by incorporating an integral propulsive unit into the spacecraft or by using a separate upper stage. Separate upper stages that are currently in use include the Payload Assist Module (PAM-DII), developed by McDonnell Douglas; The Transtage, developed by Martin Marietta; the Transfer Orbit Stage (TOS), developed by Orbital Sciences Corporation and Martin Marietta; the Inertial Upper Stage (IUS), developed by the Boeing Company; and the Centaur, developed by General Dynamics. The basic design characteristics of these upper stages are shown in Figure 3-3.

a. Payload Assist Module (PAM-DII)

The PAM-D (D for Delta class) is one of a family of single-SRM, spin-stabilized stages used to place moderate-sized payloads into transfer orbits. PAM-DII is a larger version of the same stage. An apogee kick motor (AKM) is required to place payloads into high-energy circular orbits. In addition to Delta, the PAM family of vehicles is compatible with Titan III and the space shuttle.



	PAM-DII	Transtage	TOS	IUS Stage I	IUS Stage II	Centaur
Manufacturer	McDonnell Douglas	Martin Marietta	OSC/Martin Marietta	Boeing		General Dynamics
Length	6.6 ft (2.0 m)	14.8 ft (4.5 m)	11 ft (3.3 m)	17 ft (5.2 m)		29.5 ft (9.0 m)
Diameter	5.25 ft (1.6 m)	10 ft (3.0 m)	11.2 ft (3.4 m)	9.5 ft (2.9 m)		14 ft (4.3 m)
Propellant Mass	7140 lb (3240 kg)	22890 lb (10380 kg)	21400 lb (9710 kg)	21400 lb (9710 kg)	6060 lb (2750 kg)	44800 lb (20320 kg)
Gross Mass	7695 lb (3490 kg)	29780 lb (13510 kg)	23800 lb (10800 kg)	23960 lb(10965kg)	8600 lb (3900 kg)	52600 lb (23860 kg)
Propellant	Solid	N2O4-Aerozine 50	Solid / HTPB	Solid/HTPB	Solid/HTPB	LOX/LH2
Engine Company	Thiokol	Aerojet	CSD	CSD	CSD	Pratt & Whitney
Engine Designation	STAR-63D	AJ10-138 (2 units)	SRM-1	SRM-1	SRM-2	RL-10A-3-3A (2 unit)
Avg. Thrust	17.6K lb (78.3K N)	2 x 8K lb(35.6K N)	45K lb (200K N)	45K lb (200K N)	18.3K lb (81.2KN)	2 x 16.5K lb(73K N)
Isp	281.7 sec vac	309.1 sec vac	294 sec vac	292.9 sec vac	300.9 sec vac	444 sec vac
Restart Capability	No	Multiple	No	No	No	Multiple
Control	Spin stabilized	2 nozzles & 3-axis ACS	1 nozzle & 3-axis ACS	1 nozzle & 3-axis ACS	1 nozzle & 3-axis ACS	2 nozzles & 3-axis ACS
Nominal Burn Time	121 sec	449 sec	150 sec	153 sec	104 sec	600 sec

Remarks: PAM-DII requires a spin table during deployment. Transtage replaces Stage II avionics and ACS and can deploy one or two payloads. IUS is two stages for its geosynchronous missions.

Figure 3-3. Upper stage characteristics and capabilities.

b. Transtage

The Transtage was first flown on the Air Force Titan vehicle in the early 60s. It contains 22,890 lb of N_2O_4/Aerozine 50 liquid propellant and uses two Aerojet AJ10-138 engines. It has a total thrust of 16,000 lb and has multiple restart capability. Its role for DOD and NASA missions has been taken over by the IUS, but it is still available for civil missions.

c. Transfer Orbit Stage (TOS)

The Transfer Orbit Stage is a three-axis, stabilized, solid propellant orbit transfer stage with a 13,000-lb GTO payload capability from a 300-km circular orbit at 28.5 deg inclination. Its first flight occurred on 25 September 1992 when it was successfully launched from a Commercial Titan III for the Mars Observer mission.

The TOS has only one impulse burn, and therefore high-energy circular orbits are not attainable from LEO without an AKM. The TOS will be compatible with both Titan III and the space shuttle.

d. Inertial Upper Stage (IUS)

Initially, the IUS was planned as a family of vehicles, including a two-stage, a twin-stage, and a three-stage version, but the twin-stage and the three-stage programs were cancelled. The surviving IUS is a two-stage solid propellant upper stage consisting of a large first-stage motor containing 21,400 lb of propellant and a smaller second-stage motor containing 6,060 lb of propellant, both manufactured by Chemical Systems Division (CSD). It was developed by the Air Force for DOD, NASA, and civil use and is designed to be compatible with both the space shuttle and the Titan launch vehicle. The reference STS mission is 5,000 lb to GEO.

e. Centaur

The Centaur is a high-performance, cryogenic upper stage that has gone through a number of evolutionary performance and reliability upgrades, particularly in the areas of electronics and software. The two basic configurations exist: a 10-ft diameter stage for the Atlas family and a 14-ft diameter one for Titan IV. Both have two 16,500-lb thrust Pratt and Whitney RL-10A-3-3A engines, LOX/LH_2 propellants, high specific impulse (446), ar are capable of multiple restarts.

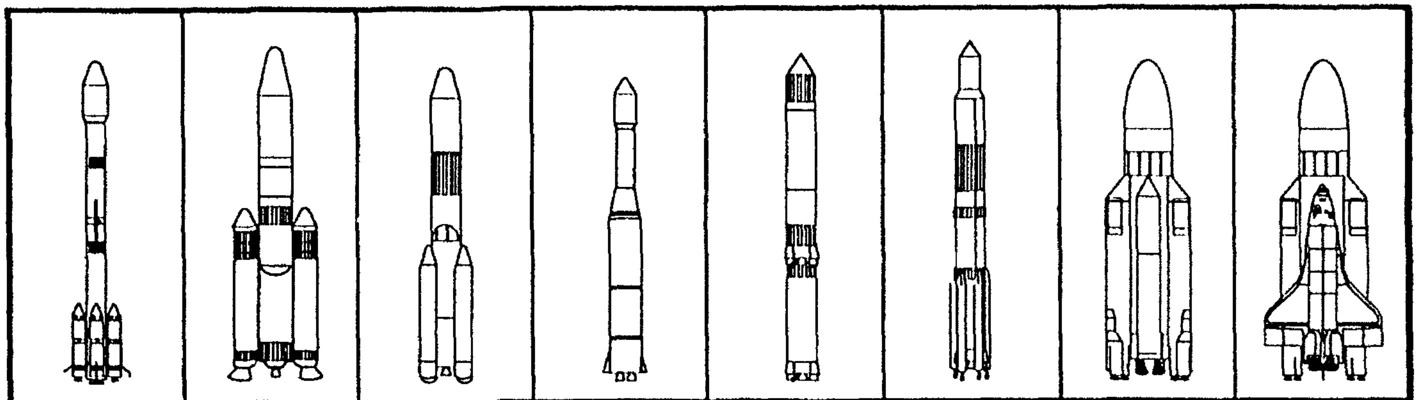
The larger Centaur on Titan IV is capable of delivering 13,000 lb to GEO; however, current structural capabilities permit a weight of only 11,500 lb.

3. Foreign Vehicles

Figure 3-4 shows current and programmed non-USA expendable launch vehicles with greater performance than Delta II. A more comprehensive description of all major foreign launch vehicles is given in Reference 3-1.

a. Europe (Ariane Series)

The first attempt by Europe to develop a European launcher, the Europa Project, was cancelled in 1973 after a series of launch failures. The second attempt, led by the French space agency Centre National d'Etudes Spatiales (CNES), was more successful, resulting in the current Ariane family of vehicles. The Ariane was developed by a European multinational space organization and had its first flight in 1979, with a LEO capability of approximately 9000 lb. Within five years, two upgraded versions, Ariane 2 and Ariane 3, with enhanced performance up to 13,500 lb to LEO, became operational. This first series of three vehicles has been replaced by the Ariane 4 series, which includes six vehicles varying in performance from 10,800 lb to 21,100 lb to LEO and, at the same time, providing an increased length and diameter for the payloads. The core vehicle has three stages, two employing storable propellants and the third being a cryogenic LOX/LH_2



LAUNCH VEHICLE	ARIANE 4	ARIANE 5	H-2	LM-3	ZENIT	PROTON	ENERGIA	ENERGIA BURAN
COUNTRY	FRANCE	FRANCE	JAPAN	CHINA	CIS	CIS	CIS	CIS
PERFORMANCE, lb LEO POLAR LEO DUE EAST 1) GTO GEO	 16,900 21,000 (5.2°) 9,260 (7°) 2) —	 26,400 39,600 15,000 2) —	 14,500 23,000 (30°) 8,800 4,800	 — 11,000 (31.1°) 3,100 (31.1°) 1600 2)	 25,090 (99°) 34,600 (12°) 12,900 (12°) 2) 5,380 2)	 — 44,000 (51.6°) 12,100 2) 4,850 2)	 176,000 194,000 (51.6°) — 40,000 2)	 35,000 66,000 (51.6°) — —
RELIABILITY 3) SUCCESS RATE ~%	87.5	—	—	92.0	92.3	87.7	100.0	100.0
PAYLOAD ACCOMMODATION PAYLOAD DIA ~ft PAYLOAD LENGTH ~ft CONE LENGTH ~ft	 12.0 12.9; 162; 21.2 14.0		 12.1 24.9 14.4	 7.6; 8.9 6.7; 9.7 7.1; 8.3	 11.2 19.2; 27.4 12.7	 10.8; 12.5 13.7; 16.7; 17.4 8.7	 22.0 121.0 —	 14.8 56.0 —
COST/FLT ~$91 M	115–125	105–115	105–125	35	65–70	35–70	~110	—
INITIAL LAUNCH CAPAB	1989	1995	1993	1984	1985	1968	1987	1988
GLOW ~klb	1,040	1,570	582	445	1,012	1,550	5,300	—

1) ~ 28.5°
2) WITH KICK MOTOR OR UPPER STAGE
3) INCLUDES DEVELOPMENT FLIGHTS

Figure 3-4. Foreign launch vehicles.

stage. Additional performance from this baseline Ariane 4 vehicle is obtained by adding strap-on solid, liquid, or combination boosters. The Ariane 44L shown in Figure 3-4 has four liquid strap-ons and has the greatest lift capability of the Ariane 4 series.

The Ariane 5, which is planned to be operational in 1995, is a completely new design intended to provide capabilities similar to the U.S. Titan IV in performance and payload dimensions. It is also designed to be capable of launching the planned European-manned space shuttle vehicle Hermes. The Ariane 5 has a large diameter core stage with cryogenic LOX/LH_2 propellants, a pair of large, segmented, solid rocket strap-ons and an upper stage. The LEO due-east performance of 46,200 lb compares favorably with the U.S. Titan IV with SRMUs. The current Ariane fleet is marketed and operated by a private corporation, Arianespace, and has proven to be a highly successful commercial space launch system.

b. Japan (H-Series)

Japan launched their first space launch vehicle, the N-1, in 1975. The development and launch of this vehicle, derived from a version of the Thor Delta launcher, was accomplished with the assistance of McDonnell Douglas, Rocketdyne, and other United States companies. The N-1 payload capability was quite limited and was replaced by the more capable N-2 vehicle in 1981. In 1986, the N-2 was replaced by the much more capable H-1 launcher, with 7000 lb to LEO due east and 1200 lb to GEO. This vehicle has the same nine solid Castor II strap-ons and a LOX/J-1 (kerosene) first stage as the N-2, a new Japanese-developed LOX/LH_2 second stage, an inertial guidance system, and a third-motor stage solid rocket motor. This vehicle is the Japanese launcher currently in production. Under development at the present time is the larger H-2 vehicle using an entirely Japanese technology. First launch is projected for 1993, although problems with the development of the LE-7 LH2/LOX engine may delay this event. The H-2 consists of a larger-diameter, two-stage cryogenic core with a pair of large strap-on solids. The H-2 is designed with both 13-ft and 15-ft diameter payload fairings. It is expected to represent Japan's entry into the commercial space launch marketplace. The prime contractor for the H-series vehicles is Mitsubishi Heavy Industries.

c. People's Republic of China (Long March Series)

China's first space launch occurred in 1970 when their Long March 1 (also called CZ-1) placed a 600-lb satellite into low-Earth orbit. By 1975, the Long March 2, with significantly greater capability, became operational and formed the basis of the core vehicle for the current Long March fleet. The Long March 2 vehicle with two storable propellant (N_2O_4/UDMH) stages is based on China's ICBM design. With the addition of a LOX/LH_2 third stage, the Long March 3 (CZ-3) achieved GEO capability in 1984. In 1985, China announced that the Long March family would be available commercially. An improved version of this three-stage vehicle, designated CZ-3A, will also be available to the commercial market in 1993, with delivery capability of 5500 lb to GTO, similar to the United States' Atlas II. With the addition of four liquid strap-ons, the vehicle designated Long March 2E (CZ-2E) flew successfully in 1990. By 1995, a version of this vehicle (CZ-2E/HO) with a cryogenic upper stage will have capabilities similar to the United States Titan III.

d. Commonwealth of Independent States (CIS)

The CIS, formed after the breakup of the Soviet Union, has a wide range of expendable launch vehicles at present (see Reference 3-1). Two of the vehicles, the Proton and the Zenit, are available for commercial launches.

1. Proton
The Proton became operational in 1967 and has been a workhorse vehicle for the Soviet's heavier missions to space stations and GEO with more than 300 launches. The four-stage D-1-e version has a payload capability of 44,100 lb to LEO at 51.6 deg inclination and 5000 lb to GEO. This performance is comparable to the U.S. Titan IV/IUS, although the Proton fairings are smaller in diameter than the Titan fairings. Noteworthy is the fact that the fourth stage has a LOX/RP-1 engine which accounts for the lower payload weight to GEO compared to the Titan IV/Centaur. The first three stages employ storable N_2O_4/UDMH propellants and have six, four, and one engine(s) on the first, second, and third stages, respectively. This vehicle was offered as a commercial launcher in 1983.

2. Zenit
The Zenit, first launched in 1985, is a totally new design. There are two versions: a two-stage vehicle for LEO missions and a three-stage vehicle for higher orbits and interplanetary missions. All three stages use LOX/kerosene propellants. The first stage has one turbopump and four combustion chambers, each of which produce very high thrust (1.63 Mlb), comparable to the U.S. Saturn V first stage F-1 engines. The Soviet technology is more advanced than the F-1, developing a chamber pressure of 3500 psia. The third stage is similar to the Proton fourth stage. Overall performance is in the range of the commercial Titan III. The Zenit-3 was selected to support commercial launches from a proposed privately funded spaceport at Queensland, Australia, originally planned to start in 1995; however, this project is now on hold.

B. PARTIALLY RECOVERABLE LAUNCH VEHICLES

1. Domestic (STS Space Shuttle)

The space shuttle is composed of two recoverable solid rocket boosters and an expendable external tank containing liquid propellants for the three main engines mounted on the manned, delta-winged Orbiter. The Orbiter, in addition to providing astronauts' requirements for 7 to 10 days on orbit, contains a cargo bay 60 ft long by 15 ft in diameter. The shuttle payload weight to LEO has been reduced from the projected 65,000 lb to approximately 51,000 lb as a result of design changes following the Challenger accident. There are four Orbiters in the fleet, and all launch from the two launch pads at Kennedy Space Center in Florida.

Current emphasis is on launching payloads requiring manned presence or other unique capabilities offered by the shuttle. Examples would include complicated, exclusive, one-of-a-kind payloads with a need for extensive predeployment checkout, with man available to correct any anomalies; or for space-based experiments employing the Spacelab and astronaut participation.

There are opportunities for having small payloads carried on a shared ride basis with other cargo on the shuttle. Some of these are described briefly at the end of the Launch Systems Integration section (Section IV).

2. Foreign (CIS Energia Buran)

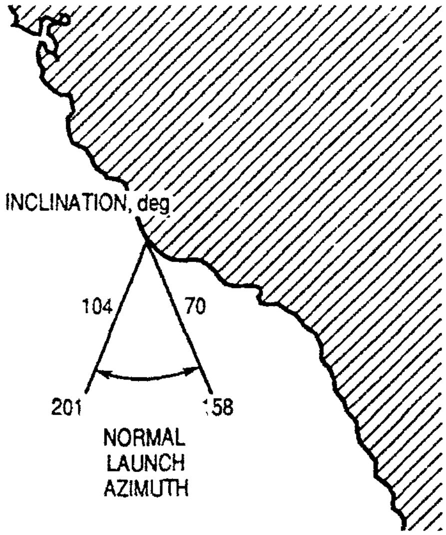
The CIS Energia, which became operational in 1987, was developed to launch a variety of heavy payloads including the Buran Orbiter. The launch vehicle and winged orbiter have some similarities to the U.S. Space Transportation System. However, the Buran lands in an automated mode. The system is also designed such that it can either carry cargo to orbit by replacing the Buran with a cargo carrier side mounted on the propellant tank or perform manned missions using the Buran. The design difference permitting this dual application was to put the four LOX/ LH_2 engines onto the central core tank, instead of on the Orbiter as in the United States design. Another difference is that, whereas the U.S. system employs two solid rocket boosters for its first or booster stage, the Energia has four LOX/kerosene liquid strap-ons, each with the same four large-thrust engines used by the Zenit. The second stage, or core, contains four state-of-the-art LOX/LH_2 high-pressure staged combustion engines. These are ignited in a parallel burn configuration together with the 16 first-stage engines at liftoff.

The reliability of the Energia Buran had been enhanced with significant application of redundancy for flight-critical elements and by providing engine-out capability on the two stages. The Buran, as stated earlier, is capable of automated unmanned landing.

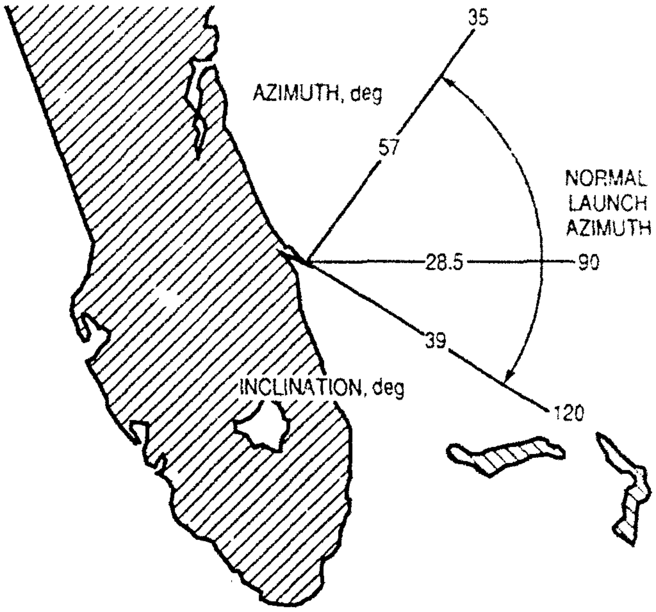
The significant payload capability of 194,000 lb to LEO at an orbit inclination of 51.6 deg is expected to be employed for the Soviet Space Station and interplanetary applications. Provision has been made for further growth to achieve 440,000 lb to LEO by adding four additional strap-ons and other modifications.

C. LAUNCH SITES

This subsection discusses the impact of the launch site on the mission and launch vehicle selection. Launch sites are typically chosen to provide the widest coverage of potential payload orbital inclinations. But, when convenience of location and appropriate range safety factors are considered, frequently they are limited to specific orbital bands of interest. Such is the case for most United States space launches, which use the Eastern Launch Site (ELS) for equatorial and low-inclination orbits and the Western Launch Site (WLS) for near-polar launches. Figure 3-5 portrays the normal band of launch azimuths and orbital inclinations achievable from each of these two launch sites. The ELS normal azimuth launch band, based on range safety considerations, extends from 35 to 120 deg from due north. This would provide orbital inclination ranging from 28.5 to 57 deg, ignoring possible launch vehicle or upper stage delta-velocity-induced orbital inclination changes. Extension of the normal azimuth band may, however, be achievable by launch vehicle dogleg maneuvers with the approval of Range Safety. The WLS normal azimuth launch band extends from 158 to 201 deg from due north. The 158-deg azimuth is based on range safety considerations, while the 201-deg azimuth is based on normal usage and can be extended up to approximately 300 deg before range safety considerations apply. This normal azimuth band



WESTERN LAUNCH SITE
VANDENBERG AFB



EASTERN LAUNCH SITE
KENNEDY SPACE CENTER AND
CAPE CANAVERAL AIR FORCE STATION

Figure 3-5. Normal-band launch azimuths and orbital inclinations.

reflects orbital inclinations ranging from 70 to 104 deg. In addition to these two primary launch sites, the Scout also uses the Wallops Island Flight Facility and the San Marco Range operated by the Italian Government in cooperation with NASA. The air-launched Pegasus can also provide more extensive orbital inclination coverage than fixed launch sites. A strong competitor to U.S. space launchers, the Ariane uses a launch site in Kourou, French Guiana, which is 5.2 deg north of the equator.

The way in which a space launch complex is designed strongly influences its maximum launch rate capability. A complex designed for integrate-on-pad (IOP) assembles the entire launch vehicle and payload on the launch pad, thereby extending on-pad operational times, which results in low turnaround rates. A complex designed for integration-transfer-launch (ITL) uses off-launch-pad facilities to integrate the vehicle and the payload and transfers the entire assembly to the launch pad. This significantly reduces the on-pad time and provides the potential for much higher turnaround rates and, consequently, higher launch rate capabilities. The more complex the launch vehicle (for example, those using strap-on solids and upper stages), the greater the benefit derived from an ITL complex. Most U.S. launch complexes, however, are of the IOP variety. To further minimize the launch vehicle turnaround time, especially at an ITL complex, an off-line facility should be provided to integrate, encapsulate, and test the upper stage, spacecraft, and fairing prior to stacking on the launch vehicle. This approach minimizes time spent on the launch pad. The maximum annual ELS and WLS facility launch rate capacities for Delta, Atlas, and Titan vehicles are presented in Table 3-1, with the specific launch complex identified for each

Table 3-1. Launch Rate Capacity Circa 1995

Launch Vehicle	Launch Complex		Launch Capacity	
	ELS	WLS	ELS	WLS
Titan IV/Centaur, IUS, NUS	LC-40/41	-	6	-
Titan IV/NUS	-	SLC-4E	-	2
Titan IV Centaur		SLC-7		*
Titan II	-	SLC-4W	-	3
Atlas II and IIAS	LC-36 A/B	SLC-3	9	*
Delta II	LC-17 A/B	SLC-2	12	6

*No current funding for WLS capability

vehicle. The vehicle maximum launch capability cannot exceed this maximum facility capacity, which is partly a function of the facility manpower availability. Launch vehicle processing timelines, which have direct bearing on the launch rate capability, vary considerably for individual launch vehicles and specific launch sites. For example, a Titan IV/Centaur takes approximately 175 days to process at ELS, with roughly 80 days on the pad—considering typical payload testing. Higher launch rates may be achieved by adding crew shifts and also by duplication of critical in-line processing facilities.

The Delta and Atlas processing times are direct functions of the number of crew shifts used. The Delta processing time from start to finish can be reduced from 89 to 45 to 30 days, respectively, when the crew shifts increase from one to two to three. The Atlas processing time correspondingly reduces from 77 to 38 to 26 days for increasing crew shifts from one to two to three. The respective facility processing capacity and manpower capabilities, however, may further restrict the overall launch capacity. Any extended on-pad payload-processing requirements can extend the pad time, further restricting turnaround times and maximum launch-rate capabilities. Some off-line payload-processing facilities exist at the ELS to alleviate on-pad processing time.

The Spacecraft Payload Integration Facility (SPIF) at CCAFS is used to process shuttle and Titan payloads, the Satellite Assembly Building (SAB) to process DSCS nonhazardous payloads, and a separate facility to process Navstar payloads. In addition, the ELS payload processing capabilities at the NASA Kennedy Space Center (KSC) or the offsite Astrotech facility can be used on an as-needed (and as available) basis for processing commercial payloads. At WLS, however, payload processing is predominantly accomplished on the launch vehicle at the pad.

D. PAYLOAD COMPARTMENT ENVIRONMENTS

The payload is required to endure a variety of changing environmental conditions at launch vehicle ignition, liftoff, ascent, stage/fairing separation, and finally, spacecraft separation at orbit destination. This section characterizes typical critical environmental parameters that form the basis for payload design and test criteria. They include thermal, pressure, dynamics (acoustics, vibration, and shock), electromagnetic compatibility/electromagnetic interference, and contamination. Specific values for a given payload/launch vehicle combination should be obtained through reference to the appropriate user guides and implementing the integration process described in Section IV in cooperation with the launch vehicle operators.

1. Thermal

Typical thermal conditions mandate prelaunch payload fairing (PLF) temperature and humidity control capability along with satellite power dissipation and other environmental conditions. Worst-case thermal analysis includes the effects of fairing ascent temperatures, free molecular heating, vehicle cryogenic impact through the interface, and parking orbit heating effects. STS-launched systems must also include conditions in the Orbiter with the cargo-bay doors opened.

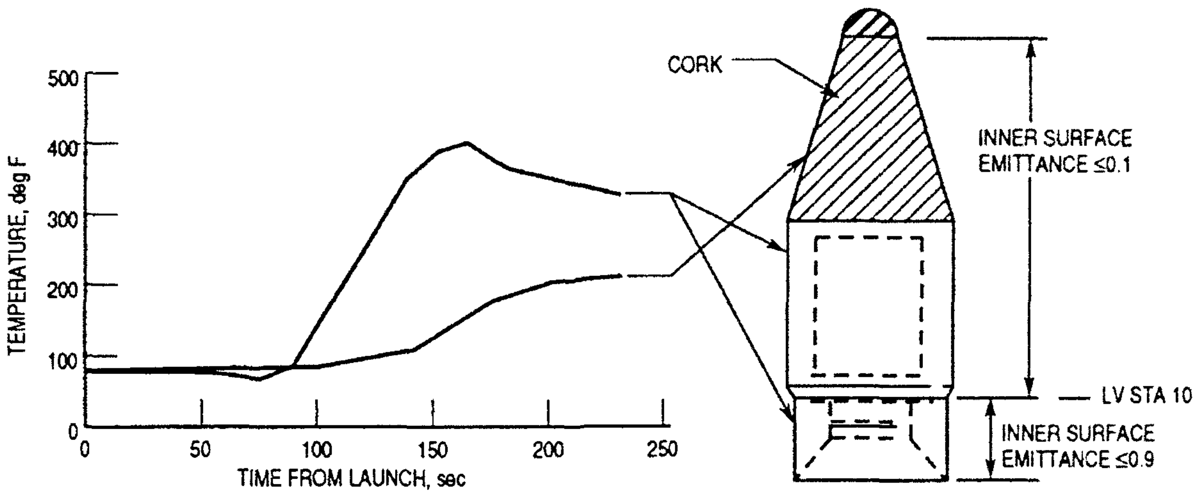
Prelaunch air conditioning is used to control temperatures of encapsulated hardware on the launch pad. This impacts the design in areas involving flow rate, duct access, intermittent interruptions of conditioned flows, and termination of air flow minutes before liftoff. Typical temperature control can range from 4–10°C minimum levels to a maximum of 25–43°C. Relative humidity control of the air ranges from 28–60 percent, which is sometimes expressed in dew point temperature (e.g., Titan II at 26°F). Static pressure, flow rates, noise level, and air velocities need to be resolved between spacecraft and launch vehicle contractors. During countdown, satellite power dissipation may be necessary.

Spacecraft must be designed to take account of payload-fairing internal surface temperature and thermal flux during ascent. Fairing temperatures can range up to 200–300°C at approximately launch +3 min and are highest in the nose section. Insulation materials are sometimes used on conical or nose sections, limiting heat emission concerns from these areas. Figure 3-6 illustrates an Atlas II case. Ariane 4 lists the fairing temperature at 61°C at launch +200 sec. with a maximum flux of 500 watts/m^2. Thermal profiles should be obtained from the fairing contractor along with emissivity values. After the fairing is ejected, free molecular heating (FMH) occurs with typical peak values around 1135 watts/m^2 (excluding solar and albedo radiation).

For a short time ($\leq$ 1 sec), plume heating from retro motors or spin motors can occur in predeployment maneuvers. These conditions are application-dependent, and handbook values are often only specified at the separation plane. Cryogenic upper-stage cooling and parking-orbit heating impacts need to be determined on an individual mission basis.

2. Pressure

During transportation and ascent, static pressure changes occur where black boxes and structures with cavities (such as tubes and honeycomb) are required to vent and/or survive pressure differential deflections. Figure 3-7 shows various profiles for payload compartment pressure. Maximum rate and time duration are critical parameters to consider for venting the component.



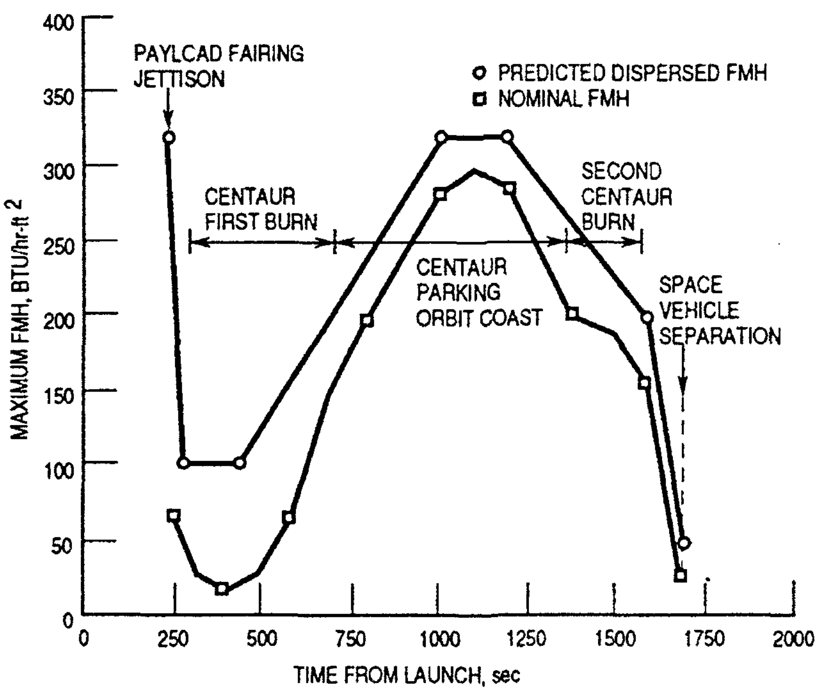


Figure 3-6. Sample fairing temperature history and maximum space vehicle heating rates.

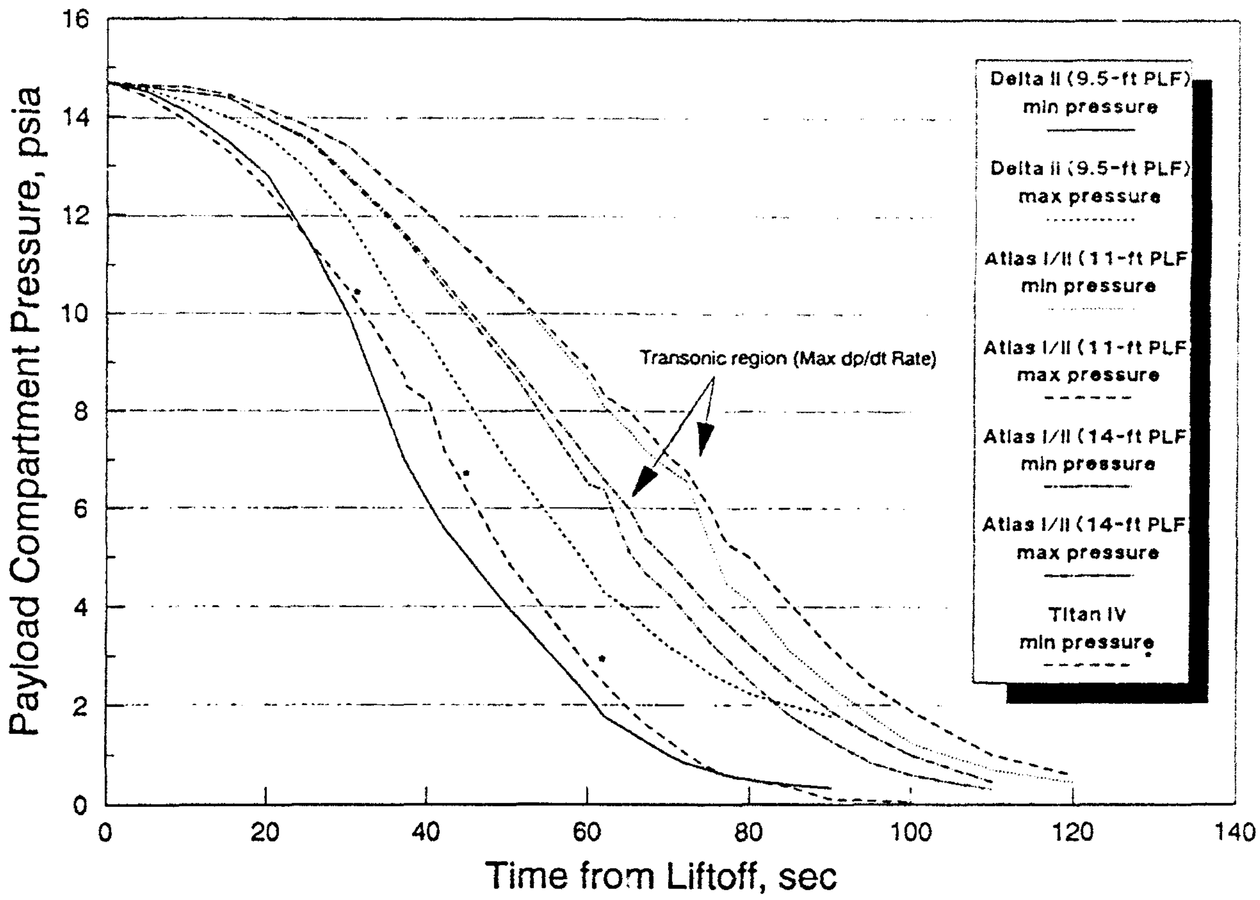


Figure 3-7. Payload compartment pressure profiles.

Values depend on the ratio of payload volume to fairing volume and other conditions. Consideration also should be given to the impact on hardware of pressure changes occurring during air transportation.

Typical fairing post-jettison pressure is < 0.48 Pa (0.01 psf). Usually free molecular heating limits the payload fairing jettison time.

3. Dynamics

The reader is cautioned to treat the plots in this section as representative only and to consult with the appropriate program office for specific launch vehicle dynamic data.

a. Acoustics

Liftoff and ascent subject the spacecraft to acoustic levels within the payload fairing that vary significantly in amplitude (Figure 3-8) during the ascent. The overall sound pressure level

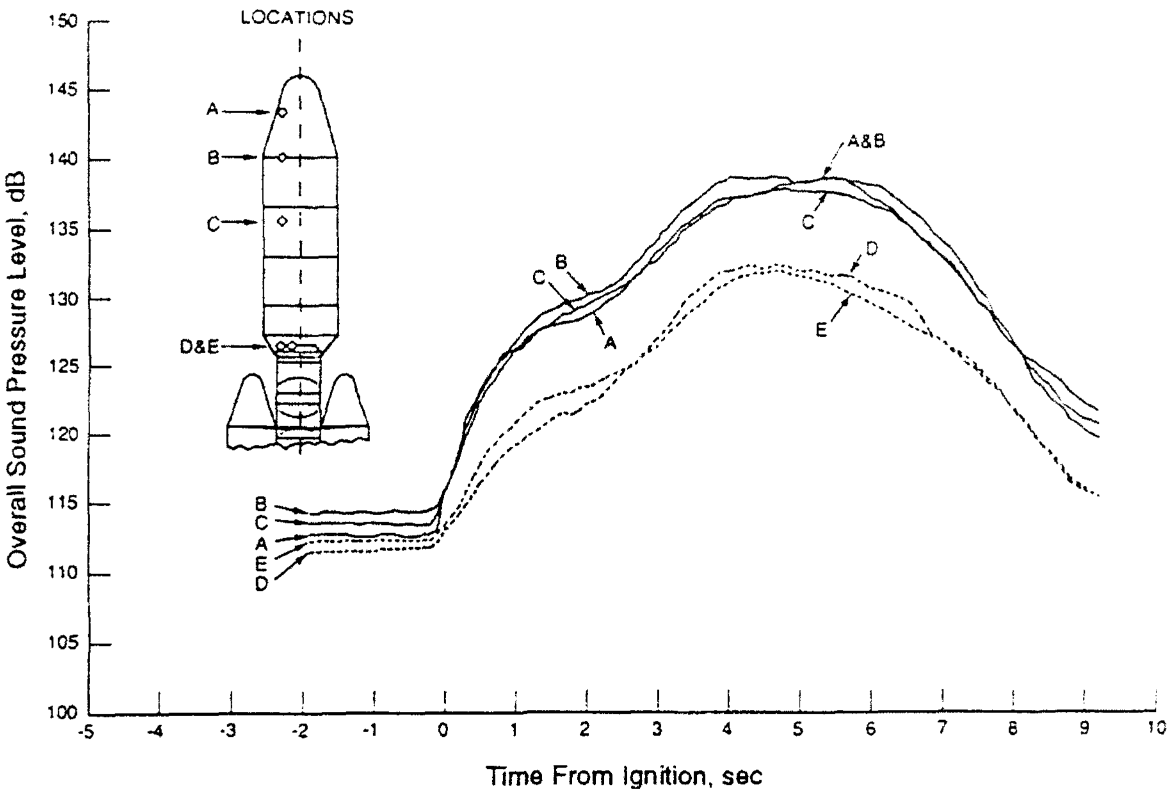


Figure 3-8. Acoustic amplitude levels during liftoff.

during liftoff is a function of the total engine mechanical power modified by the launch pad configuration, distance of the spacecraft from the exhaust, and structural details of the payload fairing. Such factors as covered exhaust ducts, the type of exhaust turning vanes, and use of water injection into the exhaust stream modify the acoustic levels. During transonic/maximum dynamic pressure portions of ascent, the spacecraft acoustic levels depend upon the nature of the aerodynamic flow field (shock, turbulent boundary layer) and the transmission of sound through the fairing. Duration of significant energy varies with each launch vehicle but is less than 10 sec during liftoff and as much as 45 sec during the transonic/maximum dynamic pressure period. The acoustic energy inside the fairing can be mitigated by acoustic blankets or increased by the presence of vents. An additional influence is generated by the proximity of spacecraft surfaces to the payload fairing wall. As the surface gets closer to the fairing wall, the acoustic level is increased as shown in Figure 3-9.

Acoustic specifications are usually given in terms of an empty fairing and the user must make the necessary adjustment for the spacecraft spacing. The resulting acoustic energy interacts with the spacecraft and results in the vibration response of surface and components. In addition, vibration of the fairing can be transmitted to the spacecraft via structural connection. Spacecraft acoustic test levels are usually given in terms of one-third octave frequency bands and the resulting overall level. A range of overall and one-third octave band sound pressure levels for small,

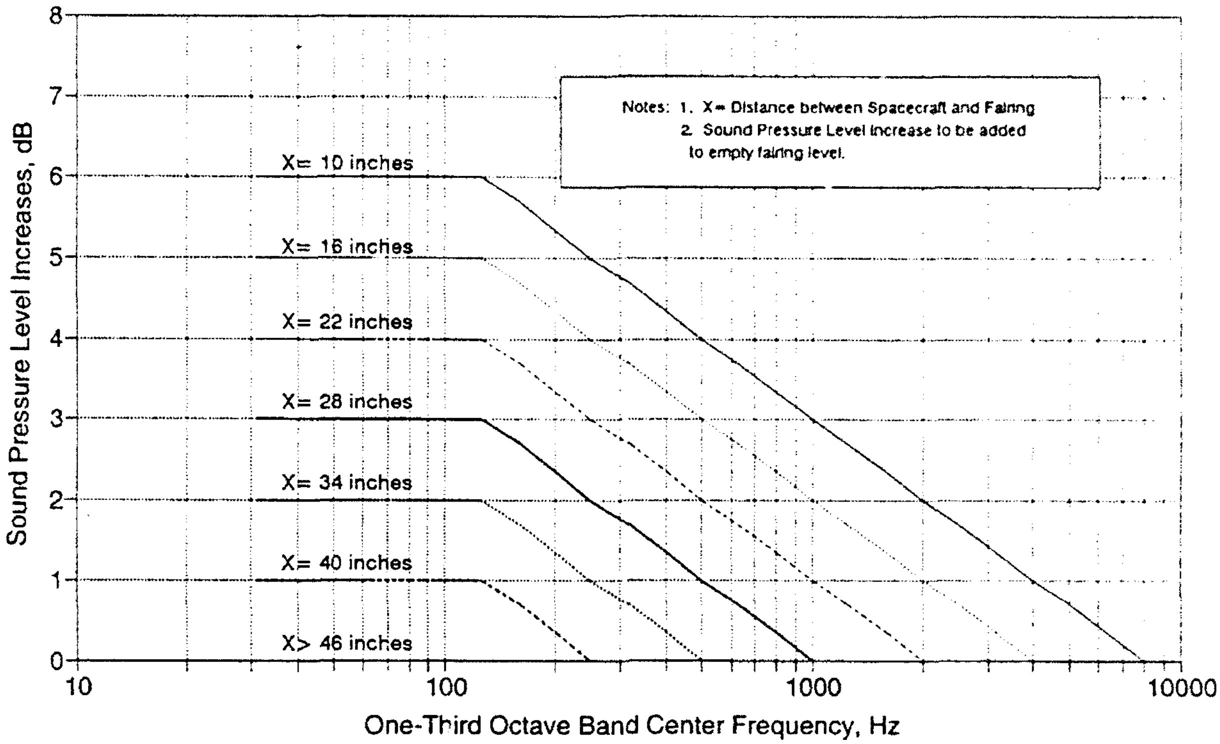


Figure 3-9. Spacecraft fill factor. Adjustment to be added to empty fairing sound pressure level.

medium, and large launch vehicles are shown in Figures 3-10, 3-11, and 3-12. Spacecraft are initially designed for such levels, and their capability to withstand the acoustic levels is usually verified by ground testing in a reverberant chamber.

b. Vibration

Vibration response at any location on the launch vehicle is the sum of responses due to direct acoustic field interaction with the structure and vibration energy transmitted through the structure from other locations. In many cases, the primary source is the direct acoustic field. Usually, flight data are obtained at specific locations for verification of component design/test requirements and at the launch vehicle/spacecraft interface for use in validating interface requirements. The values are expressed in terms of an acceleration spectral density (g^2/Hz) value over a frequency range of 20 to 2000 Hz with an overall level in terms of root mean square acceleration (grms). Typical ranges are given in Figure 3-13. These values represent the maximum vibration caused by the launch vehicle at the spacecraft interface. At locations on the spacecraft distant from the interface, the vibration response will be due to the direct acoustic field interacting with the spacecraft structure. It is for this reason that interface vibration levels are of limited application, and care in using these values as a test requirement for the spacecraft must be exercised. When spacecraft are small and will not couple well with the acoustic field, the spacecraft may be

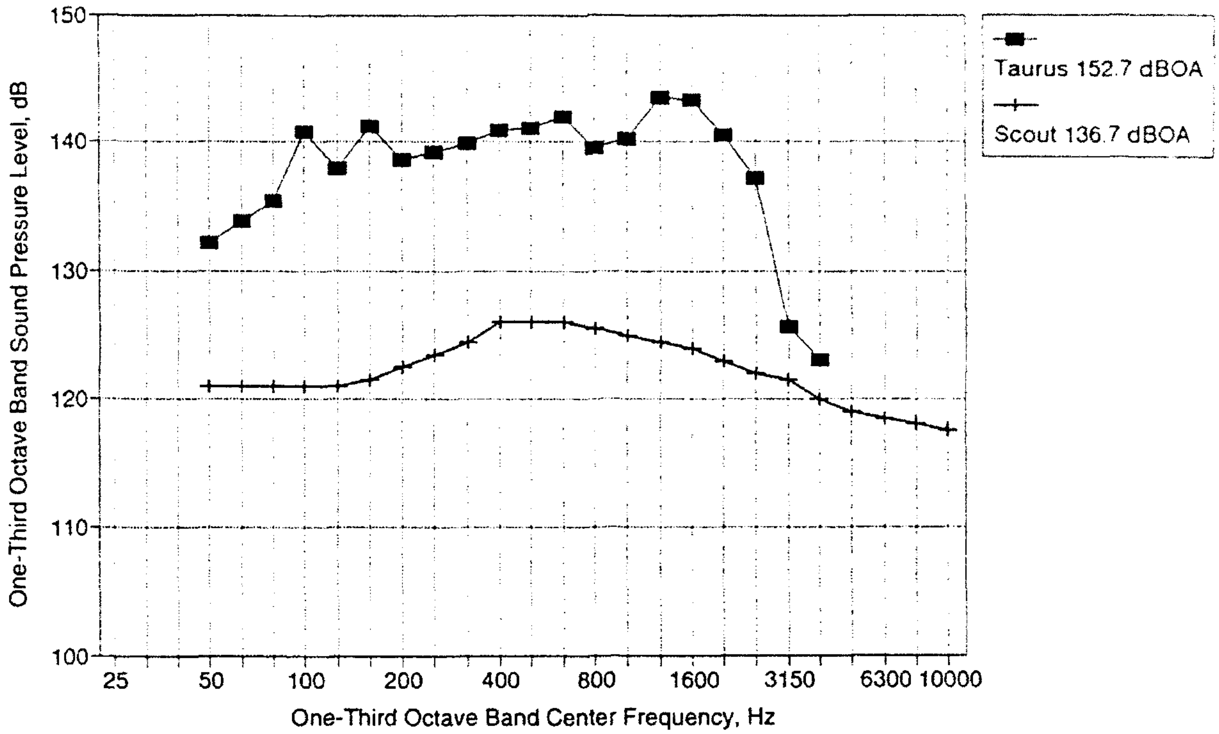


Figure 3-10. Acoustic environments for small launch vehicles. Levels may be less for fairings having incorporated noise mitigation measures.

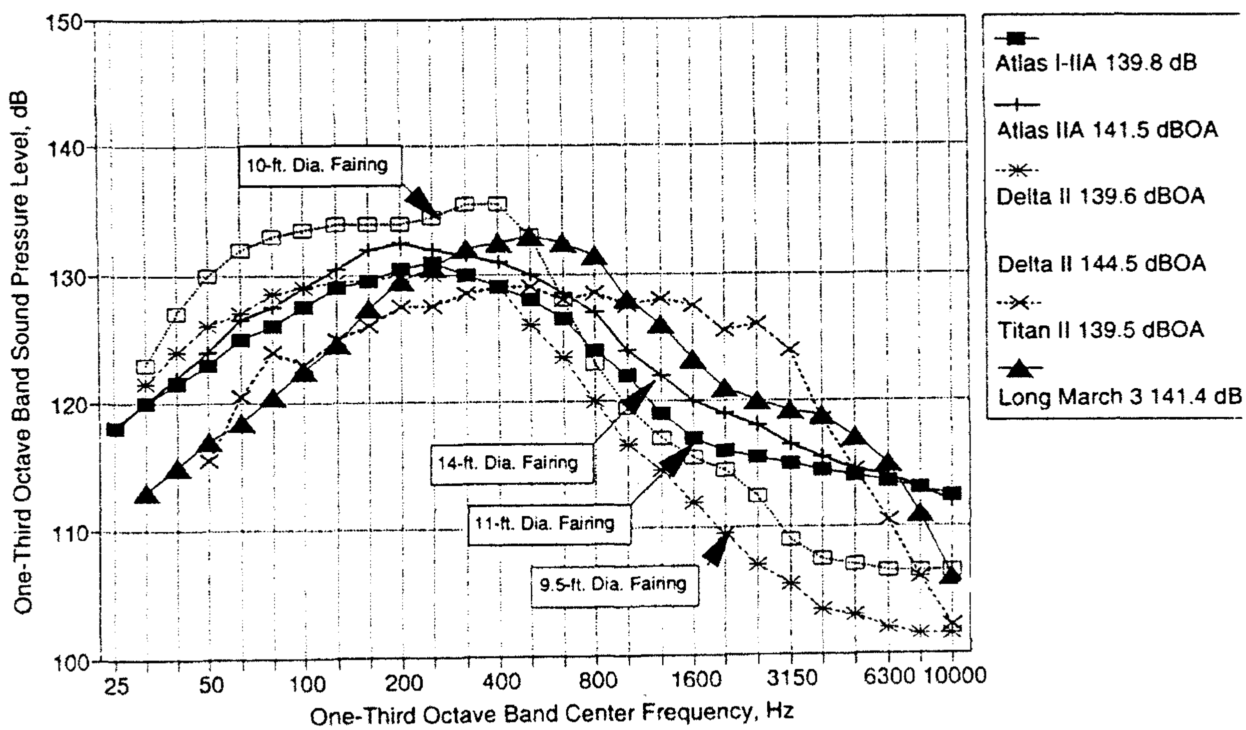


Figure 3-11. Acoustic environments for medium launch vehicles. Levels may be less for fairings having incorporated noise mitigation measures.

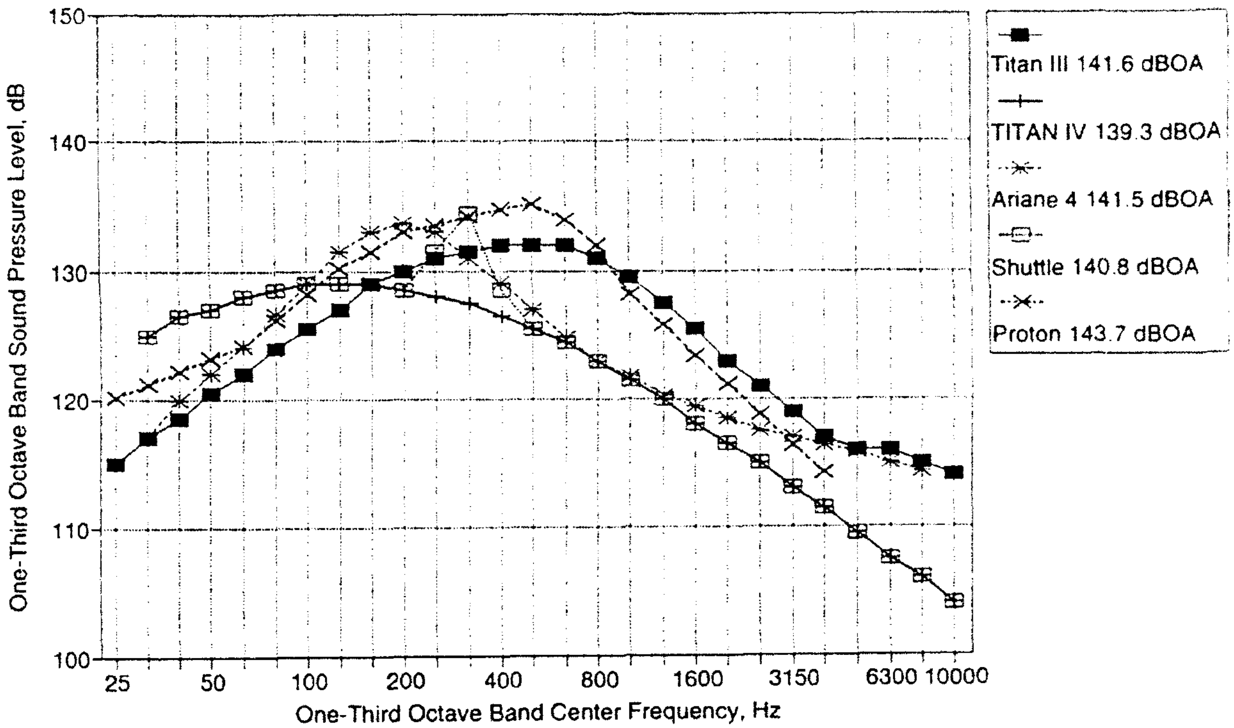


Figure 3-12. Acoustic environments for large launch vehicles. Levels may be less for fairings having incorporated noise mitigation measures.

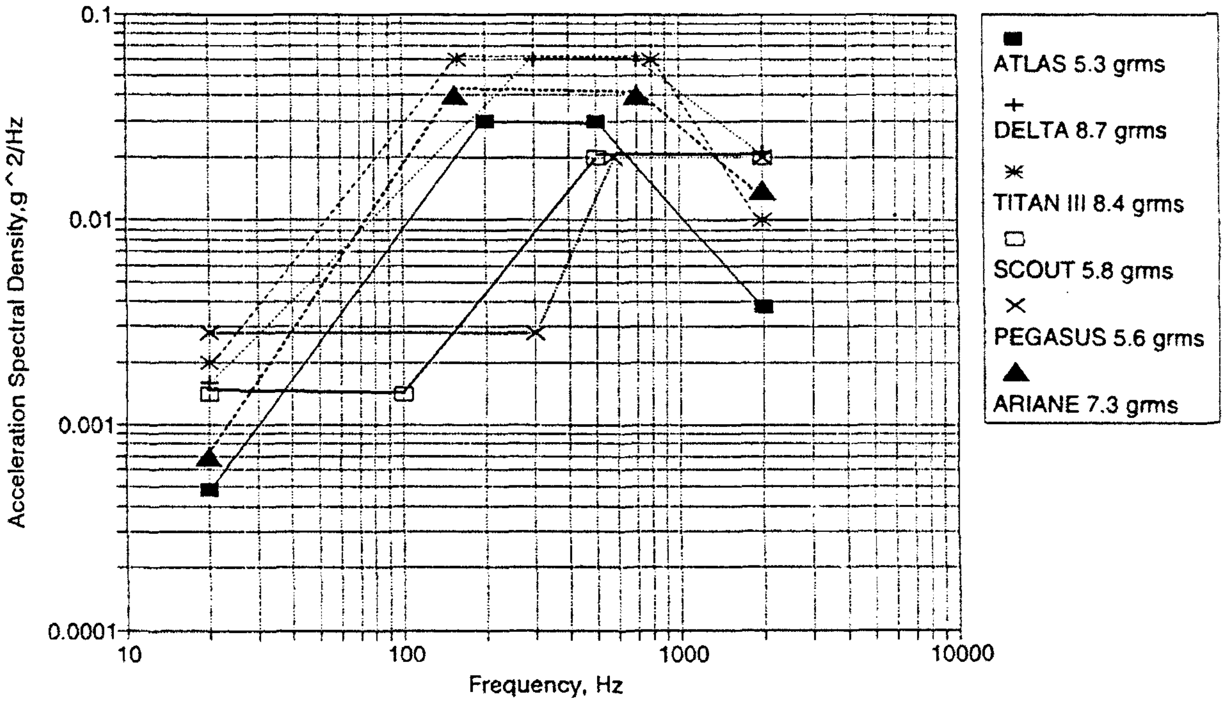


Figure 3-13. Vibration environments for launch vehicles. Levels at the spacecraft/launch vehicle interface.

mounted on vibration shakers and tested to the interface vibration spectrum. However, for large spacecraft, such testing is usually accomplished by the acoustic test.

c. Shock

During ascent, spacecraft are subjected to self-induced shock events as well as those associated with events on the booster, such as SRM separation, fairing separation, and upper stage separation from the core vehicle. Such shocks occur due to the sudden release of energy stored in the bolts, nuts, and clampbands that connect structure and the energy related to fracturing structural joints. Shock interface levels are specified in two ways: (1) launch-vehicle-induced shock at the launch vehicle/spacecraft interface, and (2) spacecraft-induced shock at the launch vehicle/ spacecraft interface. Shock levels are expressed in terms of a spectrum representing an envelope of the positive and negative peak acceleration response for a single-degree-of-freedom system assuming damping of 0.05 percent ($Q = 10$). It is not uncommon, however, for shock levels to be expressed as a single-response level representing the highest level as depicted in Figure 3-14. Evaluation of component capability to function after exposure to shock is accomplished by testing the affected components to a shock-response spectrum that duplicates the response within specified test tolerances. In general, the launch-vehicle-induced shock on the spacecraft is less than that imposed by the spacecraft upon the launch vehicle. The need for launch vehicle equipment to withstand the typical spacecraft-induced shock levels shown in Figure 3-14 depends upon whether the launch vehicle must execute a collision avoidance maneuver after separation.

The prediction of shock levels is not a well established science, being highly dependent upon structural parameters and joint design; thus, the characterization of levels is often established by test. Verification of spacecraft capability to withstand its own induced shock is demonstrated by separation tests. Verification of launch vehicle capability is usually performed by testing at the component level.

4. Electromagnetic Compatibility/Electromagnetic Interference (EMC/EMI)

Electromagnetic compatibility covers radio frequency (RF) radiation-induced phenomena as well as electrical bonding, grounding, isolation, shielding, and lightning issues. Criteria are based on spacecraft self- or multiple-vehicle compatibility, and compatibility with various facilities. Levels and margins are set from safety, noninterference, and susceptibility requirements.

Table 3-2 illustrates some of the payload EMC requirements. For DOD programs, test conditions and levels are usually defined in military standard documents* with adjustments or tailoring defined by agencies procuring the launch vehicle and/or the spacecraft. The specific electromagnetic environment of the launch facility, the launch vehicle transmitters for ranging and data, and satellite electric field radiation can impact the levels in the MIL-STDs.

* Military standards are developed and published under the authority of various Government agencies and cover a wide range of technical subjects. They may be found through bibliographical search in major library databases.

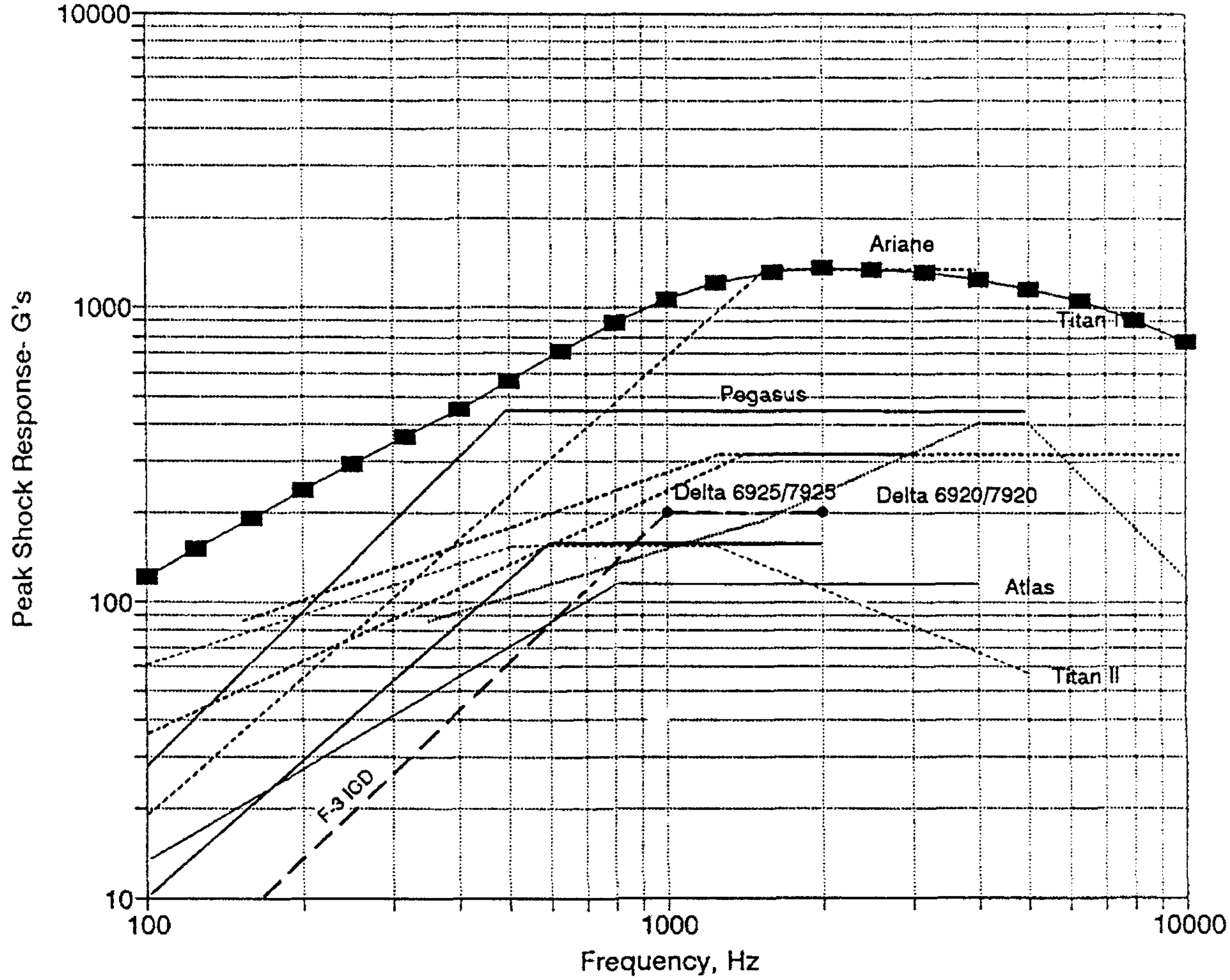


Figure 3–14. Shock environments from launch vehicle separation.

Electrical bonding between the satellite and launch vehicle separation plane is specified to some milliohm level (typically 10 mΩ) after the vehicles are mated. The path is required through a conductive interface "frayed" surface. Electrostatic discharge (ESD) protection is often specified for both vehicles. Static charge can be built up from friction resulting from launch-induced vibration wherein two surfaces rub together and by the charge built up on the outer surfaces when the vehicle flies through clouds bearing ice crystals. This charging phenomenon is called triboelectric electrification. Charging also occurs on upper stages and spacecraft in parking orbit to final on-orbit conditions. This involves vehicle interaction with the space plasma. Discharging interference has been particularly observed from night to day transitions for spacecraft in geosynchronous orbits and from passage of spacecraft in low-Earth orbits at high inclinations through the aurora in electrical storms. Discharging can interfere with RF and sensitive digital circuits.

Table 3-2. General EMC Requirements

Type	Test	ELV	Component	Instrument	S/C	Comments
CE	DC power leads	X	Sb, Rb, R	Sb, Rb, R	Sb	
CE	AC power leads		Sb, Rb	Sb, Rb	Sb	
CE	Spikes on Orbiter DC power lines		Sb	Sb	Sb	
CE	Spikes on Orbiter AC power lines		Sb	Sb	Sb	
CE	Antenna terminals	X	R	-	-	
RE	Magnetic field		-	-	Sb	
RE	AC magnetic field	X	Rb, R	Rb, R	Sd	
RE	E-fields	X	Rb, R	Rb, R	Sd, Rb, R	
RE	Payload transmitters	X	-	-	Sd	If on during launch
RE	Spurious (transmitter antenna)	X	-	Rb, R	-	
CS	Power line	X	Rb, R	Rb, R	Rb	
CS	Intermodulation products	X	Rb, R	-	-	
CS	Signal rejection	X	Rb, R	-	-	
CS	Cross modulation	X	Rb, R	-	-	
CS	Power line transients	X	Rb, R	Rb, R	Rb	
RS	E-field	X	Rb, R	Rb, R	Rb, R	
RS	Compatibility with Orbiter		-	-	Rb	
RS	Orbiter unintentional E-field		-	-	Rb	
RS	Magnetic-field susceptibility	X	Rb, R	Rb, R	Rb, R	
ESD	Triboelectric discharge	X	-	-	R	Checks digital units
CCST	CW response	X	R	-	-	Critical circuit
CCST	Pulse response	X	R	-	-	Susceptibility

R = Test to ensure reliable operation/compatibility
Sb = Test items which interface with Orbiter power
Rb = Test to ensure reliable interface
Sd = Test items operating near Orbiter

List applicable to shuttle, X = items applicable to expendable launch vehicles. See NASA-Goddard, GEVS-SE, "General Environmental Verification Specification for STS and ELV." Defined per MIL-STD-1541A.

Lightning is a special case and requires design provisions in the launch vehicle, PLF, and umbilical cable to prevent damage from a direct hit to the payload or the launch vehicle. Numerous electrostatic discharge specifications exist on subjects from the handling of piece parts to protecting the stacked vehicle from lightning strikes on the launch pad.

Vehicles can have single or multipoint grounds. Care needs to be taken not to use the structure as an intentional current load path. Typically, a 1-megΩ resistance between primary power input leads and the structural ground needs to exist. Grounding between the facility, the electronic ground support equipment (EGSE), and the vehicle needs to be handled with care; special constraints may be established on a mission-specific basis.

5. Contamination

Typically, a top-level contamination control plan is required which levies requirements on all contractors' launch vehicle and spacecraft processing equipment. The requirements can vary, depending on the satellite's mission equipment, from optical systems to communication systems. General requirements will involve cleanliness on interior surfaces to level 600 (visibly clean) for particles and less than 1.0 $\mu g/cm^2$ of nonvolatile residue (see MIL-STD-1246B). Vehicles are either under a controlled transportation environment or are uncovered in a Class 100,000 or better facility area. During operations in transit to and at the pad, the encapsulated hardware is maintained with a positive pressure, purge-conditioned air, gaseous nitrogen (GN_2). Special nonvolatile residue deposit prevention is specified for ascent and collision avoidance maneuvers after separation. Selection of material for flight and ground support equipment (GSE) has to consider outgassing, wear products, or flaking. This is why zinc, cadmium, and electroplated tin finishes are banned from flight hardware or from hardware that comes in contact with the flight hardware. Material contamination control plans are used to specify criteria in this area.

E. RELIABILITY ASSESSMENT

Space system planners have to account for the possibility of launch failures no matter how mature their launch vehicles. After discussing typical causes of failure, this subsection treats risk reduction approaches and other planning implications of imperfect launch reliability.

While conducting hundreds of flights, the U.S. space program has encountered dozens of launch failures. This experience exhibits three main features:

1. Failures occur more frequently early in booster development and operations; during a learning interval of 20 or so launches, the failure rate drops and then levels off, typically at a value ranging from 3 to 5 percent. (Figures 3-1 and 3-4 show reliability values for specific launch vehicles.)

2. About half of mature-vehicle failures are caused by human error, with the others attributed to design flaws.

3. Failures most frequently originate in the propulsion subsystem (55 percent) and avionics (16 percent), followed by separation devices (7 percent), structures (3 percent), and other less-frequent contributors. Approximately 9 percent are failures of unknown origin.

The probability of launch failure is often the chief threat to a space mission. Strategies for reducing the risk of mission loss combine measures that increase redundancy, robustness, and attention to detail. Redundancy enables systems to be fault-tolerant. Such approaches as triple-string redundancy can, in principle, prevent essentially all avionics-related mission failures. Providing the capability for engine-out operation (i.e., the mission can be successfully completed despite shutdown of one engine) is a form of redundancy that has the potential to cut propulsion-related failures about in half by making the propulsion system tolerant of "fail-to-function" faults. However, propulsion explosions and fires can have effects too severe to permit alleviation by redundant systems.

Robustness—that is, the provision of generous design margins—promotes increased reliability through fault avoidance. An important example of launch vehicle robustness is the use of liquid propulsion engines operated well below the thrust levels demonstrated during their qualification tests.

Detailed design and process audits can provide increased reliability. These audits verify important facets of launch vehicle hardware and processing design, including the derivation of component, subsystem, and interface specifications; the identification of critical components and confirmation of their design margins; and the existence of suitable tests at the system and integrated vehicle level that encompass, as far as practical, all flight events and conditions. Such audits also validate, for the as-built flight hardware, the resolution of all anomalies encountered during manufacturing, acceptance testing, and preflight processing.

Risk-reduction choices take many forms. A space program that launches a constellation of satellites may simply budget for one or more spare spacecraft to replace those lost, or it may choose to invest in booster improvements to obtain higher launch reliability through increased redundancy and robustness. In contrast, a program conducting a single launch of a one-of-a-kind spacecraft may seek higher launch reliability by sponsoring special reviews or other systems engineering enhancements intended to uncover design flaws and to prevent or correct human errors before a launch. The mix of risk-reduction measures chosen for a particular space system is based on affordability and cost/benefit perceptions. Large uncertainties hamper efforts to quantify the benefit; i.e., the avoided failure cost, from any candidate risk-reduction measure. Estimates are often inflated for the reliability improvement offered by a specific measure, whereas the total cost of a launch failure may be understated. Costs of multiprogram inactivity and reprogramming result when delays are caused by a major failure. If these indirect costs are included, the total cost of a failure can be many times larger than the cost of the lost hardware and the recovery effort.

When estimating the practical launch rate capacity of launch systems, launch planners must account for failure-related launch schedule delays. Occasionally, failures force delays because of damage to the launch complex. In any event, manufacturing and launch processing activities usually are halted after a failure and are not resumed until the cause is established and corrective actions are instituted. This "standdown" in activity can last for several months to a year or more.

The "availability" of a launch system is the fraction of time it is engaged in normal launch activity in contrast to being unavailable while recovering from a failure. The availability achieved depends on the launch failure percentage, the standdown time, and the ratio of the launch rate to the maximum (i.e., failure-free) launch capacity. Decreasing any of these three controlling parameters will increase availability. Near-perfect availability is sometimes essential. It can be achieved by proceeding with launches without waiting for a failure to be understood and corrected. This approach, termed "flying through failure," incurs the risk of repeated failures that have a common cause. For this reason, the approach is rarely used.

A satellite program can alleviate the penalty (schedule and cost) of waiting for the chosen launch vehicle to return to service following a failure. This can be done by integrating the satellite with more than one type of launch vehicle, if a second vehicle type has sufficient payload capability. The option of a backup launch vehicle can be evaluated through cost/benefit trade studies early in the payload integration cycle.

F. LAUNCH SYSTEM COSTS

This subsection discusses the influence of cost on launch vehicle selection. The cost of the space transportation system selected to deliver the payload to its desired orbital destination is of major concern. Careful consideration of all the program requirements, including total number of payloads, permissible payloads per launch, orbit destination, schedule limitations and spacecraft costs; and pertinent launch vehicle considerations, including whether the system is a new, modified, or existing launch vehicle system, its operational capability, its payload delivery capability, and launch transportation system costs, should enable determination of the most cost-effective space transportation system for a particular payload program.

1. Life-Cycle Costs

The launch vehicle selection process should involve a comparison of competitive space transportation systems to determine which one results in the lowest total program life-cycle cost. The life-cycle cost of a space transportation system encompasses all costs incurred from initiation of launch system development or modification (for new or modified systems), or start of operations for an unmodified existing system, through completion of the desired system operational phase-to-payload deployment. This includes both the one-time-only costs (nonrecurring costs) and the continuing-per-flight costs (recurring costs). The nonrecurring costs can be quite large for an entirely new launch system They include the launch vehicle development costs, the required production facility costs (either new or modified facilities), the launch facility development and construction costs, the manufacturing cost of any launch vehicle reusable hardware (i.e., the space shuttle), and the cost associated with the first-time payload-to-launch-vehicle integration. For an existing vehicle, the nonrecurring costs can be small to zero, depending on the extent of any vehicle or facility modifications required and whether an entirely new or recurring payload is being flown. The recurring costs are defined in the following paragraph; these recurring costs, though usually much lower for a single flight than the nonrecurring costs, become the determining factor in the life-cycle costs when many flights are considered.

2. Typical Launch Vehicle Cost or Price

The simplest life-cycle cost comparisons involve operational launch vehicle systems where vehicle plus production and operational facilities already exist, and additional capabilities are not necessary. Only recurring operational costs need be considered; that is, the cost per launch and the number of launches required. The operational cost per launch should include the cost of the following elements:

- Booster and upper stage hardware, including structure, avionics, and rocket engines.
- Payload fairing.
- Launch operations, including launch services, propellants, and government services if necessary (to account for transportation and facility-use reimbursements).
- Other government costs,* including government technical and program management procurement, contract administration, logistics, general systems engineering and integration (GSE&I), independent validation and verification (IV&V), and travel.
- Range operations, including tracking, telemetry, and communications (TT&C) and range safety.

The resultant cost (or price) per launch, while including most of the above five elements, varies depending on whether it is a government-controlled launch (i.e., controlled by DOD or NASA) or a commercial launch. The cost of a government-controlled buy and launch considers the total lot buy, the production and launch rates, prior unit buys and assurances of further buys, unique specifications, and special requirements. Depending on the above considerations, the user cost can vary considerably. A commercial buy reflects a contractor price based on his total anticipatory buy of materials and expected launch rates. The commercial contractor takes a risk when he sets a fixed price. A bad cost estimate, in which the price is set optimistically low, can result in a financial loss; setting the price too conservatively high can result in loss of sales. With these qualifications in mind, rough order recurring costs for several existing launch vehicle systems are shown in Figures 3–15 and 3–16 in terms of total cost per launch and cost per pound delivered to orbit. The data for Figure 3-15 were extracted from Reference 3-1. For high-fidelity cost data, the analyst is urged to contact the appropriate launch vehicle SPOs.

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*Not required for commercial launches.

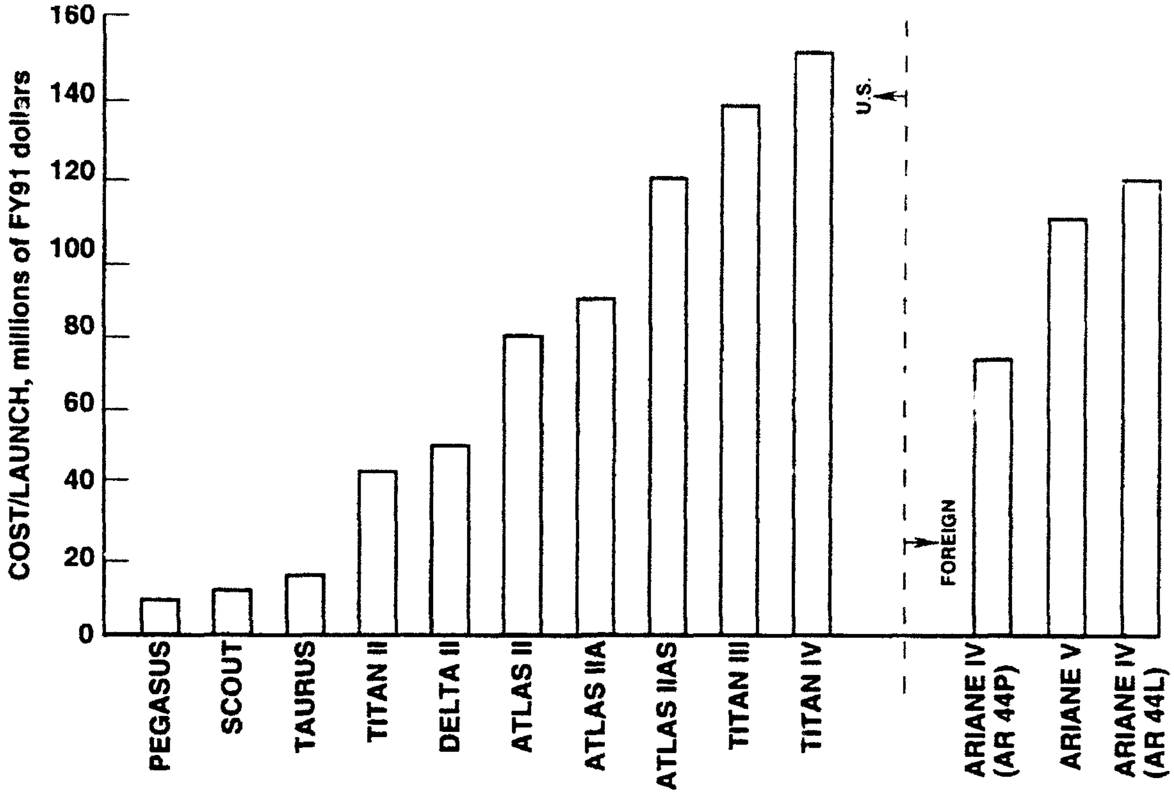

Figure 3-15. Total recurring cost per launch.

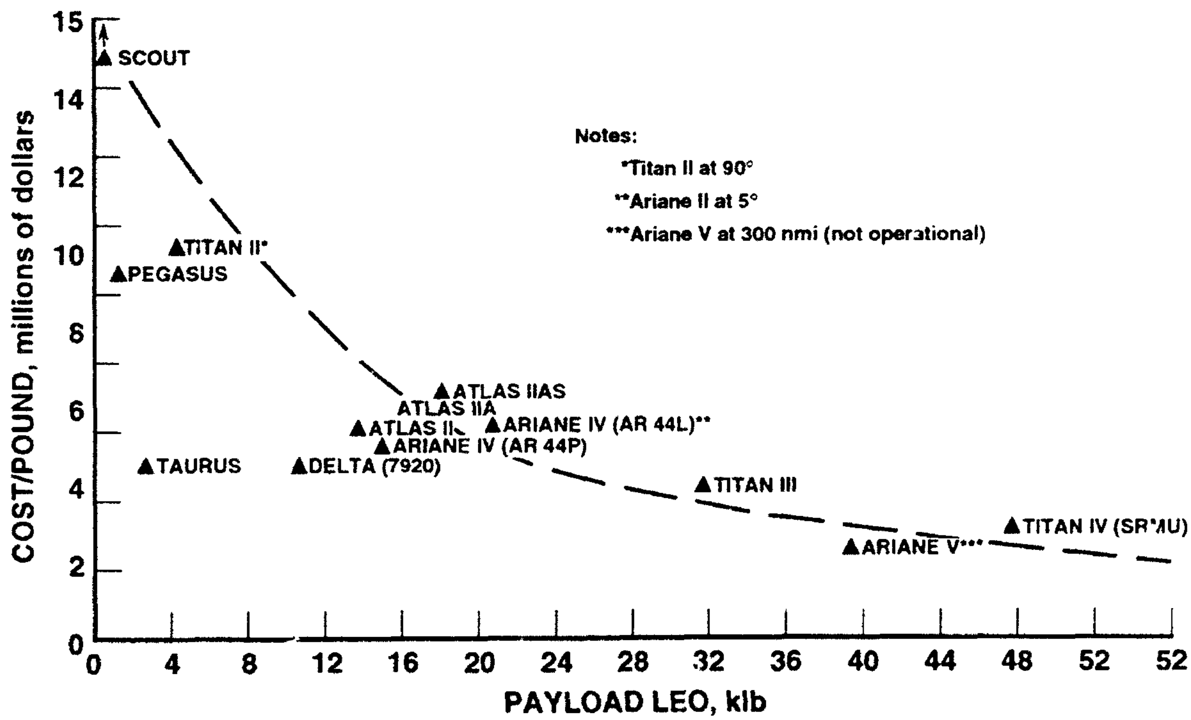

Figure 3-16. Recurring cost per pound (100 nmi at 28.5 deg).

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3-1. J. Isakowitz, "International Reference Guide to Space Launch Systems," *AIAA*, Washington, DC (1991).

3-2. NASA/Goddard, GEVS-SE, General Environmental Specification for STS and ELV.

3-3. MIL-STD-1541A, Electromagnetic Capabilities Requirements for Space.

3-4. MIL-STD-461C, Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference

3-5. MIL-STD-462, Measurement of Electronic Interference Characteristics.

IV. LAUNCH SYSTEM INTEGRATION (LSI)

Leslie E. Lundquist

This section is concerned with the process that establishes the interface requirements and subsequent verification of these interfaces between all elements involved in the ground processing and launch of a spacecraft. The major elements involved include the space vehicle, the launch vehicle, and their respective ground support systems. This integration process covers thousands of items, many of which could cause a mission failure if executed incorrectly and the error not discovered in a subsequent test. The words "attention to detail" and "discipline" are particularly applicable to this area.

Examples of the interfaces between the major elements and the necessary procedures include the following:

- Physical (Mechanical and Electrical)
- Functional
  - Loads
  - Mass Properties
- Avionics
  - Discretes
  - EMI/EMC
- Environmental
  - Thermal
  - Contamination
  - Vibration and Shock
  - Acoustics
- Mission and Performance Analysis
  - Launch Window
  - Targeting Guidance Software
  - Mission Orbits
  - Injection Accuracy, Accelerations, etc.
- Ground Systems SV Handling, Hoisting, etc.
- Aerospace Ground Equipment
  - Test Equipment
  - Fueling Requirements
  - Communications
- Interface Verifications
  - Requirements Matrix
  - Verification Methods Description
  - Test Definition

The basic function of Launch System Integration is the management, engineering, and validation of all the system interfaces to assure timely, complete ground processing and a successful mission. There are many participating organizations involved in the integration process. For Air Force satellites and launch vehicles, the integration responsibility resides within the program management function. This usually involves dedicated people in the satellite and launch vehicle system program offices (SPOs).* For commercial launches, the satellite and launch vehicle prime contractors take on this management function. Additional support is provided by a number of organizations, typically including:

*A SPO is the group or office charged with the overall responsibility for the development and operation of a space or launch system.

- Satellite and launch vehicle contractors
- Launch site organizations
- Satellite control facilities/operators
- Safety organizations
- Security support

The primary "hands-on" work in the entire process is accomplished by the satellite and launch vehicle contractors who produce and maintain the Interface Control Document (ICD), which defines all the interface requirements and the methods that will be used to verify that they have been satisfied. The verification analyses or test results are maintained in lower tier documents. The contractors also maintain traceability of the verification of every requirement in the ICD.

With better understanding on what launch system integration is and who the main participants are, the following subsections will describe the time phasing of activities, a management approach found to get the task done efficiently, and more details of the important technical interfaces contained in a typical ICD.

A. INTEGRATION PROCESS

The integration process is the implementation of a series of activities that are initiated in the early stages of system design and continued throughout the design, development, and acquisition phases. A typical process flow for a first-time integration is divided into phases as illustrated in Figure 4-1, which is representative of the Titan program.

Three phases of integration activities are illustrated, stretching over three to four years. Phase 0 follows the development of a mission/satellite concept, and the tentative selection of a launch vehicle, and starts with go-ahead authority to proceed with the program. This "Predefinition" period includes tasks that define the specifics of the satellite mission, perform preliminary feasibility assessments, identify major incompatibilities, and produce a draft set of interface specifications. Also typically generated are a reference ascent trajectory, preliminary design loads, organizational responsibilities, and overall development schedules.

Phase I, "Definition," is where the detailed system engineering takes place to produce the specific hardware and flight (functional) specifications that enable procurement go-ahead. The activities include negotiations to finalize solutions to incompatibilities, firming up flight/trajectory planning, performing design analyses (e.g., loads, thermal, stability, contamination, and others), and designing mission-peculiar software. Mission-peculiar hardware kits and facility modifications are also defined, and long-lead hardware items are initiated as needed to support the program schedule.

The products of the above efforts are documented in a comprehensive Interface Specification Document, including operating/functional requirements, in addition to the obvious hardware interfaces. Also in Phase I, items of analysis, test, and software that are critical to mission success are identified to enable future verification to be made that they have been correctly accomplished. Such verification tasks are also typically documented in a section of the interface specification.

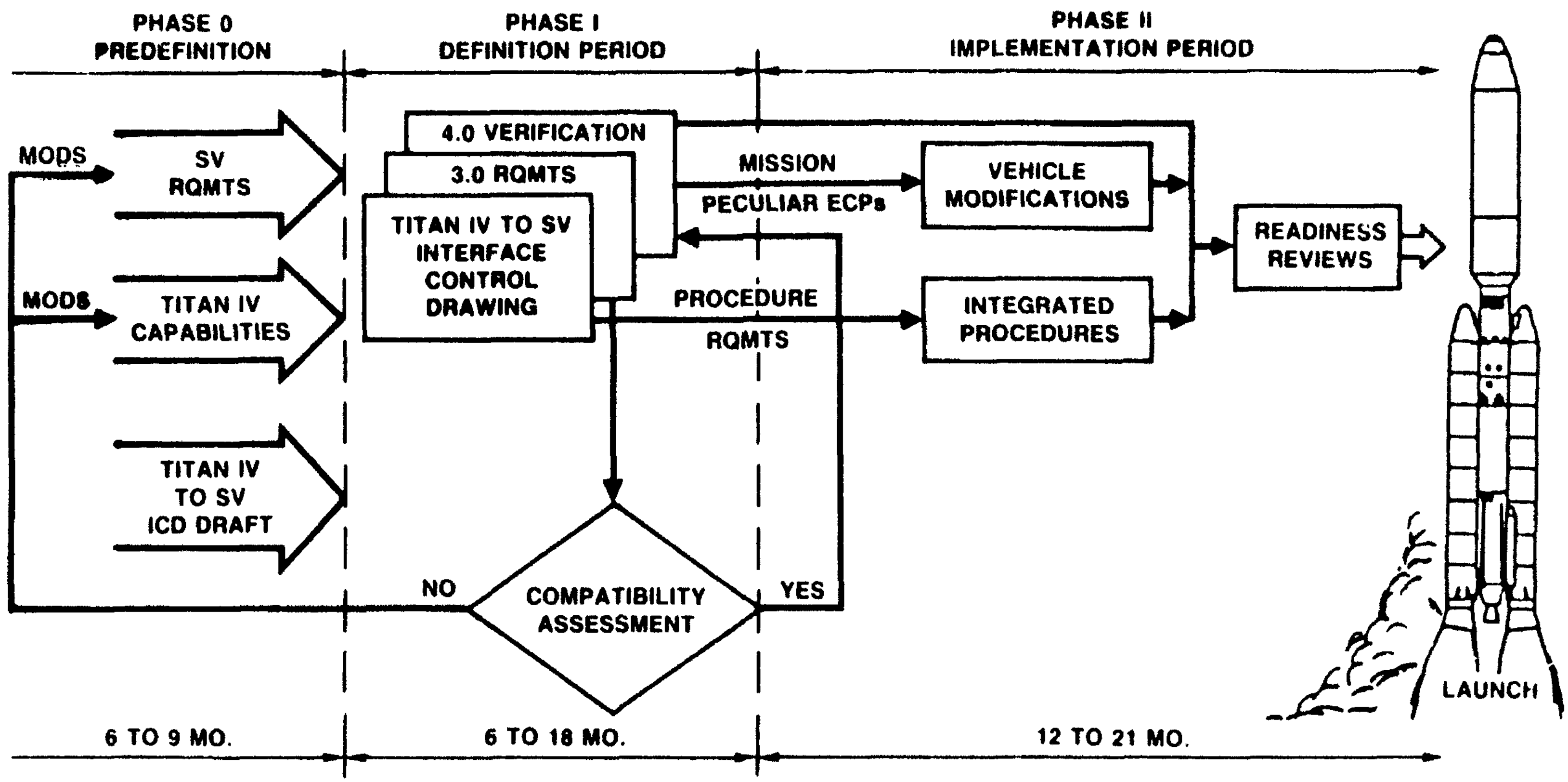

Figure 4-1. First-time payload integration process overview.

Phase II, "Implementation," is when the plans and designs are carried forward to produce the hardware, software, procedures, tests, and other activities to culminate in an actual launch operation. Vehicle-peculiar modifications are made, typically to payload fairing (PLF) access doors, umbilical cables, air handling ducts, and other items as required. Detailed procedures must be written and checked in anticipation of satellite mating, PLF installation, combined system tests, and countdown operations. Significant efforts are also expended on the verification tasks and readiness reviews (often by independent agencies/contractors) to assure the success of the mission to the maximum extent possible.

There is an iterative character to the entire process providing increasing refinement and added detail of the analyses as the program progresses. Note the "compatibility assessment" feedback loop that can result in modifications to the launch and/or the satellite vehicle.

It is clear from the foregoing that the integration process is a lengthy one, at least for a new, first-time integration. For such systems, it is absolutely essential to identify all long-lead items required, whether flight or ground systems hardware or software. For repeat launches of the same satellite and launch vehicle, it still requires approximately 18 months to accomplish all the integration.

B. IMPLEMENTATION

Experience has shown that a good management approach for achieving efficient integration is to establish a set of "working groups" to resolve difficulties incurred while developing the interface designs, operating procedures, and test requirements.

A typical working group structure is illustrated in Figure 4-2. The responsibilities of each working group are described below, and additional detail on the technical issues involved in system integration is provided.

1. Structural/Mechanical/Environmental Working Group

This group is predominantly concerned with:

- Loads and Dynamics
  - Mathematical models of the launch vehicle, space vehicle, and forcing functions for critical events
  - Loads for all structural elements
  - Load cycles—preliminary/design/validation
  - Uncertainty factors
- Mass Properties
  - Defining and complying with launch vehicle constraints
- Vibro-acoustics and Shock
  - Defining and analyzing impacts
- Thermal
  - Constraints and interactions

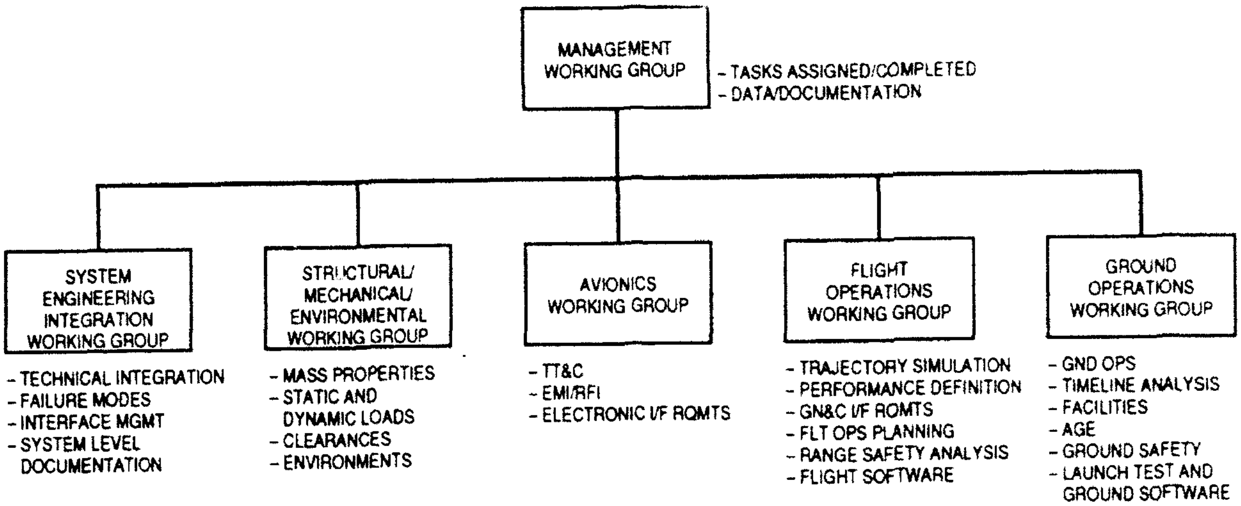

Figure 4-2. The working group structure.

The Structural/Mechanical/Environmental Working Group coordinates and manages these areas. The dynamic loads analyses are usually the most costly of the activities because of their iterative nature, complexity of the dynamic models, and need for extensive use of computer facilities.

Typically, three load cycles are planned, but often additional cycles are required. This is due to changes that occur late in the development effort, often driven by the concurrent development of the space vehicle and a new or modified launch vehicle. Model changes come about due to design changes or when testing does not corroborate the analytic models. Forcing functions may also change as actual flight data show differences from early estimates. Initial designs and analyses apply an "uncertainty factor" (typically at least 1.5) on top of the familiar safety factors in order to encompass the effects of the model and forcing function changes that are almost certain to occur.

The liftoff and staging events are usually critical (worst-case loads) and drive the design of the spacecraft. Critical vibro-acoustic environments tend to exist at launch, where the reflected energy from the surrounding terrain and launch facilities also impinge on the payload fairing, and during ascent at the high dynamic pressure/transonic buffet regions of flight. The separation event is usually an interface shock problem where structural release from the launch vehicle may be accomplished by explosive bolts or equivalent devices. Environmental issues were discussed in detail in Section III.

Thermal constraints and interactions arise from operation of heat-generating spacecraft components, aerodynamic heating of the fairing during ascent, Earth and solar radiation, and free molecular heating after fairing separation. The working group will generally address all thermal control requirements from prelaunch operations through spacecraft separation from the upper stage. During ground processing, the spacecraft will undergo function testing that may require air conditioning systems or limits on the operating time of certain components to avoid overheating. After the spacecraft is mated to the launch vehicle and enclosed in the payload fairing, the temperature and flow rate of the conditioned air needed to cool the spacecraft and protect it from fluctuations in outside air temperature and humidity must be monitored. Some sensors may also require loading of cryogenic coolants shortly before launch. The temperature response of spacecraft components to the radiant heating from the hot payload fairing after lift-off, followed by the heating effects of the sun, Earth, and the upper atmosphere after fairing jettison, must be predicted by analysis. The results of that analysis will determine the need for trajectory or mission timeline changes, attitude constraints, battery sizing to power heaters, and limits on transfer orbit eclipse durations.

2. Avionics Working Group

The avionics working group is typically responsible for:

- Avionics Interfaces
  - Command and Control
  - Telemetry
  - Power
  - Connectors, Cables, and Switching
- Electromagnetic Compatibility (EMC)
  - Radiated Fields—Effects/Control
  - Conducted Emissions
  - Power Quality
- Flight Control
  - Launch Vehicle Attitude Control
  - Control Software/Stability

The flight control function is included here for convenience. For some programs, it might be delegated to a separate software working group. The attitude control software often must be tailored to be stable when coupled to a spacecraft with unique flexibility and/or center of gravity characteristics.

Communication links must be acquired and readied to support the training, test, launch, and flight operations. These links may be very complex and extensive in order to support the numerous channels and links to the command centers at the launch base, satellite control, and contractor facilities. Telemetry is channeled to the users, while command and control links are provided to respond to the data received and contingencies that might occur. In addition, the terminal facilities and computer software that reduce and display the data must be defined, developed, checked out, and ready to support training as well as actual operations.

3. Flight Operations Working Group

This group is responsible for:

- Ascent Trajectory Development
  - Shaping within loads and thermal constraints
  - Performance
  - Propellant loadings and margins
  - Guidance interfaces and associated software development and validations
- Range Safety
  - Trajectory dispersions (due to winds and malfunctions)

"Day-of-launch" constraints and placards (criteria for holding or continuing operations) must be developed to control or curtail activities where contingencies or environmental factors intervene. For example, excessive wind shears aloft (measured by weather balloon releases) near the high dynamic pressure portion of the ascent may cause excessive loads or loss of control authority. The launch may be delayed or terminated in such situations.

Range safety organizations at the launch bases need to evaluate the proposed ascent profile with dispersions of all kinds. Data (including wind effects on pieces of a destroyed vehicle) to develop "Impact Limit Lines," "Destruct Lines," and safety criteria must be supplied to support their activities. This is necessary to assure the safety of populated and sensitive ground areas in the event of a guidance or control malfunction.

4. Ground Operations Working Group

This group is concerned with numerous aspects of the prelaunch activities, including:

- Transport from Factory to Launch Site
  - Associated containers and handling equipment
- Launch Base Flow
  - Receiving/handling
  - Final assembly (of space vehicle), test planning, associated unique test equipment/facilities, and test operations
- Hazardous Preparations
  - Propellant loading
  - Explosive device installations
  - Transport to launch pad
  - Mating of launch vehicle and space vehicle
  - Final interface checkouts, combined system tests, and preparations
  - Fairing installation
- Planning and Procedure Documents for All of the Above

Because of the high degree of coordination and planning involving contractors, supporting agencies, and other working groups (e.g., the Range/Host, communications agents, use of preparation buildings/facilities, handling/hoisting equipment, interface test planning with the launch vehicle contractors, ground safety agents), this effort must start essentially at program "go-ahead."

The associated documentation to identify, approve, and coordinate support requirements, action directives, and implementation procedures is very extensive. Much engineering effort is expended in the development, review, validation, and approval cycles necessary to assure efficient, successful, and safe ground flow/preparation activities.

5. Management and System Engineering Integration Working Groups

These groups are responsible for ensuring that task assignments are carried out and for control and scheduling of the activities of other working groups. They are concerned with making compromise decisions between the diverse disciplines and organizations.

C. MANNED LAUNCH SYSTEMS

The presence of astronauts in the NASA Space Transportation System (STS) program generates special launch systems integration problems. Space shuttle integration is far more complex than any expendable launch vehicle integration, since it involves manned interfaces, unique/extra safety, extended time on orbit for the Orbiter, and recovery and abort operations. Use of a flight crew for checkout, deployment, and numerous optional contingency actions (including hardware retrieval, extra-vehicular activity, and reentry and landing) requires extensive planning, training, and rehearsal. Figure 4-3 provides an overview of the many and diverse aspects of the STS integration activities.

A great deal of formal documentation is generated and falls into the following groups:

- Functional requirements
- Implementation planning
- Action directives
- Detailed procedures

Although much more complex, the integration process for the STS launch system is fundamentally similar to the expendable launch vehicle case. Figure 4-4 illustrates typical schedule, activity requirements, and the numerous reviews/milestones that pace the process.

System integration activities continue throughout the entire program. They may go on for four or five years prior to first launch and continue with subsequent launches for the entire life of the program

An early key STS program milestone is the Payload Integration Plan (PIP). When signed, it is effectively the contract with NASA covering how and when a specific payload will be flown, the major interface agreements, and the services to be supplied. The PIP covers:

- Roles and responsibilities
- Scope of agreements and requirements
- Baselines, guidelines, and constraints for the integration process
- Schedule of integration activities

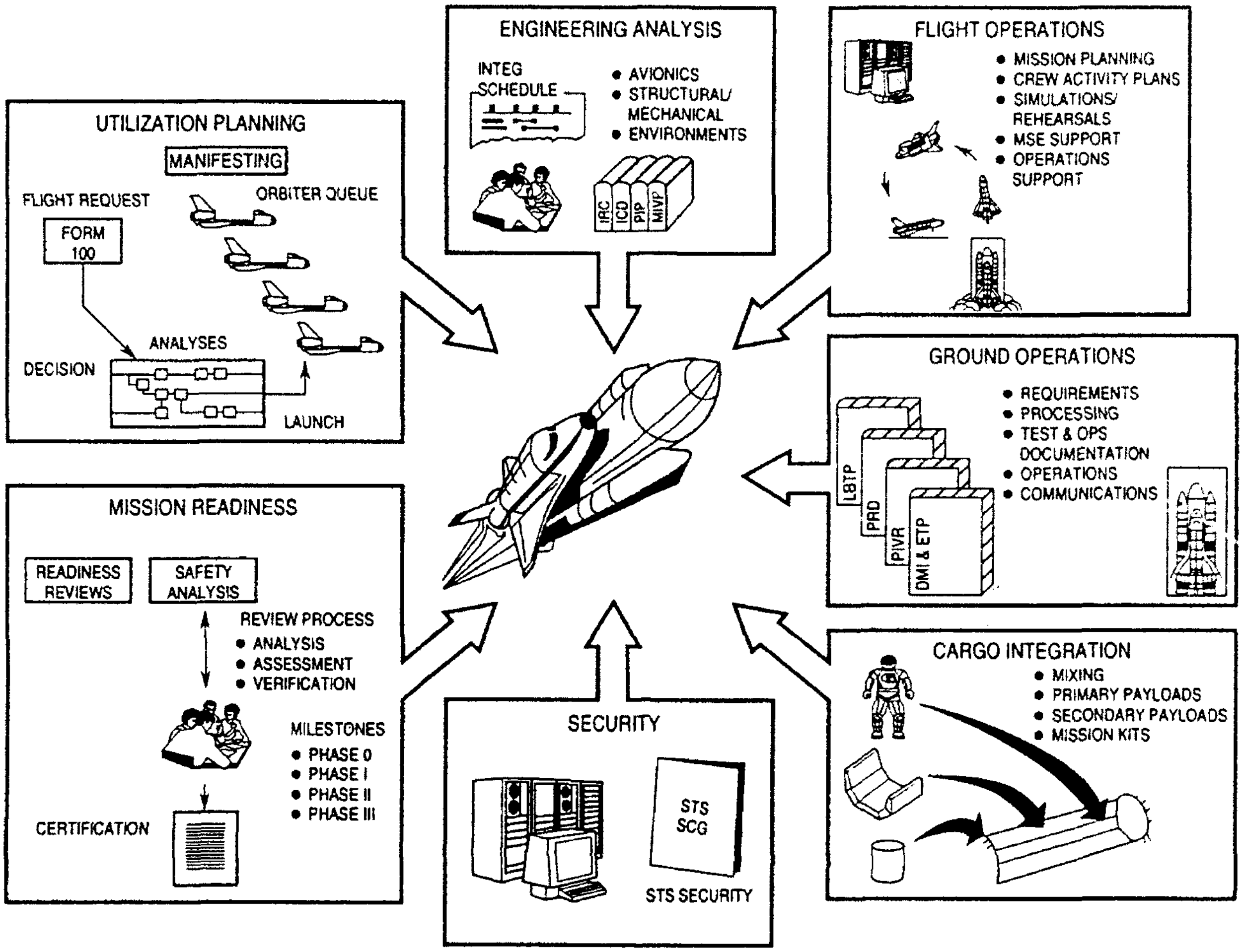

Figure 4-3. STS integration and operations.

Subsequently, PIP annexes are generated, providing much more extensive and detailed information. These documents are published and controlled by NASA. However, much of the content material is written by the program offices and the associated working group representatives, based upon NASA-specified formats. The standard set of PIP annexes as defined by NASA are:

- Payload data
- Flight planning
- Flight operations support
- Command and control data
- Payload operations control center
- Launch site support plan
- Interface verification
- Intravehicular activity
- Extravehicular activity
- Orbiter crew compartment
- Training

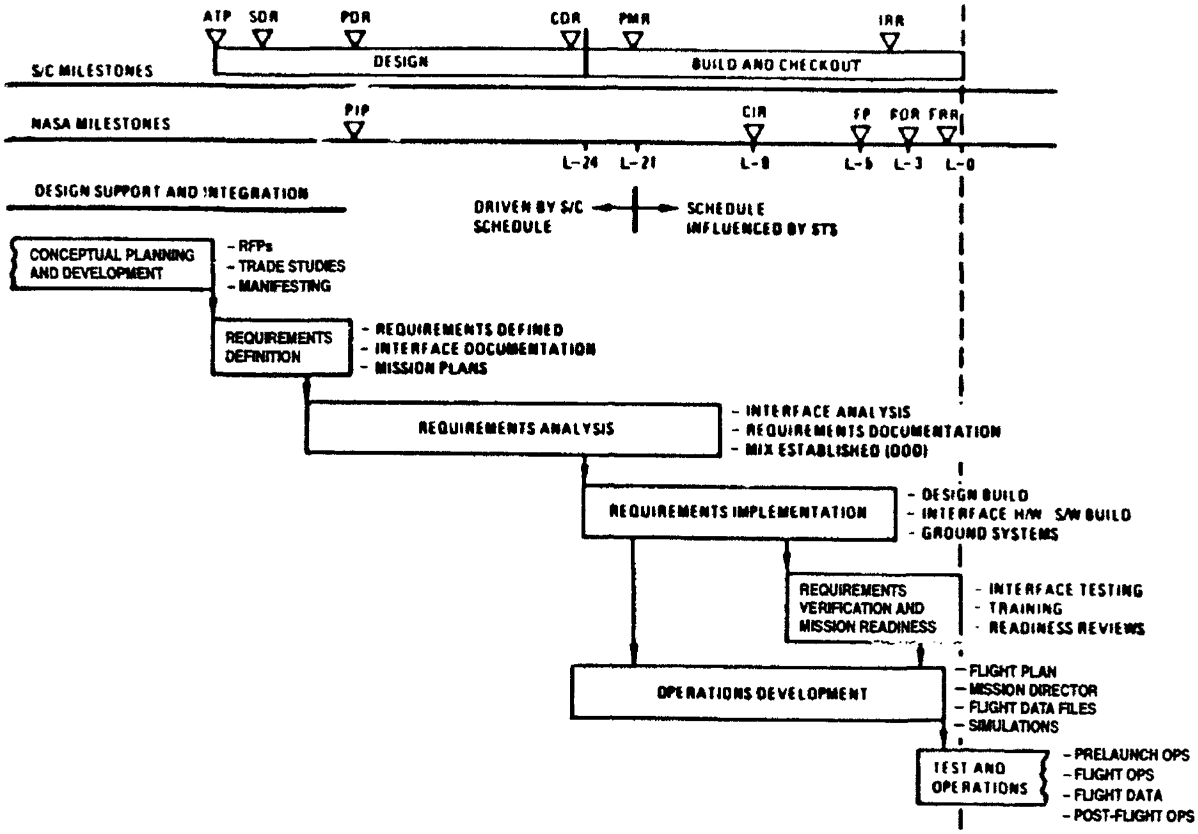

Figure 4-4. STS payload integration and operations process.

The working group approach is also used by NASA to develop the PIP Annexes that detail data and planning. However, additional groups may be added to address safety and software, and the flight operations effort is much expanded as compared with that for an expendable vehicle. Many months of preparation are necessary; personnel and equipment may be rehearsed and put through several simulation exercises (stand-alone and joint), where numerous contingencies are simulated before conducting an actual flight. Safety issues are major design drivers; if they are not addressed early, they can cause significant cost and schedule impacts. The major driver is the NASA requirement that any system whose failure could cause catastrophic failure of the Orbiter must be two-fault tolerant. The key reference document for interfacing with the STS is NASA's "Core ICD."*

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*NASA Core Interface Control Document (ICD). Attachment I to NSTS 07700, Volume XIV, Payload Accommodations, ICD-2-19001, "Shuttle Orbiter/Cargo Standard Interfaces." Also known as Attachment 1 to NSTS 07700, Volume XIV, "Payload Accommodations."

The NASA STS offers some unique opportunities for flying small or mid-sized payloads that do not need to occupy a major part of the Orbiter's cargo bay. These include:

- Payload installed in the cabin or lockers
  - Usually small experiments
  - Potential for much manned interface/operation.
- Get-away-special (GAS) containers
  - Mounted in the payload bay
  - Small payloads
  - Simple standardized interfaces
  - Potential to deploy a small satellite
- Shared ride in the cargo bay

Integration activities with the in-cabin/locker or GAS-type payloads are generally much simplified and standardized, as compared to those activities associated with payloads carried in a cradle in the cargo bay. Early planning and contact with NASA are still needed, however, since a backlog of such payloads is typically awaiting flight assignment. For shared cargo bay operations, three other issues arise:

- The geometric, structural, thermal, and power interactions between the shared payloads must be taken into account. The program offices must work together to conduct integrated analyses.
- Mass properties (CG location) and other constraints apply to both normal and abort operations. Also, for shared cargo, contingencies such as deployment of one and the hang-up of the other must be considered.
- Dealing with mixed cargoes, along with its scheduling and integration, is difficult at best. Compatibility of orbital parameters and dependence on all partners being ready at the same time can challenge practicality.

V. ROCKET PROPULSION

Eusebio Suarez-Alfonso

A typical space launch vehicle contains a variety of rocket propulsion elements. For example, the space shuttle has two solid rocket boosters (SRBs), which are ignited on the ground and jettisoned about 2 min later; three space shuttle main engines (SSMEs), which are also ignited on the ground and shut down near orbit injection; two orbital maneuvering engines (OMEs) used for orbit injection and deorbit; and an attitude control system (ACS) containing 32 bipropellant thrusters. The Titan IV vehicle has two Stage 0 solid rocket motors (SRMs) for initial boost, two Stage I liquid rocket engines, and a single Stage II liquid rocket engine. The Titan may or may not have an Orbital Transfer Stage, such as the Centaur or the Inertial Upper Stage (IUS), depending on the mission requirements.

Propulsion systems can be classified by the type of primary energy used for conversion to vehicle thrust. These include chemical, electrical, nuclear, solar, laser, and others. Chemical propulsion systems, the most common type at the present time, can be further classified into liquids, solids, or hybrids.

Another way of classifying rocket propulsion systems, which is adopted herein, is based on the type of vehicle they service. Launch vehicle propulsion is used to propel a vehicle from Earth to a transfer orbit or to a low-Earth orbit (LEO). Orbital transfer vehicle propulsion transfers the vehicle from one orbit to another orbit, as, for example, the transfer from LEO to GEO. Satellite propulsion systems perform satellite control and stationkeeping functions. In general, propulsion systems for launch vehicles are large and heavy, those for orbital transfer vehicles are smaller and lighter, while satellite propulsion systems are usually very small and very light.

A. GENERAL PRINCIPLES

From an overall system standpoint, the function of the propulsion system is to impart velocity to the vehicle as efficiently as possible. An important measure of propulsion system performance is specific impulse.

Specific impulse can be simply defined as the propulsion system thrust produced per unit propellant mass flow rate

$$I_{sp} = \frac{F}{\dot{W}} \tag{5-1}$$

where:

I_{sp} = specific impulse, sec

F = thrust, lb

$\dot{W}$ = total propellant weight flow rate, lb/sec

Values of I_{sp} can range from a low of 235 sec for a satellite hydrazine monopropellant system to a high of as much as 10,000 sec for an ion satellite propulsion thruster.

The specific impulse developed by a propulsion system is dependent upon several factors, such as the propellant or propellant combination (combustion product temperature and molecular weight), propellant mixture ratio, propellant temperature, combustion chamber pressure, and nozzle expansion area ratio. Of these factors, the propellant combination and the nozzle expansion ratio are the most significant.

The second most important propulsion system performance parameter is thrust, which is directly related to specific impulse, as shown in Eq. (5-1). This parameter, whose value is dictated only by vehicle requirements, can be expressed as

$$F = \dot{M}\,Ve + (Pe - Pa)Ae \tag{5-2}$$

where

F = thrust, lb

$\dot{M} = \frac{\dot{W}}{g}$ = propellant mass flow rate, lb–sec/ft

Ve = gas exhaust velocity at nozzle exit, ft/sec

Pe = gas exhaust pressure at nozzle exit, lb/ft^2

Pa = ambient pressure, lb/ft^2

Ae = nozzle exit area, ft^2

Another important parameter applicable to bipropellant liquid propulsion systems and which also has a significant effect on specific impulse, is the propellant mixture ratio. This parameter can be expressed as

$$MR = \frac{\dot{W}o}{\dot{W}f} \tag{5-3}$$

where

MR = mixture ratio

$\dot{W}o$ = oxidizer weight flow rate, lb/sec

$\dot{W}f$ = fuel weight flow rate, lb/sec

At stoichiometric mixture ratio conditions, specific impulse is a maximum. References 5-1 and 5-2 address vehicle and propulsion system design and performance. References 5-3 and 5-4 provide a summary of existing propulsion systems and the technology requirements for future systems.

B. LAUNCH VEHICLE PROPULSION SYSTEMS

Launch vehicle propulsion refers to the propulsion systems needed to transfer a payload from the Earth's surface to a mission orbit (e.g., a LEO). The propulsion systems are usually large and heavy. These systems are typically solid rocket motors and liquid rocket bipropellant engines, although air breathing engines are also receiving attention.

Factors that influence the design and performance of launch vehicle propulsion systems include atmospheric pressure, drag, aerodynamic heating, gravity, and vehicle acceleration. All of these factors are interrelated.

For the first stage of launch vehicles, where gravity and drag effects are very significant, achieving high thrust is a dominant factor. Both liquid and solid propulsion systems are very competitive for these applications. The most significant characteristics of solid and liquid systems are compared in Table 5-1.

1. Liquid Rocket Propulsion

Liquid propellants can be storable, cryogenic, or a mixture of both types and are fed into the rocket combustion chamber either by turbopumps (pump-fed) or by pressurized tanks. Liquid-engine nozzles can be cooled regeneratively (using propellant flowing through passages in the nozzle wall), ablatively, or by radiation.

Pump-fed liquid rocket engines can be classified by the power cycle used to drive the engine turbopumps. Typical power cycles are the gas generator, the expander, and the staged combustion. These systems are depicted in Figure 5-1. The gas generator cycle is the most common, and has been used in the Atlas and Titan launch vehicles. It is also planned to be used in the space transportation main engine (STME) for the National Launch System (NLS). It utilizes combustion products from a gas generator to drive the engine turbine(s). The expander cycle, which is used by the Pratt and Whitney Centaur engine, utilizes the heat absorbed by the combustion chamber cooling propellant to drive a turbine. In the staged combustion cycle, which is used by the SSME, preburned propellants are first used to drive the engine turbine(s) and then injected into the combustion chambers for complete combustion.

Table 5-1. Comparison of Liquid and Solid Propulsion Systems

Liquid	Solid
Higher specific impulse	Higher performance in volume-limited applications
Can be tested and calibrated prior to launch	Fewer parts/faster checkout
Amenable to verification of manufacturing processes	Shorter development time
Suitable for launch under widely different temperature conditions	Rapid readiness for launch

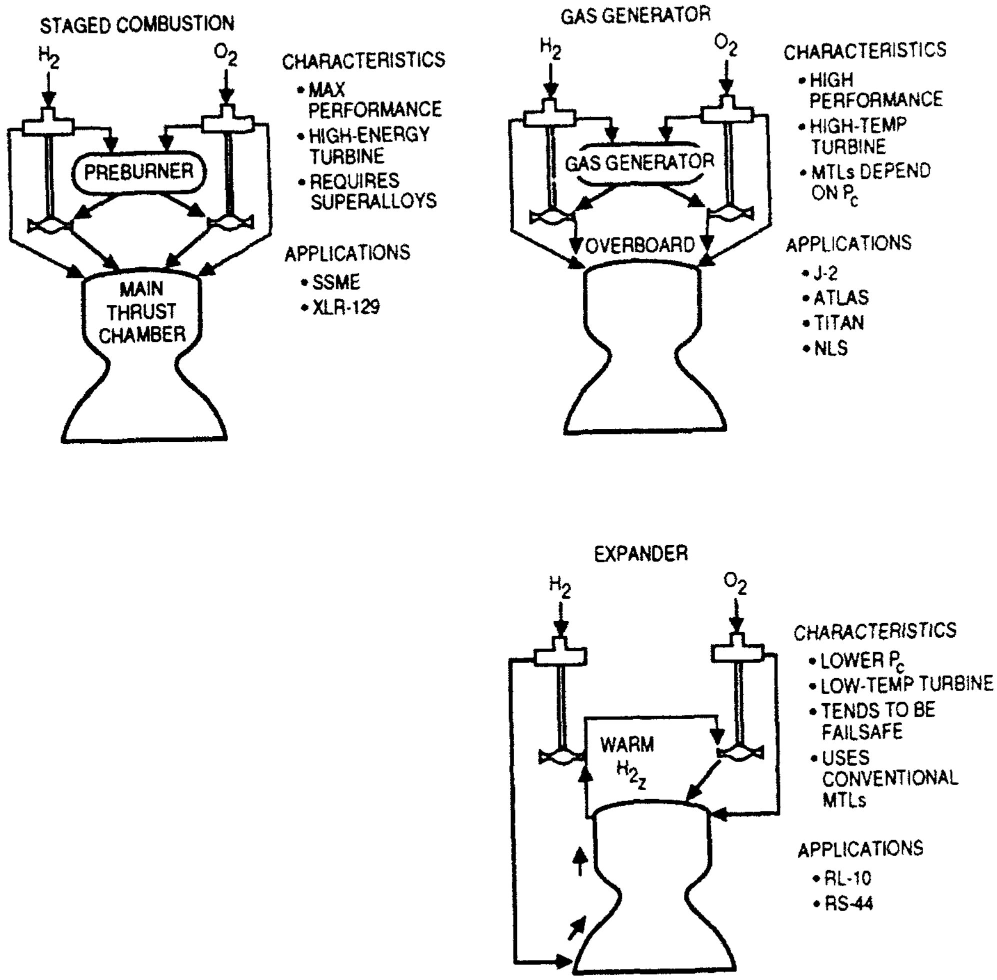

Figure 5-1. Turbopump power cycles.

Each power cycle has advantages and disadvantages. For example, the staged combustion cycle is most suitable for engines with very high combustion chamber pressures (about 3000 psi and higher), whereas the expander cycle is applicable to lower chamber pressures (about 1200 psi and lower). The gas generator cycle is generally used for intermediate chamber pressures. Combustion chamber pressure has a significant effect on the size of the liquid engine; higher pressures result in the use of smaller engines for the same thrust.

Propellant combinations currently employed for launch vehicle propulsion include nitrogen tetraoxide (N_2O_4)/Aerozine 50 (50 percent hydrazine – 50 percent unsymmetrical dimethylhydrazine, by weight), liquid oxygen (LO_2)/liquid hydrogen (LH_2), and liquid oxygen (LO_2)/RP-1 (rocket propellant 1, a kerosene-type hydrocarbon fuel). Each propellant combination has advantages and disadvantages, but LO_2/LH_2 is a favored choice for future applications because of its environmentally clean exhaust product (water) and its very high specific impulse performance. A disadvantage is that liquid hydrogen has a very low density (~4.4 lb/ft^3), which results in a need for large and, therefore, heavy propellant tanks. Theoretical specific impulse values for the propellant combinations are 338 sec for N_2O_4/Aerozine 50, 363 sec for LO_2/RP-1, and 450 sec for LO_2/LH_2. These values are based on a chamber pressure of 2000 psia, a nozzle expansion ratio of 50, an optimum mixture ratio, and vacuum conditions.

Thrust vector control is required for directional guidance and control of the vehicle. In the case of liquid rocket engines, thrust vector control is usually provided by swiveling the engine using yaw and pitch plane actuators, which may be driven hydraulically or electromechanically. Electrically redundant electromechanical actuators are favored for future systems.

2. Solid Rocket Propulsion

Solid rocket motors (SRMs), because of their simplicity, compactness, and low development cost, are very suitable for launch vehicle applications. An SRM consists of a pressure vessel that acts as the combustion chamber, a fixed or movable nozzle for directing the discharge of combustion gases, a solid propellant charge that contains both fuel and oxidizer, and an igniter which initiates the combustion process. The propellant grain, which contains the fuel, oxidizer, and various other additives, is bonded internally to the combustion chamber, more generally referred to as the motor case. Once the propellant grain is ignited, the solid motor burns to provide a prescribed pressure versus time history dictated by the selected configuration of the propellant charge (grain geometry). Solid rocket motors are ablatively cooled.

Solid rocket motor cases are fabricated of metal (generally titanium or steel) or are filament-wound composite structures. Composite structures may utilize fiberglass, kevlar, or graphite, although graphite is a preferred choice because of its superior strength and low weight. Solid rocket motor cases are insulated with a particle-filled (silica, asbestos, or kevlar) rubber that is bonded internally to the motor case. A sprayed or brushed-on liner is applied to the insulation prior to propellant casting to facilitate bonding of the cured propellant to the insulation. Typical motor case operating pressures are from 200 to 1500 psia, depending on the application.

Solid rocket motors generally employ composite propellants for space applications. These propellants contain a crystalline oxidizer (usually ammonium perchlorate) and a metal fuel (typically aluminum) with processing aids and a liquid polymer, such as polybutadiene acrylonitrile (PBAN) or hydroxy-terminated polybutadiene (HTPB). The polymer binds the ingredients together and also acts a fuel.

Performance of SRMs is generally lower than other propulsion systems. The average specific impulse performance of the large, seven-segment Titan IV SRM is, for instance, about 270 sec.

The current state of the art in large solid motors for launch vehicles is best represented by the solid rocket motor upgrade (SRMU), an improved version of the Titan IV SRM. The 120-in. diameter, three-segment SRMU uses a graphite epoxy motor case loaded with a composite HTPB propellant. Thrust vector control is provided by a movable nozzle. Delivered average specific impulse performance is predicted to be 286 sec.

An evolving concern about solid motors is the impact of their toxic effluent on the environment. Research efforts are examining "clean" propellant alternatives.

3. Hybrid Rocket Propulsion

The hybrid motor combines some of the best features of SRMs and liquid engines. In a typical arrangement, a liquid oxidizer (such as liquid oxygen) is injected into a combustion chamber containing a fuel-rich solid propellant. The solid grain is ignited by a separate source prior to liquid oxidizer injection. Hybrid motors combine the simple, compact, SRM design with the higher performance of a liquid fuel. At the time of this writing, there are no operational hybrid motors. A variant of the typical hybrid motor is one where the exhaust gases from a fuel-rich solid gas generator are reacted in a separate chamber with a liquid oxidizer. Hybrid motors, like liquid engines, provide operational flexibility, such as abort, start/stop, and thrust throttling.

4. Orbital Transfer Vehicle Propulsion Systems

Propulsion systems for orbital transfer vehicles are typically designed for light weight (which generally means low combustion chamber pressures), very high specific impulse performance (which favors the LO_2/LH_2 liquid propellant combination, the expander power cycle, and large nozzle expansion ratios), and relatively low thrust (typically less than 20,000 lb).

a. Chemical Propulsion

Current orbital transfer propulsion systems in the liquid propellant class include the RL-10 engine used in the Centaur vehicle (LO_2/LH_2 propellants) and the twin AJ10-138 engine system used in the Transtage vehicle (N_2O_4/Aerozine 50 propellants). Another liquid bipropellant system that has been considered for orbital transfer vehicle use is N_2O_4/MMH (monomethylhydrazine) because of its space-storable capability. Current orbital transfer propulsion systems in the solid propellant class include the Stage I and Stage II IUS motors, and the PAM-DII and TOS motors.

Another type of liquid rocket engine that is receiving attention is the integrated modular engine (IME). The engine is arranged so that it is short, modular, and compact. Instead of the conventional cylindrical combustion chambers with long expansion nozzles for liquid rocket engines, the IME contains a series of small rectangular combustion devices arranged in any convenient geometric form, such as rectangular or circular, around a center structure. The combustion exhaust gases are commonly expanded through a single plug nozzle, which is very short and provides a continuous area for gas confinement and expansion. The outside of the gas stream is a free surface that allows for optimum expansion to ambient pressure conditions. The main advantages of the IME are compactness and high specific impulse performance. In addition, the segmented combustion chambers could be individually controlled to provide thrust throttling and/or thrust vector control.

Solid rocket motors for orbital transfer application are typically light in weight, have very low structural weight ratios, must provide for high nozzle expansion ratio, and be capable of operating in the space environment. There are a variety of motors in current use, ranging in weight from 13 to 40,000 lb, with average thrusts of 400 to 60,000 lb. To perform a complete orbital transfer mission, two SRMs are required. One is to place the vehicle in the transfer orbit and the other to place the vehicle in the final orbit. Solid rocket motors for orbital transfer are either spin stabilized or use a liquid propellant reaction control system.

b. Nuclear Propulsion

Nuclear propulsion powers an engine in two ways. In one approach, called nuclear thermal, a fission reactor is used to heat a propellant such as liquid hydrogen. Hydrogen would reach a temperature of up to 3000 K before being ejected through the expansion nozzle. The specific impulse for a nuclear thermal rocket engine is in the range of 800 to 1000 sec. In another approach, nuclear electric propulsion, either argon or hydrogen is heated up to temperatures at which the gas turns into plasma, which is then ejected using electrostatic or electromagnetic forces. A nuclear electric engine can achieve specific impulse values of up to 30,000 sec. However, additional technology developments in thermal control, advanced materials, nuclear containment, and other areas are needed before the high performance potential of nuclear propulsion concepts can be fully exploited.

c. Laser and Solar Propulsion

Laser propulsion has received attention, especially for orbital transfer missions. In the typical approach, the laser energy is focused on a flowing stream of hydrogen gas that is then expanded through a nozzle. Relatively high thrust levels can be developed with a specific impulse of about 850 sec.

Solar propulsion is similar to laser propulsion, except that solar energy in the infrared is optically concentrated to heat the hydrogen working fluid. The specific impulse performance is also about 850 sec.

5. Satellite Propulsion Systems

Satellite propulsion systems provide energy in the form of momentum exchange to control the position or attitude of the spacecraft. Depending upon the mission, the propulsion system may be required to perform various vernier velocity maneuvers to maintain or change a satellite's orbital parameters. The reaction control system (RCS) also uses momentum exchange to provide attitude control. For example, operations may require sun/Earth point, spinup, spindown, and control of body disturbances such as on-orbit center-of-mass shifts that occur as onboard propellants are consumed.

In newer satellite systems, the need for large amounts of chemical energy for control has been declining. Instead, available electrical energy from solar arrays is used to operate momentum wheels, control moment gyroscopes (CMGs), or magnetic torquers for control.

Satellite propulsion systems must be highly reliable. The systems must operate thousands of times and even hundreds of thousands of times for periods of up to 10 years. Reliability is enhanced by using redundant propulsive thrusters and control valves. Like other propulsion systems, satellite propulsion favors high specific impulse performance, low structural weight, and simplicity.

Thrusters for satellite propulsion are very small and light (typically 2 lb or less) and produce low thrusts (typically less than 50 lb and, in many cases, less than 10 lb). Cooling is accomplished by radiation. A summary of future propulsion options for satellites and their expected specific impulse performance is provided in Table 5-2.

a. Chemical Propulsion

The most commonly used satellite propulsion system employs monopropellant hydrazine as the working fluid. Although its specific impulse performance is very low, about 235 sec, it is a simple system with a proven record of high reliability. In its most simple form, the hydrazine fuel and the pressurant gas (typically nitrogen) are stored within a single tank that has a bladder to separate the gas from the hydrazine. The tank is then pressurized with sufficient gas to feed the hydrazine into the thrusters for the entire mission. Each thruster has a catalyst to decompose the hydrazine at about 1700°F to produce a gas mixture of mainly hydrogen and nitrogen. The propulsion system is equipped with heater systems to ensure that the freezing point of hydrazine, about 35°F, is not approached.

In recent times, there has been a trend toward increasingly heavier payloads, longer missions, and more ambitious orbital operations. This trend has resulted in the use of storable bipropellant systems, especially N_2O_4/MMH with its higher specific impulse performance. No ignition system is required for this propellant combination.

Two other chemical satellite propulsion options are listed in Table 5-2, augmented hydrazine and water electrolysis. These systems are hybrid chemical–electrical systems. In the former, the hydrazine decomposition products are subsequently heated through a high-resistance heater element. In the latter, water is decomposed electrolytically, and the resulting hydrogen and oxygen are reacted as a bipropellant system. Significant development work has been performed on both systems.

b. Electric Propulsion

Although the promise of high performance from electric propulsion devices has been known for many years, the application of the various concepts to flight vehicles has been minimal. All systems require a significant amount of electrical power, a small amount of a working fluid, and typically produce very low thrusts (much less than 1 lb). Ion and pulsed plasma devices have been flown as experiments, and resistojets have been flown on commercial satellites.

A resistojet is basically the same device as described above for augmented hydrazine, but the working fluid is directly heated through an electric resistance element. Many different working fluids have been considered, such as ammonia, water, hydrazine, and methane. A somewhat similar electrothermal device is the arcjet, in which a gaseous working fluid is heated using an electric arc and then discharged through a nozzle.

In the pulsed plasma system, a plasma is generated by flowing a gas across a high-current-density electric arc or is vaporized from a solid block of propellant. The plasma is then electromagnetically accelerated. Gaseous fluids, such as hydrogen, argon, xenon, carbon dioxide, and solid Teflon have been tested.

Table 5-2. Propulsion Options for Satellites

Concept	I_{sp} (sec)
• Chemical	
1. Monopropellant hydrazine	235
2. Augmented hydrazine	300
3. Storable bipropellant	315
4. Water electrolysis	340
• Electric	
1. Resistojet (electrothermal)	600
2. Pulsed plasma (electromagnetic)	1200
3. Arcjet (electrothermal)	1500
4. Colloid (electrostatic)	1500
5. Pulsed inductive (electromagnetic)	2000
6. Magnetoplasmadynamic (electromagnetic)	2500
7. Ion (electrostatic)	3000

Ion and colloid thrusters are electrostatic devices. In the ion devices, efforts have concentrated on two types of thrusters: mercury bombardment and cesium contact, with mercury bombardment being preferred. Colloid thrusters, which have received very little attention, use a doped colloidal working fluid. In both systems, fluid acceleration is achieved by the interaction of electrostatic fields on charged propellant particles, such as ions and colloids. The particles are neutralized prior to discharge from the thrusters.

Pulsed inductive and magnetoplasmadynamic thrusters are electromagnetic devices. Both provide very high specific impulse values, and both operate on the principle of the interaction of the magnetic and electric field on a propellant plasma. The exhaust beam from these devices is neutral. Electromagnetic thrusters have the potential to produce relatively high thrusts, up to 10 lb, in practical applications.

Additional information on electric propulsion systems can be found in References 5-2 and 5-3.

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5-2. G. Sutton, "Rocket Propulsion Elements, An Introduction to the Engineering of Rockets," Fifth Edition, *Wiley–Interscience* (1986).

5-3. R. L. Doebler, F. Suarez–Alfonso, K. A. Turner, and H. A. Croft, "Propulsion Technology for Space Systems, Current State of the Art and Projection of Trends," The Aerospace Corporation, TOR-0083(3909-63)-2 (29 October 1982).

5-4. "The Propulsion Challenge," *Aerospace America* (July 1990), pp. 23–86.

VI. LAUNCH VEHICLE SIZING/PERFORMANCE ANALYSIS

Lester Forrest

The ready availability of high-speed computers has fostered the proliferation of sophisticated computational tools for the conceptual design of launch vehicle systems. Table 6-1 is a partial list of computer programs that have been developed by the aerospace industry for this purpose. At the high end of the spectrum are trajectory simulation programs, which, combined with vehicle sizing routines, enable the designer to establish optimum vehicle configurations with high precision. Such computer tools, however, are often complex, expensive to operate, and usually demand a vast array of input data that are ill-defined or not available in the preliminary or conceptual phase of launch vehicle planning. For this purpose, most industry organizations resort to simpler computational techniques that are trajectory-approximate, may be partially automated, or are amenable to manual execution. PREVAIL, developed by The Aerospace Corporation, is an example of the latter type of vehicle sizing routine (Reference 6-1).

The conceptual design process can be characterized as a preliminary definition of near-optimum vehicle configurations that can accomplish the mission objective(s). Subsequent studies will isolate the best possible choice among several alternatives, using more precise performance and weight estimation techniques and taking into consideration additional factors such as cost,

Table 6-1. Launch Vehicle Design Programs

Program	Organization	Trajectory Simulation	Features	Comments
FONSIZE	The Aerospace Corp.	Yes	Many vehicle types and configurations	In development
GTS—Size	The Aerospace Corp.	Yes		Solid rocket vehicles only
AVID	NASA Langley	Yes	Shuttle and other launch configurations	Prescribed ascent profile
HAVCD	Boeing	Yes	Hypersonic winged vehicles	Prescribed ballistic ascent profile
FLYIT	Boeing	Yes	PC-based trajectory optimization program	
POST	Martin	Yes		
FASTPASS	General Dynamics	Yes	Various vehicle types and configurations	Vehicle synthesis/trajectory optimization
PREVAIL	The Aerospace Corp.	No	Computer mechanized (VAX) size optimization equations	Prescribed ballistic ascent loss parameters
BP	The Aerospace Corp.	No	PC-based interactive sizing routine	Prescribed ballistic ascent
CONSIZ	NASA Langley	No		

reliability, operability, flexibility, etc. The earliest phase of design starts with a mission orbit and payload objective and seeks to define only the most elemental aspects of the vehicle configuration: i.e., number of stages, velocity split among stages, stage weight, and propellant weight. Gross dimensions, component weights, and other preliminary design details may then be estimated and vehicle performance recalculated. This early design activity is frequently referred to as vehicle sizing.

This section provides a brief description of some basic techniques that are used for vehicle sizing and performance analysis of rocket-powered space launch vehicles. The term launch vehicle, as used here, includes all of the vehicle elements required to insert a payload into its mission orbit. The elements involved may include one or more rocket-powered stages to transfer a payload from Earth to LEO, plus a rocket-powered transfer vehicle, if needed. A transfer vehicle is used to take a payload from one orbit to another, generally from a low-Earth parking orbit to some higher circular or elliptical mission orbit, which may also lie in a different plane.

The relationships for vehicle sizing presented in the following paragraphs are general in nature and may be found in any textbook on launch vehicle design. The particular form of the equations used reflects the methods of approach and nomenclature contained in PREVAIL, a preliminary design tool developed by The Aerospace Corporation (Reference 6-1).

A. THE BASIC ROCKET EQUATION

The heart of the conceptual design process is the basic rocket equation. This equation is derived from Newton's second law of motion as

$$\Delta V_{Ideal} = gI_{sp}\ln\frac{W_{initial}}{W_{final}} = gI_{sp}\ln r \tag{6-1}$$

where

ΔV_{Ideal} = velocity imparted to the rocket (ft/sec)

I_{sp} = specific impulse of the rocket (sec)

g = gravitational constant (32.174 ft/sec^2)

r = $W_{initial}/W_{final}$

$W_{initial}$ = vehicle weight at propellant ignition (lb)

W_{final} = vehicle weight at propellant burnout (lb)

Equation (6-1) assumes a single-stage rocket operating without drag, gravity, or other loss influences with I_{sp} constant at a representative mission value.

The variable r in Eq. (6-1) includes the weight of the payload and may be expanded to

$$r = \frac{W_{initial}}{W_{final}} = \frac{W_b + W_{PL} + W_p}{W_b + W_{PL}} \tag{6-2}$$

where

W_b = weight of vehicle at burnout excluding payload (lb)

W_p = weight of propellant consumed (lb)

W_{PL} = weight of payload (lb)

The burnout weight (W_b) includes all elements of weight that comprise the rocket stage at burnout. This may include tanks, pumps, nozzles, engines, guidance and control components, residual fluids, unburned propellants, and structure.

Another important term used in vehicle design is propellant mass fraction (MF_p). The propellant mass fraction does not include the weight of the payload and is defined as

$$MF_p = \frac{W_p}{W_p + W_b} \tag{6-3}$$

Propellant mass fraction may be thought of as a measure of how well the design achieves a low burden of stage inert weight (burnout weight).

1. Using the Rocket Equation for Vehicle Sizing

The basic scheme for the conceptual design of rocket-powered space transportation vehicles proceeds as follows. Equation (6-1) is solved for r to give

$$r = \exp \frac{\Delta V_{ideal}}{g I_{sp}} \tag{6-4}$$

The designer determines how much ΔV must be imparted to the payload to deliver it to a desired position and state in space. Estimates are made of the I_{sp} and the propellant mass fraction MF_p. Then Eq. (6-4) is used to determine the value of r, and Eqs. (6-2) and (6-3) are used to determine the total vehicle weight. With total weight established, subsystem weights are calculated. Then the cost of the subsystems are determined, and these costs are summed to obtain the cost of the total vehicle. Vehicle cost combined with other cost parameters may then be used to calculate the cost of the system over its lifetime of use (life-cycle cost). These results, along with other requirements and information available to the designer, may suggest refinements to the parameter values originally estimated, so that the calculations may be repeated several times to produce an appropriate vehicle design.

One factor complicating the use of the basic rocket equation is that space transportation vehicles are subject to the effects of external forces such as Earth gravity and atmospheric drag. These forces, which prevent the vehicle from achieving the magnitude of velocity gain predicted by Eq. (6-1), are accounted for in the vehicle sizing process by evaluating ΔV_{Ideal} in Eq. (6-4) as the sum of a true (inertial) velocity gain and the ΔV equivalent of various losses that require the use of additional impulse propellant to offset:

$$\Delta V_{Ideal} = \Delta V_{True} + \Delta V_{Losses} \tag{6-5}$$

The notion of an ideal velocity expressed in terms of a loss may be understood by recognizing that the loss term added to the true term yields the maximum velocity that the propellant would impart if the rocket were operated under lossless (ideal) conditions. Estimating the loss effects is treated later in this section.

For multistage vehicles, a decision must be made as to the basis or method for apportioning ΔV_{Ideal}, the ideal velocity gain for the total vehicle, among the individual stages (often referred to as the ΔV split). Whatever the method used to establish the ΔV split, the sum of the velocity contributions by the individual stages must numerically equal the above stipulated ΔV_{Ideal} needed to achieve the launch objective. That is,

$$\Delta V_{ideal} = \sum_{i=1}^{NS} \Delta V_i \tag{6-6}$$

where ΔV_i is the velocity increment added by Stage i, and NS is the total number of vehicle stages.

2. Sizing of Various Types of Vehicles

There are many launch vehicle configurations and sizing options that may have to be considered in a conceptual design exercise. A comprehensive listing would most certainly include single-stage vehicles, optimized and nonoptimized multistage vehicles with series-burn or parallel-burn propulsion elements, multistage series and parallel-burn vehicles that incorporate existing stages, reusable and partially reusable stages with lifting body or winged designs, and many more possible stage and vehicle configuration alternatives.

This document provides design algorithms for a few of these options, with the objective of illustrating the kind of techniques and the level of detail that are likely to be used for vehicle sizing in the preliminary or screening phase of the vehicle definition process.

a. Single-Stage Vehicle

A single-stage launch vehicle may be sized as follows. The input quantities required to be known are the ideal velocity gain (ΔV_{Ideal}), the propellant mass fraction (MF_p), and the specific impulse of the rocket (I_{sp}). The mass ratio (r) is obtained from Eq. (6-4). The propellant weight (W_p) is expressed in terms of known parameters by solving Eqs. (6-2) and (6-3) simultaneously to yield

$$W_p = \frac{MF_p W_{PL}(1 - \frac{1}{r})}{MF_p + \frac{1}{r} - 1} \tag{6-7}$$

The payload weight W_{PL}, as defined for the launch vehicle, is all the weight delivered to orbit less the burnout weight of the vehicle. W_{PL} may include the weight of a satellite, a payload fairing (if not staged earlier), and a transfer vehicle.

With the propellant weight W_p determined and the propellant mass fraction MF_p known, the burnout weight of the vehicle is computed by solving Eq. (6-3) rearranged as

$$W_b = \frac{W_p(1 - MF_p)}{MF_p} \tag{6-8}$$

b. Nonoptimized Multistage Series-Burn Vehicle

A series-burn vehicle is one in which each stage is burned sequentially. When one stage has consumed all of its impulse propellant, the empty structure (stage burnout weight) is dropped off, and the remaining stages and payload continue along the mission trajectory. For the purpose of conceptual design, it is assumed that the losses incurred in the staging process are small and can be neglected.

The nonoptimized multistage series case presumes that the ΔV split among stages is already established, based on previous calculations or other information. If, for example, all stages are assumed to have identical I_{sp} and mass fraction (MF_p) characteristics, then the velocity contribution for all stages properly would be taken as equal in order to achieve a maximum performance/ minimum weight vehicle design. These conditions notwithstanding, the equal velocity assumption is frequently suitable for a first-cut trial analysis.

The vehicle sizing process is carried out beginning with the last stage and working toward the first stage, with W_{PL} in Eq. (6-7) redefined as W_{Di}, the dead weight above Stage i, the stage currently being sized. The dead weight includes the mission payload and payload fairing plus the total weight (fueled weight) of each of the stages above Stage i. Using this method, all of the variables needed to solve the sizing relations [Eqs. (6-4), (6-7), and (6-8)] will be known, permitting the propellant weight and the burnout weight for each stage to be determined in sequence.

c. Optimized Multistage Series-Burn Vehicle

For a multistage vehicle with stages that have dissimilar I_{sp} and MF_p values, the design goal is to find the optimum velocity split among stages that will maximize vehicle performance or minimize weight. In general, the optimized configuration results in higher velocity gains for those stages that have superior I_{sp} and mass fraction characteristics.

As developed here, the sizing procedure for the optimized multistage series-burn vehicle provides the analyst with the option of designing the vehicle for minimum gross weight at a specified payload capability or for maximum payload capability at a specified gross weight. The design optimization procedure uses the Lagrange multiplier technique, a standard mathematical procedure for finding the maxima and minima of functions of several variables. The process as applied to the multistage series-burn vehicle consists of an iterative search for the Lagrange multiplier that satisfies the following equation.

$$\prod_{i=1}^{NS}\left\{\frac{1}{\lambda g I_{spi} MF_{bi}} + \frac{1}{MF_{bi}}\right\}^{\frac{I_{spi}}{I_{sp1}}} = \exp\frac{\Delta V_{Ideal}}{g I_{sp1}} \tag{6-9}$$

where

$$MF_{bi} = \frac{W_{bi}}{W_{bi} + W_{pi}} \tag{6-10}$$

and

MF_{bi}	= burnout mass fraction of stage i
W_{bi}	= burnout weight of stage i
W_{pi}	= propellant weight of stage i
I_{spi}	= specific impulse of stage i
I_{sp1}	= specific impulse of Stage I
ΔV_{Ideal}	= ideal velocity gain (total vehicle requirement)
g	= gravitational constant
λ	= Lagrange multiplier

As indicated by the above list, the method presumes that the basic characteristics of the vehicle are known, along with the ideal velocity gain needed to achieve the mission objective. All of these quantities may be adjusted, and the optimization may be repeated if a first trial produces results that are less than satisfactory in terms of vehicle proportions, performance, or other considerations of special concern to the designer.

After solving for the Lagrange multiplier as described above, the computational procedure develops the characteristics of each stage of the optimal vehicle using the following relationships

$$r_i = \frac{1}{\lambda g I_{spi}\ MF_{bi}} + \frac{1}{MF_{bi}} \tag{6-11}$$

$$W_i = \frac{W_{Di}(r_i - 1)}{1 - MF_{bi} r_i} \tag{6-12}$$

$$W_{bi} = W_i(1 - MF_{pi}) = (W_i MF_{bi}) \tag{6-13}$$

$$W_{pi} = W_i MF_{pi} \tag{6-14}$$

$$\Delta V_i = g I_{spi} \ln r_i \tag{6-15}$$

To further elucidate the method of solution, Figure 6-1 is provided, showing the sequence of steps in the sizing procedure as it is executed in The Aerospace Corporation's PREVAIL program (Reference 6-1). If the option selected is to find the minimum weight vehicle for a given payload, then W_{PL} is an input quantity, and the characteristics of the individual stages starting from the uppermost stage may be computed directly from the above relations. Then the gross weight of the vehicle W_o is obtained from

$$W_o = \left[\sum_{i=1}^{NS} W_i \right] + W_{PL} \tag{6-16}$$

If the option selected is to find the maximum payload for a specified gross vehicle weight W_o, then the procedure is to estimate W_{PL} and to iterate on this quantity until the sum of the stage weights plus W_{PL} matches the value of W_o specified. For either option, the final outputs of the calculation are the characteristics of the optimal vehicle, including the ΔV split and the associated stage weight quantities.

B. QUANTIFICATION OF INPUT PARAMETERS

1. Velocity Gain Requirements for Launch

The ideal velocity gain (ΔV_{Ideal}) required of a launch vehicle to deliver a payload to a specified orbit is defined as the sum of (a) the inertial velocity gain needed to accelerate the payload from velocity conditions at the launch site to velocity conditions on orbit, and (b) the velocity gain equivalent of various energy losses incurred by the launch vehicle on its ascent trajectory. The required gain was expressed in Eq. (6-5) as

$$\Delta V_{Ideal} = \Delta V_{True} + \Delta V_{Losses}$$

where ΔV_{True} represents the inertial velocity gain requirements and ΔV_{Losses} is the sum of the various loss effects which operate on the vehicle. The ΔV_{True} term is defined by the relation

$$\Delta V_{True} = V_p - V_o \tag{6-17}$$

where

V_p = velocity required to achieve/maintain the circular parking orbit (ft/sec)

V_o = Earth velocity component in the direction of the launch azimuth (ft/sec)

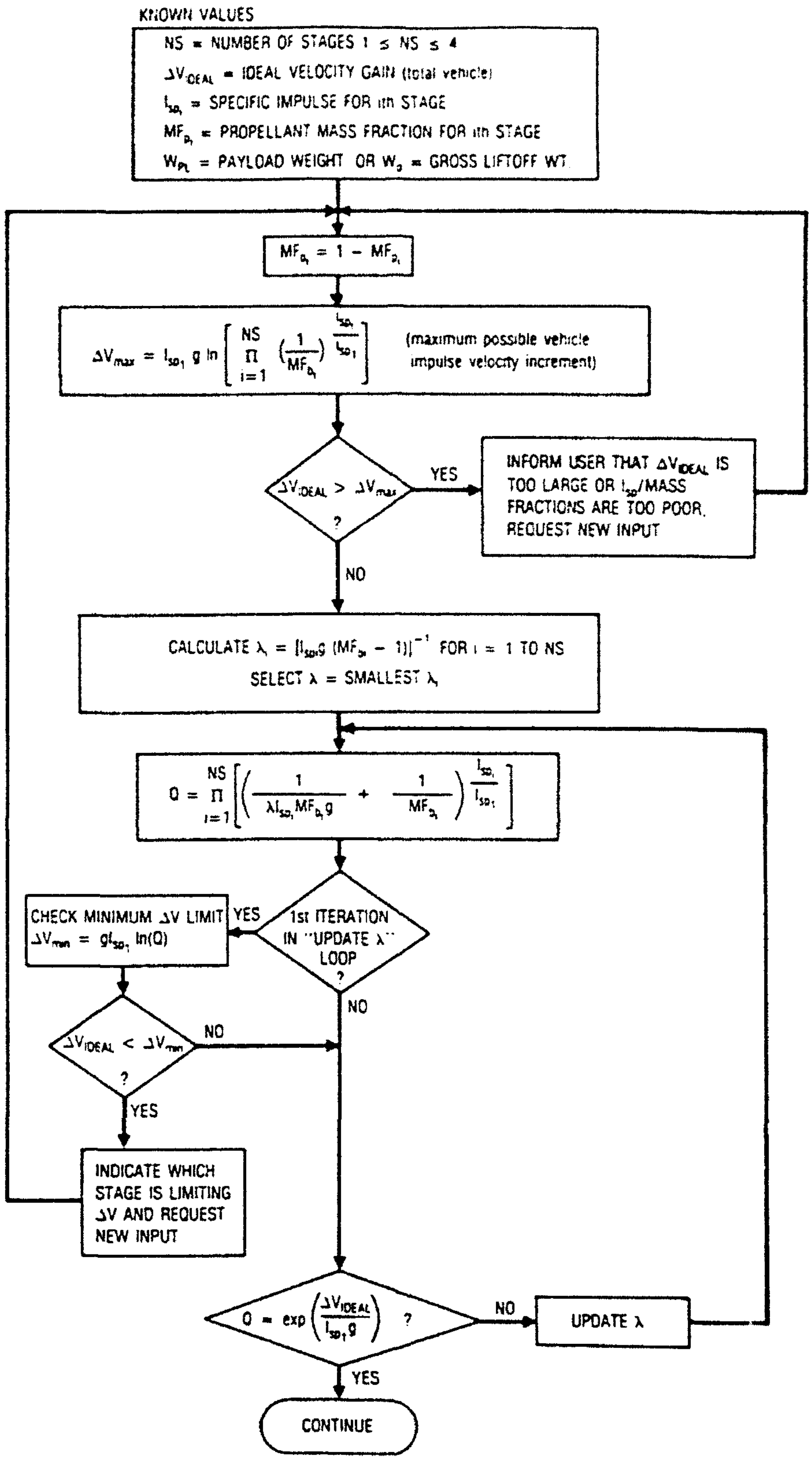

Figure 6-1. Sizing procedure for optimized multistage series-burn vehicle.

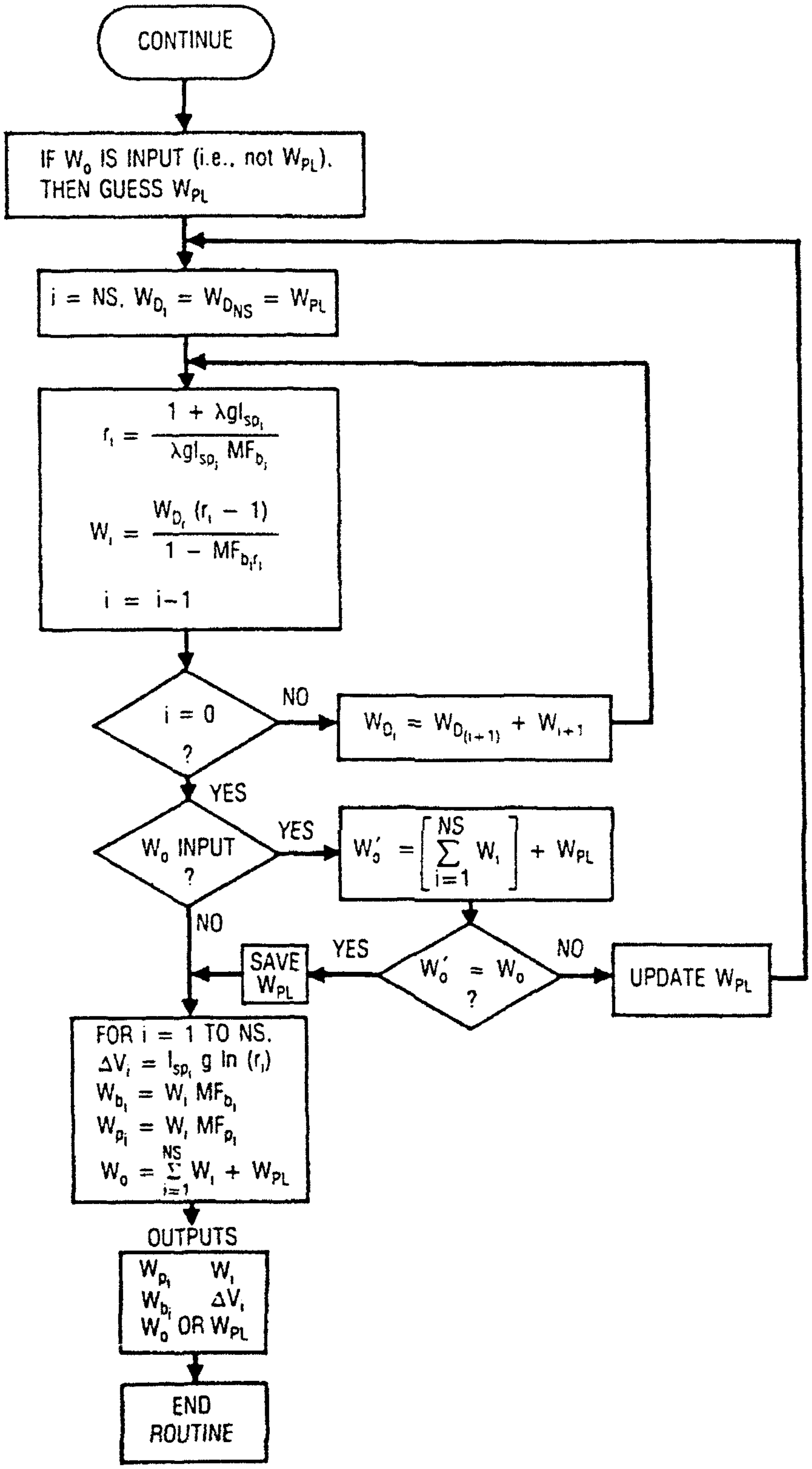

Figure 6-1. Sizing procedure for optimized multistage series-burn vehicle (continued).

The term V_p specifies the burnout velocity requirement for the last vehicle stage and is evaluated from

$$V_p = k_2 R_e \sqrt{\frac{g}{R_e + h}} \tag{6-18}$$

where

R_e = mean equatorial radius of the Earth (3444 nmi)

g = gravitational constant (32.174 ft/sec^2)

h = altitude of the circular parking orbit (nmi)

k_2 = units conversion constant (78)

The term V_o in Eq. (6-17), representing a contribution to the vehicle velocity at launch, is given by

$$V_o = V_e \sin Az \cos L \tag{6-19}$$

where

V_e = Earth tangential velocity at the equator (1520 ft/sec)

Az = launch azimuth angle; i.e., angle measured clockwise from due north (deg)

L = latitude of launch site (deg)

Thus, a launch heading due east (90 deg azimuth) gets the maximum benefit of the Earth's local tangential velocity, and a launch from a site at 0 deg latitude gets the maximum possible Earth velocity contribution at any launch heading.

The variables in Eq. (6-19) can be expressed in terms of orbit inclination i_o using the relation

$$\cos i_o = \sin Az \cos L \tag{6-20}$$

where i_o is the angle between the orbital plane and the Earth's equatorial plane. Then by combining Eqs.(6-17) through (6-20), the true velocity gain, ΔV_{True}, can be expressed as

$$\Delta V_{True} = k_2 R_e \sqrt{\frac{g_o}{R_e + h}} - V_E \cos i_o \tag{6-21}$$

All of the variables in Eq. (6-21) are known or can be derived from given launch site and parking orbit characteristics or specifications.

In the ideal velocity gain relationship, Eq. (6-5), ΔV_{Losses} may be expanded to

$$\Delta V_{Losses} = \Delta V_{Gravity} + \Delta V_{Drag} + \Delta V_{Thrust} \tag{6-22}$$

showing that the loss effects that operate on the vehicle are gravity, drag, and engine thrust (loss due to atmospheric pressure effects). Stage separation losses are relatively small and may be neglected for the purpose of a conceptual design performance analysis study.

The explicit loss terms in Eq. (6-22) are a function of many variables, and it is not practical to include a detailed approach for the evaluation of these terms in this document. An approximate method for specifying the total loss (ΔV_{Losses}) is shown in Figure 6-2. The curve represents composite results for the velocity losses of a number of expendable launch vehicles launched out of Eastern Test Range (ETR) to parking orbits of 100 nmi altitude. The input required is liftoff thrust-to-weight ratio, which for most launch vehicle designs lies in the range above 1.25. Among the many factors that determine the magnitude of this ratio are propellant type and stage burn arrangement (series vs parallel). Evolutionary-type designs with strap-on solid rocket boosters (exemplified by Titan IIIC) tend toward higher numbers in the range illustrated. Values of the ratio for a number of DOD and NASA launch vehicles are provided in Table 6-2 for the user's guidance in selecting a reasonable number. A similar chart for foreign launch vehicles is provided in Table 6-3.

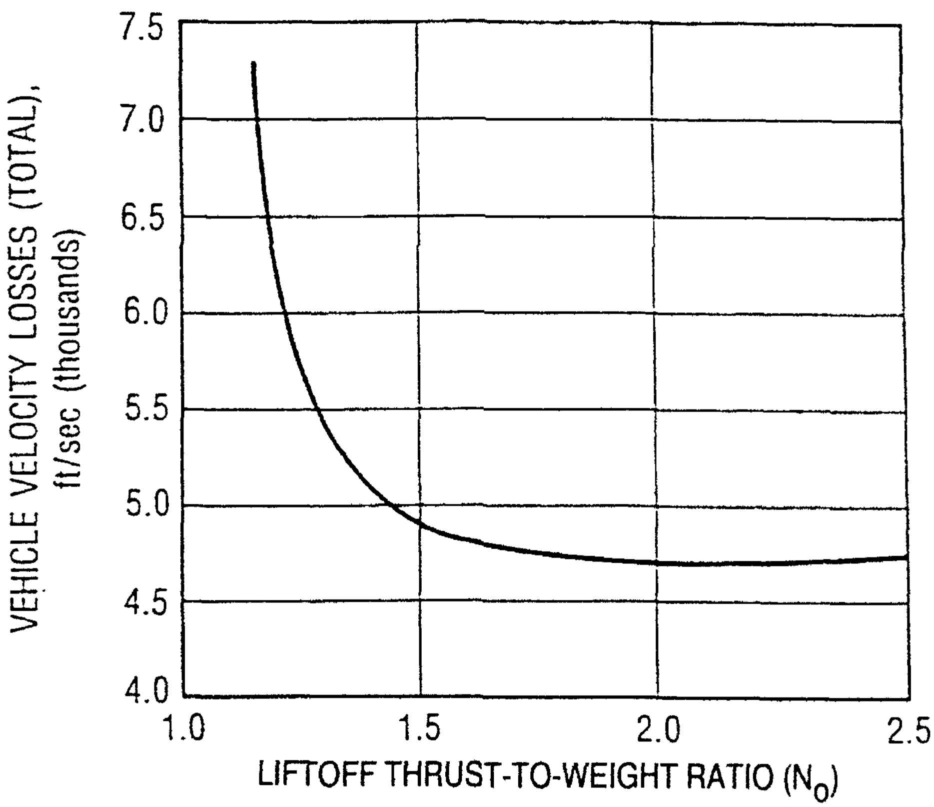

Figure 6-2. Relationship for ΔV_{Losses}

Table 6-2. Liftoff Thrust-to-Weight Ratio, U.S. Launch Vehicles

Vehicle Name	Launch Weight, lb	No. of Engines and Type	Burn Sequence	Thrust/Weight
Saturn I	1,165,000	8 liquid	Series	1.29
Saturn IB	1,292,000	8 liquid	Series	1.24
Saturn V	6,100,000	5 liquid	Series	1.23
Atlas (SLV-3)	260,000	3 liquid	Parallel	1.46
Atlas–Centaur	300,000	3 liquid	Parallel	1.23
Atlas II	293,000	3 liquid	Parallel	1.29
Titan II–GLV	300,000	2 liquid	Series	1.43
Titan II–SLV	340,000	2 liquid	Series	1.26
Titan IIIC	1,400,000	2 solid*	Series	1.71
Titan 34D/Transtage	1,514,000	2 solid*	Series	1.44
Titan 34D	1,492,200	2 solid*	Series	1.46
Titan III	1,492,200	2 solid*	Series	1.46
Titan IV/Centaur G	1,910,449	2 solid*	Series	1.36
Titan IV/IUS	1,885,525	2 solid*	Series	1.38
STS	4,523,000	3 liquid 2 solid*	Parallel	1.53
Delta 3914	420,500	1 liquid 9 solid*	Parallel	1.68
Delta 6920/PAM-D IMLV	483,500	1 liquid 9 solid*	Parallel	1.71
Scout SLV-1A	47,200	1 solid	N/A	2.27

*Strap-on SRMs

2. Velocity Gain Requirements for Orbital Transfer

The calculation procedures presented for launch vehicle sizing assume direct injection of the payload into circular orbit at 100 nmi. For higher orbits, it is generally more energy efficient to first insert the payload element of the vehicle into a circular parking orbit and then to execute a transfer maneuver to the final mission orbit using a rocket-powered transfer vehicle. Propulsive requirements for the transfer vehicle are minimized using a two-burn Hohmann trajectory in which the ΔV provided by the first burn is just enough to raise the vehicle to its final orbit altitude or apogee, at which point a second rocket burn is executed to insert the vehicle into its final mission orbit. The mission orbit may be either circular or elliptical.

In general, the velocity gain and associated vehicle sizing requirements for the transfer maneuver are relatively simple to calculate since gravity, drag, and thrust losses (particularly for short-duration impulsive burns) are negligibly small; i.e., $\Delta V_{Ideal} = \Delta V_{True}$. Sizing calculations may be accomplished using the same relations shown for launch vehicle stages; e.g., Eqs. (6-4),

Table 6-3. Liftoff Thrust-to-Weight Ratio, Foreign Launch Vehicles

Vehicle Name	Launch Weight, lb	No. of Engines and Type	Burn Sequence	Thrust/Weight
Soyuz SL-4 (USSR)	720,000	16 liquid 4 liquid	Parallel	1.56
Ariane 2 (France)	490,000	4 liquid	Series	1.22
Ariane 3 (France)	530,000	4 liquid 2 solid*	Parallel	1.60
Ariane 4 (France)	1,033,000	4 liquid 4 solid*	Parallel	1.32
N-2 (Japan)	297,600	1 liquid 9 solid*	Parallel	1.63
H-1A (Japan)	306,460	1 liquid 9 solid*	Parallel	1.58
H-2 (Japan)	528,000	1 liquid 2 solid*	Parallel	1.46
Mu-3S-2 (Japan)	136,400	1 solid 2 solid*	Parallel	2.62
SLV-3 (India)	37,500	1 solid	Parallel	2.53
FB-1 (China)	420,000	4 solid	Series	1.47

*Strap-on SRMs

Note: The systems equipped with nine SRMs use only six of them to produce liftoff thrust, and the other three are fired later.

(6-7), and (6-8) for an expendable single-stage design, taking ΔV_{Ideal} as the sum total of incremental velocity gains required to execute the transfer maneuver. Figure 6-3 shows the ΔV requirements for transfer without plane change between the assumed 100-nmi parking orbit and various mission orbits, circular and elliptical. The parameter h in the figure refers to the apogee height of the final mission orbit. Figure 6-4 shows the additional velocity increment required to accomplish the transfer maneuver with change of plane from parking to mission orbit. Figure 6-4 assumes that plane change is accomplished at apogee of the transfer ellipse where the ΔV requirement for the plane change maneuver is at a minimum.

a. Specific Impulse (I_{sp})

Specific impulse, I_{sp}, is one of the most important performance parameters used in rocket design. It is the thrust delivered by the rocket per unit of propellant flow rate and is usually expressed in the literature in terms of lb force/lb mass per sec. The magnitude of I_{sp} is principally determined by the chemical nature of the rocket propellant(s) but it is also affected by the design of the rocket nozzle, the combustion chamber pressure, and the ambient pressure in which the system operates.

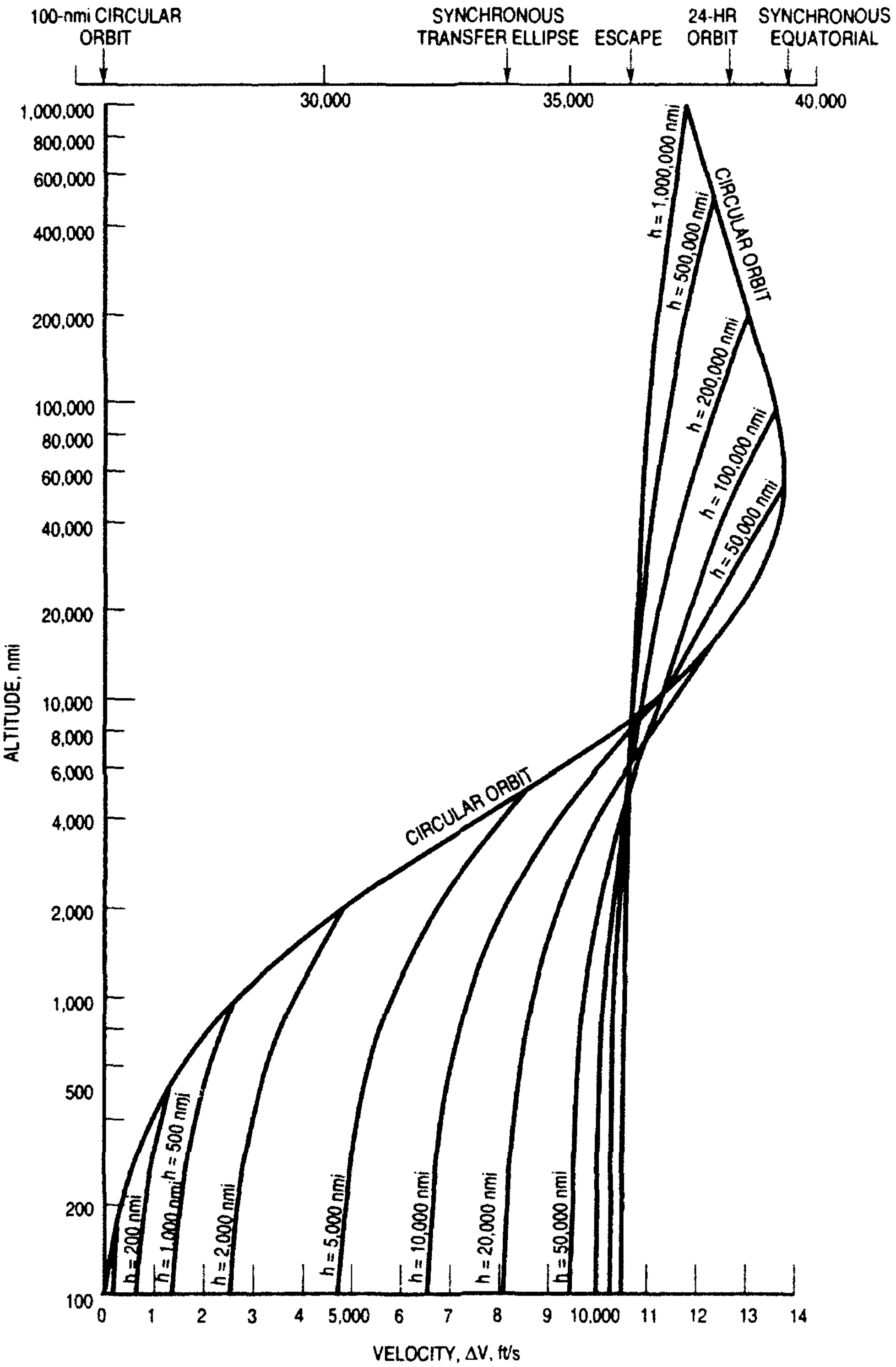

Figure 6-3. Orbit transfer ΔV requirements—no plane change.

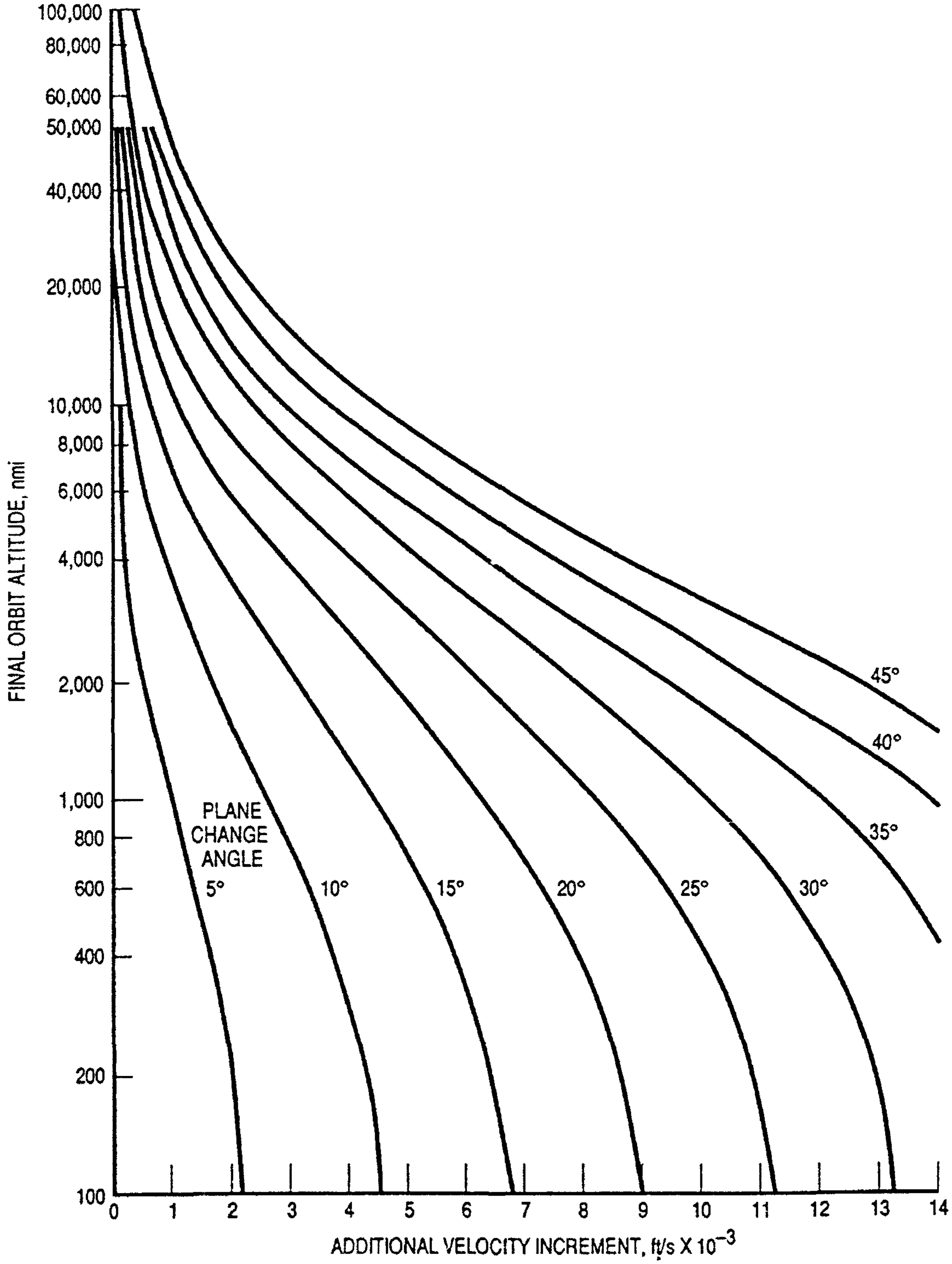

Figure 6-4. Additional velocity increment required for circularizing maneuver, simultaneous with a plane change at apogee.

To aid the designer in selecting an appropriate I_{sp} for input to the vehicle sizing relationships, Table 6-4 is provided showing representative values for various propellant types. Two sets of values are given: one for current technology and one for advanced technology. The advanced technology values are based on projected developments that might be implemented in the next 10 to 20 years.

For liquid propellants, the sea-level values indicated in the table (I_{spsl}) can be used for all Stage I designs, and the vacuum values indicated (I_{spv}) can be used for all upper stage designs. For solid propellant Stage I boosters, an average of the sea-level and vacuum values shown may be used. These curves assume that all stages have like parameters

b. Parametric Analysis

The sensitivity of vehicle size and performance to changes in the basic parameters of vehicle design can best be illustrated by the use of the dimensionless factors: initial gross weight to payload weight ratio, W_o/W_{PL}, and the ideal velocity ratio, $\Delta V_{Ideal}/gI_{sp}$. Figures 6-5 and 6-6, taken from Reference 6-3, show the influence on the weight ratio due to changes in the ideal velocity ratio and in the number of stages used. These curves assume that all stages have like parameters for MF_p and I_{sp} and are drawn for a structure factor σ of 0.10, where

$$\sigma = 1 - MF_p \tag{6-23}$$

The structure factor is usually defined as that fraction of the stage weight that is not usable propellant. It reflects both the efficiency of propellant utilization and the efficiency of stage structural design, so that structure factors lower than the curve value tend to reduce the gross weight of the vehicle for a given payload.

The curves indicate that multiple staging tends to reduce gross vehicle weight, with substantial benefit resulting from the addition of a second stage and with diminishing, though not insubstantial, benefits resulting from the addition of higher stages.

To demonstrate the effect of staging, consider the requirements for a minimum energy low-Earth orbit. For this case, using the highest I_{sp} presently attainable with chemical rocket propulsion, the parameter $\Delta V_{Ideal}/gI_{sp}$ is on the order of 2.1. Then from Figure 6-5, a single-stage launch vehicle design would have a gross-weight-to-payload ratio of 35, compared to a two-stage vehicle design which would have a ratio of 13. Thus, the curves suggest that a single-stage launch vehicle would not be practical for an orbital injection mission. It must be noted, however, that the weight ratio is sensitive to structure factor, σ. Therefore, advances in technology leading to improved structural efficiency (reduced σ) would give greater credibility to a single-stage-to-orbit approach. Indeed, a number of such vehicle designs have been proposed recently.

Table 6-4. Representative Specific Impulse Values for Various Propellant Types

LIQUID PROPELLANTS

TECHNOLOGY LEVEL	ITEM	STORABLE				LO_2/LH_2	
		STAGE I	STAGE II	STAGE I	STAGE II	STAGE I	STAGE II
CURRENT	PROPELLANTS MR Isp_v (sec) Isp_{sl} (sec)	NTO/A-50 1.90 301 254	NTO/A-50 1.79 316 —			LO_2/LH_2 6.00 453 365	LO_2/LH_2 5.50 436 —
ADVANCED	PROPELLANTS MR Isp_v (sec) Isp_{sl} (sec)			NTO/MMH 2.30 328 264	NTO/MMH 2.30 348 —	LO_2/LH_2 6.00 455 396	LO_2/LH_2 6.00 466 —

TECHNOLOGY LEVEL	ITEM	LO_2/HYDROCARBON					
		STAGE II	STAGE II	STAGE I	STAGE II	STAGE I	STAGE II
CURRENT	PROPELLANTS MR Isp_v (sec) Isp_{sl} (sec)	LO_2/RP-1 2.27 304.00 265.00	LO_2/RP-1 2.27 309.00 —				
ADVANCED	PROPELLANTS MR Isp_v (sec) Isp_{sl} (sec)	LO_2/RP-1 2.90 339 274	LO_2/RP-1 2.90 365 —	LO_2/CH_4 3.50 360 323	LO_2/CH_4 3.50 380 —	LO_2/C_3H_8 3.00 352 305	LO_2/C_3H_2 3.00 375 —

NOTE: Values shown for Stage II apply also to Stages III and IV

SOLID PROPELLANTS

TECHNOLOGY LEVEL	ITEM	VALUE
CURRENT	Isp_v Isp_{sl}	261 233
ADVANCED	Isp_v Isp_{sl}	273 245

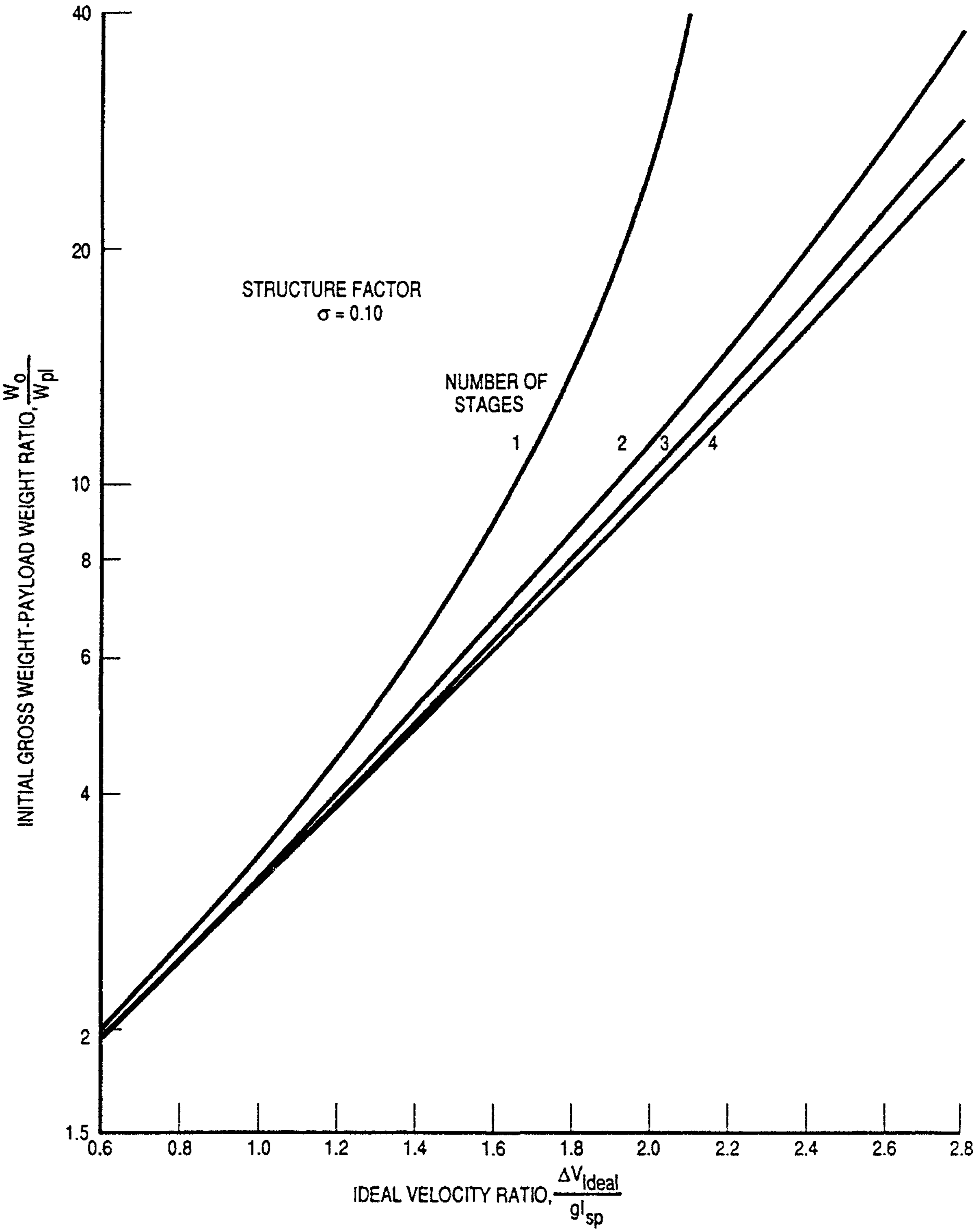

Figure 6-5. Payload ratio for like parameters (I).

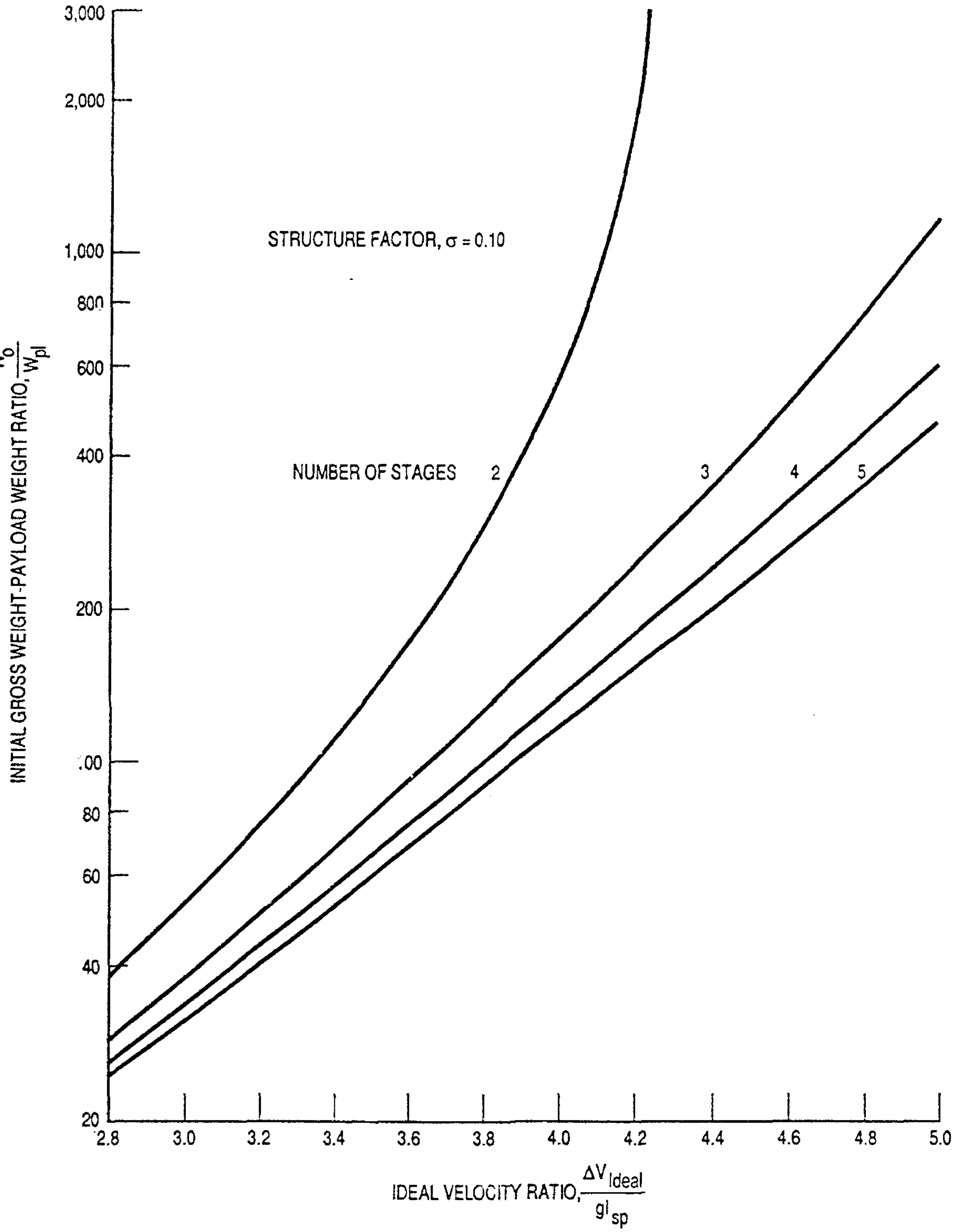

Figure 6-6. Payload ratio fo. ..ke parameters (II).

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VII. FUTURE SPACE LAUNCH VEHICLE CONCEPTS

Malcolm G. Wolfe, Anthony T. Zachary, Harry Bernstein

Numerous solutions have been proposed to satisfy future space transportation requirements. Projected requirements have generally been the outcome of a series of National Security Decision Directives (NSDDs) and a variety of broad national studies aimed at predicting the future of the U.S. role in space. (See References 7-1 through 7-7.)

Five major programs that are considered important to United States space transportation planning for the future are:

1. The National Launch System (NLS)
2. The Heavy-Lift Launch Vehicle (HLLV)
3. The Thermonuclear Transfer Vehicle
4. The Single-Stage-to-Orbit (SSTO) Vehicle
5. The National Aerospace Plane (NASP)

This section gives the current status and a brief description of proposed concepts for each of these programs.

A. NATIONAL LAUNCH SYSTEM (NLS)

1. Design Evolution

The Space Transportation Architecture Study (STAS) was initiated early in 1985 to identify the U.S. space transportation needs (manned and unmanned launch vehicles, orbit transfer stages, facilities, and mission control systems) of the future. The study, conducted under the joint management of the U.S. Department of Defense (DOD) and the National Aeronautics and Space Administration (NASA), established the need for a new heavy-lift, unmanned cargo vehicle. It was determined that to best fulfill space transportation objectives, primary focus must be directed toward more effective operations, more operability, and a significant reduction in recurring costs. It was concluded that a new launch vehicle program should be initiated, and a primary element of any new program would be an early investment in technology necessary to meet the nation's space transportation goals. The study established the broad systems architecture needed to deliver a wide range of anticipated payloads.

A program was initiated in 1987 called the Advanced Launch System (ALS) as a logical follow-on to the STAS effort. Under congressional mandate, the program was to be a jointly managed NASA/DOD effort to meet the future needs of the nation. ALS was to be oriented toward improved reliability, operability, and economy. Congress established a cost-reduction goal from the current recurring cost of approximately \$3000 to \$5000 per pound to \$300 per pound of payload in orbit. Of necessity, these objectives dictated a "clean sheet" approach and major changes in the government's and aerospace industry's way of doing business. In support of this new effort. Total Quality Management (TQM) methods (management techniques proven successful in Japanese industry) were adopted and associated educational processes initiated.

The ALS program consisted of three major parts: the operations and vehicle systems studies, engine design concept studies, and a technology development program focusing on engine technology, but with many tasks directed toward other key vehicle and operations technologies. The engine technology tasks emphasized cost reduction through simplification (parts reduction), near-net-shape fabrication (e.g., castings in lieu of machining), and less demanding operating conditions. The latter is reflected in the adoption of a gas-generator engine cycle operating at a lower chamber pressure than the more complex staged-combustion space shuttle main engine (SSME). Vehicle-related technology tasks were directed toward a new tank material (aluminum lithium), elimination of hydraulic systems, increased reliability through redundancy, margins/reserves, recoverability, etc. Operations-related technology tasks pertained to paperless management concepts, laser-initiated pyrotechnics, automated checkout, and reduced manpower for launch processing, mission planning, and mission control. This technology program is still in progress with a scope commensurate with technology needs identified in system concept studies.

A series of ALS trade studies was conducted to define not only the launch vehicle configurations but also manufacturing concepts, operating concepts, processing facilities, and techniques for fulfilling reliability and operability requirements. The trade studies outlined the need for a family of vehicles to cover the 30,000 lb to 220,000+ lb payload to LEO range, as envisioned for meeting the nation's space transportation needs to LEO (low inclination and polar), geosynchronous orbit, other high-energy orbits, space-station resupply, and deep-space missions. System studies showed economies attainable with family-of-vehicles common elements to provide a "building block approach." The baseline configuration utilized a common core with strap-on, parallel-burn liquid boosters providing improved reliability via the startup of all engines on the ground, with vehicle release when start of all engines had been verified. Technology work on moderate-sized (approximate weight of 300,000 lb), monolithic solid strap-on boosters was carried on as an alternative approach to provide cost competition to the liquid booster.

Economic considerations dictated the need for developing a single, new liquid engine for the various launch vehicle applications. The engine configuration selection was based on priorities of low cost, without compromising reliability, while providing adequate performance (in contrast to the performance maximization objective of prior launch vehicles). The analysis performed for propellant combination selection, in following the ground rule of trading performance for cost, resulted in a detailed comparison between oxygen/hydrogen and oxygen and each of three hydrocarbon fuels. The three hydrocarbon fuels evaluated were methane, propane, and RP-1. Oxygen/hydrogen was selected for the core vehicle based on the need for adequate performance and for liquid boosters based on commonality with the core vehicle requirements.

To enhance vehicle reliability, health management concepts were adopted to indicate unsatisfactory conditions, to enable vehicle systems to terminate subsystem operations (e.g., engine) before an unsatisfactory condition turned into a catastrophic failure, and to provide redundancy management capabilities. With "engine-out" capability (and fault-tolerant avionics) to fulfill the mission, the reliability of multiengine configurations would be substantially increased.

The use of common tankage and engines was seen as a very cost-effective method of providing boost capability if the engine cost remained as projected. The modular approach provided increased payload capability with additional strap-on boosters. By utilizing two boosters and the common core, a payload of 220,000 lb in LEO could be achieved. A key decision related to recurring cost that has yet to be settled is the desirability and practicality of recovering the engines from flight vehicles. The recovery of the booster engines was generally accepted as not only feasible but also highly desirable. However, core engine and avionics recovery still remains a hotly debated issue.

Early in the study process, the manufacturing and operational aspects of the program were seen as major contributors to significant cost reduction. Enhancements such as "on-launch-site" fabrication and assembly, automated checkout, and improvements in leak detection were investigated. Major operation concept and facilities studies were also conducted, resulting in the selection of clean, multipurpose launch pads, integrate-transfer-launch concepts featuring the use of a mobile launch platform with an umbilical mast, and no fixed towers on the pad. Such concepts are also necessary for achieving high launch rates, schedule dependability, and operational flexibility. The result of these studies has been a significant database for future decisions.

Concurrent with the ALS studies, NASA's Marshall Space Flight Center initiated a parallel effort to define a heavy-lift capability for limited use in deploying and servicing Space Station Freedom (SSF), achievable with minimum up-front investment by utilizing space shuttle facilities and components. The vehicle would provide unmanned cargo capacity by substituting a side-mounted cargo module in place of the Orbiter or a cargo module in tandem on top of the external tank (ET). The advanced solid rocket motors (ASRMs) planned for future shuttle operations would also be utilized. New space shuttle main engines, which are fully reusable, were considered too costly to use in an expendable mode. However, if the specific engines had already been utilized on prior shuttle flights, it was believed that the per-mission cost would be substantially reduced, thus allowing them to be expended.

In late 1990, the decision was made to merge the two competing approaches into a single program having a family of vehicles fulfilling both the DOD and NASA needs. Thus, the National Launch System (NLS) was born. The approach was to be evolutionary in contrast to the "clean-sheet" approach that was being pursued by ALS. To minimize up-front cost, NLS would utilize some elements of existing systems and infrastructure, while simultaneously adopting key elements of the "clean-sheet" approach such as the development of a new low-cost engine, the introduction of concepts and new facilities to provide improved reliability and operability, and other cost-saving concepts of the ALS program. Also, NLS is to have the potential for future introduction of new technologies.

2. Family of Vehicles

The family of vehicles currently serving as points of reference, and now under study, range from a small, two-stage vehicle capable of putting approximately 20,000 lb in LEO; a one-and-one-half-stage vehicle utilizing a shuttle external-tank-derived tank section (common core) for moderate-sized payloads to LEO and capable of being upgraded to a two-and-one-half-stage vehicle (by adding an upper stage) with GEO capability; and the heavy-lift configuration using strap-on ASRMs for boosters, the common core, and a cargo transfer vehicle (CTV) for cargo delivery to SSF, as shown in Figure 7-1. A new NLS upper stage used with the one-and-one-half-stage vehicle for GEO missions also is being considered for use as the final stage of the 20,000-lb vehicle to further provide commonality. Vehicles with increased payload capability attainable via modular growth to meet heavy-lift requirements up to 124,000 lb are also under study. Payload estimates for the various members of the NLS family are given in Figure 7-1; both NASA and DOD requirements can be satisfied. The primary interest of the DOD in the one-and-one-half-stage is to place 50,000 lb into an 80- x 150-nmi orbit; NASA's interest is to use it to deliver 18,000 lb of net payload to SSF. DOD plans to use the two-stage vehicle to deliver 20,000 lb to an 80- x 150-nmi orbit, 4000 lb to GEO, or 8,000 lb to GTO; NASA would use it to deliver 55,000 lb of net payload to the SSF. The two-and-one-half-stage vehicle would deliver 97,000 lb to the 80- x 150-nmi orbit or 15,000 lb to GEO; it could also deliver 83,000 lb of gross payload to the SSF orbit. The HLLV option could deliver 135,000 lb to the 80- x 150-nmi orbit or 124,000 lb of gross payload to the SSF orbit.

NLS launch capabilities will be first implemented at NASA/KSC, using the shuttle vehicle assembly building (VAB) and LC 39. Next, NLS will be implemented at CCAFS with a single launch pad and processing facilities compatible with the ITL concept. NLS expansion planning envisions additional launch capabilities at VAFB.

In April 1991, the NLS program was reviewed and approved by the National Space Council as a major initiative for fulfilling space transportation lift capability, operability, and cost-reduction needs.

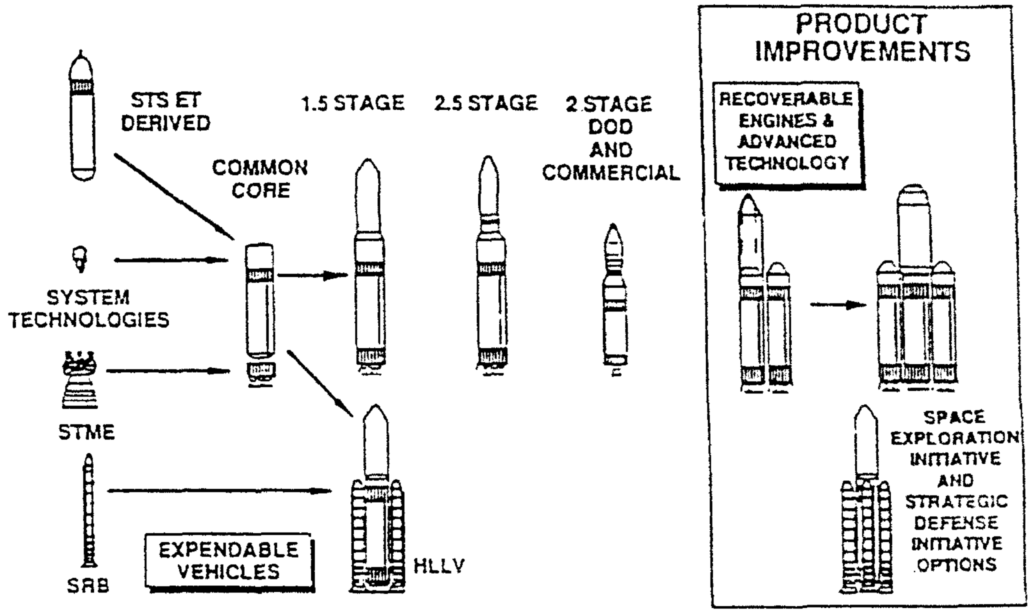
B. HEAVY-LIFT LAUNCH VEHICLE (HLLV)

1. Space Exploration Initiative (SEI)

Apollo II placed the first humans on the moon on 20 July 1969. On the 20th anniversary of that event, President George Bush announced a new vision in the 21st century—to return and establish a permanent presence on the Moon and to land a human on Mars by the year 2019. This vision, the Space Exploration Initiative, represents probably the greatest technological challenge the world has ever faced.

The Moon, at a quarter of a million miles from the Earth's surface, is the nearest object in space where people can live under conditions similar to those to be faced on other planets. Thus, the Moon is a natural test bed to prepare for missions to Mars through simulation, systems testing, operations, and studying human capabilities. Of all the planets in our solar system, Mars is the most like Earth. With a thin atmosphere, weather, seasons, and a 25-hour day, Mars has a dense and complex surface, including ice and evidence of water. Although conditions on Mars cannot support life now, available data suggest that Mars was warmer, wetter, and had a much denser atmosphere early in its history and may have been able to support life as we know it.

VEHICLE EVOLUTION



NET PAYLOAD DELIVERY (Klb) ESTIMATES

	ORBITS			
VEHICLES	80 X 150 nmi	220 nmi-SSF	GEO	GTO
1 1/2 STAGE	50 (57)	18 / 46 (24) / (62)		
2 STAGE	20		4	8
2 1/2 STAGE	97 (105)	55 / 83 (62) / 90	15	
HLLV - ASRMs / 4 STMEs	135 (135)	96 / 124 (96) / (124)		

NET DOCKED AT SSF

GROSS TO ORBIT

WEIGHTS IN PARENTHESES ARE FOR NO-ENGINE-OUT CAPABILITY

DOD PRIMARY INTEREST - $\square$

NASA PRIMARY INTEREST - ⬭

Figure 7-1. NLS vehicle evolution and payload delivery estimates.

At its closest point, Mars is 35 million miles from Earth. This distance increases to 230 million miles when the Earth and Mars are on opposite sides of the sun. The journey to the Moon takes about three days. Using conventional chemical propulsion, a mission to Mars will take about 230 days one way and require surface stay times of about 500 days to allow the planets to realign before returning home. Advanced nuclear propulsion technology can shorten the transit time, provide flexible surface stay times, significantly reduce the propellant mass to LEO, and increase the available launch opportunities.

The Lunar/Mars mission will require the transfer of several hundred tons of equipment and fuel; thus a heavy-lift launch capability will be required to minimize assembly in Earth orbit. The use of nuclear propulsion for Earth–Mars transfer (Reference 7-8) would permit the weight to be reduced to approximately one-half that of chemical systems and achieve faster interplanetary trip times. Both of these capabilities were available in the United States in the 1970s.

A schematic for a typical lunar landing mission is illustrated in Figure 7-2. The delta velocity budget for this mission is given in Table 7-1.

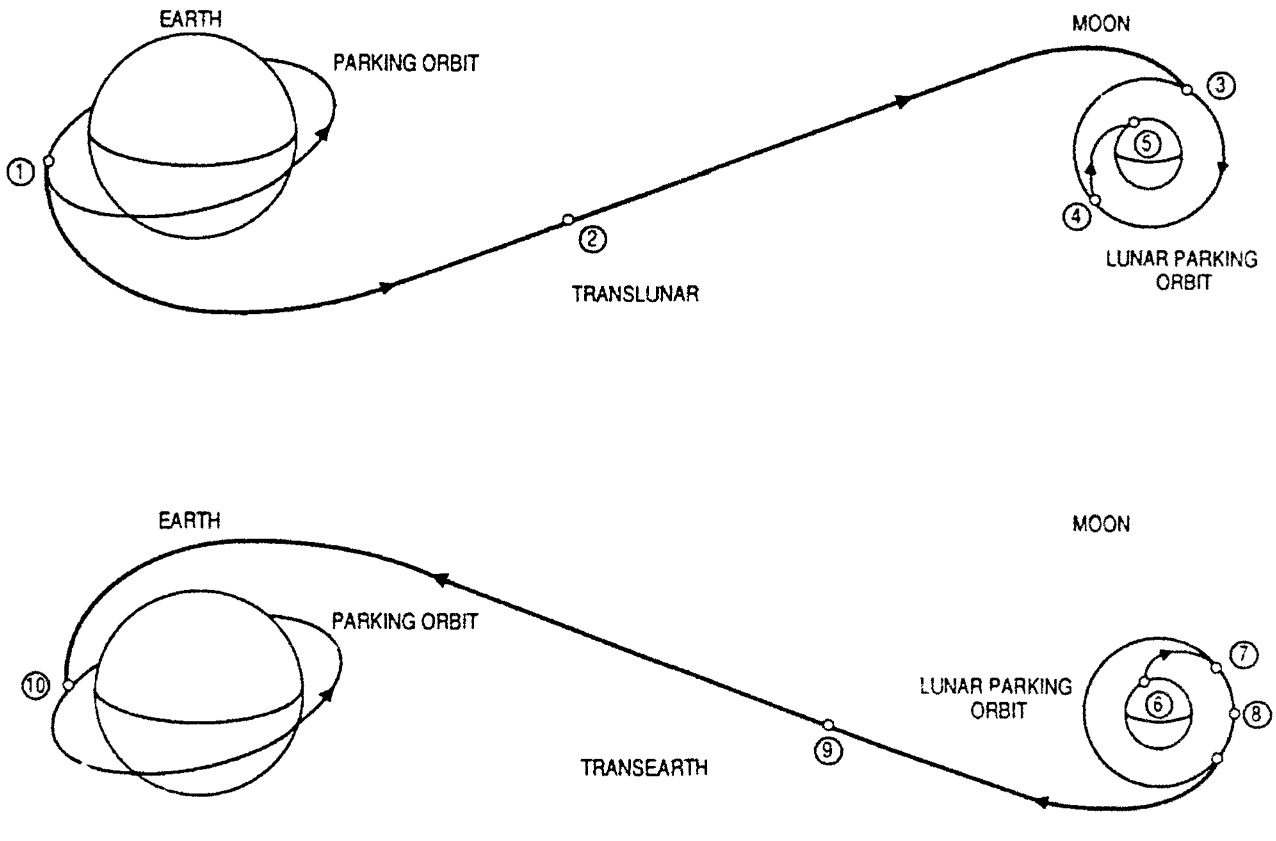


Figure 7-2. Schematic of typical lunar landing mission.

Table 7-1. A Typical Lunar Landing Mission
(270 x 270 x 28.5 deg Earth orbit to 60 x 60 x 90 deg lunar orbit. No plane changes)

Event	ΔV, ft/sec
1. Translunar Injection	10,700
2. Midcourse Maneuver	100
3. Lunar Orbit Insertion	3,100
4. Deorbit Burn	200
5. Lunar Landing Burn	6,000
6. Launch From Surface	5,800
7. Circular Orbit Insertion	200
8. TransEarth Injection	3000
9. Midcourse Maneuver	100
10. Earth Orbit Insertion	10,700
Total Mission	39,900 ft/sec
Mission Without Lunar Landing	27,700 ft/sec

2. Current Heavy-Lift Capability

The Saturn V is representative of a heavy-lift capability that existed in the late 1960s and was used for the United States manned missions to the Moon (the Apollo mission). The Saturn V could lift 280,000 lb to LEO. Versions of the Saturn/Apollo vehicle included Saturn I, Saturn IB, and Saturn V. Several derivatives of the Saturn V, with payloads up to 700,000 lb for a Mars mission, were proposed but never built.

The CIS Energia is representative of current heavy-lift capability and was first flown on 15 May 1987 carrying an unknown unmanned payload. On 15 November 1988, it was used to launch the Soviet equivalent of the United States space shuttle Orbiter, the Buran, which translates to "Snowstorm." The Buran, unlike the shuttle Orbiter, is capable of automated reentry and landing.

The Energia was flown in 1989 with four liquid strap-on boosters capable of lifting almost 200,000 lb to LEO. The Energia can accommodate varying numbers of strap-on boosters from two to eight to provide different lift capabilities up to more than 400,000 lb to LEO.

3. Proposed Heavy-Lift Capability

In the far term, the U.S. has made a national commitment to develop heavy-lift capability (see Reference 7-9). Numerous advanced, partially reusable launch-vehicle concepts have been proposed for advanced programs that require heavy lift, such as the Solar Power Satellite Program and manned planetary missions. The gross liftoff weight of such vehicles is typically in the range of 10 to 12 million pounds, compared to 6.4 million pounds in the case of the fully expendable Saturn V.

Recent heavy-lift studies have focused on three approaches:

1. Shuttle-derivative concepts, which make maximum use of the shuttle external tank and substitute liquid rocket boosters for the current SRBs. Payloads on the order of 100,000 to 150,000 lb are the focus of interest for such designs.

2. "Clean-sheet" concepts, which attempt to minimize structural weight, but which also make provision for recovery of high-value items, such as propulsion and avionics. These are items installed in propulsion/avionic (P/A) modules that are recovered after each launch, refurbished, and reused. The recoverable designs examined have about the same payload capacity as the shuttle-derivative concepts, but operational costs are reduced.

3. Very heavy lift "clean sheet" concepts, which are designed to satisfy SEI (Space Exploration Initiative) high launch capacity requirements in the range from 400,000 to 700,000 lb and higher. Shuttle-derivative vehicles tend to reach a lower limit of payload capability and therefore are not suitable for this purpose. Three very heavy lift options that have been studied by the NASA Marshall Space Flight Center are illustrated in Figure 7-3.

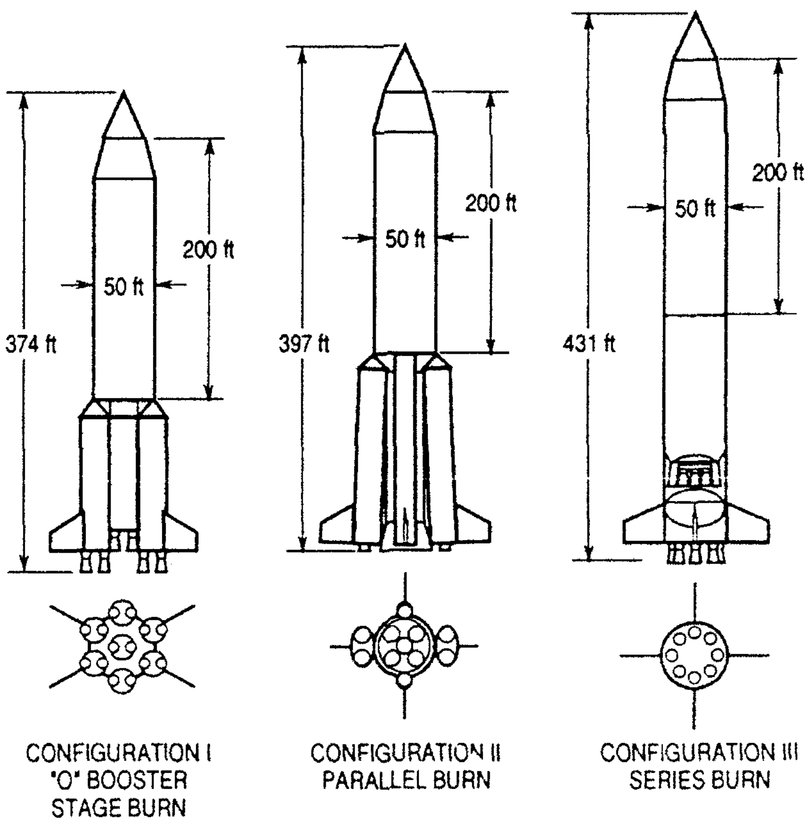


Figure 7-3. HLLV options comparison.

C. THERMONUCLEAR TRANSFER VEHICLE

A nuclear propulsion system offers the advantage of not being dependent on the limited energy available in chemical reactions, since it uses the energy release (ΔE) obtained in the mass changes (ΔM) that occur in nuclear reactions according to Einstein's equation

$$\Delta E = \Delta mc^2$$

where c is the velocity of light.

Nuclear propulsion systems have the potential to provide much more energy than chemical systems, although this is achieved at the expense of added mass for shielding and added complexity. Further information on nuclear propulsion is provided in References 7-10 through 7-13.

1. Nuclear Thermal

The nuclear-thermal rocket is a device that uses a nuclear reactor to heat the propellant to high temperatures. The propellant is then expanded by a supersonic nozzle to produce thrust, similar to a conventional chemical rocket. The nuclear-thermal rocket uses a low-molecular-weight propellant (probably hydrogen), to give a substantial increase in performance over chemical systems.

Nuclear-thermal rockets underwent substantial development in the 1960s and the early 1970s under the Reactor Inflight Test Vehicle (RIFT) program and the Rover/Nuclear Engine for Rocket Vehicle Applications (NERVA) program. A series of full-power reactor/engine tests resulted in propellant temperatures in excess of 2,700 K and a specific impulse of 845 sec. One 1,125-MW reactor power test was run continually for one hour. In addition, a reactor power test demonstrated 28 automatic startup/shutdown sequences, and a thrust level of 250,000 lb was demonstrated. Concepts have been proposed using a 5,000-MW Nerva 2 engine with a specific impulse goal of 925 sec.

Although no integrated rocket system was ever flight qualified or flown, the program did generate substantial test experience prior to program termination in 1972. Nuclear thermal rockets, with further development, are an attractive option for the interplanetary phase of the Mars mission.

2. Nuclear Electric

In common with nuclear-thermal propulsion, nuclear-electric propulsion depends on nuclear reaction as a source of energy. However, nuclear-electric propulsion uses the reactor to power electromagnetic thrusters that electromagnetically accelerate the propellant. A schematic of a nuclear-electric-propelled vehicle is shown in Figure 7-4.

Nuclear-electric systems provide higher specific impulses than nuclear-thermal systems but at the cost of much lower thrust. The lower thrust mandates a longer transfer time.

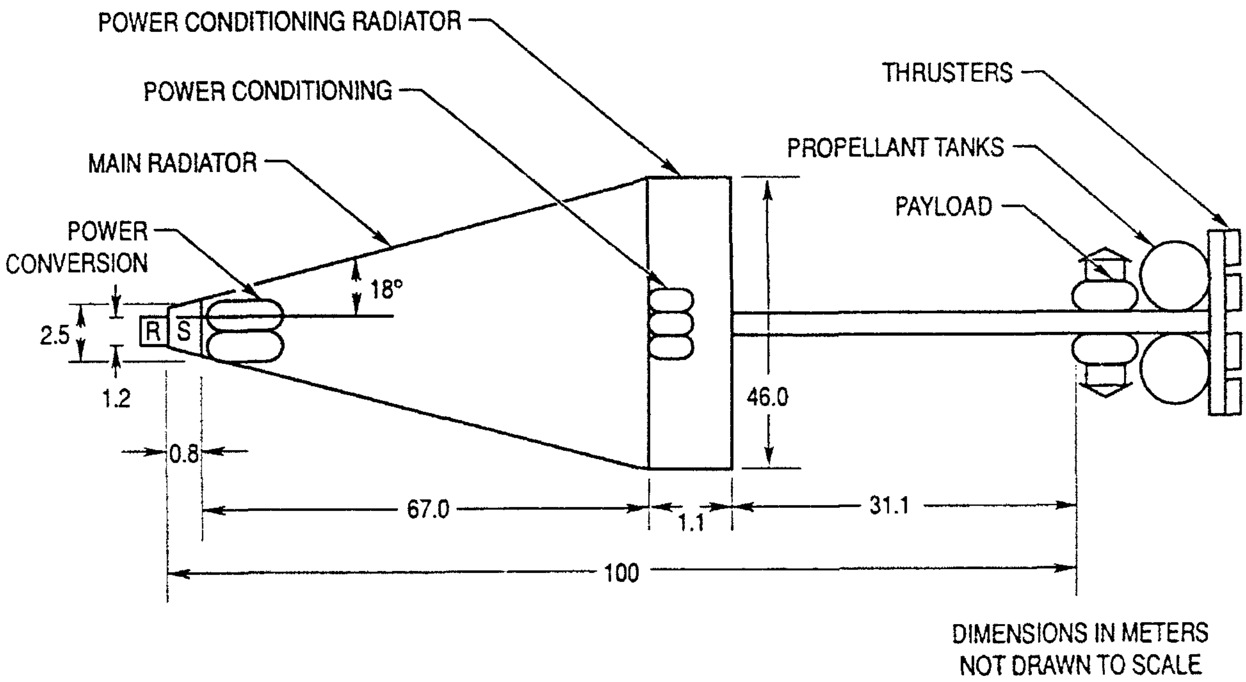


Figure 7-4. Schematic of a nuclear-electric propulsion vehicle/system.

3. Technology Requirements

Since 1972, advances in materials and fuel technology promise higher temperatures leading to higher performance engines. Newer concepts, such as the compact-particle-bed reactor, offer the potential for high-power-density reactor cores, which could lead to substantially higher integrated thrust-to-weight ratios. A list of nuclear-thermal propulsion concepts that were presented at a 1990 workshop is given in Table 7-2.

Table 7-2. Nuclear-Thermal Propulsion Concepts

• Dual mode	• Wire Core Reactor
• Gas Core—Open Cycle A	• Advanced DUMBO
• Gas Core—Open Cycle B	• Pellet Bed Reactor
• Gas Core—Light Bulb	• Foil Reactor
• Enabler (NERVA-based)	• Liquid Annulus Reactor
• Low-Pressure Core	• Droplet Core Reactor
• Particle Bed Reactor	• Boiling Metal Reactor*
• Cermet Reactor	• Tungsten Reactor*
• Nuclear Rocket Using Indigenous Martian Fuel (NIMF)	

*These two concepts were considered after the workshop.

A high thrust-to-weight engine would be particularly attractive for a second-generation upper stage of an advanced heavy-lift launch vehicle. To provide propulsion for Moon and Mars cargo missions, where transit time is not an important constraint, low-thrust nuclear-electric propulsion systems are attractive because of their very high specific impulses, which range from 3,000 to 10,000 sec. The impact of specific impulse on trip times is shown in Figure 7-5. While the development of nuclear-electric thrusters is moderately well advanced, the main issue in the technology status of these systems is the lack of space-qualified nuclear power systems in the 1- to 5-MW range. About 30 electric thrusters have already been flown in space.

Promising nuclear-electric thrusters include ion and magnetoplasma-dynamic engines. Ion engines use a noble gas, such as xenon or argon, as a propellant. Ion systems have specific impulses approaching 10,000 sec but have low thrust levels. Magnetoplasma-dynamic thrusters have demonstrated high performance with specific impulses ranging from 3,000 to 6,000 sec.

4. Safety

Using a thermonuclear rocket raises the safety issue concerning the accidental release of radioactive material and the radioactivity produced by fission by-products in the engine exhaust. The decay of these radioisotopes releases secondary radiation, such as gamma rays, and poses a threat to the crew if the vehicle is manned. The issue of meeting all the necessary safety and environmental standards will be a substantial challenge.

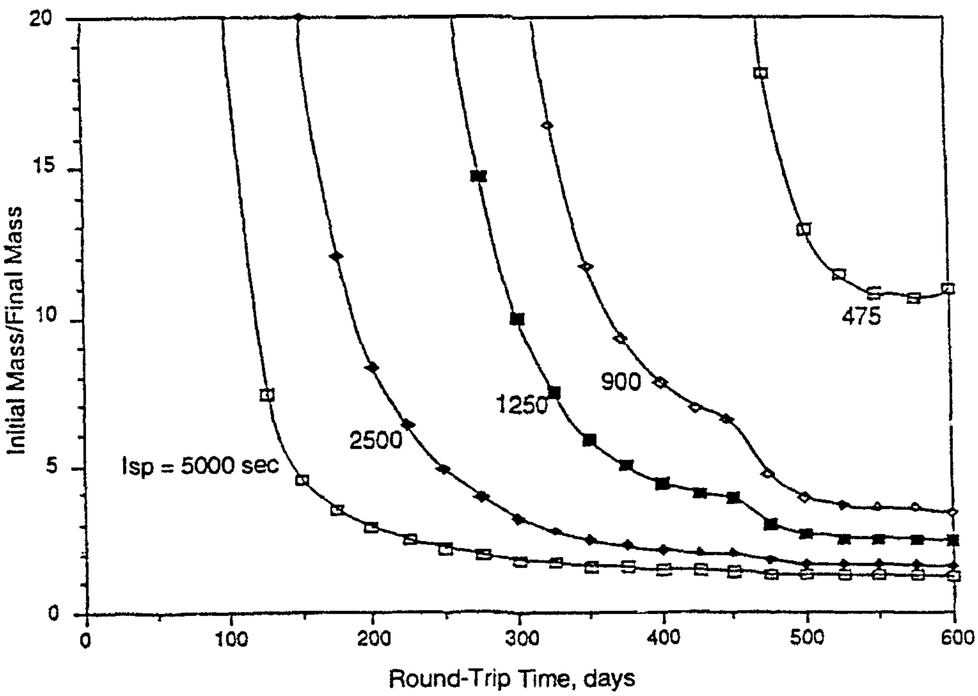


Figure 7-5. Propulsion system impacts on total mission trip time.

D. SINGLE-STAGE-TO-ORBIT (SSTO) VEHICLES

A viable single-stage-to-orbit vehicle has long been the desire of many space transportation planners because of its potential for reduced operational complexity and cost and its launch-on-demand capability for critical military systems. The Space Defense Initiative Office (SDIO) has examined a number of SSTO configurations to satisfy SDIO mission launch requirements. The SSTO program has some technological aspects in common with the National Aerospace Plane (NASP) program described below. Major differences between the two programs are:

1. The NASP effort (the X-30 program) is focused on building an air-breathing SSTO technology demonstrator, with plans for a NASP-derived operational vehicle, whereas the SSTO effort is focused on building a prototype, operational, rocket-powered SSTO vehicle.
2. The X-30 vehicle will carry test instrumentation but have no discretionary payload capability, whereas the SSTO is required to deliver 15- to 20,000-lb payloads to LEO.
3. The X-30 vehicle will take off and land horizontally, whereas an SSTO may take off and land either horizontally or vertically.

The performance goals of the SSTO are:

1. 7-day turnaround
2. 10,000-lb polar, 15,000 to 20,000 lb due east
3. 3,000 ft^3 containerized payload bay
4. Return-to-launch-site and abort-once-around capability ($<$ 1,300 nmi cross range)
5. 3 g maximum acceleration/low dynamic pressure

1. SSTO Configurations

Four different configurations were studied in Phase I of the SSTO program. These are illustrated in Figure 7-6. The Vertical-Takeoff-and-Landing (VTOL) concept has been selected for study in Phase II.

2. Key Enabling Technologies

The achievement of single stage to orbit is primarily dependent on reducing the structure factor of the vehicle and increasing the specific impulse of the propulsion system. Figure 7-7 illustrates the sensitivity of gross liftoff weight (GLOW) to propellant mass fraction and specific impulse. Advanced oxygen/hydrogen engines, such as those used by the space shuttle, for instance, have a specific impulse on the order of 400–450 sec. Current expendable vehicle propellant mass fractions are about 0.90. If a mass fraction of 0.92 and a specific impulse of 460 sec can be achieved, the SSTO design will be able to move away from the dangerous knee of the curve

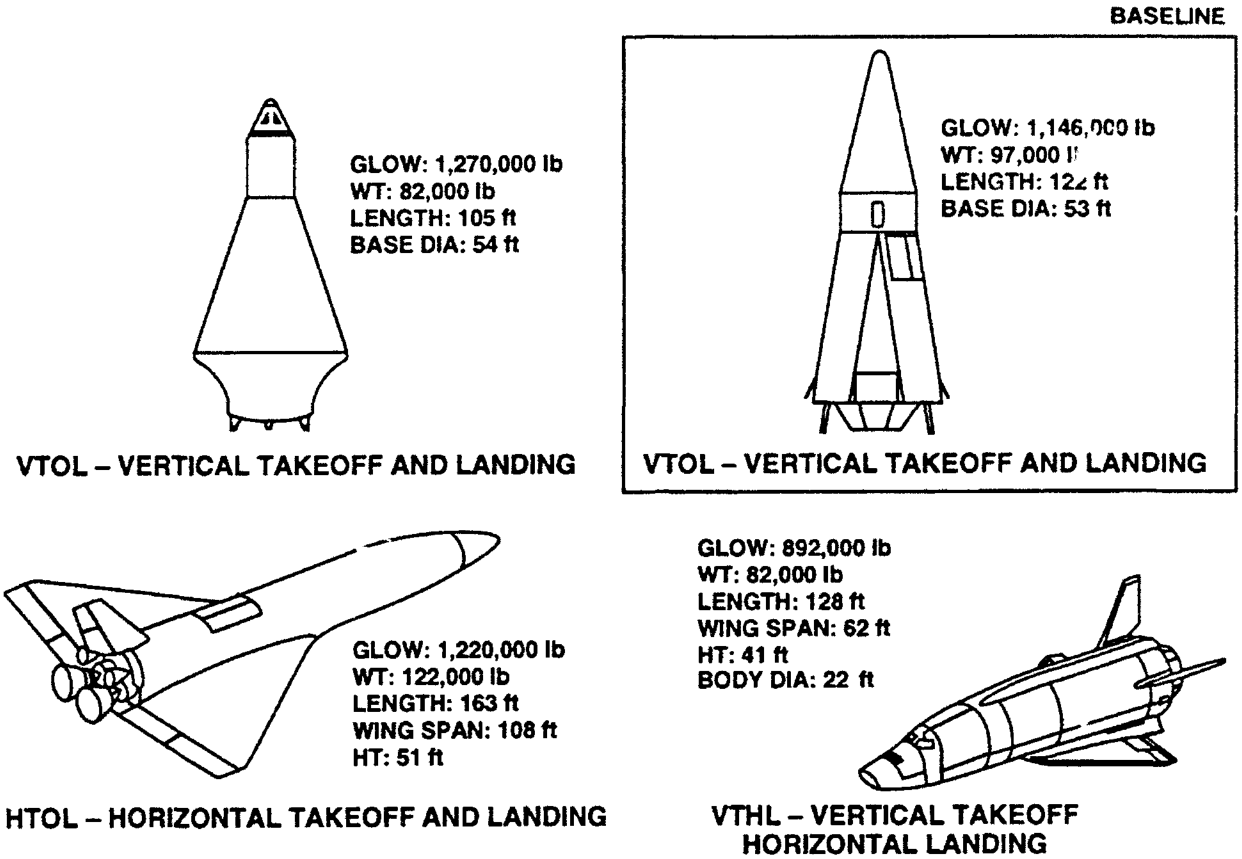


Figure 7-6. SSTO options.

where a very slight increase in structural weight could propel the design into the negative payload regime. A number of key technologies are steadily evolving, leading to the conclusion that the SSTO could become a viable concept.

These technologies include the following:

- Propulsion
  - Aerospike engine
  - Twin spool turbopumps
  - Integrated main propulsion and orbital maneuvering system (OMS)
  - Leak detection and isolation
- Structural
  - Titanium metal matrix composites
  - Carbon/carbon with coatings
  - Titanium
  - Aluminum-lithium

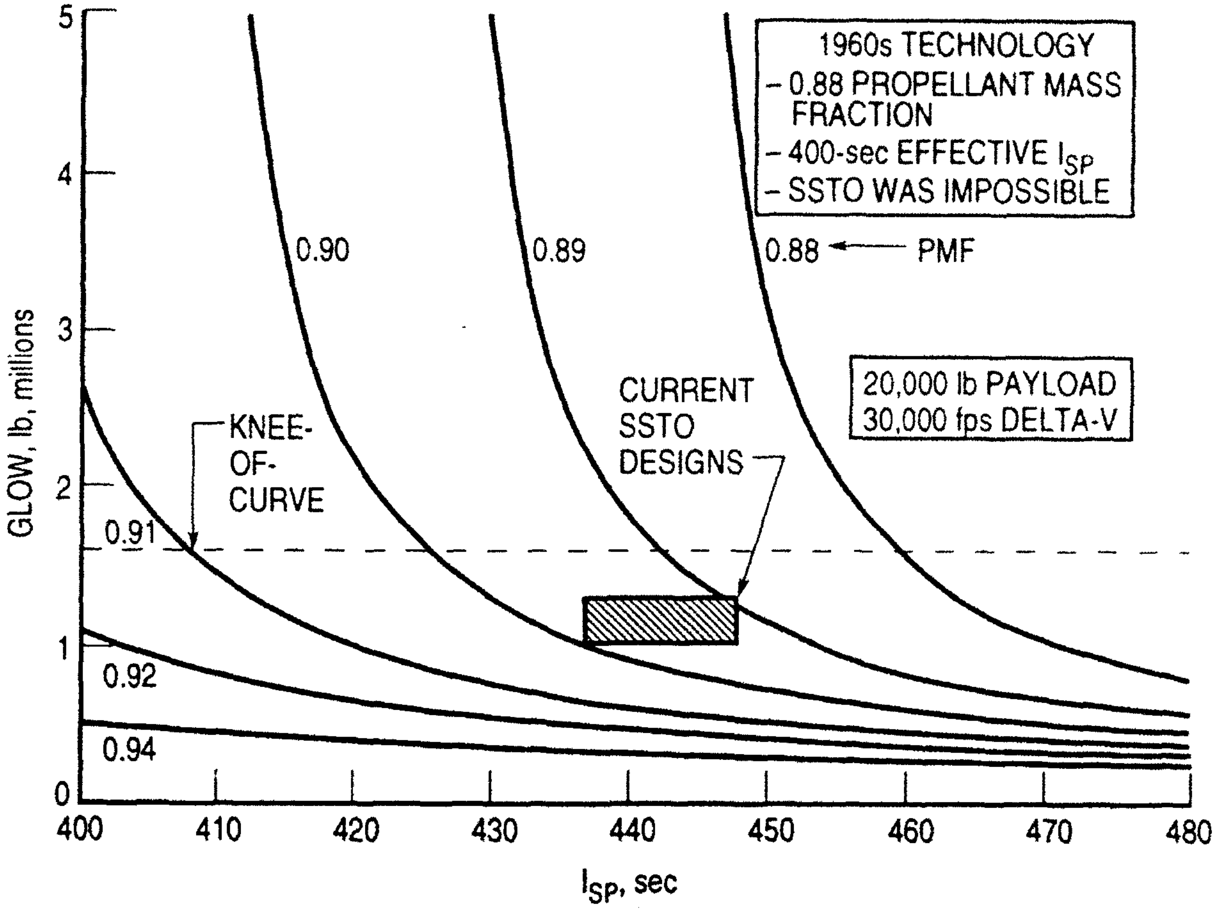


Figure 7-7. SSTO gross liftoff weight vs I_{sp} and propellant mass fraction.

- Tanks
  - Graphite epoxy/thermoplastic
  - Aluminum-lithium LOX tank
- Operations
  - Electromechanical actuators
  - Laser ignition
  - Inflight health and monitoring
  - Built-in test
- Optimization
  - Computational fluid dynamics
  - Vehicle and trajectory optimization codes

If the maturity level of the technology is inadequate to support the development of a pure SSTO, two optional fallback concepts can be considered. One is to provide solid rocket augmentation at liftoff, the rockets being jettisoned as soon as they are expended. Another option is to launch the SSTO vehicle from an air-breathing platform, an approach being taken by the German Sanger concept and the joint British/CIS program, which proposes to launch the British interim HTOL vehicle from an Antonov AN-225 transport aircraft.

E. NATIONAL AEROSPACE PLANE (NASP)

In early 1986, the United States initiated a government/industry effort focused on those technological advances that will permit development of future air-breathing, single-stage-to-orbit hypersonic cruise vehicles. These future vehicles will require advanced aerodynamics and propulsion systems, resulting in a need for a high effective specific impulse, reduced structural weight, highly integrated vehicle systems, and validated computational methods for system design at the high Mach numbers beyond current ground test capabilities ($M > 8$). The importance of developing these technologies is not only recognized in the United States; British, French, German, Japanese, and CIS hypersonic programs are also in various stages of development.

The NASP program has the overall goal of providing the technological basis for future hypersonic flight vehicles for application to both civilian and military systems. Currently, the X-30 experimental flight test vehicle is being developed by a consortium of industrial contractors. This is not a prototype of any specific civilian system but will be used to demonstrate the requisite technology for such future systems, including airplanes with hypersonic cruise capability and launch vehicles for payload delivery to orbit. These vehicles will use conventional runways for takeoff and landing.

Hypersonic aircraft can reduce the trip times between points on the globe (e.g., London to Tokyo) by 75 percent compared to the Boeing 747. The improved operational efficiency of a single-stage, horizontal-takeoff-and-landing launch vehicle is expected to reduce the recurring cost of placing payloads into orbit by an order of magnitude.

The achievement of high effective specific impulse is dependent on the design and optimization of all the interrelated vehicle components, including the airframe and the engine. Because the airframe forebody provides the aerodynamic compression for the propulsion system and the afterbody provides the expansion surface (the nozzle), the optimum integration of airframe and engine is a key technology for these vehicles. Reduction in structural weight and improvements in fuel weight fraction require the development of new, high strength-to-weight materials that can maintain their characteristics over a wide temperature range. Also required is the development of innovative structural concepts that will withstand the heat and aerodynamic loads, provide cooling for critical parts of the airframe and propulsion system, and be fully reusable. Recent advances in computational capability will permit the design and analysis of vehicle and vehicle systems for flight regimes where ground simulation is impractical. Computational codes require as much validation as possible through existing simulation capabilities tailored directly to NASP-related analysis. In addition, the effective integration of thermal management, flight controls, propulsion controls, and other vehicle subsystems is important for achieving optimum vehicle design.

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