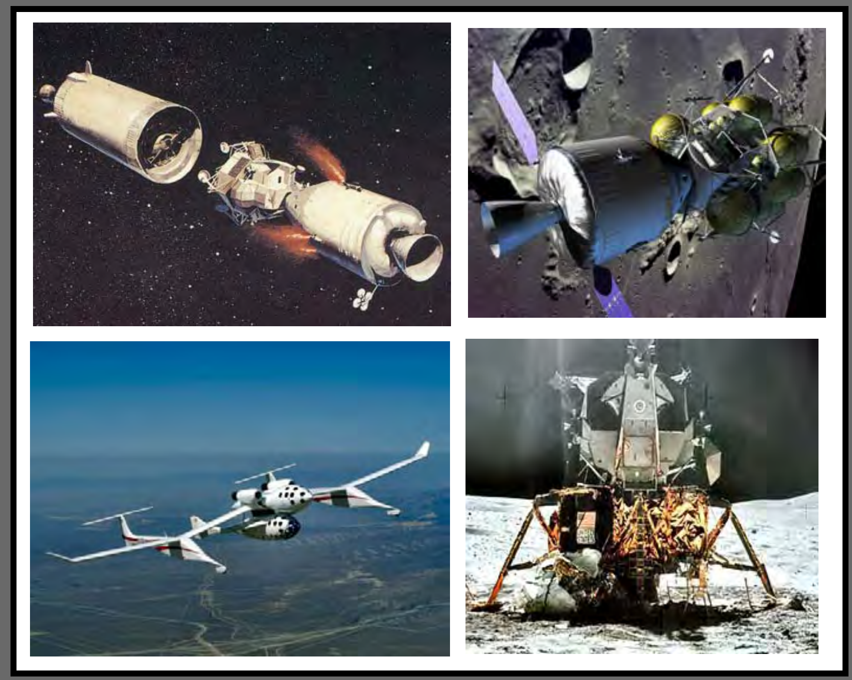
Go to the list of contents

SICSA SPACE ARCHITECTURE
SEMINAR LECTURE SERIES

PART III : SPACE TRANSPORTATION, PROPULSION AND PATHWAY OPTIONS



www.sicsa.uh.edu

LARRY BELL, SASAKAWA INTERNATIONAL CENTER FOR SPACE ARCHITECTURE (SICSA)
GERALD D.HINES COLLEGE OF ARCHITECTURE, UNIVERSITY OF HOUSTON, HOUSTON, TX

The Sasakawa International Center for Space Architecture (SICSA), an organization attached to the University of Houston's Gerald D. Hines College of Architecture, offers advanced courses that address a broad range of space systems research and design topics. In 2003 SICSA and the college initiated Earth's first MS-Space Architecture degree program, an interdisciplinary 30 credit hour curriculum that is open to participants from many fields. Some students attend part-time while holding professional employment positions at NASA, affiliated aerospace corporations and other companies, while others complete their coursework more rapidly on a full-time basis.

SICSA routinely presents its publications, research and design results and other information materials on its website (www.sicsa.uh.edu). This is done as a free service to other interested institutions and individuals throughout the world who share our interests.

This report is offered in a PowerPoint format with the dedicated intent to be useful for academic, corporate and professional organizations who wish to present it in group forums. The document is the third in a series of seminar lectures that SICSA has prepared as information material for its own academic applications. We hope that these materials will also be valuable for others who share our goals to advance space exploration and development.

The SICSA Space Architecture Seminar Lecture Series is divided into two general Lecture Groups :

GROUP ONE:	GROUP TWO:
Part I : Space Structures and Support Systems	Part V : The History of Space Architecture
Part II : Human Adaptation and Safety in Space	Part VI : The Nature of Space Environments
Part III : Space Transportation, Propulsion and Pathways	Part VII : Environmental Planning and Systems
Part IV : Space Mission and Facility Architectures	Part VIII : Shelter Design and Construction

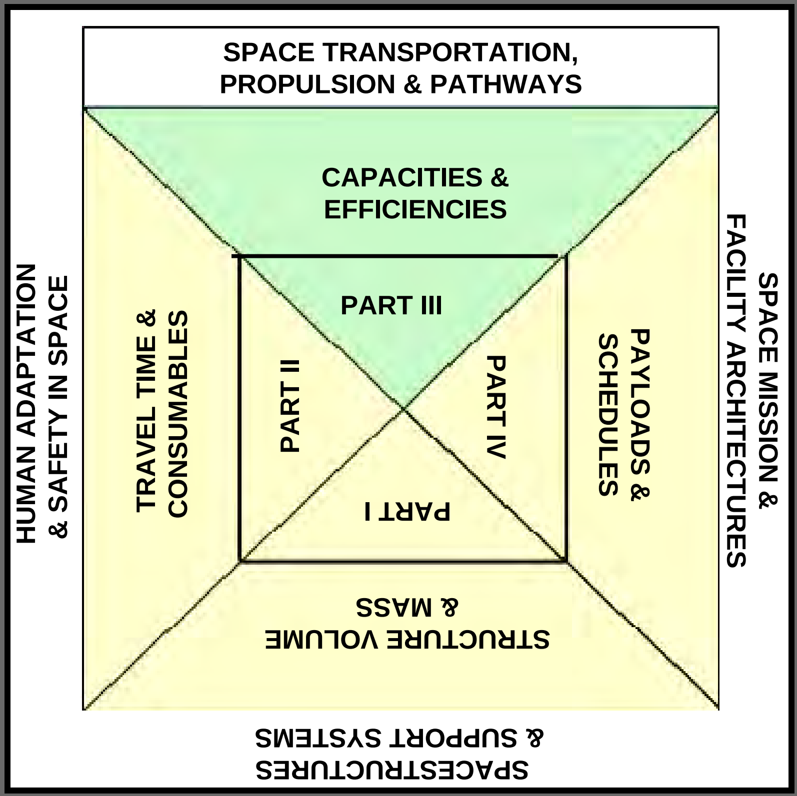
This lecture series provides comprehensive information, considerations and examples to support planning of human space missions and facilities:

- Part III (this report), discusses capacities and efficiencies of different transportation vehicles, propulsion and pathway options that enable and constrain planning decisions associated with the other three parts:

- Launch, transfer and landing vehicle selection and design must be correlated with structure.

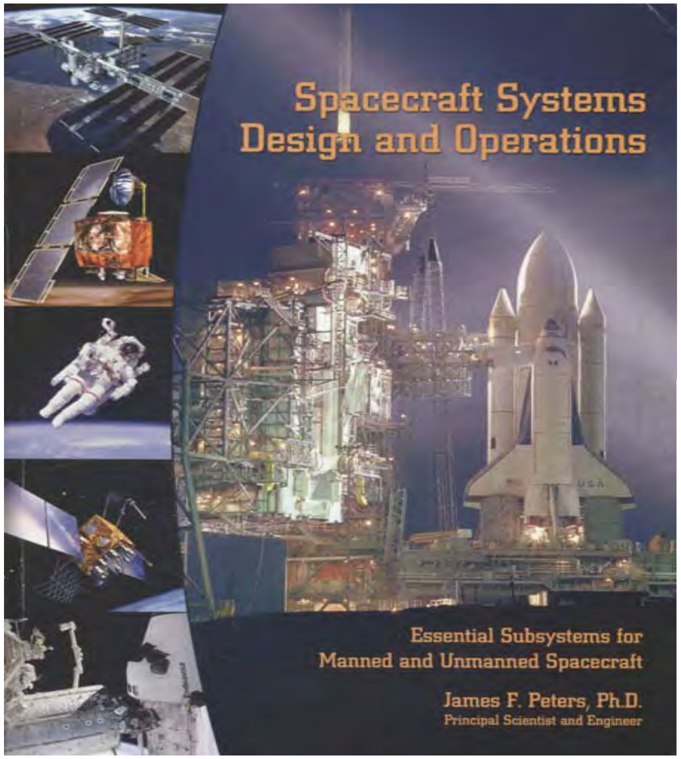
-Vehicle capacities, propulsion systems and destination pathways will directly influence crew accommodation/ consumable payloads, and radiation exposure risks (travel times/ schedules) associated with Human Adaptation and Safety in Space (Part II).

- Transportation, propulsion and pathway selection is a fundamental priority in Planning in Space Mission and Facility Architectures (Part IV).



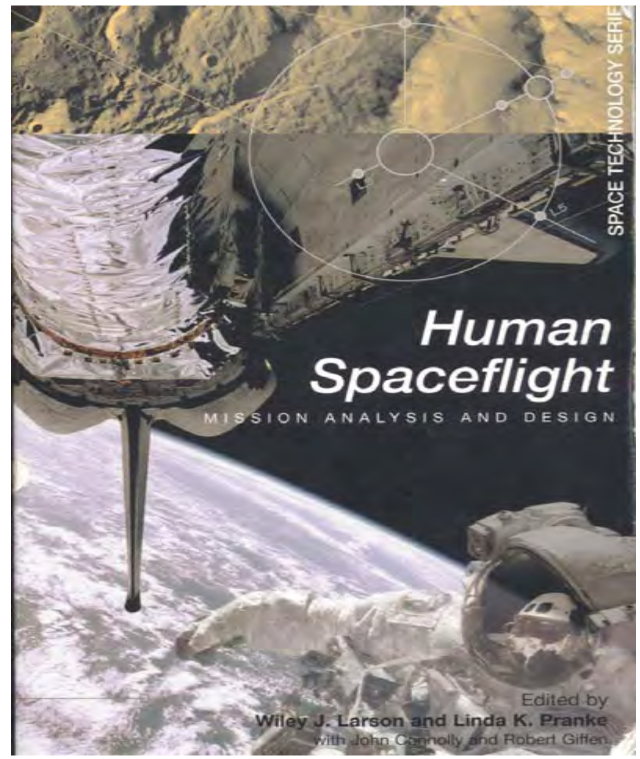
Key Relationships to Other Lectures

We are very grateful to Dr. James F. "Jim" Peters who has generously made a large body of material he has developed and collected available to us. This report draws extensively from his work. Much additional material can be obtained from his book, "Spacecraft Systems Design and Operations", which can be obtained from the Kendall/Hunt Publishing Company, 4050 Westmark Drive, Dubuque, Iowa 52202. This excellent publication is used as a primary text for the SICSA MS-Space Architecture curriculum, and is highly recommended as a valuable reference document for students and professionals at all career stages.



Key Reference Book

"Human Space Flight: Mission Analysis and Design" is a comprehensive and substantial book that should be in the library of any organization and individual involved in space project management, research, design or operations. The document was edited by Wiley J. Larson of the US Air Force Academy and Linda K. Pranke of LK Editorial Services as part of a Space Technology Series through a cooperative activity of NASA and the US Department of Justice. Text materials were contributed by 67 professional engineers, managers and educators from industry, government and academia. It is available through the Higher Education Division of McGraw-Hill.



Important Resource Book

It would be difficult or impossible to find anyone more knowledgeable about the subject of his book, “Space Stations and Platforms”, than Gordon Woodcock from Boeing. “Gordy” has enormously broad experience and expertise, and we are all fortunate he has made the effort to share it. As noted by Edward Gibson in the book’s forward, “Over the coming years, this work should become a classic space station reference. It has high value for those who desire to understand, appreciate or contribute to our first permanent settlement in New Earth”. It can be obtained through the publisher: Orbit Book Company, Inc., 2005 Township Road, Malabar, Florida 32950.

Important Resource Book

The Cambridge Encyclopedia of Space edited by Fernand Verger, Isabelle Sourbes-Verger and Raymond Ghirari has been a key reference source for Section I of this report, and is highly recommended as an important publication for information about transportation orbits vehicles, and a broad range of other topics. The scope is comprehensive, the writing is clear, and the graphics are exceptional. This book is more than an encyclopedia of facts and data. It also offers the reader insightful commentaries about circumpherences surrounding successful and failed developments around the world that have brought us to where we are in space and can guide us to where we must go in the future.

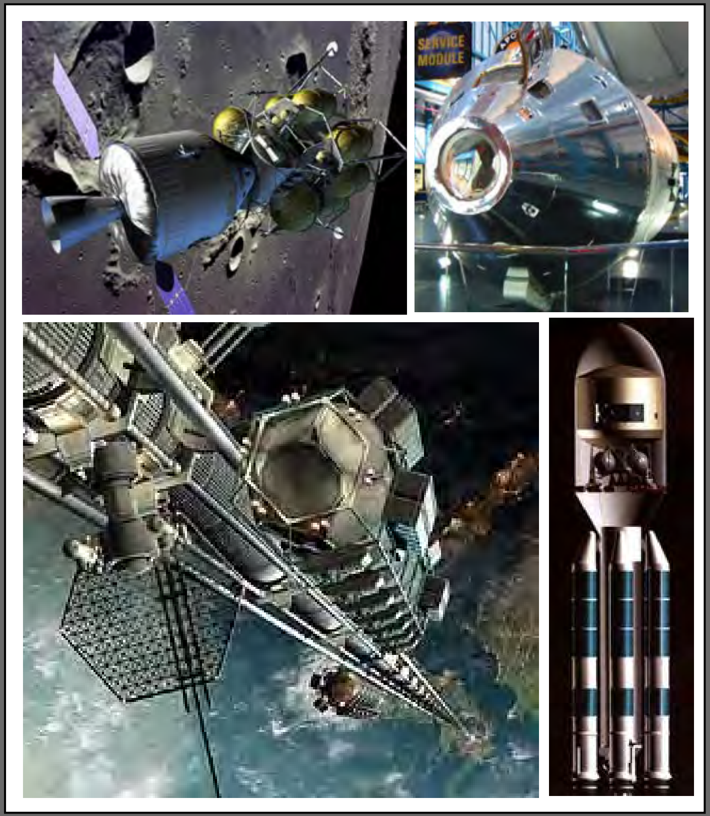
Important Resource Book

Section A: Transportation Systems

•Lunar/Mars Landers and Ascent Vehicles

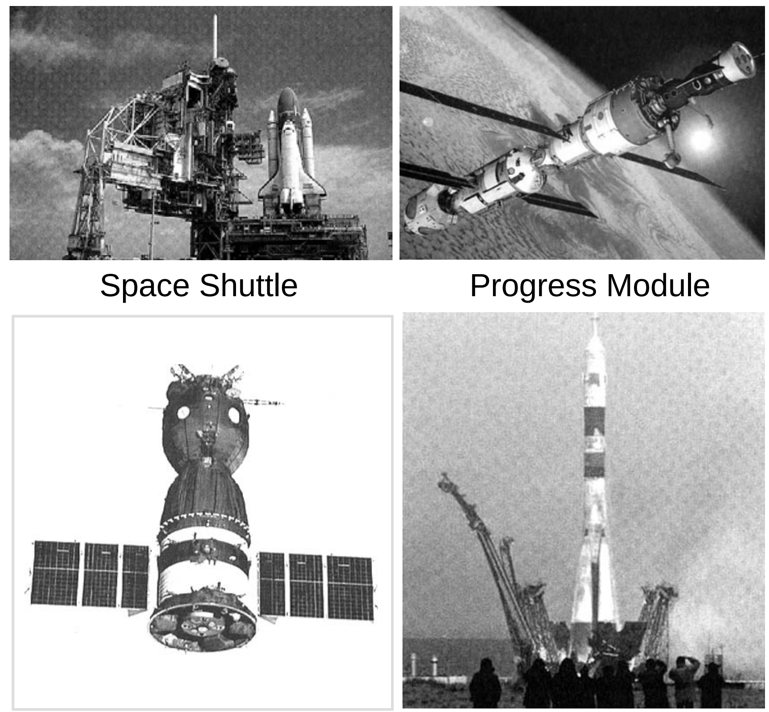
BACK TO THE LIST OF CONTENTS

SECTION A : TRANSPORATION SYSTEMS



Current and future space transportation systems potentially include a variety of types and functions:

- Earth launch and/or return vehicles:
  - -Reusable spacecraft (e.g. Shuttle)
  - -Expendable launch vehicles (ELVs)
- Orbital intercept and maneuvering vehicles:
  - -Earth-Mars trajectory sprint vehicles
  - -Orbital maneuvering vehicles
- Lunar/planetary trajectory vehicles:
  - -Pressurized spacecraft and transfer modules
  - -Unpressurized cargo systems
- Lunar/planetary descent/ascent vehicles:
  - -Surface delivery landers
  - -Surface delivery and orbital return systems
- Crew Earth reentry spacecraft/capsules
  - -Winged spacecraft
  - -Ballistic capsules



Space Shuttle — Progress Module — Soyuz Spacecraft — Soyuz Launch Vehicle

Vehicle Examples

NASA

Affordable space access and return comprises the greatest challenge for science, exploration and commercialization.

- Current costs of about $10,000/payload pound to LEO far exceed budgets of most private organizations.
- Getting people and scientific or commercial equipment to orbit is only half of the problem, since participants and products must also be returned.
- Space tourism can only occur if travel costs are dramatically reduced, and safety is greatly increased.
- Production of many proven space vehicles has been or will be discontinued and will be costly to reestablish.
- Advanced launchers and engine technologies are vitally needed to solve current problems.
- New programs in the U.S, Europe and Asia offer hope that this can be accomplished.

Making Space Accessible

The Soyuz ("union") Vehicle appeared in 1966 and has been the world's most frequently flown launcher (40-45 launches/year during the 1970s and 1980s):

- It is based upon the Russian R7 Sapwood (or "Semyorka" ICBM design, a one-stage rocket with 4 strap-ons that launched Sputnik into orbit on October 4, 1957.
- The R7 is also the backbone of other Korolev launchers still in use (the Vostok and Molniya).

Soyuz comprises a Vostok with a second stage:

- While the Russians class this as a 2-stage assembly, the west describes it as 1½ stages (the strap-ons and core engines fire simultaneously).
- The Soyuz sister vehicle Molniya ("lightning") launched the USSR's Luna 4-14, Venera 1-8, 3 Mars probes, and is still in production.
- Soyuz-U is used to launch progress supply ship to the ISS and Voskhod manned vehicles.

Payload Capacity :

- Mass to LEO (51.8^{o}) : 5MT
- Fairing Diameter : 2.35m
- Fairing Length : 9m

Propulsion :

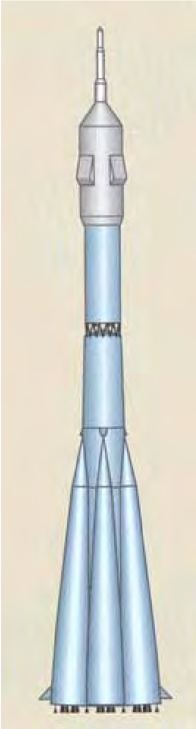
- LOX / Kerosene

Launch Site :

- Baikonur, Kazakhstan (45^{o}54'N / 63^{o}18'W)

Production :

- Developer : Central Specialized Design Bureau / Energomash
- Status : Supports ISS ops.



Russian Soyuz - U

Conventional Launch Vehicles

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE

NASDA's H-2 series went into service in 1994:

- It was the first Japanese rocket that wasn't based upon US technology.
- The system used 2 cryogenic stages built by Mitsubishi, and 2 large SRBs produced by Nissan.

After an H-2 launch failure in 1999, new efforts shifted to develop the H-2A

- Two models are being planned, with 2,4 or 6 boosters, and 2 and 3-stage versions.
- There will also be a 1-stage variant of the 2A design to launch an unmanned mini-shuttle ("HOPE").
- The H-2A is expected to be able to lift 8MT into GTO.
- The H-2 has been the most expensive launcher on the market, and NASDA's goal is to reduce pricing by 55%.

Payload Capacity :

- Mass to LEO (30^o) : 10MT
- Fairing Diameter : 4.6m
- Fairing Length : 9.2m

Propulsion :

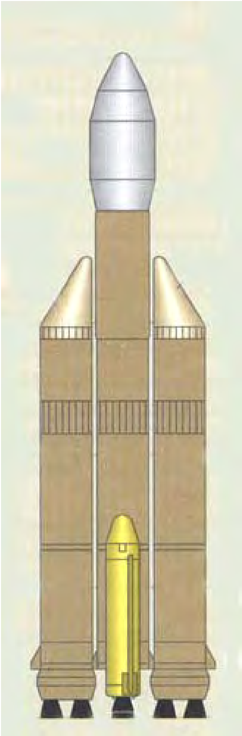
- Solid Propellants

Launch Site :

- Kagoshima, Japan (31^o14'N / 131^o05'W)

Production :

- Developer : NASDA
- Status : Under development



Japanese H - 2A

Conventional Launch Vehicles

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE

Two Long March CZ (Chang Zheng) launch family initially derived from 1950s Soviet ICBM technology:

- The CZ3 was the first to carry GEO satellites.
- Using a LOX/H_2 third stage, China became the third space power to use this technology.
- Solid boosters can be stretched to the length of the 1st stage, potentially enabling 11.8 MT to be put in LEO, and 7 MT into GTO.
- A faring similar to the Russian Soyuz design has been developed for future manned flights.
- China has announced interests in developing a Long March X heavy-lift launcher capable of placing 23 MT in LEO.
- They are also considering future reusable vehicles.

Payload Capacity :

- Mass to LEO (57^o) : 11.8MT
- Fairing Diameter : 3.8m
- Fairing Length : 10m

Propulsion :

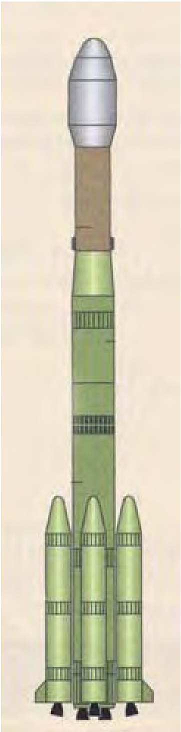
- Nitrogen tetroxide – Hydrazine
- LOX / LH2 3rd stage

Launch Site :

- Xichang, China
  (28^o06'N / 102^o18'W)

Production :

- Developer : Great Wall Company
- Status : Operational



Chinese Long March CZ3 – B/C

Conventional Launch Vehicles

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE

The Ariane 5 ESC-B is aimed at capturing new commercial launch markets:

- Capabilities have been enhanced using a new Vinci engine which adds power to the earlier series and is capable of multiple restarts in flight.
- The enhanced version provides a 15MT capacity to geostationary transfer orbit (GTO).
- CNES has proposed upgrades that may provide a 15MT capacity to GTO and / or reduce cost of 12MT Ariane ESC-B launches by 30%.
- CNES has established a Future Launcher Technology Program (FLTP) aimed at demonstrating reusable launch capabilities.

Payload Capacity :

- Mass to LEO (65^o) : 18MT
- Fairing Diameter : 4.75m
- Fairing Length : 10.35m

Propulsion :

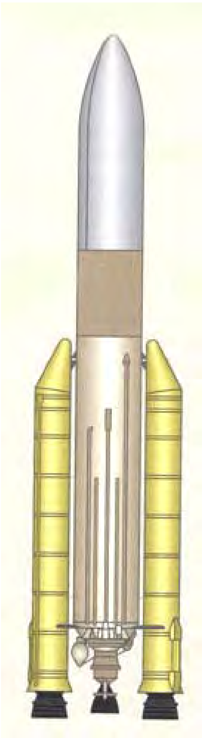
- LOX/LH_2
- Solid boosters

Launch Site :

- Lourou, Giana (5^o32'N / 52^o46'W)

Production :

- Developer : ESA / CNES
- Status : Operational



European Ariane 5 ESC-B

The first Proton heavy-lift vehicle appeared in 1965 as a 2-stage launcher, and as a 4-stage version in 1967:

- It was developed as a commercial GEO vehicle, and is currently the only CIS launcher capable of placing satellites in GEO.
- A Russian-U.S. joint venture (Lockheed-Khrunichev-Energia) was created in 1992 to commercialize services.

Proton's key advantage is modular flexibility:

- There are three variants (2, 3 and 4 stages).
- The 4^{th} stage (Block-DM) used for high orbits and probes can use either LOX/Kerosene or Nitrogen tetroxide – Hydrazine.
- Protons have launched the Zond and Luna lunar probes, Venera to Venus, Vega to Halley's Comet, Mars and Phobos to Mars.
- A Proton mission was planned to carry cosmonauts around the Moon and back in December 1968 (prior to Apollo 8) but scrubbed when an unmanned Zond 6 suffered a depressurization the month before.

Payload Capacity :

- Mass to LEO (51.6^{o}) : 20MT
- Fairing Diameter : 4.1m
- Fairing Length : 14.6m

Propulsion :

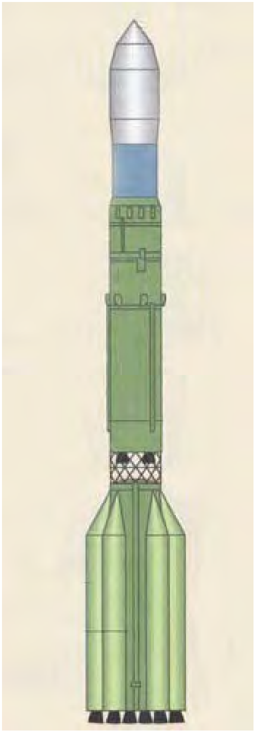
- Nitrogen tetroxide - Hydrazine
- LOX / Kerosene

Launch Site :

- Baikonur, Kazakhstan (45^{o}54'N / 63^{o}18'W)

Production :

- Developer : RSC Energia
- Status : Operational to GEO



Russian Proton D-1

Conventional Launch Vehicles

The Titan IV is used exclusively for military purposes due to high costs which are poorly suited for commercial uses:

- It began with the adaptation of Titan II ICBMs used for Gemini missions (1964-1966).
- Deconditioned Titian II missiles have been reworked since 1988.
- Titan IV was first launched in 1989 and derived from Titan 34 D with extended boosters and 1st and 2nd stages.
- The central core can utilize a cluster of 2-8 Castor 4A solid boosters (Titan 2L).

Payload Capacity :

- Mass to LEO (28.6°) : 21.9MT
- Fairing Diameter : 4.57m
- Fairing Length : 12.2m

Propulsion :

- Nitrogen tetroxide - Hydrazine
- Solid boosters

Launch Site :

- Vandenberg AFB, California (34°36'N / 120°36'W)

Production :

- Developer : Lockheed Martin
- Status : Operational (military)

US Titan IV

Conventional Launch Vehicles

The Energia ("energy") is 5 times more powerful than any other CIS booster, and can generate nearly 8 million pounds of thrust at lift-off:

- It was designed in the 1970s to carry payloads of 65-200MT to LEO.
- The modular design can accommodate a variety of payloads, including the Buran ("snowflake") orbiter which is similar to the U.S. Shuttle.
- It is comprised of a central cryogenic core flanked by 2-8 LOX/ Kerosene boosters (all fire simultaneously at lift-off).
- Buran was launched on November 15, 1988, returned to Earth after 2 orbits, and landed automatically.

The system was "mothballed" due to financial problems in post-Soviet Russia:

- Commercial payloads haven't required such large capacities.
- A technical team has remained in service and rockets already built have been retained.

Payload Capacity :

- Mass to LEO (57^o) : 65MT
- Mass with 8 boosters : 200MT
- Fairing Diameter : 5.5m
- Fairing Length : 37m

Propulsion :

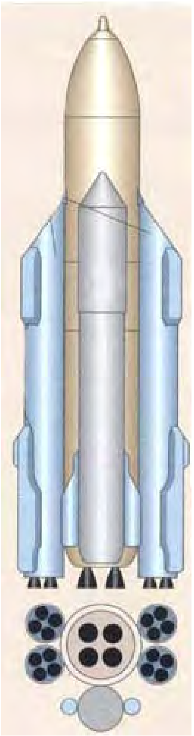
- LOX/ / LH_2
- LOX / Kerosene boosters

Launch Site :

- Baikonur, Kazakhstan (45^o54'N / 63^o18'W)

Production :

- Developer : RSC Energia
- Status : Mothballed



Russian Energia

Heavy Lift Vehicle (HLV)

In response to the new EELV program, Boeing is developing a new basic rocket that uses a modified McDonnell Douglas Delta 3 cryogenic stage with 2-4 solid fuel strap-ons to diversify capabilities:

- The Delta 3 stage is a link between the Delta 2 and EELVs that use a wider Delta 1st stage (shortened at the top with longer boosters).
- The cryogenic LOX/ LH2 2nd stage will be similar to the one used on the ATLAS-5 EELV.

Payload Capacity :

- Mass to LEO (27^{o}) : 15.6MT
- Fairing Diameter : 3.7m
- Fairing Length : 12m

Propulsion :

- LOX/ / LH_2
- Solid boosters

Launch Site :

- Vandenberg AFB, California (34^{o}36'N / 120^{o}36'W)

Production :

- Developer : Boeing / McDonnell Douglas
- Status : Under development

US Delta 4-Heavu

Evolved Expendable Launch Vehicles

Lockheed Martin and Boeing were jointly selected by the US Pentagon in 1999 to develop launchers for a new Evolved Expendable Vehicle Program (EELV).

- Lockheed Martin is pursuing gradual transformation of its Atlas series line.
- The Atlas-5 EELV will be a Heavy-lift Vehicle (HLV) incorporating RD-180 variable-throttling engines developed by the Russian company Energomash and manufactured by Pratt & Whitney as a Common Core Booster (CCB).
- The Atlas-5 series will incorporate 1-5 strap-ons, the CCB, and a Common Element Centaur as the 2nd stage.

Payload Capacity :

- Mass to LEO (28.5^o) : 19MT
- Fairing Diameter : 4.57m
- Fairing Length : 10.35m

Propulsion :

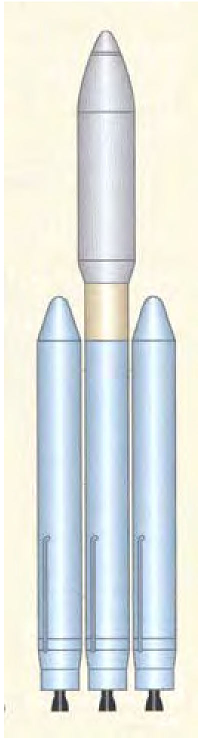
- LOX/ / Kerosene
- LOX / LH_2

Launch Site :

- Vandenberg AFB, California (34^o36'N / 120^o36'W)

Production :

- Developer : Lockheed Martin
- Status : Under development



US Atlas-5 EELV-HLV

Evolved Expendable Launch Vehicles

The Space Shuttle consists of 2 solid propellant rocket boosters (SRBs) which are recovered and reused, an expendable cryogenic LOX/ LH2 External Tank (ET), and an Orbiter crew cabin:

- Six different Orbiters have been built:
- Enterprise has served only for non-orbital tests, Challenger and Columbia have been lost, leaving Discovery, Atlantis and Endeavor (which replaced Challenger).
- The Shuttle can transport 2-8 astronauts to LEO for missions lasting about one week.
- It has a 4.5m x 18 m cargo bay which can carry pressurized labs as well as a large variety of equipment and supplies.
- NASA has assigned Shuttle operations to the United Space Alliance, a private company, in the interest of reducing costs.

Payload Capacity :

- Mass to LEO (28.45^o) : 25MT
- Fairing Diameter : 4.5m
- Fairing Length : 18m

Propulsion :

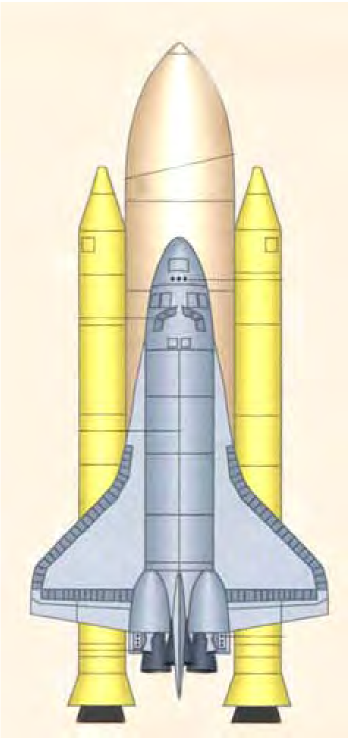
- LOX / LH_2
- Solid boosters

Launch Site :

- Cape Canaveral, Florida (28^o30'N / 80^o33'W)

Production :

- Developer : Rockwell
- Status : To be retired

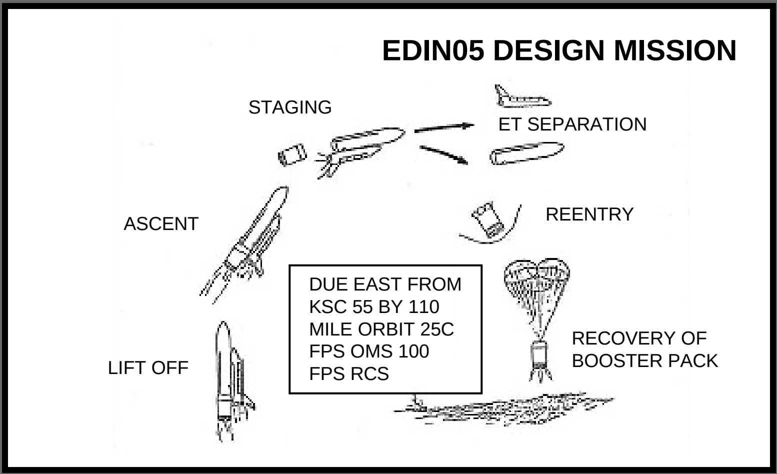


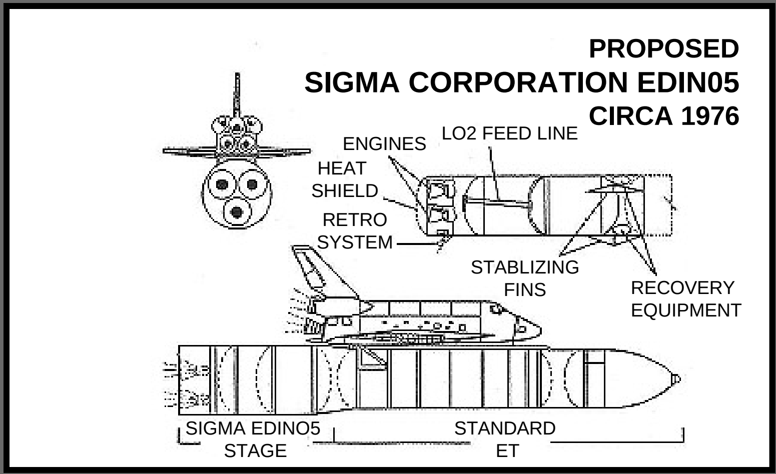
US Shuttle STS

Reusable Launch Vehicles

Many studies have investigated possible ways to improve the Shuttle's payload capacity and operating efficiencies beginning soon after the development program first began. Most have centered upon enlarging the cargo bay and replacing the SRBs with longer ones or with recoverable Liquid Rocket Boosters (LRBs).

The Sigma Corporation presented a "EDIN05" concept in 1976 which proposed enlarging the external tank and replacing the SRBs with liquid booster packs and engines from the Saturn first stage. A major draw back was a need to redesign the Cape Canaveral launch facility to accommodate the variant vehicle interface.





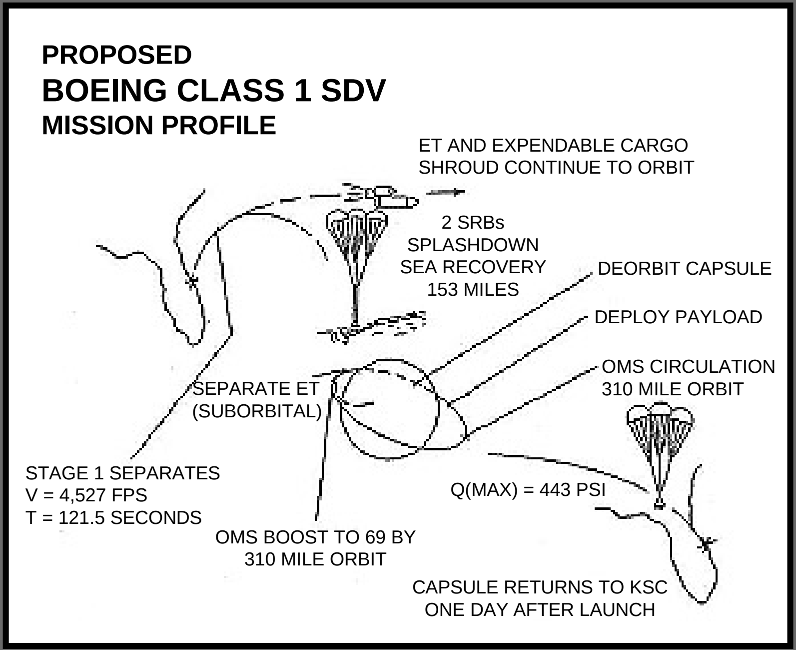
Proposed Shuttle Upgrades

NASA and many aerospace companies are considering design options for a Crew Exploration Vehicle (CEV) which will replace the Shuttle and support future Moon/ Mars missions:

- “Super ELVs”:
  - Some companies are considering “clean sheet” designs for new expendable launch vehicles capable of delivering 75 MT or more to LEO.
  - Some concepts envision development of derivative EELVs which would upgrade launchers similar to the proposed Boeing/ McDonnell Douglas Deta-4 Heavy and the Lockheed Martin Atlas-5 Heavy to deliver LEO payloads of 45 MT or more.
  - Others are advocates for various versions of Shuttle Derived Vehicles (SDVs) which would use the existing External Tanks and/ or SRBs minus the Orbiter.

Boeing proposed SDV concepts that would supplement Shuttle flight operations, using various expendable cargo shrouds and recoverable Space Shuttle Main Engines (SSMEs).

NASA



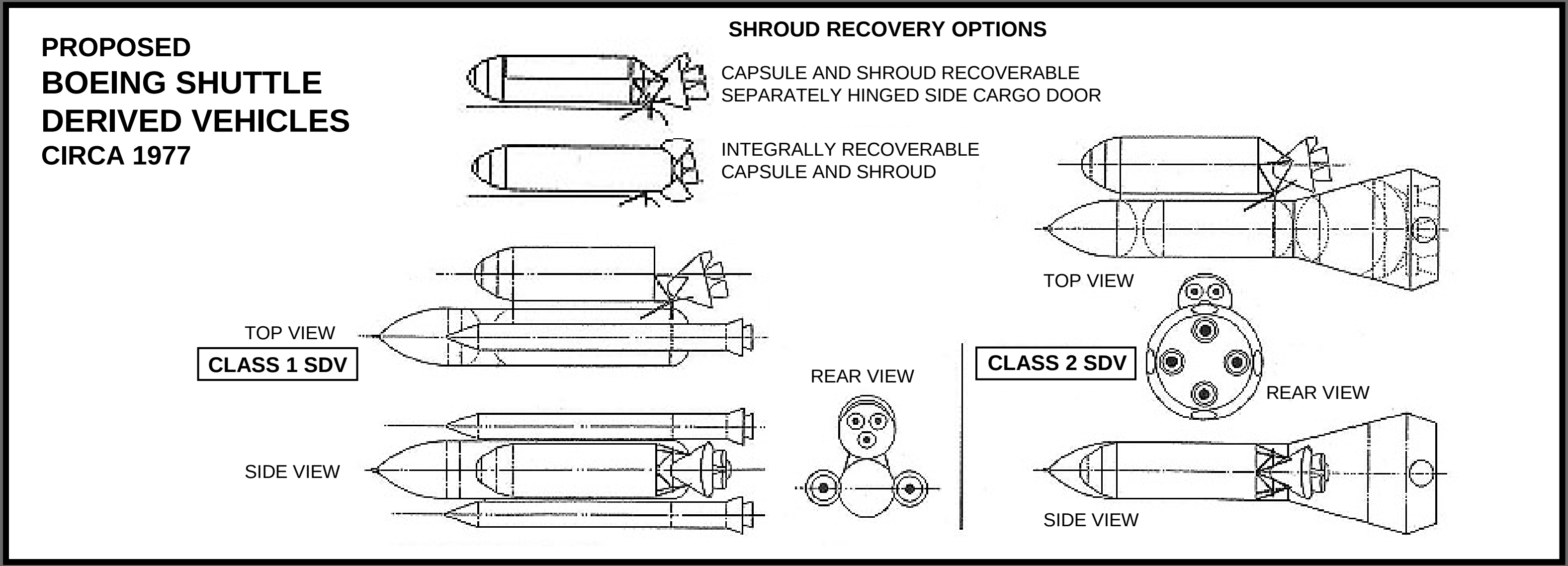
Shuttle Derived Vehicles (SDVs)

The Class I SDV concept attached SRBs to opposite sides of the cargo shroud and the ET at the top:

- A capsule containing the SSMEs could be recovered separately or as integral part protected within the cargo shroud.

The Class II SDV concept eliminated the SRBs and modified the ET to incorporate a bell-shaped first stage housing propellant tanks and 4 engines similar to Sigma's EDIN05:

- Like Sigma's proposal, new Mobile Launch Platforms (MLPs) would need to be created for this approach.



Boeing SDV Concepts

Boeing's SDV proposals provided alternative cargo shroud/ engine combinations:

- The Class I SDV had 2 general options:
  - A long cargo shroud (about 40 m long x 7.5 m diameter) with no payload doors, and a capsule containing clamshell doors that protected 3 engines during reentry.
  - A shorter shroud (about 22 m long x 7.5 m diameter) with 3-4 SSMEs that folded into a payload bay for protected recovery.
- The Class II SDV used cargo shroud and propulsion modules that were similar to but larger than those used for Class I:
  - The shrouds were slightly longer, but substantially greater in diameter (about 9 m).

The proposals envisioned greatly increased LEO delivery capabilities:

- Mass to orbit:
  - Boeing projected that the 3-engine derivative of either concept could place between 60-100 MT payloads into 310 mile 28.5° orbits.
  - Larger 4-engine Class I versions could boost over 110 MT to the same orbits and Class II vehicles might lift up to 150 MT.
  - Boeing studies projected combined launch rates of 45 Orbiter and 22 SDV flights/ year.
  - A major launch limitation was that non-man-rated SDVs on Shuttle pads would interfere with capabilities to launch crew rescue missions within 24 hr periods mandated at the time.

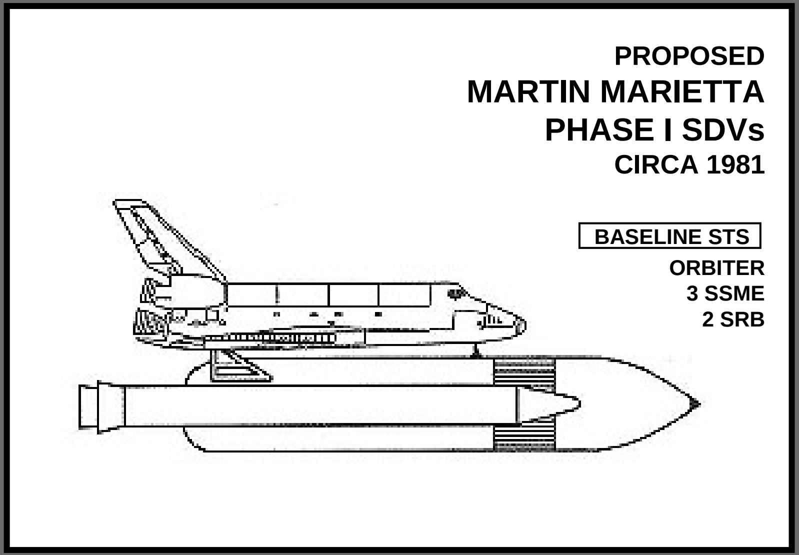
Boeing SDV Concepts

The Michoud Division of Martin Marietta (responsible for design/ production of Shuttle ETS) undertook studies in the 1980s which considered 4 SDV options:

- A Class I option simply replaced the Orbiter with a Cargo Element to greatly increase payload-to-orbit capacity by substantially reducing weight.
- Class II configurations replaced SRBs with 2 LRBs, each using 4 SSMEs capable of carrying 50 MT to LEO (twice the Shuttle capacity).
- Class III combined improvements of Class I and Class II, affording payload capacities up to 120 MT and enabling the US to recover HLV capacities abandoned after Saturn V.
- Class IV removed SSMEs and associated avionics from the Orbiter and installed them on a recoverable ET, freeing up some additional room on the Orbiter but offering no real performance benefits.

Martin Marietta SDV studies baselined most primary Shuttle elements as references, but proposed some replacements and upgrades including:

- An optional Cargo Element to replace the Orbiter.
- Replacement of solid rocket boosters with liquid boosters.
- A reusable Propulsion/ Avionics package attached to the ET to convert the Orbiter to a glider.



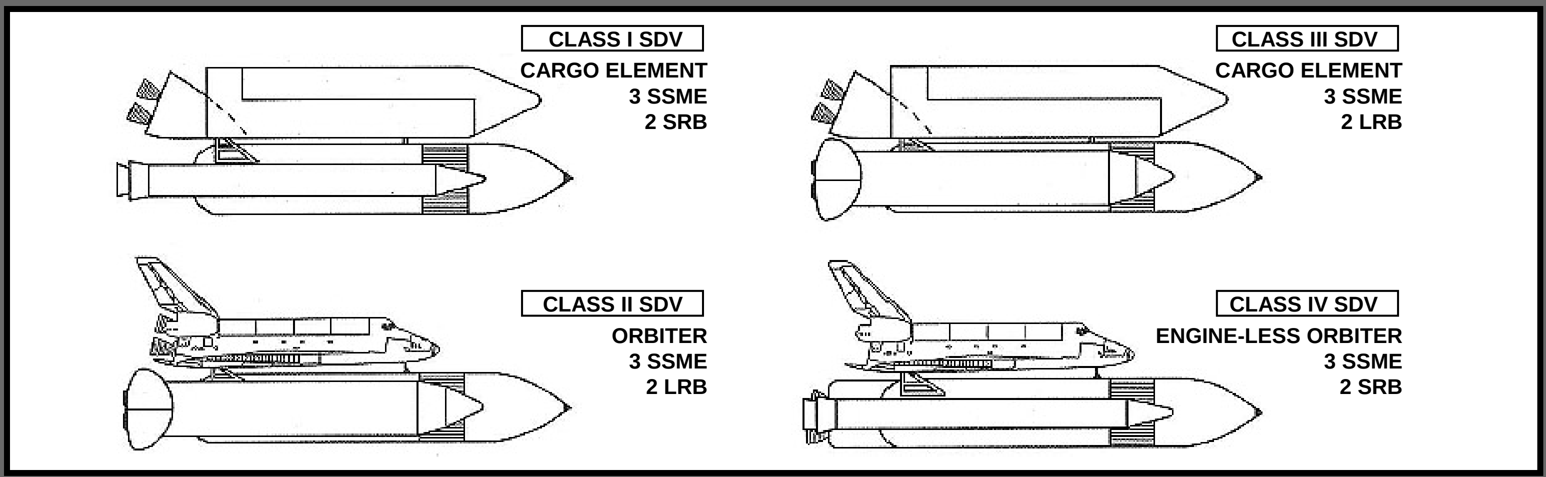
Martin Marietta SDV Concepts

Class I: Orbiter is replaced with a Cargo Element (CE) and a reusable Propulsion/ Avionics (PA)

Class II: The Orbiter is retained and the SRBs are replaced with LRBs.

Class III: The Orbiter is replaced with a CE and SRBs are replaced with LRBs.

Class IV: The Orbiter becomes a glider (propulsion system/ avionics are removed and installed under the ET).



Martin Marietta SDV Concepts

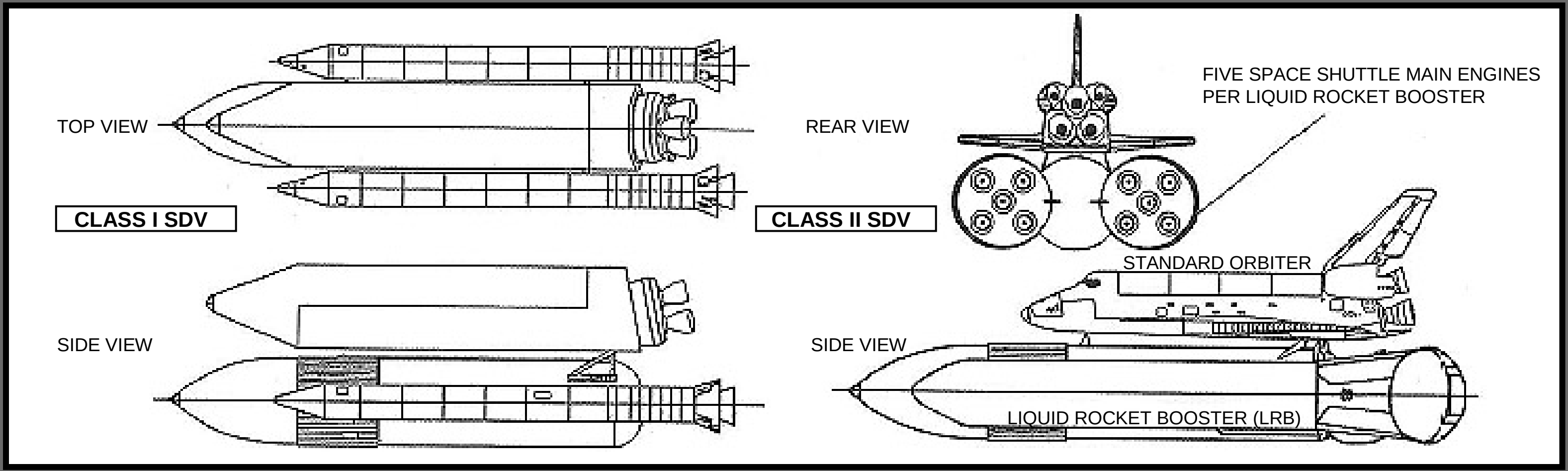
Based upon option comparisons, Martin Marietta decided to concentrate additional studies on Class I and Class II:

Class I A proposal:

- The separate PA package was eliminated and engines/ avionics were installed directly on a Payload Module (formerly the CE) with Quick-Disconnect (QD) fittings enabling recovery by a co-orbiting Orbiter rendezvous for return to Earth.

Class II A proposal:

- Alternate LRB designs were proposed with a set of air-breathing engines enabling powered flight back to a recovery area near the launch site following atmosphere reentry using small deployable wings for gliding or conventional parachutes.



Martin Marietta SDV Concepts

Although the Boeing and Martin studies indicated benefits of some SDV variation, NASA was faced with difficult decisions following the 1986 Challenger accident:

- Arguments against SDV development:
  - NASA wanted to replace Challenger, and it was unlikely that Congress would approve a new program.
  - Since SDV would use major Shuttle systems, future Shuttle failures might also ground SDVs.
- Arguments for SDV development:
  - Since SDVs would be unmanned, failures would be less catastrophic to the program.
  - SDVs could boost launch rates beyond limits imposed after the Challenger disaster.
  - SDVs would reduce dependence upon a single manned vehicle fleet.
  - SDVs might require less time, money and testing than a “clean sheet” approach.

NASA and the US Air Force began to jointly investigate common Heavy-Lift Launch Vehicle (HLLV) manifest applications, requirements, concept options and costs/ benefits:

- The USAF didn’t support SDVs:
  - They were skeptical that designs based on man-rated technologies would be cost-effective.
  - They believed that DoD manifest requirements could meet better by upgraded ELV approaches.
- NASA proceeded to study SDVs independently:
  - Investigation effort was assigned to the Marshall Space Flight Center (MSFC).
  - Contracts were awarded to Martin Marietta, United Technologies (USBI) and Rockwell International.
  - Phase I studies established overall requirements, vehicle configurations and operating concepts.
  - Phase II development studies referred to “Shuttle Cargo Elements” (SCE), later called “Shuttle-C”.

Shuttle-C SDV Background

Contractor design studies were targeted on launching 100,000-150,000 lbs to orbit and considered several options:

- A “throw-away” Orbiter fuselage structure (minus wings, crew module and thermal protection system) using 2 SSMEs, a limited data system, an ET and SRBs. (This was deemed too expensive.)
- Attachment of 2 SSMEs to the bottom of an ET and placement of a large payload compartment on top (an in-line design) to be launched with the aid of 2 SRBs. (This would require special launch facilities and significant structural ET modifications.)
- Mounting engines on the payload (as a true second stage) to configure the vehicle in a traditional manner like Saturn. (This would also require significant launch pad processing facility changes.)

A side-mounted configuration similar to the earlier Martin Class I concept was recommended by all contractors:

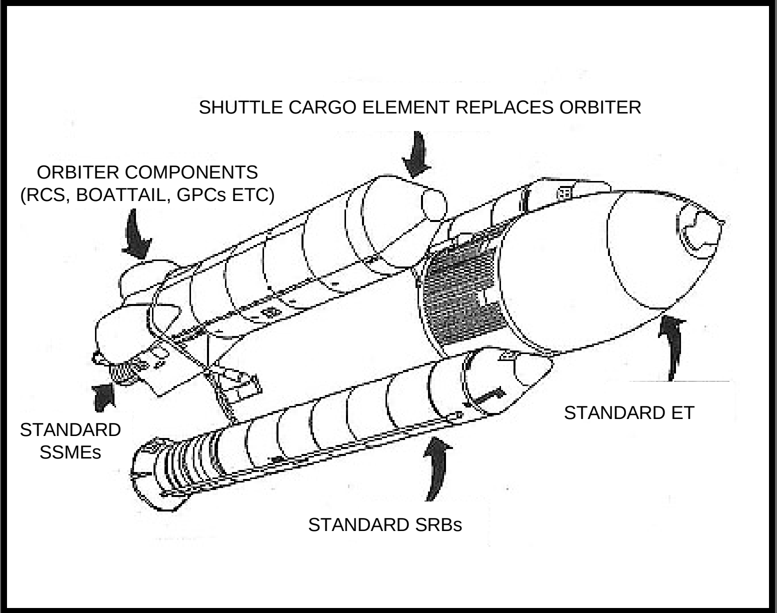
- Some studies suggested that Shuttle-C could orbit payloads for about $2,000/ lb (compared with $3,793/ lb for Delta II, and $4,100 for Titan IV).
- The reference payload was 15 ft. diameter x 72 ft. long, weighing just over 100,000 lbs (50 MT).
- The reference mission was a 253 mile orbit inclined at 28.5°.

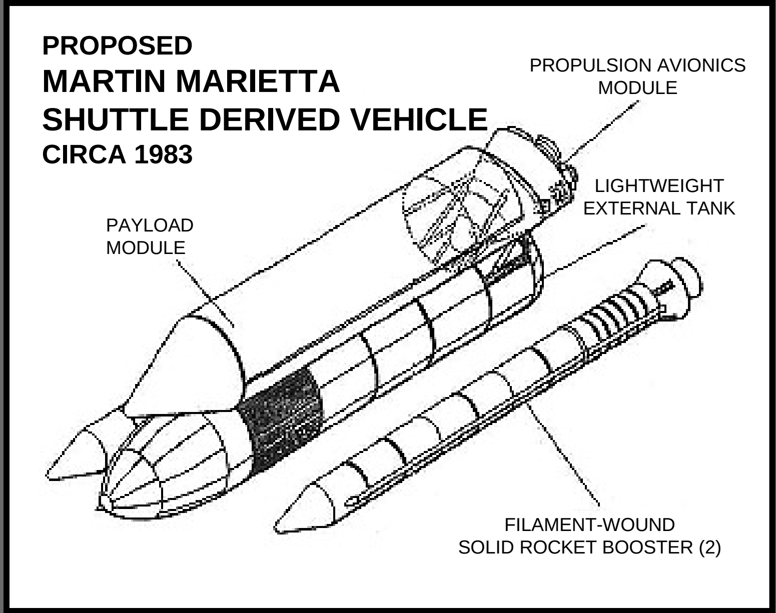
The configuration was comprised of:

- A newly designed Cargo Element (CE).
- A modified Orbiter aft fuselage (called “boattail”) with vertical stabilizer and body flap removed.
- Standard ET, SSMEs, OMS pods, Reusable Solid Rocket Motors (RSRMs) and avionics.

Shuttle-C SDV Studies

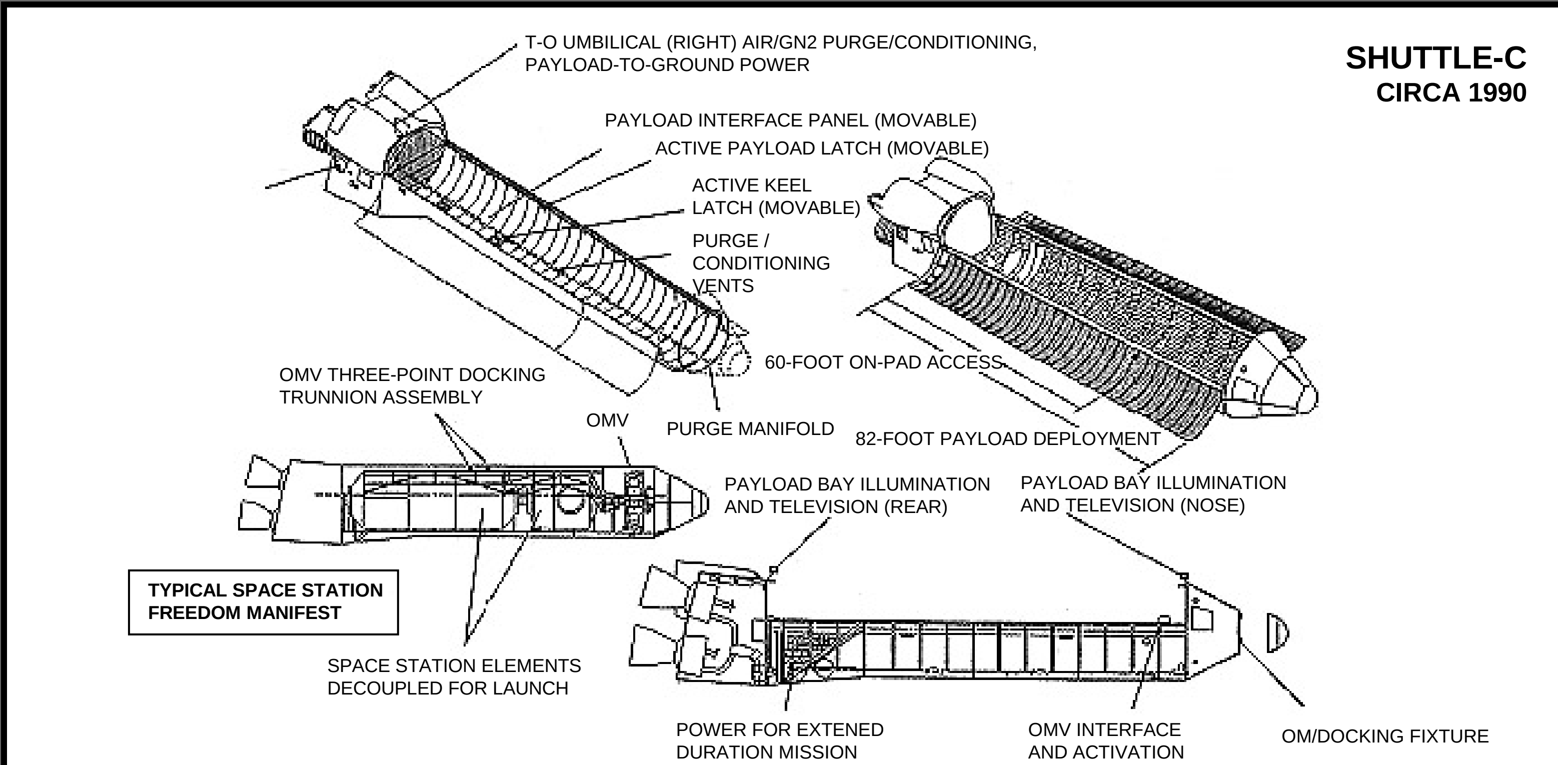
All MSFC Shuttle-C study contractors selected a configuration design very similar to Martin Marietta's side-mounted Class I concept proposal.





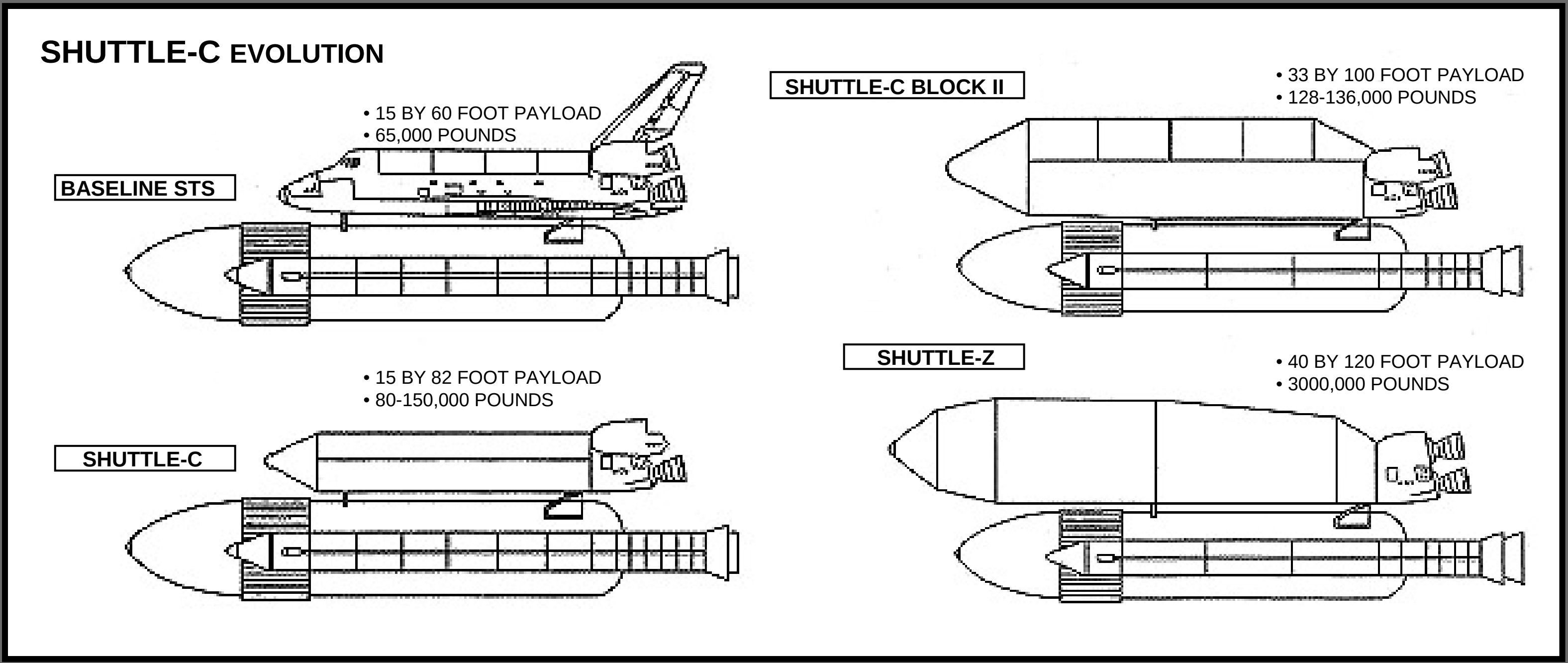
Selected Shuttle-C Configuration

NASA



Shuttle-C Cargo Element Concepts

NASA



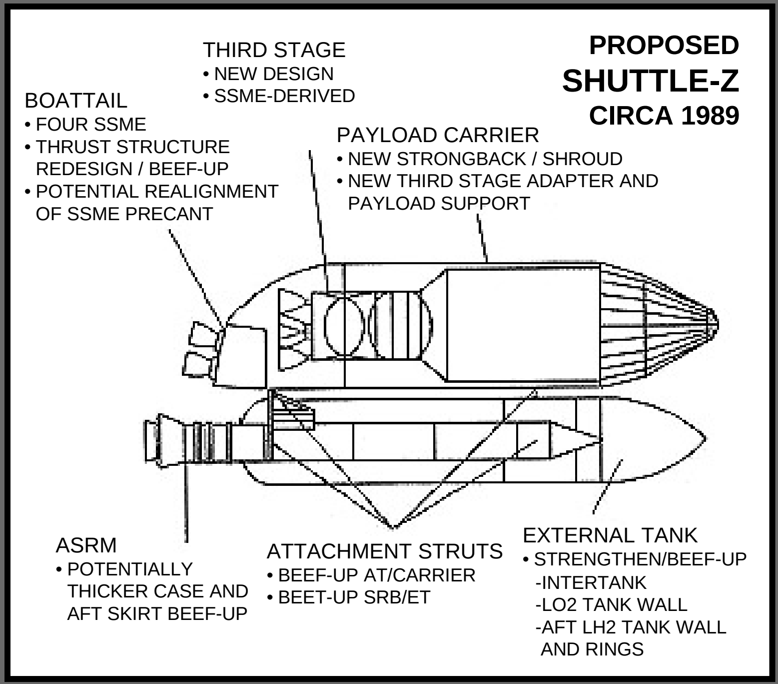
Shuttle-C Concepts

Increasingly tight NASA budgets following the Challenger accident, competition from Space Station Freedom (precursor to ISS) for development funds, and difficult FY 91 Congressional deliberations spelled the end of Shuttle-C chances:

- Much development work had already been accomplished:
  - An engineering development model had been assembled at MSFC by the Essex Corporation under contract to Boeing during 1989-90 for design/ integration of subsystems and payload compatibility studies.
  - The model was intended to be used to fit-check ground processing at the launch site.
  - A firm launch date had been set, aiming at a stream of launches planned for the mid-1990s.
  - Yet a 1990 decision by Congress cancelled the Shuttle-C, and put an end to the SDV concept

The demise of Shuttle-C also ended prospects for a proposed Shuttle-2 HLV concept, potentially capable of lifting 150 MT, 40 ft. diameter x 60 ft. long payloads. The design resembled a Shuttle-C Block II configuration "on steroids".

NASA



Shuttle-Z Concepts

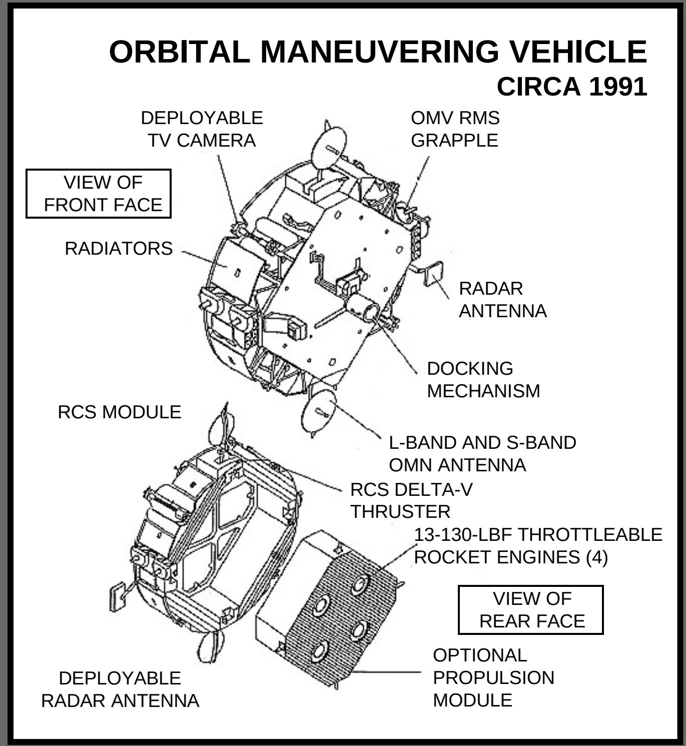
Heavy-lift capabilities are likely to be required to deliver large habitat modules, orbital transfer vehicles and other elements to high Earth orbits for planned human lunar/ Mars exploration.

- SICSA conceptualized an “ARES” Heavy-Lift launcher in 2002 in connection with a Mars mission planning project:
  - The proposal baselined a 100 MT capacity, for payload diameters up to 15 MT diameters.
  - The “Kepler Class” vehicle would use four 5-stage SRBs mounted around a stretched and structurally enhanced Shuttle ET.
  - Four MEs attached to the bottom of the ET would be sacrificed during each launch. (Alternatively, they might be attached to the payload shroud for use throughout the mission.)
  - The in-line payload alignment would keep the vehicle CG aligned with the MEs and symmetric SRBs to reduce atmospheric drag.
  - Since the vehicles would not initially be man-rated, crews would be launched by other means.

SICSA Heavy-Lift ELV Concept

Orbital flight operations for unmanned Shuttle-C applications were planned to be controlled from the ground or space station via attached Orbital Maneuvering Vehicle (OMV) which would be carried as a payload or launched separately.

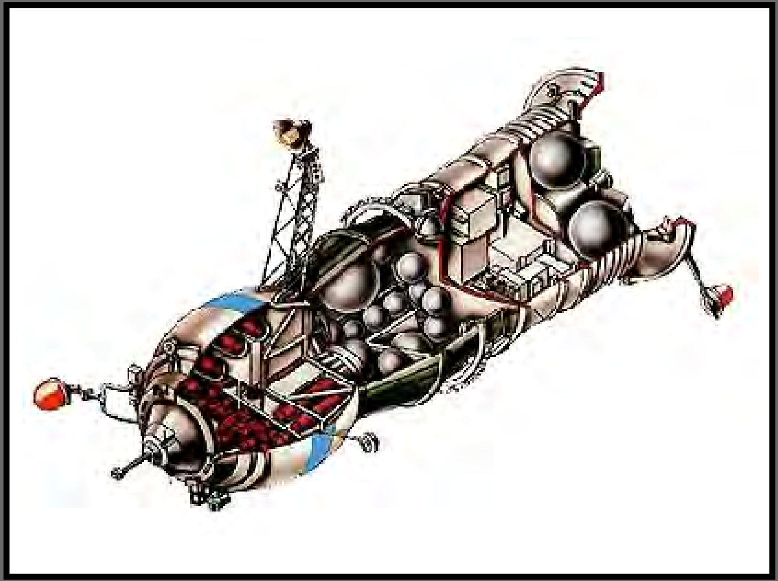
- An OMV development contract was awarded to TRW in 1986, but was cancelled as a casualty of Space Station Freedom budget problems:
  - The concept was first developed at MSFC in 1979 as the Teleoperator Retrieval System (TRS) as a revival of the original Space Tug proposed at the beginning of the Space Shuttle program.
  - The design was intended to enable Space Station Freedom to deploy and retrieve spacecraft in LEO.
  - The OMV free flyer would be carried to orbit by the Shuttle, operated from Freedom, and periodically refueled by the Orbiter or a SDV tanker.



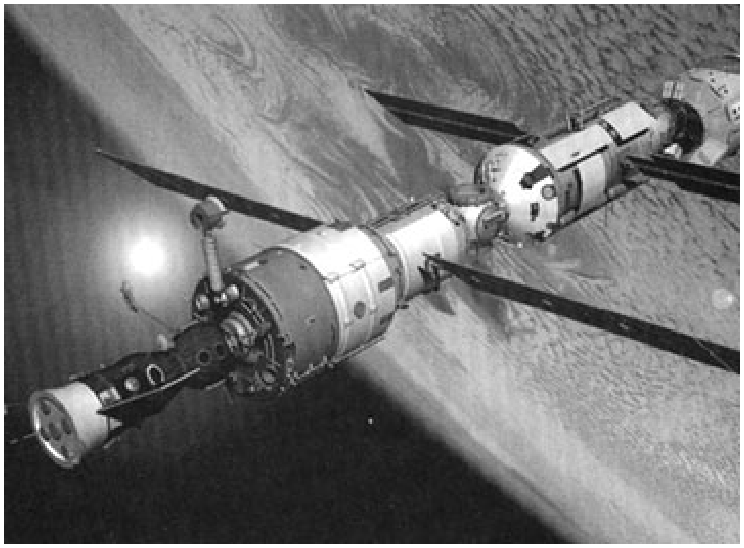
TRW's OMV Design

The Russian Progress Orbital Vehicle provides reboost for the ISS as well as logistics including water, gases and crew supplies. In addition, it acts as a "storage closet" for equipment and waste:

- The vehicle can deliver 1,920 kg of propellant, 1,800 kg of dry cargo, and 40 kg of air/ gas.
- It also has a 1,000 kg downmass capacity to return trash to Earth.
- Rendezvous, docking and departure is automated, with manual backup features.
- On-orbit stay time is limited to 180 days, and it is incinerated on reentry and replaced with another vehicle.



Progress Orbital Vehicle



Progress Orbital Vehicle

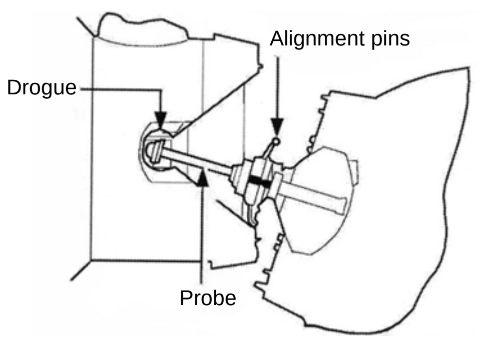
Small drag forces acting upon spacecraft that remain in LEO for long periods of time must be counteracted by some means to keep their altitudes from constantly decaying to lower levels. Skylab, the first US space station, was sacrificed because no reboost capabilities were provided.

Powered intercept vehicles that can rendezvous with and maneuver other spacecraft for reboost and other operational purposes offer one solution. Some of these vehicles, including the Russian Progress Orbital Vehicle and an Autonomous Transfer Vehicle that is being developed in Europe also provide logistics supply and waste return functions. Future automated, teleoperated and manually controlled maneuverable vehicles may also be used to transport crews and cargo elements in the vicinity of and between orbital facilities.

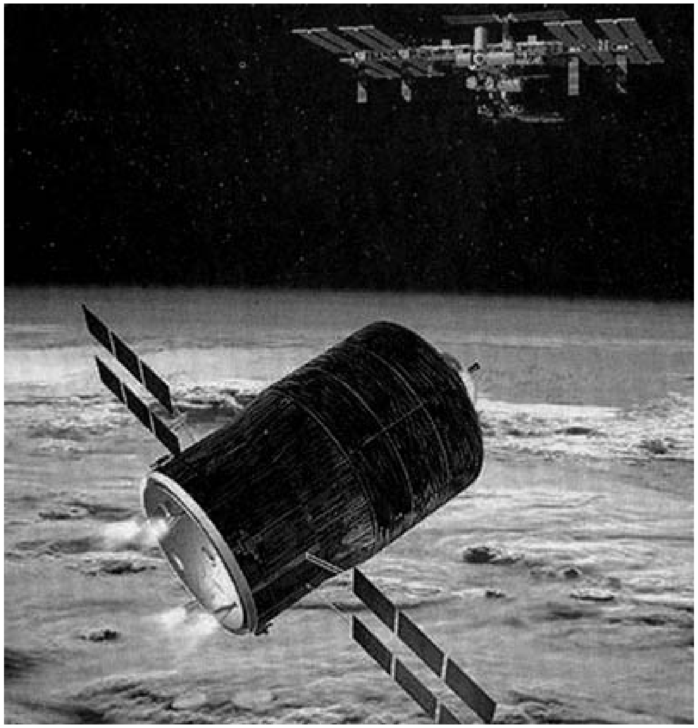
Maneuvering & Transport Services

An Autonomous Transfer Vehicle (ATV) is being developed by ESA to provide ISS reboost capabilities along with logistics support:

- The 8.5 m long x 4.25m diameter pressurized vehicle would be deployed about every 18 months on an Ariane launch vehicle.
- It would accomplish fully automated ISS rendezvous and docking, using the same Probe/ Drogue docking mechanism as the Progress.



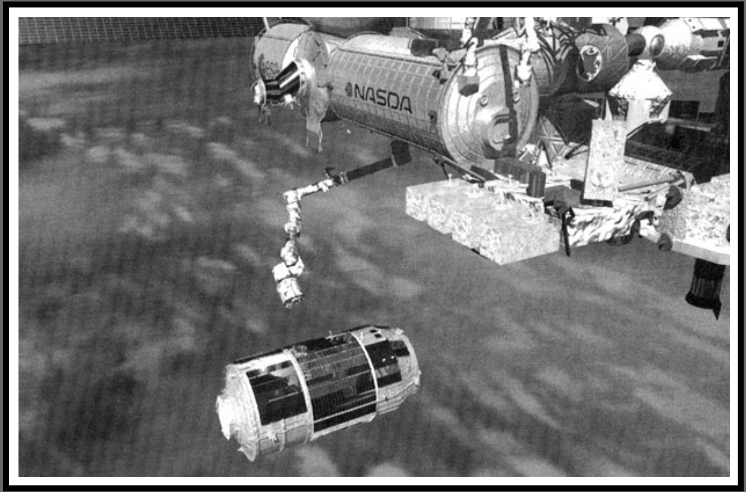
The Probe/Drogue Docking System is used for all Russian ISS Elements.

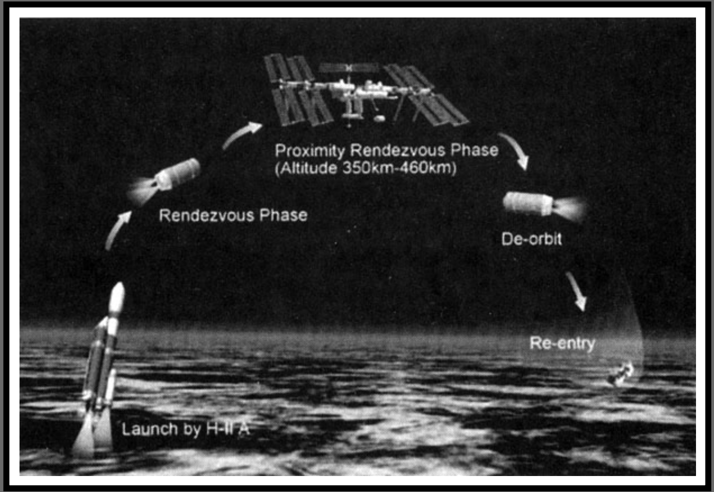


Autonomous Transfer Vehicle

The HTV-II Transfer Vehicle proposed by NASDA could also provide ISS reboost as well as pressurized and unpressurized logistics support. It would be approximately 9.2 m long x 4.4 m diameter and would be launched by a Japanese H-2A vehicle.

Unlike the Progress or Autonomous Transfer Vehicle (ATV), the HTV would not dock. Instead, the ISS Remote Manipulator System would grapple and berth the HTV to the station. The HTV removal process would be the same in reverse.

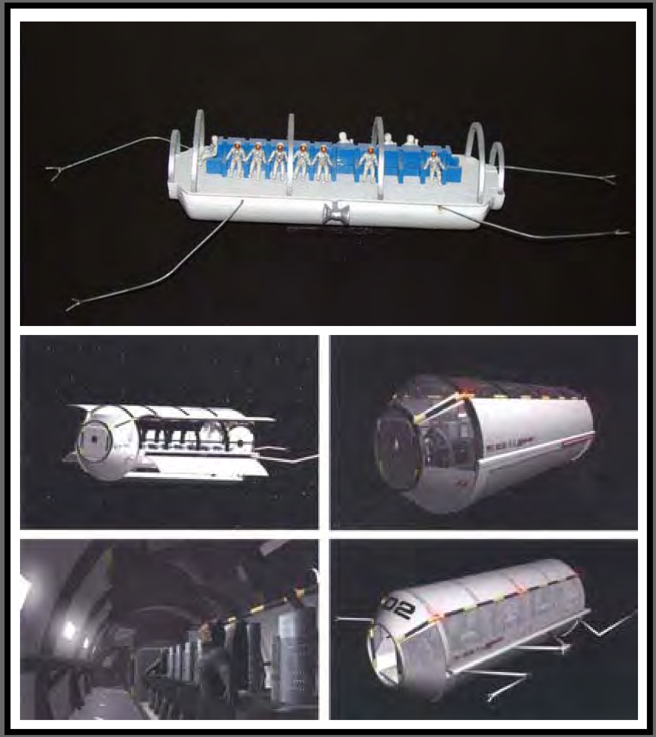




HTV-II Transfer Vehicle Concept

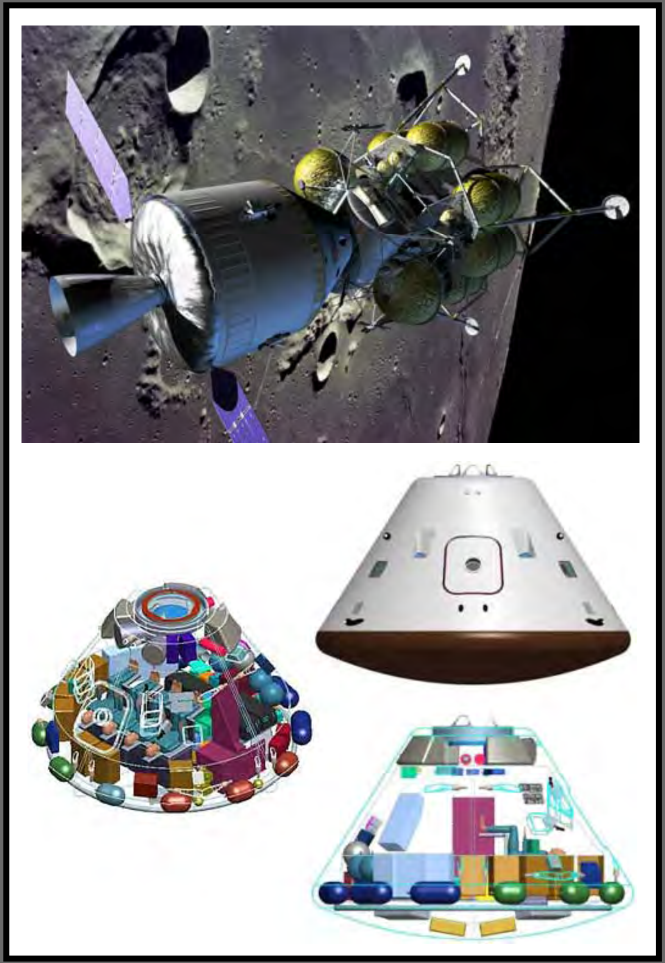
In 2001 SICSA proposed an Orbital Transfer Vehicle (OTV) concept to serve as a “pickup truck” to transport EVA-suited crews and large cargo elements between co-orbiting facilities. It could also provide ISS reboost services and support construction assembly operations:

- The OTV could be delivered by a dedicated Shuttle or Proton launch.
- It would be maneuvered and controlled through either by automated, teleoperated or manual systems as desired at any time.
- A Remote Manipulator System is incorporated for teleoperated or manual control to grapple payloads and support construction/ maintenance functions.
- Onboard tanks provide auxiliary air support and extended EVA operations.



SICSA’s LEO Transfer Vehicle Concept

NASA



Human Voyages Beyond LEO

As the US embarks upon future human voyages beyond LEO to the Moon, Mars, and perhaps further outward, new transportation, landing and return vehicles and technologies must be developed:

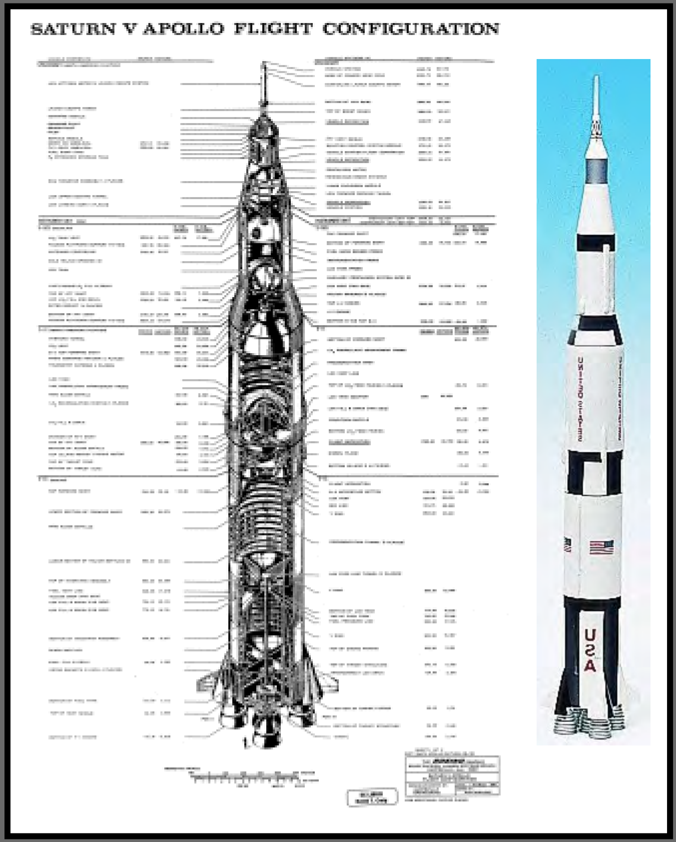
• NASA along with aerospace contractors and universities are studying requirements and concepts for a "Crew Exploration Vehicle" (CEV):

- A key purpose is to replace the aging, shrinking and technologically dated Space Shuttle fleet to support ISS and other LEO operations.
- The CEV is also intended to enable planned missions to the Moon and Mars by increasing delivery mass and volume to LEO and higher orbits/ trajectories.
- It is unlikely that a single type of CEV spacecraft will offer comprehensive crew and cargo delivery and return capabilities for human missions to the Moon, and even for less likely to Mars.
- Predictably, the CEV must comprise a fleet of different vehicle/ propulsion types that build upon Apollo lessons and legacies.

NASA

Apollo missions provide the only human flight and operational experiences to and upon an extraterrestrial surface environment to guide future lunar/ Mars planning.

- The Saturn V launch vehicle was the last of the Saturn series developed by Werner Von Braun's team at MSFC:
  - The 3-stage stack was 111 m (365 ft) tall, and produced lift-off thrust of 3.4 million kg (7.5 million lbs).
  - Five F-1 first-stage engines burned LOX and kerosene.
  - Five second-stage J-2 engines and one third stage J-2 engine used LOX/ LH2.
  - The third-stage engine fired a second time to boost the crew Command and Service Module (CSM) into a lunar trajectory.
  - The spacecraft was launched from Merrit Island, a few km from Cape Canaveral (Kennedy Space Center), the launch sites used for Projects Mercury and Gemini.



The Beginning of the Future

The design of all Apollo elements was driven by launch and orbital transfer strategies governed by technology capabilities at the time:

Three possible options were considered:

1.Direct Ascent approach:

- A huge rocket would launch all elements including a lander directly to the Moon and back again without intermediate maneuvers at either end.
- This was the simplest approach, but not feasible using available rocket technology.

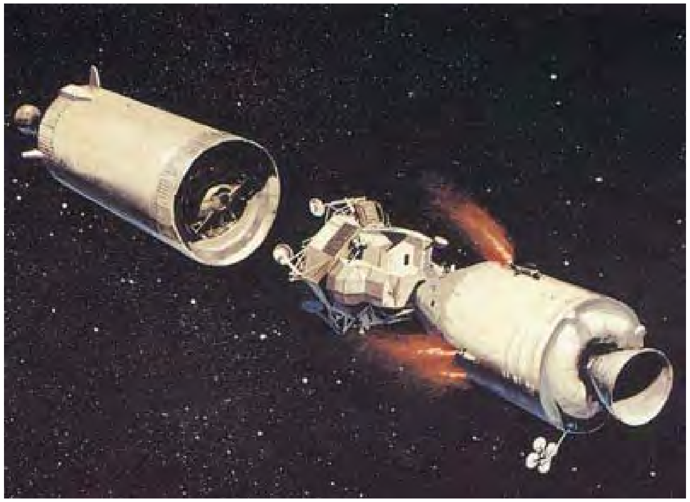
2.Earth Orbit Rendezvous (EOR) approach:

- Two rockets would be launched, one with fuel and the other with a spacecraft and crew.
- The two spacecraft would rendezvous/ dock in Earth orbit, fuel would be transferred, and the manned vehicle would proceed to the Moon and return directly to Earth orbit.

3.Lunar Orbit Rendezvous approach (selected):

- A 2-element mother ship would be launched together with a lander and ascent vehicle.
- The ascent vehicle would rendezvous/ dock with the mother ship in lunar orbit for crew return the Earth orbit.

NASA



Apollo Heading Moonward

Apollo Strategy Options Considered

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE

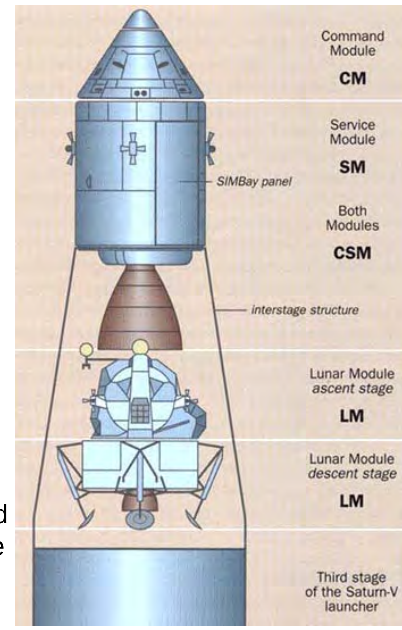
Principal Apollo spacecraft components were the Command Module (MC) and Service Module (SM) which were collectively referred to as the Command and Service Module (CSM):

- The Command Module (CM) was a cone-shaped vessel for 3 astronauts that was 3.5 m (12 ft) high with a base diameter of 3.9 m (13 ft):
  - The double-walled structure had an inner pressure hull with a pure oxygen atmosphere of about one-third Earth atmosphere.
  - The outer shell was a heat shield designed to withstand 3,000° C Earth reentry temperatures.
- The Service Module (SM) was mated to the CM during most of the journey to and from the Moon:
  - The SM supplied life support oxygen, power and water. (Power and water were produced by fuel cells.)
  - The SM also carried LOX/ LH2 tanks to fuel the cells.

The Apollo vessel included a Command Module (CM) for 3 people protected by a heat shield, and a Service Module (SM) which was xxxxxxxxx upon return to Earth with an attached Service Propulsion System (SPS).

The SM provided attitude adjustmentm crew supplies and Earth communications.

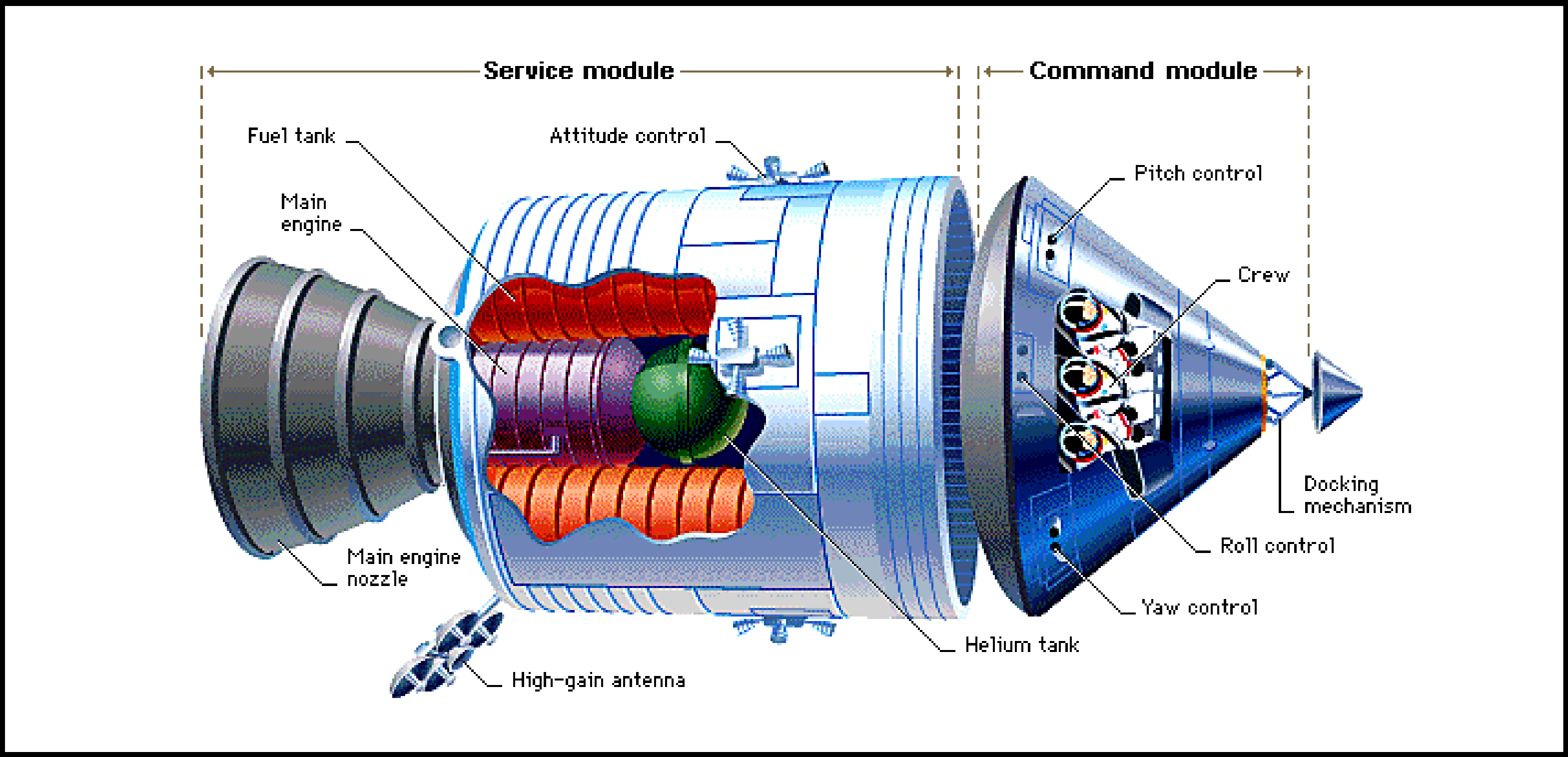
Together, the combined CM-SM were called the Command and Service Module (CSM).



Apollo Components

Apollo CSM as a Preference Vehicle

NASA



Apollo Command & Service Modules

A tragic flash fire in a pure oxygen flight simulator which occurred on January 27, 1967 took the lives of Edward White (1st Gemini Spacewalk, rookie Roger Chaffee, and Virgil (“Gus”) Grissom (2nd suborbital Gemini flight). The event delayed the Apollo Program and caused a crisis of public confidence in NASA. Several new safety measures resulted, including lowering of the oxygen atmosphere below 30 percent and the addition of the Launch Escape System.

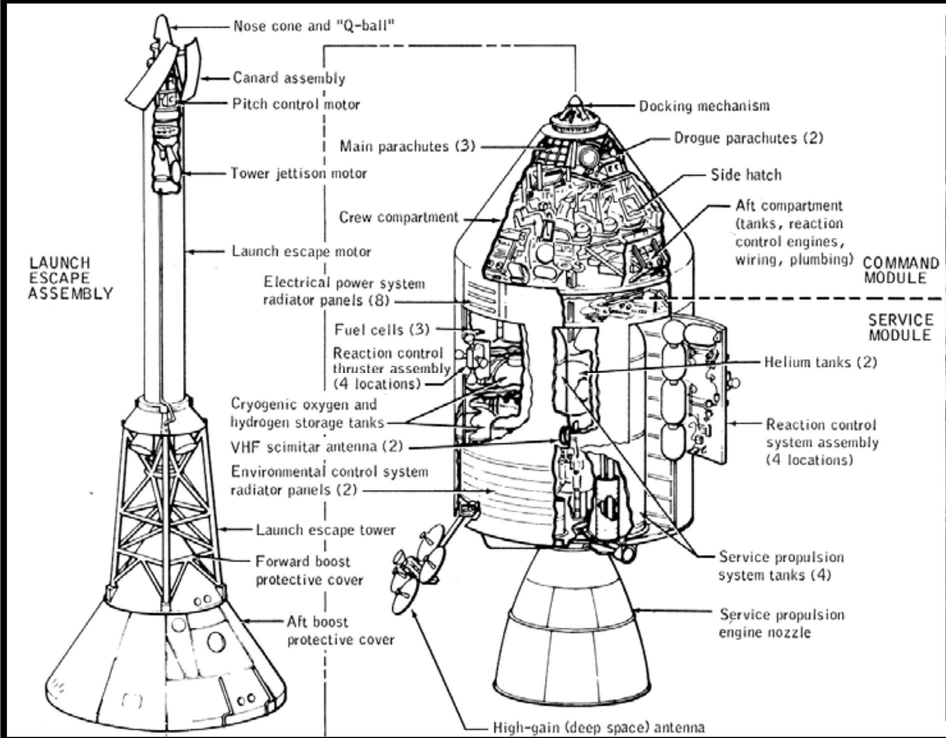
NASA

Simulator Interior Following the Accident

White, Chaffee and Grissom in Simulator

Tragic Apollo Safety Lessons

NASA

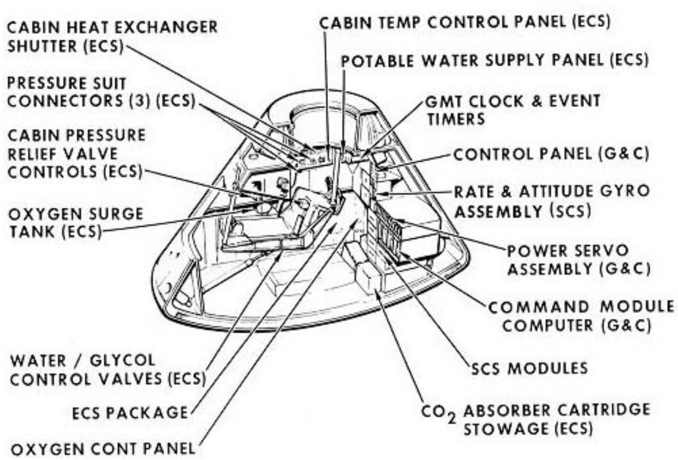


Apollo Command & Service Module & Launch Escape System

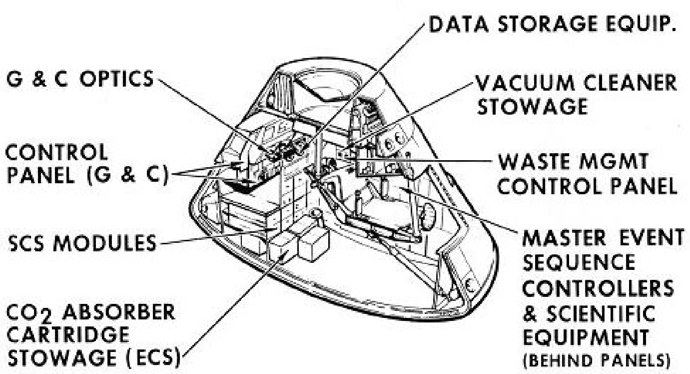
NASA

Apollo Command & Service Module

NASA



Left Side



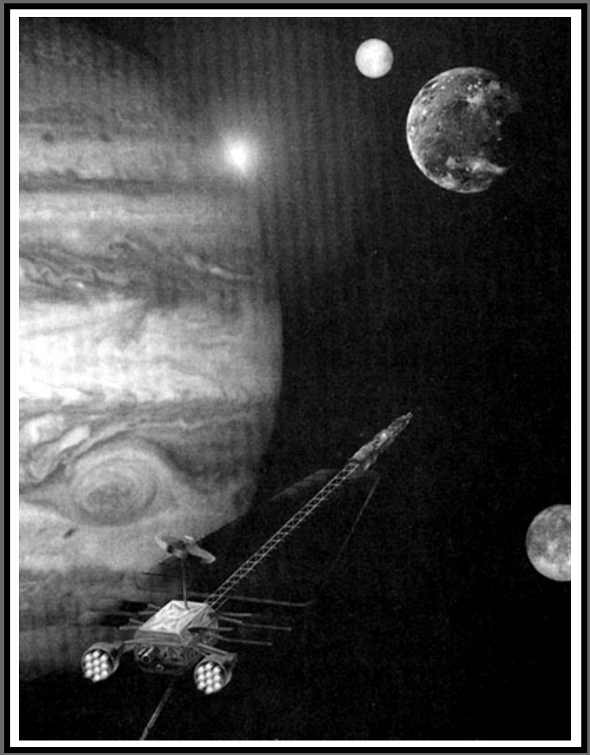
Right Side

Apollo Command Module Interior

New generations of existing spacecraft will be required to support future manned lunar missions, and entirely new breeds of transportation vehicles will be essential for human missions to Mars.

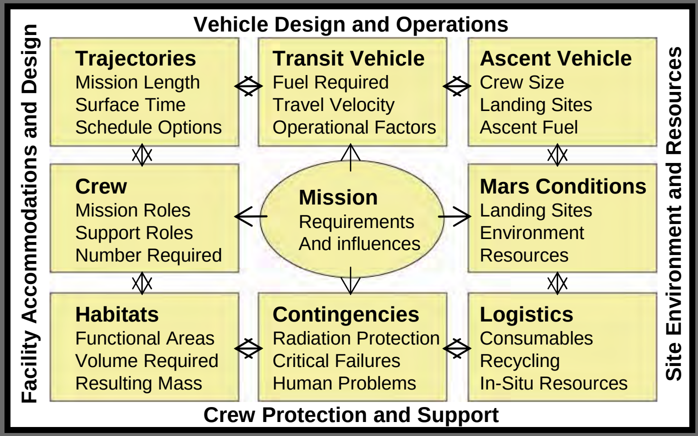
Important technology and design drivers will include:

- Substantial crew and cargo transfer accommodations associated with larger crews and longer transit times for Mars than previous Apollo missions.
- Larger cargo manifests for transits to the Moon and Mars due to accommodations for surface support influenced by larger crews, longer dwell times, and expanded activities (EVA exploration, in-situ resource collection/ processing and construction).
- Propulsive Trans-Lunar Injection and propulsive and/ or aerobraking at Mars influencing vehicle propellant and structure mass.
- Use of low-thrust nuclear engines for Trans-Mars Injection (TMI), increasing departure transit time through high radiation Van Allen Belts and influencing needs for radiation protection and/ or crew sprint intercepts.



Future Lunar/Mars Spacecraft

SICSA has conducted a variety of space exploration mission architecture and vehicle concept approaches. Some assume use of conventional Medium Lift Vehicles MLVs (Delta 4-Heavy and Atlas-5 class) for Earth launch and Earth Orbit Rendezvous (EOR) assembly and departure. Another investigation assumed availability of a Kepler class Heavy Lift Vehicle (HLV) such as SICSA's proposed ARES (100 MT+) launcher. The options also include chemical transfer and nuclear/ plasma propulsion possibilities for lunar/ Mars transit.



Facility Accommodations and Design:

- Duration of transit and surface stay for a given mission.
- The required crew size to undertake mission and support activities.
- Functional areas and requirements to accommodate all activities.
- Volume and mass of consumables needed to support crews.

Crew Protection and Support:

- All equipment and supplies for mission activities.
- Radiation countermeasures, both in transit and on Mars.
- Contingencies for critical equipment failures and crew health problems.
- Conservation through recycling and the use of in-situ resources.

Site Environment and Resources:

- Thin atmosphere and reduced gravity influencing ascent/descent.
- Seasonal temperature extremes and dust storms influencing operations.
- Surface launch access to desired orbital intercepts for return from the site.
- The type and availability of site resources to augment consumables.

Vehicle Design and Operations:

- Engineering strategies to land payloads in the thin atmosphere.
- Processes to obtain launch fuel from the Martian atmosphere.
- Crew size and sample return requirements for Ascent Vehicles.
- Safeguards to prevent contamination from dust returned to Earth.

SICSA Concept & Considerations

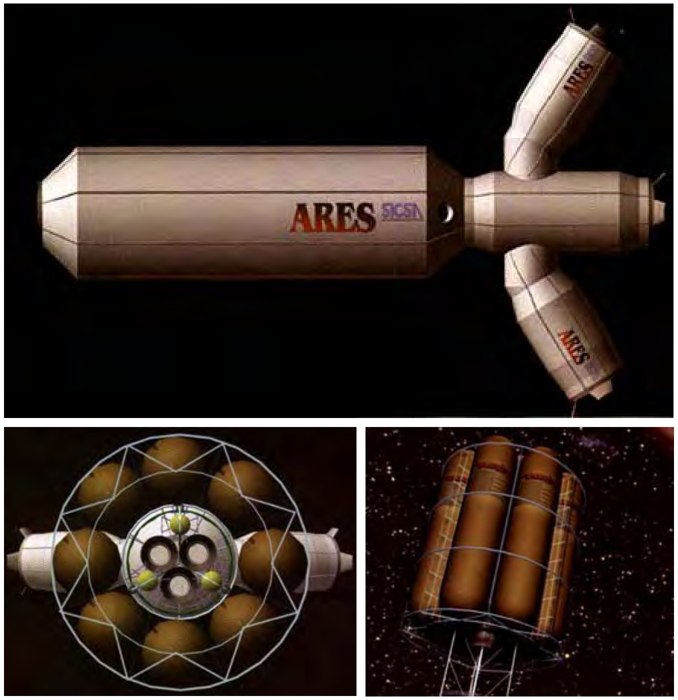
SICSA's proposed ARES transfer system is primarily intended for human missions to and from Mars orbit involving long travel times:

- The ARES Mars Transfer Vehicle (MTV) would use a nuclear powered low-thrust hydrogen plasma propulsion engine called "Variable Specific Magnetoplasma Rocket" (VASIMR):
  - The hydrogen is ionized by radio waves and guided into a central chamber threaded with magnetic fields that can eject plasma at variable velocities for thrust control.
  - LEO departure acceleration through the high radiation Van Allen Belts is relatively much slower than high thrust chemical rocket vehicles, but high transfer velocities are achieved through continuous acceleration vs. "blast and coast" systems.
  - The VASIMR-propelled MTV reverses orientation 180° at about mid-course to decelerate towards the Mars insertion orbit.

ARES Propulsion System

Crew radiation protection is a major design driver:

- The VASIMR MTV is structures in what might be termed a "Fuel Sleeve Assembly" (FSA) configuration with hydrogen fuel tanks surrounding a crew Transfer Hab (TH).
  - Following launch to LEO is a single HLV vehicle, the VASIMR power/ propulsion segment deploys the plasma engine and nuclear power elements outward behind the hydrogen tanks, thereby clearing the central area.
  - A second dedicated launch places the Transfer Hab on an intercept trajectory with the VASIMR segment, and automated rendezvous systems on the TM enable it to enter the vacated central VASIMR core and dock.
  - The hydrogen fuel and tank structures provide space radiation shielding for the TM and distance separation between the habitat and nuclear reactors mitigates radiation hazards from these sources.

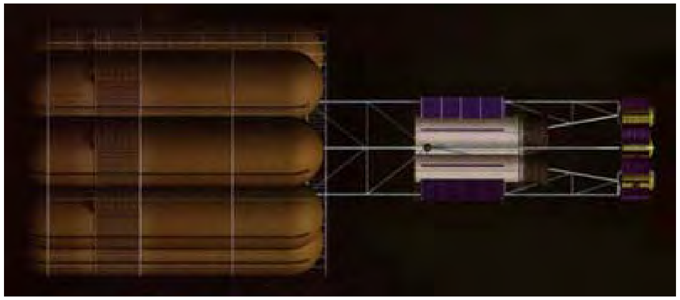


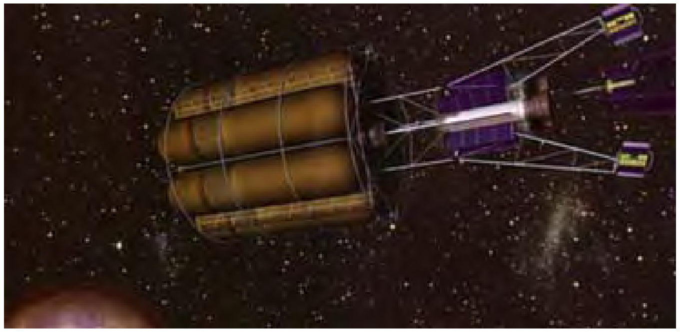
VASIMR Hydrogen Tanks Surround Crew Areas

Radiation Mitigation Strategies

While VASIMR propulsion can be expected to be much more energy-efficient than conventional chemical rockets, electrical power requirements will be massive.

- An estimated 12 mW of electricity would be produced to power the plasma process at a necessary thrust level:
  - Nuclear power is presently the only realistic option, since photovoltaic systems traditionally used on spacecraft will be grossly inadequate.
  - Power may also be required to cryogenically liquefy hydrogen fuel to avoid boil-off evaporation during extended LEO flight preparation, transit to Mars, surface operation and Earth return periods.
  - A relatively small amount of electricity (100s of kW) will be required to power habitat functions and spacecraft flight systems.





Nuclear Reactors Provide Electrical Power

VASIMR Power Requirements

The Transfer Hab (TH) module would provide living and activity accommodations for transit periods lasting approximately 500 days:

- The proposed 15 ft. diameter, 40 ft. long module is baselined to support up to 8 people:
  - Some of the crew might remain in parking orbits in the vicinity of Mars while others go to the surface.
  - The TH would also serve as a safe haven refuge for crews who may have to leave surface facilities under emergency circumstances until Earth return departure windows occur.
  - The TH will operate under very low gravity conditions similar to LEO facilities.
  - Other transfer elements including surface modules with landers, crew descent/ ascent vehicles and Earth reentry capsules attach to TH' docking ports.
  - THs are reusable for several missions, remaining in LEO or Earth/ Mars cycler orbits.

The Crew Transfer Module

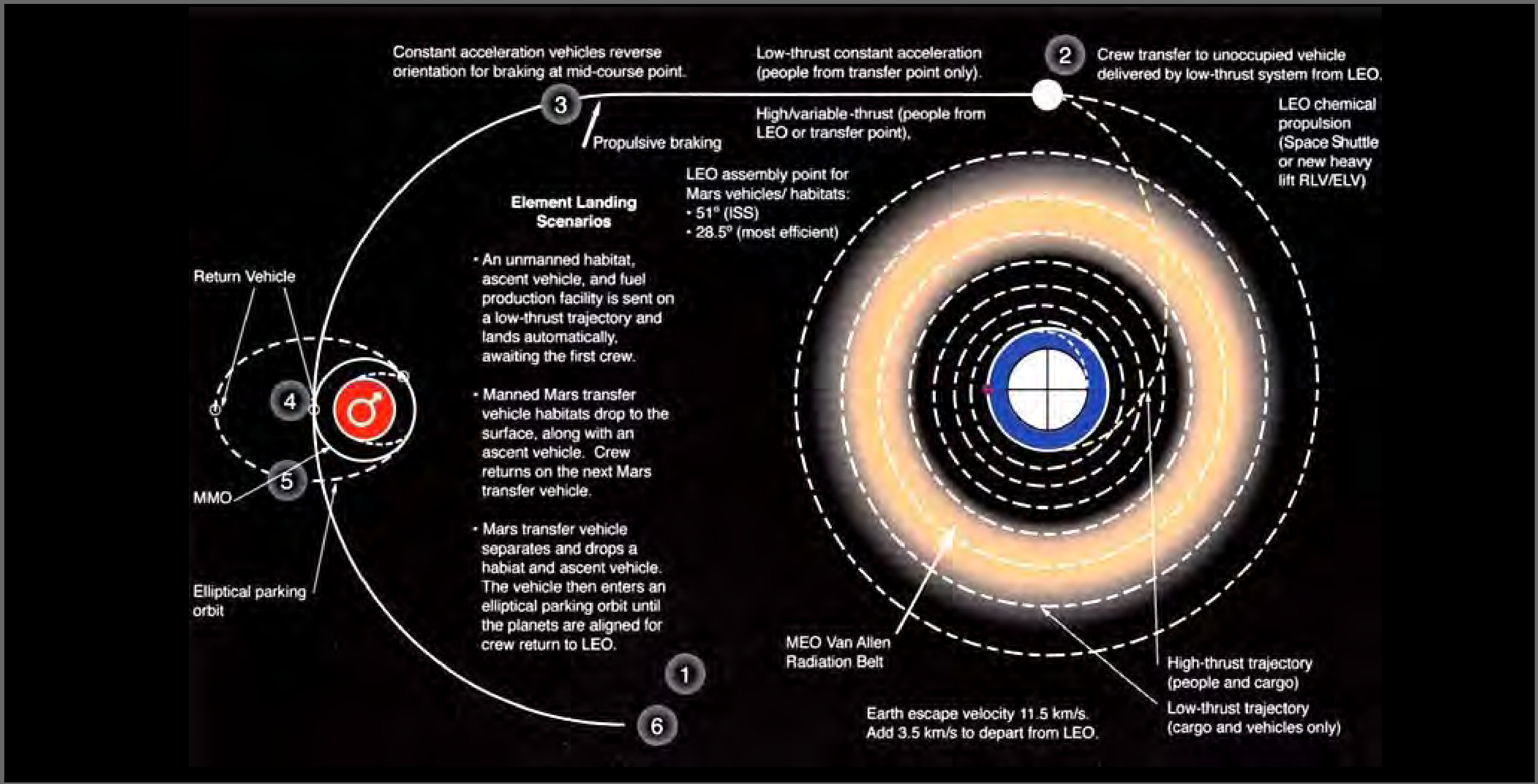
All elements proposed for the VASIMR study are designed to be compatible with the ARES launch vehicle cargo capacity:

- Key elements include surface habitat and logistics modules with integrated landing systems and crew launch, landing/ ascent and Earth atmosphere reentry vehicles:
  - An option is available for the crew to be originally launched from Earth or LEO by a “sprint” vehicle which would rendezvous and dock with the VASIMR MTV/ TM beyond the Van Allen Belts.
  - The sprint vehicle might also serve as the crew Mars surface descent vehicle, as well as the Earth atmosphere reentry and landing capsule.
  - Surface elements would be provided with “kick stage” rockets that would slow them into Mars Middle Orbit (MMO) to commence landing processes.

Crew Sprint, Surface Landing/Assent and Earth Reentry Vehicle

Surface Module, VASIMR and Crew Transferhab in Reference Shroud

VASIMR Transit Elements



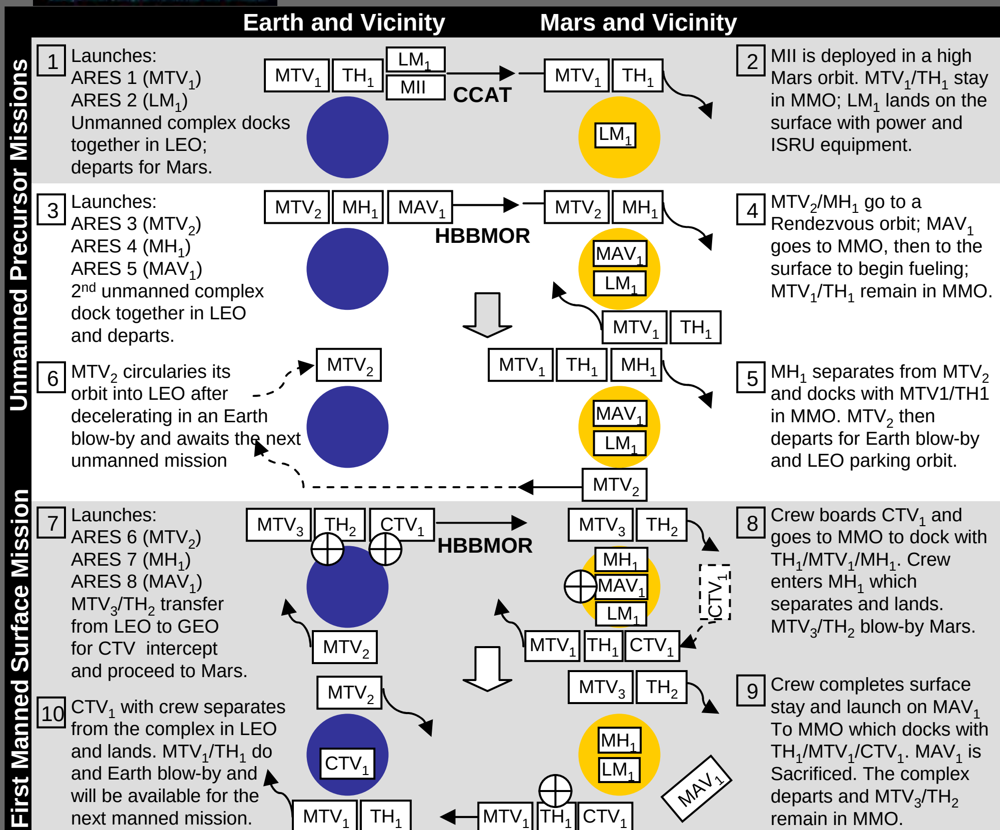
Possible VASIMR Trajectories

TRANSPORTATION SYSTEMS

EXPLORATION VEHICLES

MS-SPACE ARCHITECTURE

SICSA



Legend

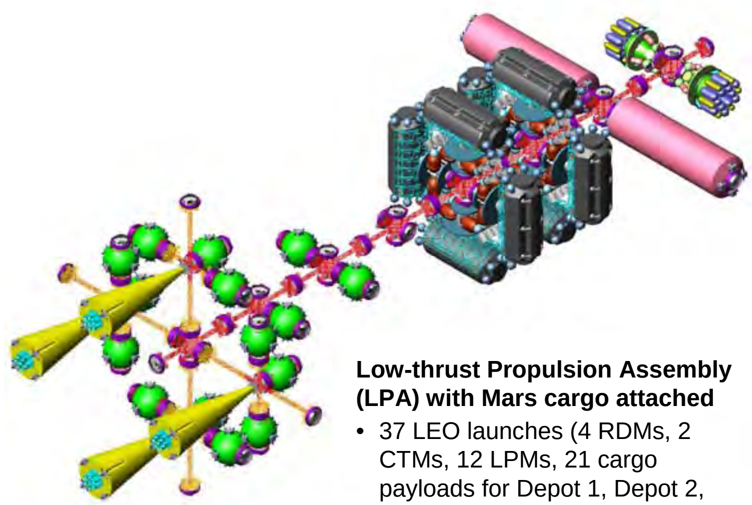
⊕	Crew: indicates manned mission segments.
LM	Logistic Module: carries power and ISRU equipment.
TH	TransferHab: modules are inserted into MTVs.
CTV	Crew Transfer Vehicle: launch/land on Earth.
MII	Mars information Infrastructure: Mars satellites.
MTV	Mars Transfer Vehicle: VASIMR spacecraft.
MH	MarsHab: deliver/support crews on the surface.
MAV	Mars Ascent Vehicle: launches crews to MMO.

Using a low-thrust engine, the MTV continues to accelerate until it reaches the halfway point between Earth and Mars, then turns around to thrust in the opposite direction in order to slow down enough to directly enter into Mars Middle Orbit (MMO).

Possible VASIMR Mission Staging

SICSA's proposed Low-thrust Propulsion Assembly (LPA) is a transport carrier to ferry payloads from LEO to lunar or Mars orbits and back which can be launched in sections by Medium Lift Vehicles (MLVs). All elements are designed to comply with 15 MT launch capacities and 3.75 m diameter by 12 m long shrouds that are expected to be compatible with planned Delta-4 Heavy and Atlas-5 capabilities.

- The LPA is essentially comprised of 3 major types of elements:
  - Reactor/ Drive Modules (RDMs), each containing a fission reactor and plasma thrusters.
  - Low-thrust Propellant Modules (LPMs) containing liquid argon or a comparable fuel.
  - Truss Segment Modules (TSMs) which form the main truss spine for payload attachments (example shown in the upper right half of the illustration).

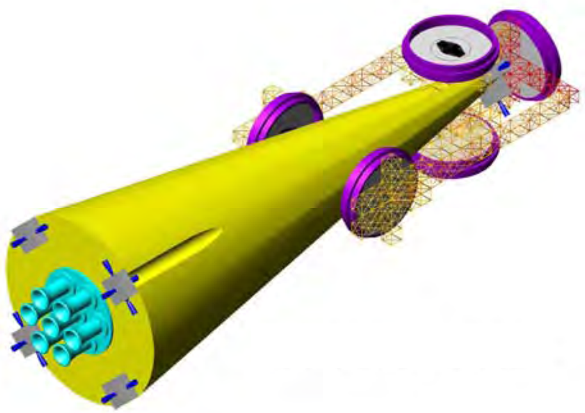


Low-thrust Propulsion Assembly (LPA) with Mars cargo attached

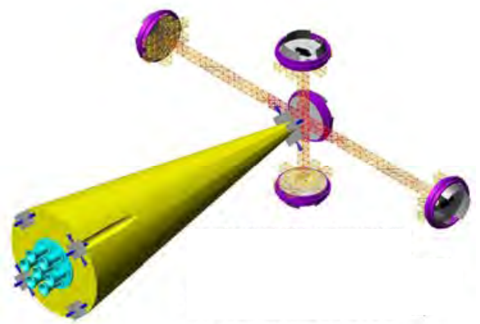
- 37 LEO launches (4 RDMs, 2 CTMs, 12 LPMs, 21 cargo payloads for Depot 1, Depot 2, and the Martian surface, and 8 TSMs "piggy-backed" on LPM launches), mass 553 tons, length 71m

Low-Thrust Propulsion Assembly (LPA)

Truss segments that connect the Reactor Drive Modules to the Truss Segment Module spine and to Low-thrust Propellant Modules fold for launch to LEO within MLV Payload shroud limitations. The RDMs are fission reactors estimated to produce about 3.5 MWe to power 7 Magneto-Plasmadynamic Thrusters (or comparable) drive units.



Reactor/Drive (RDM) – configured for launch

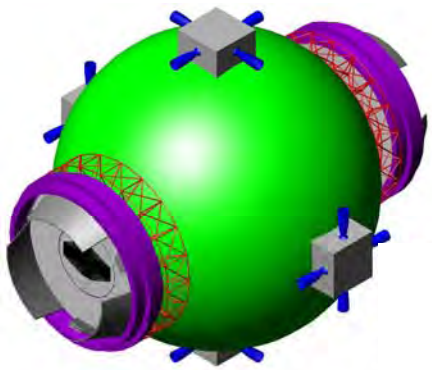


Reactor/Drive Module (RDM) – configured to flight

- One fission reactor, 3.5 MWe output (4kg/kWe)
- Seven low-thrust drive units, 0.5 MWe input each, 51%efficiency, Isp 6000 seconds, type TBD but considered to be Magneto-Plasmadynamic Thrusters for modeling
- Two long (6.4m) and two short (2.7m) folding truss arms
- Five docking ports, one on top and one on each truss arm
- 1 LEO launch; mass 15 tons, length 12.5m (unfolded)

LPA Reactor Drive Module (RDM)

Low-thrust Propellant Modules provide propellant storage for Reactor Drive Module thrusters. Fully pressurized, these 3.7 m diameter units weigh about 15 MT. Their structures contain docking ports at each end (for attachments to either Reactor Drive Modules or Truss Segment Modules) and avionics for automated orbital assembly.

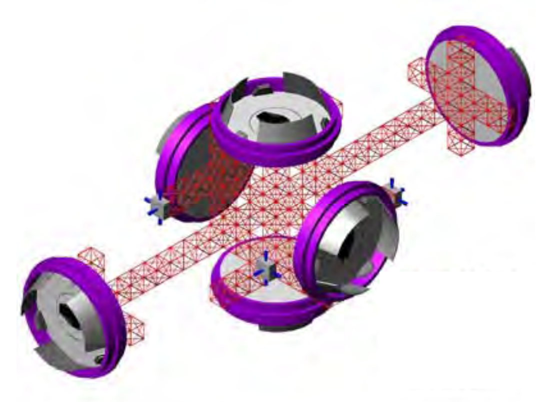


Low-thrust Propellant Module (LPM)

- Propellant storage for low-thrust drives on the RDM
- Propellant TBD, but considered to be liquid argon for modeling
- Two docking ports, top and bottom
- May be launched partly full, and/or with other components
- 1LEO launch; mass 15 tons, length 3.7m

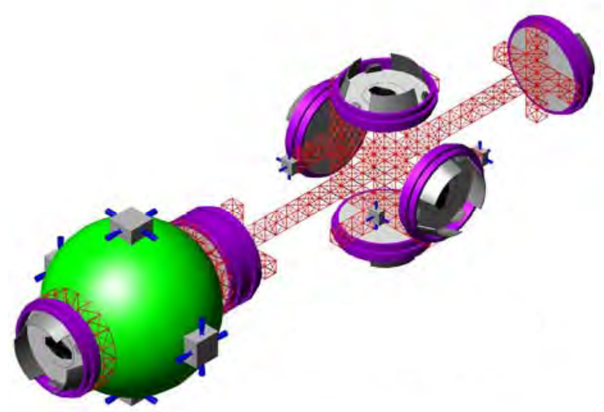
Low-Thrust Propellant Modules (LPMs)

The Truss Segment Modules that comprise the LPA's main spine each contain 6 docking ports for automated attachments of LPMs, payloads or other TSMs. Avionic systems attached to the docking interfaces enable orbital maneuvering and positioning.



Truss Segment Module (TSM)

- Fixed truss
- Six docking ports, top and bottom and four around the waist
- 0 dedicated LEO launches (launched "piggyback" with LPMs), mass negligible (<0.25 tons), length 7.9m

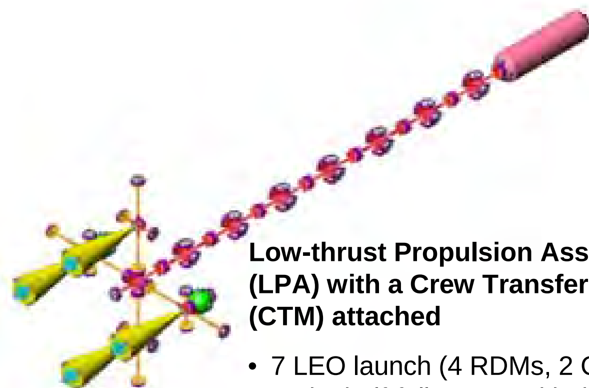


Truss Segment Module (TSM) attached to Low-thrust Propellant Module (LPM) in "piggyback" launch configuration

Truss Segment Modules (TSMs)

The baseline Crew Transfer Module (CTM) is proposed to accommodate 8 people for estimated crewed high Earth orbit-Mars orbit transits of 100 days, and Mars orbit-LEO return times which are comparable:

- Crews would access the CTM using a fast sprint vehicle which would rendezvous and dock with it beyond the Van Allen Belts:
  - The uncrewed CTM would be attached to the LPA in LEO prior to a 35 day spiral-out period to the intercept point.
  - The surface mission time might be 667 days (including a 13 day spiral-in time to Mars orbit).
  - The spiral-out time from Mars orbit for return to Earth is estimated to be 14 days.



Low-thrust Propulsion Assembly (LPA) with a Crew Transfer Module (CTM) attached

- 7 LEO launch (4 RDMs, 2 CTMs, and 2 half-full LPMs, with the 8 TSMs "piggybacked" on LPM launches), mass 90 tons.

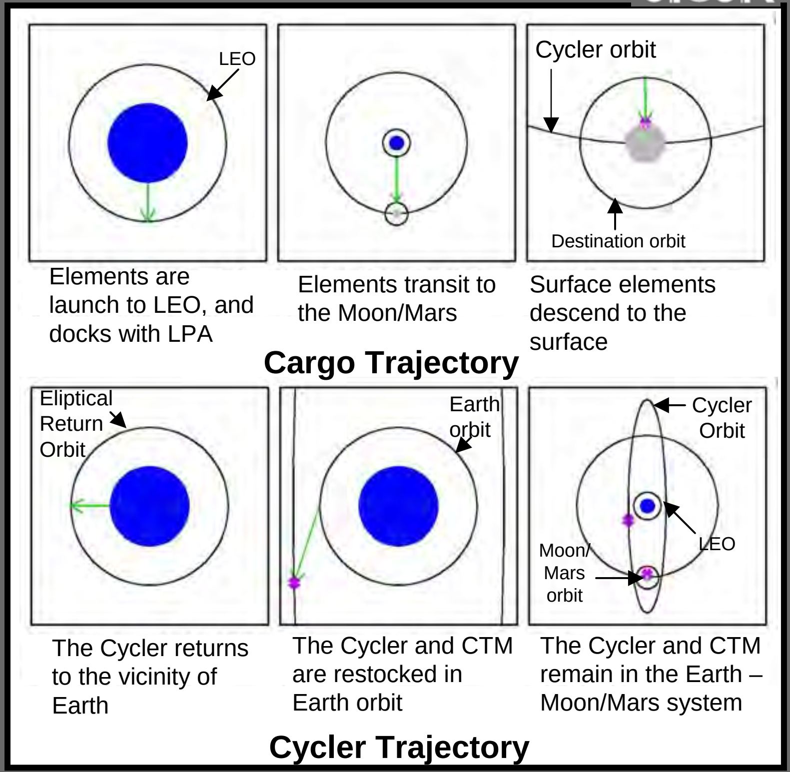
LPA With Crew Transfer Module (CTM)

Crew Transfer Module (CTM)

SICSA's low-thrust transfer strategy launches payloads to LEO together with landing stages (if required) and the elements dock to ports on the Truss Segment Modules (TSMs):

- **Attachment elements may include:**
  - A Crew Transfer Module (CTM) without crew.
  - A Surface Habitat Module (SHM) with lander.
  - A crew surface descent/ ascent vehicle.
  - A Pressurized Logistics Module (PLM) and/ or Unpressurized Logistics Module (ULM) with lander(s).
- **Cycler options:**
  - The Low-thrust Propulsion Assembly (LPA) can remain in an Earth-Moon/ Mars cycler orbit indefinitely for multiple mission applications (assuming refueling capabilities).
  - An attached CTM can remain as part of the LPA Cycler (with restocking of crew expendables in LEO).

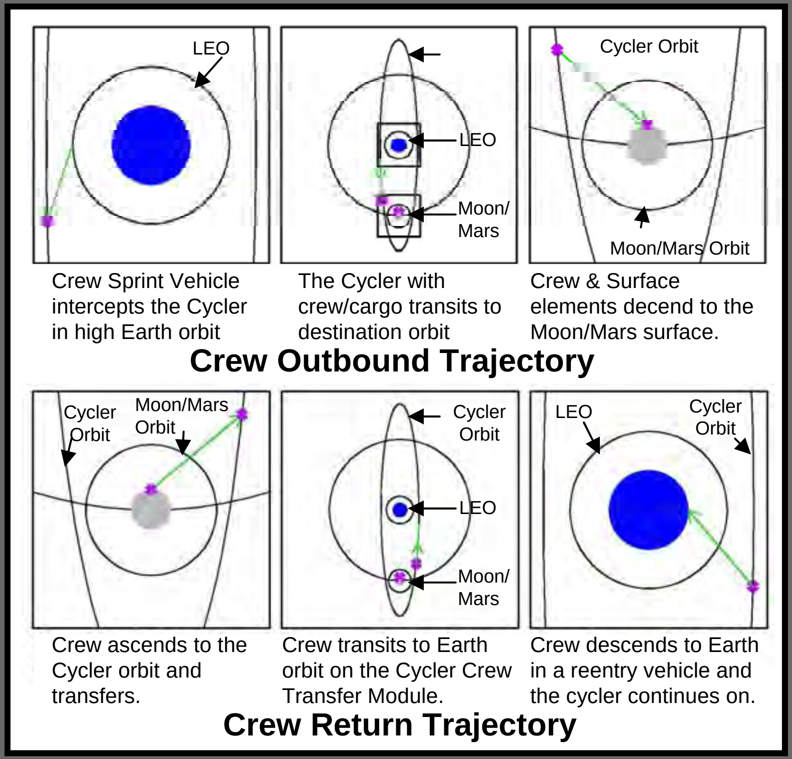
The LPA would do a "back-flip" braking maneuver at about the half-way point to the Moon and return to Earth to reduce velocity.



Low-Thrust Cargo/Cycler Transfers

The low-thrust propulsion in combination with the reusable Cycler vehicle are proposed to reduce propellant consumption over use of conventional chemical rockets for cargo and crew transfers from LEO to Moon/ Mars orbit, along with a more rapid high-thrust launch system to transfer crews to the Cycler beyond the high radiation Van Allen Belt region:

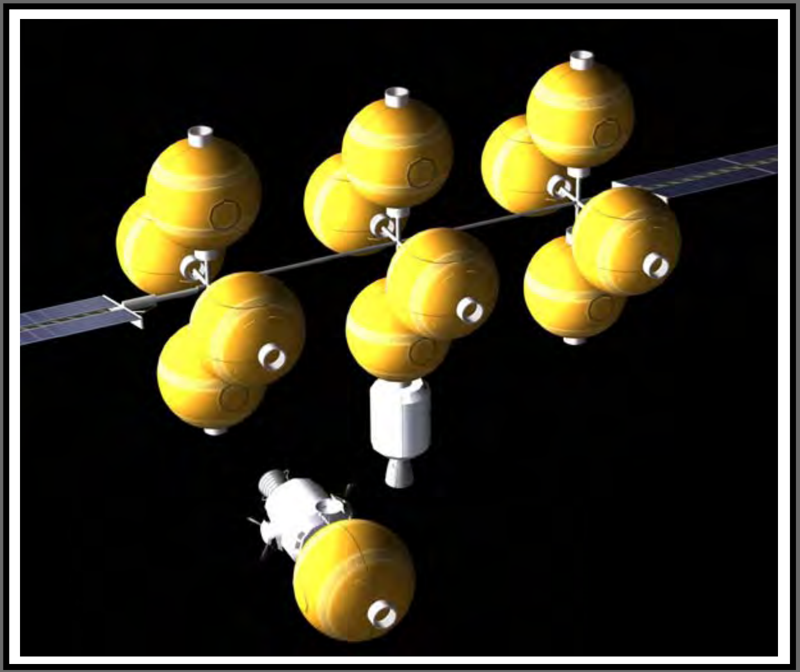
- Crew Sprint Vehicles:
  - These conventional rocket vehicles would launch crews to a high Earth orbit intercept point to dock with the Cycler.
  - As options, the Sprint Vehicles might also be designed to provide crew lunar/ Mars surface descent/ ascent functions, and/ or to serve as Earth atmosphere reentry means.



Low-Thrust and Sprint Crew Transfers

The propellant type and mass that must be carried to power a vehicle directly influences the rate of velocity change (Delta-V) and mass that is available for the payload as a fraction of total mass (the mass fraction). Even a moderate reduction in fuel mass (or improvement in mass fraction) can provide substantial economies.

- Refueling at orbital depots can reduce the amount of total propellant mass consumption:
  - This essentially breaks total Delta-V required into trip segments (analogous to staging for a conventional vehicle).
  - The vehicle only needs to carry enough propellant at one time to reach the next refueling point (rather than carry enough for the entire mission).

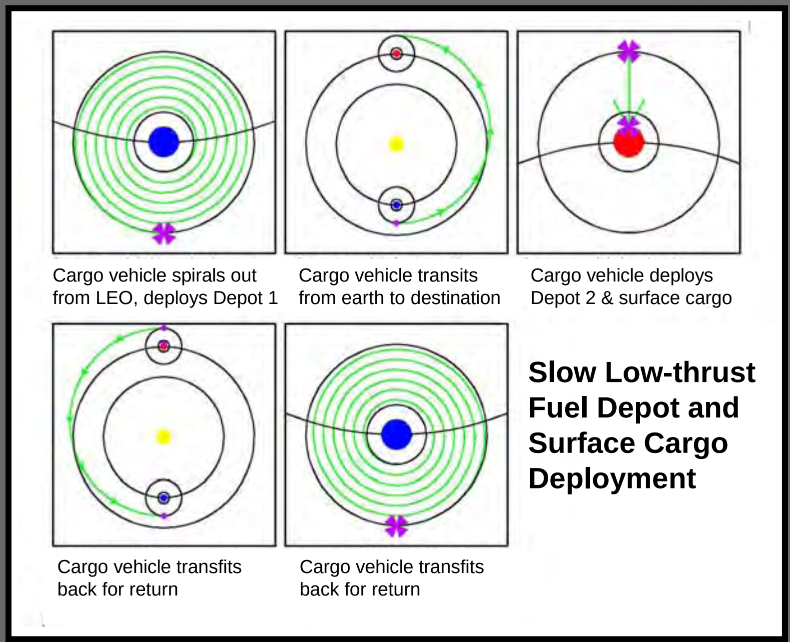


Orbital Refueling Benefits

Use of orbital fuel depots may be most beneficial for “split” missions where fuel (and other cargo) is delivered using “slow” low-thrust trajectories in advance of crews delivered by separate vehicles using “fast” low-thrust trajectories:

- Slow-trajectory low-thrust missions can minimize logistics and equipment transfer costs:
  - They offer the highest payload fractions of all known systems.
  - They are well suited to delivering orbital depots to refuel faster crew trajectories.
- Fast low-thrust trajectories are best suited for crew transfers:
  - They are less fuel efficient than slow trajectories, but much better than conventional systems.
  - They afford much quicker flight times than slow trajectories, offering many benefits.

Each cargo mission would deliver propellant to orbital depots to refuel crew mission vehicles.

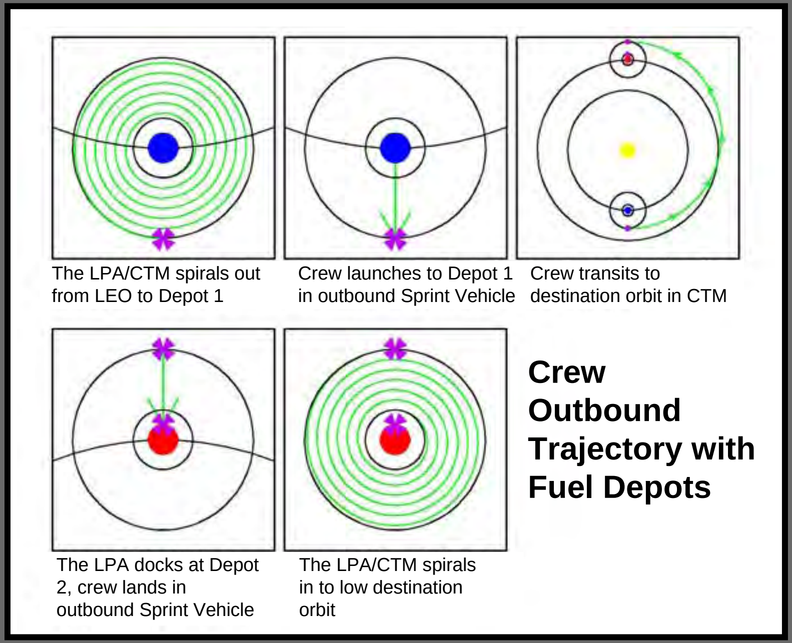


Split Cargo/Crew Mission

Split cargo/ crew missions will enable fuel economies afforded by slow trajectories to benefit shorter travel time advantages of faster trajectories. Examples are provided for Mars missions:

- Mission travel days:
  - Slow low-thrust (“conjunction class”) trajectories will require about 493 days outbound and 493 days inbound (986 days total), compared with about 165 days outbound and inbound (330 days total) for fast trajectories.
  - Slow trajectories will offer about 200 days on the surface (estimated by launch windows) compared with about 644 days for fast (“opposition class”) trajectories.
- Payload/ total vehicle mass fractions:
  - Slow trajectories will offer round-trip mass fractions of about 0.327, compared with about 0.027 for fast trajectories (assuming no orbital refueling to reduce propellant mass).

Crew Sprint vehicles might intercept the LPA at the outbound fuel depot location outside LEO



Crew Transfer with Fuel Depots

Mars Cargo Mission Time

Slow trajectory low-thrust cargo mission periods are estimated to be 1,016 days:

- 259 days of LEO launch and assembly (37 launches at 1 week intervals)
- 757 flight days:
  - -192 days Earth-out spiral
  - - 242 days Mars transit (no Mars-in/ out spirals)
  - - 200 days wait at Mars
  - - 100 days Earth transit
  - - 23 days Earth-in spiral

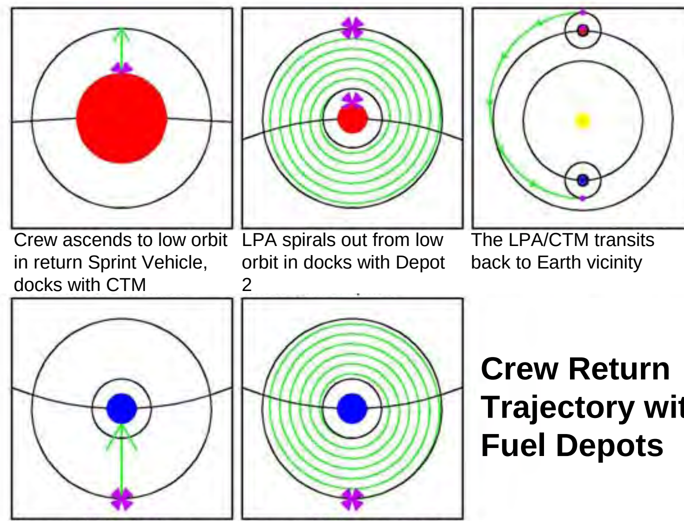
Mars Crew Mission Time

Fast trajectory low-thrust crew Mars mission periods are estimated to be 894 travel and surface days:

- 0 days during 35 day Earth spiral-out
- 100 days Mars transit
- 13 days Mars-in spiral (partial crew on board)
- 667 days surface mission (partly concurrent with Mars-in spiral)
- 14 days Mars-out spiral
- 100 days Earth transit
- 0 days during 35 day Earth spiral-in.

Estimated Mars Mission Days

Crew ascent vehicles might intercept the LPA at an inbound fuel depot location in the Moon/Mars orbit.



Crew ascends to low orbit in return Sprint Vehicle, docks with CTM

LPA spirals out from low orbit in docks with Depot 2

The LPA/CTM transits back to Earth vicinity

The LPA docks at Depot 1 & crew lands in return Sprint Vehicle

The LPA/CTM vehicle spirals in to low Earth orbit for reuse.

Crew Return Trajectory with Fuel Depots

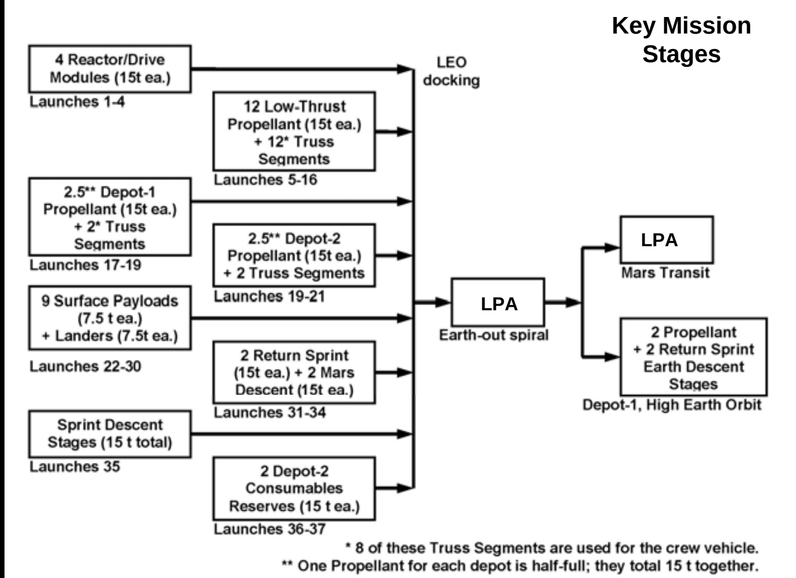
Crew Transfers with Fuel Depots

The LPA and attached elements are assembled in LEO:

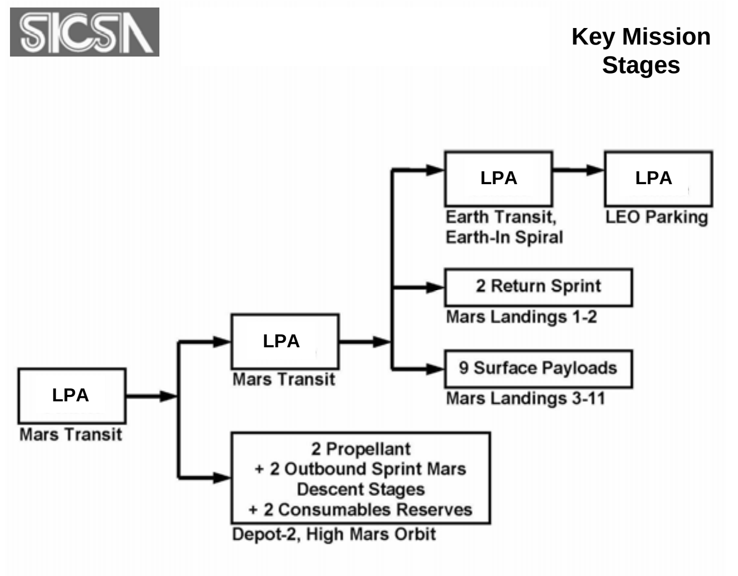
- Reactor Drive Modules (RDMs), Low-thrust Propellant Modules (LPMs) and Truss Segment Modules (TSMs).
- Depot 1 and 2 LPMs, structures and transfer systems.
- Surface payloads and landers.

Cargo elements are delivered to orbital locations:

- LPMs and associated systems are placed in high Earth orbit (Depot 1) and Mars orbit (Depot 2).
- Surface elements are deployed to landing orbits.
- The LPA may remain in a Mars parking orbit.



Cargo – Earth Orbit Departure



Cargo – Mars Transit/Return

Low-thrust Cargo Mission Example

The LPA is placed and assembled in LEO and crew elements are attached, including:

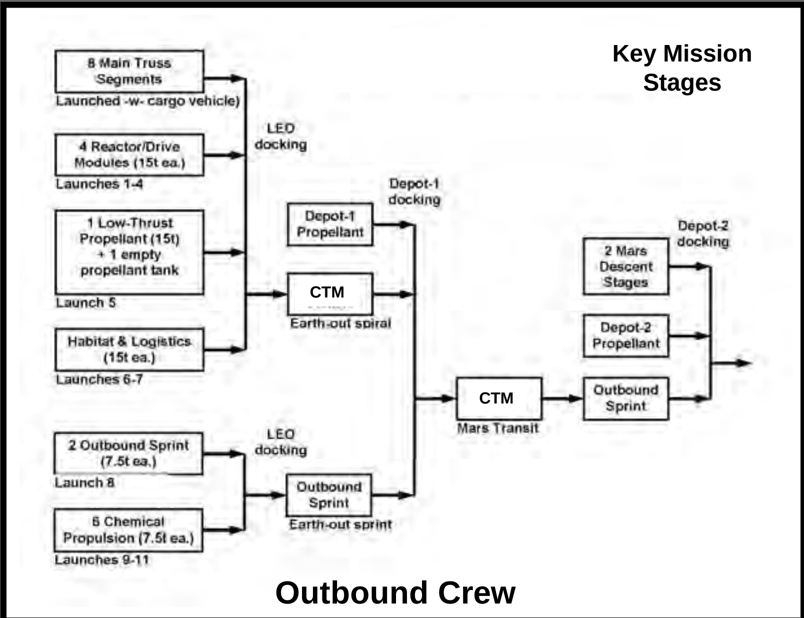
- A Crew Transfer Module (CTM)

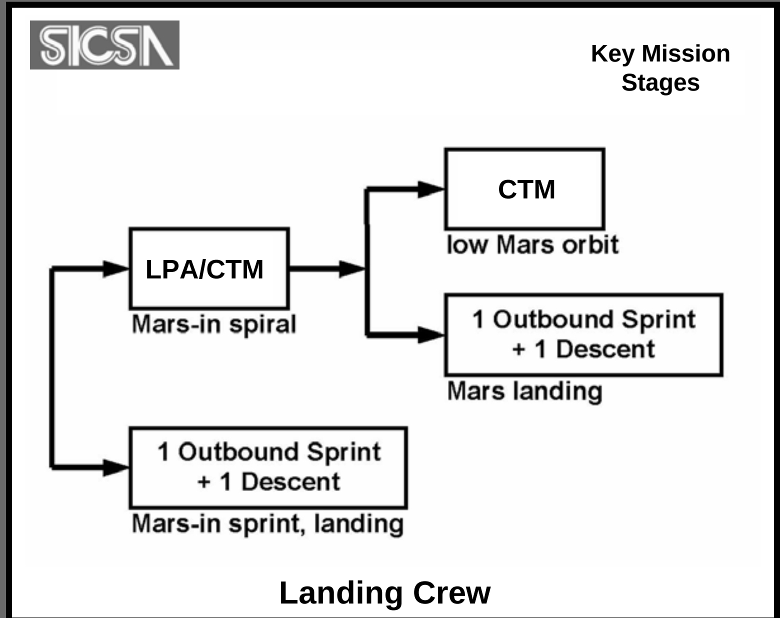
- Partly full propellant tanks (to be refueled)

The LPA proceeds to the vicinity of Mars and propulsively slows by reversing orientation:

- The LPA spirals-in to a low Mars orbit.

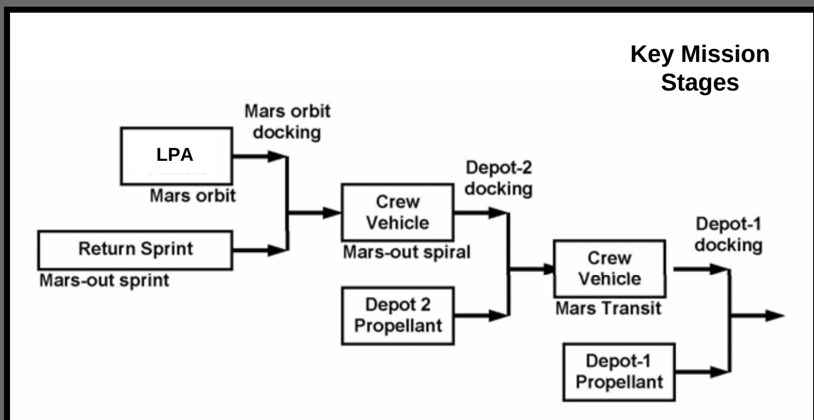
- The crew descends via the vehicle to the surface.





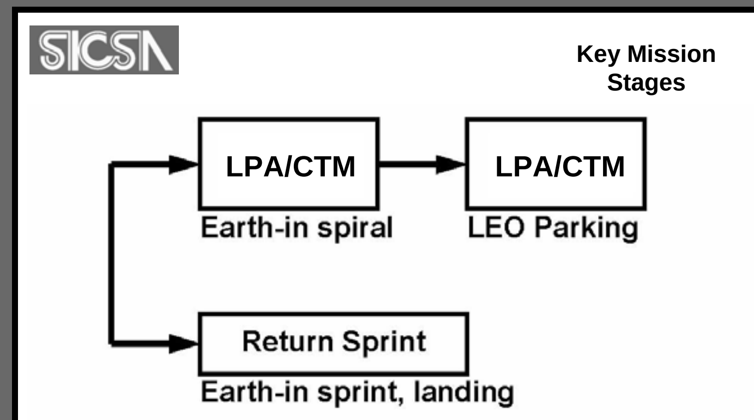
Low-thrust Mars Crew Mission Example

The LPA/ CTM decelerate and spiral in to LEO for crew landing and proceed to Depot 1 for refueling.



Inbound Crew

The crew assent vehicle intercepts the LPA in the Depot 2 refueling orbit and transfers to the CTM.



Crew Earth Return

Low-thrust Mars Crew Mission Example

Mars Cargo Vehicle Masses

	Dry	Initial
total initial mass (tons)	60	553
drive propellant (tons)	0	179
structure (tons)	60	60
payload to Earth orbit (tons)	0	43
payload to Mars orbit (tons)	0	272

Launches Required: 37
4 structure
12 propellant
21 payload

Cargo Vehicle Masses

Mars Crew Vehicle Initial Masses

	Dry	Earth-Out Spiral	Mars Transit	Mars In & Out Spirals	Earth Transit	Earth-In Spiral
total initial mass (tons)	75.0	85.3	98.4	83.4	98.4	85.3
propellant (tons)	0.0	10.3	15.9	8.4	15.9	10.3
structure (tons)	45	45	45	45	45	45
payload (tons)	30	30	37.5	30	37.5	30

Launches Required: 8 (11 with sprint vehicles and crew)
4 structure
1 propellant
2 payload
(1 sprint vehicle)
(3 chemical propulsion)

Crew Vehicle Masses

Low-thrust Mars Vehicle Mass Estimates

SICSA proposed a MLV-compatible Earth to Moon/ Mars orbit transportation system in 2004-2005 which would use stacked stages of chemical Propulsion Modules (PMs). This work was undertaken in support of a NASA space exploration “Concept Evaluation and Refinement” (CE&R) study contract granted to SPACEHAB.

- Baseline requirements include:
  - Maximum single-element launch capacities limited to 15 MT, and 3.75 m diameter x 12 m long payload shrouds (for all orbital and surface facilities)
  - All element assembly operations to be conducted autonomously and independent of ISS.
  - Initial crew accommodations (orbital and surface) for 4 people to support surface stays of 14 days extendable for 100 days (Moon) and 500 days (Mars).

MLV Chemical Propulsion Stack

Each of the MLV-compatible Propulsion Modules would require a dedicated 15 MT launch to LEO where they would be assembled into “trains” using a maneuvering vehicle for rendezvous positioning and docking alignments:

- The PMs would use LOX/ LH2 propellant and would provide several advantages:
  - Integrated thruster engines in each module would provide a high level of redundancy in the event of failure.
  - Engine re-starts would not be required since each module is used only once.
  - Total thrust available can be tailored to mission requirements through the modular approach.
  - PM stages are ejected following use to continuously reduce transfer mass as a mission progresses.
  - No fuel line connections or propellant transfers are needed between modules or the vehicle.

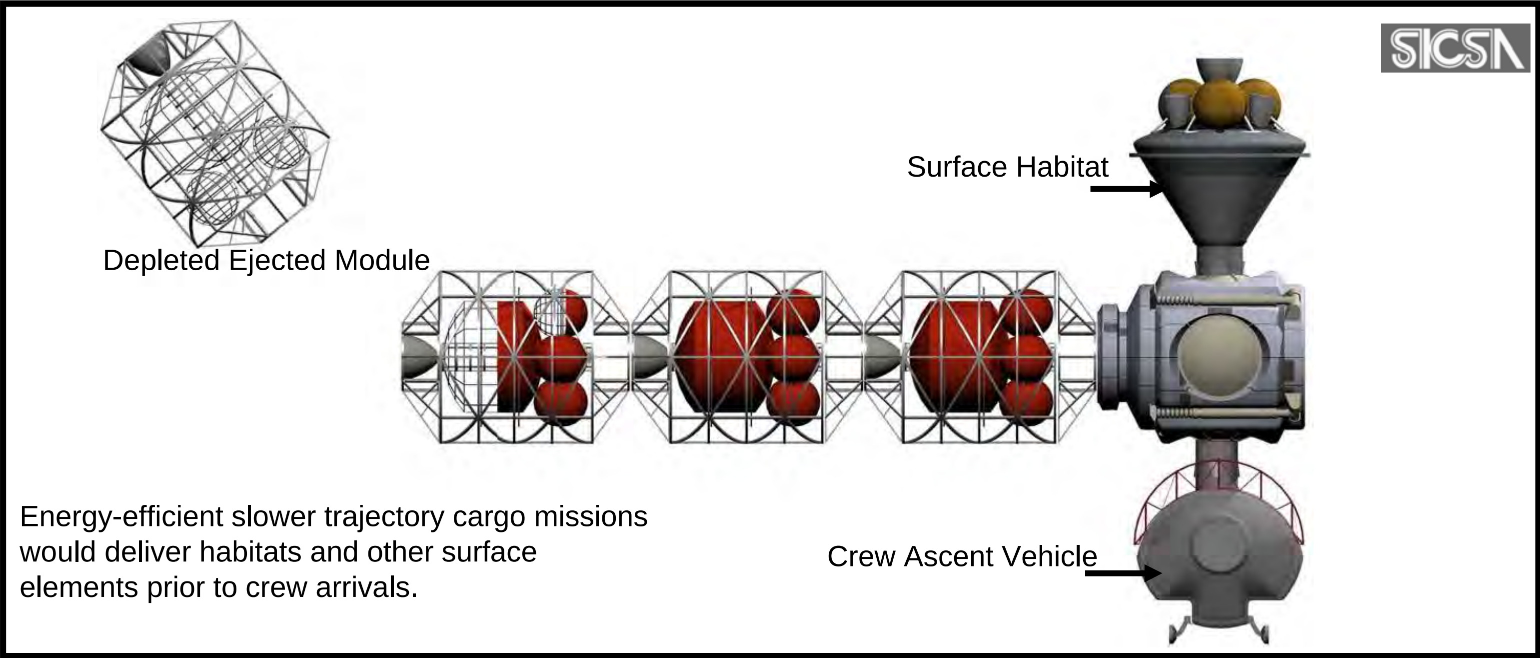
MLV Chemical Propulsion Module

Key goals of the SPACEHAB study were to determine requirements and concepts for human lunar/ Mars exploration using MLV LEO launch capabilities:

- Important study conclusions:
  - Use of Medium Lift Vehicles (MLVs) will require means to assemble flight elements and some payloads in orbit, including attachment of surface modules with their landers and connection of Propulsion Modules together with the Crew Transfer Module and payloads.
  - Smaller launch vehicles will necessitate more LEO deliveries and assemblies, but will also reduce cost and schedule delay impacts of a single launch failure, and can utilize multiple conventional launch site options.
  - Autonomous and remotely monitored/ controlled maneuvering vehicles can be used to minimize on board element avionics requirements for rendezvous and docking.

MLV Element Assembly in LEO

The cargo train would reverse orientation near the destination insertion orbit and propulsively brake using the remaining unspent Propulsion Module to place surface payloads into their lower altitude landing orbit.

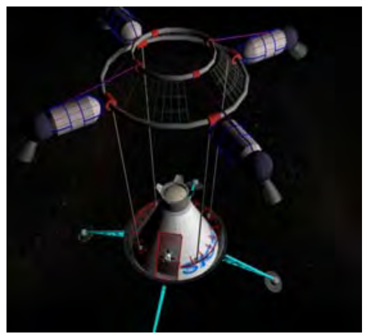


MLV Transfer Cargo Mission

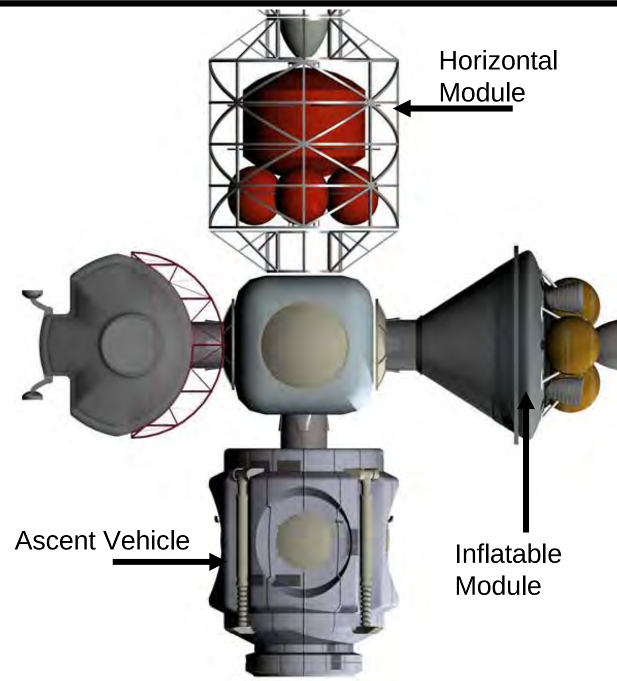

Horizontal Module


Inflatable Module


Logistic Module

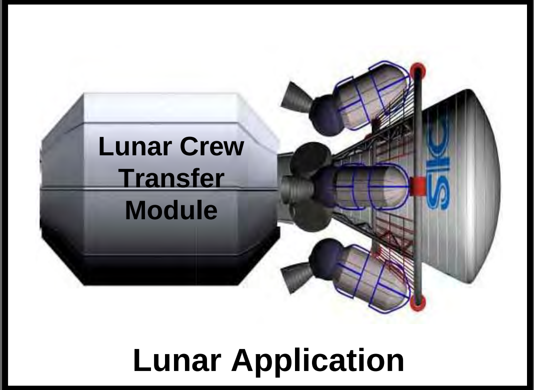

Ascent Vehicle

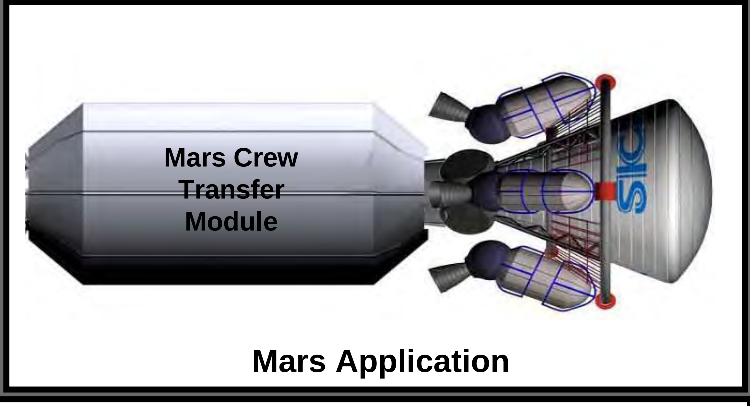
Surface Elements in Deployed Status

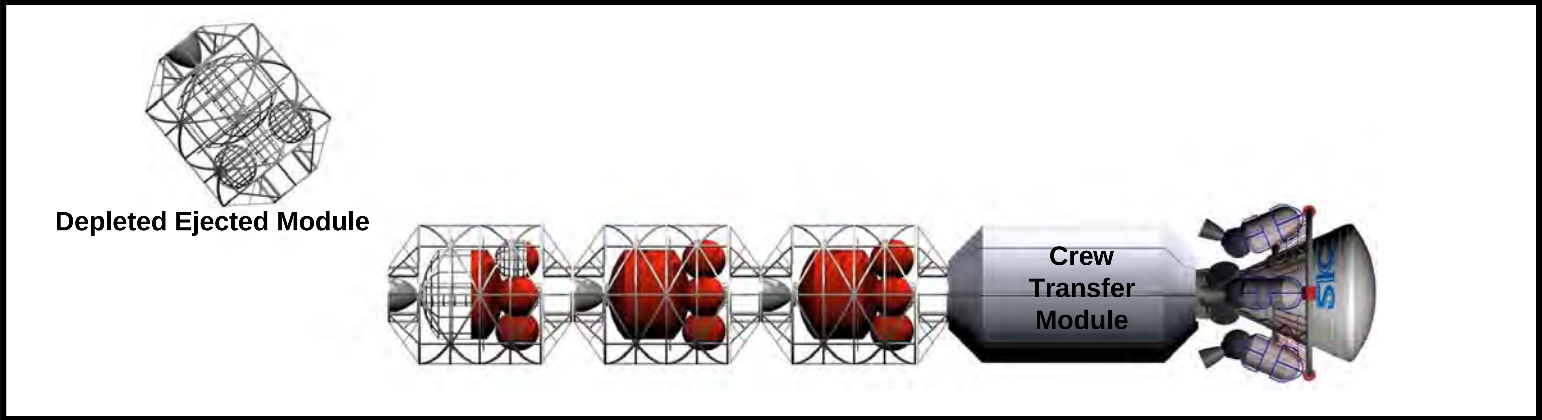


Example Surface Elements

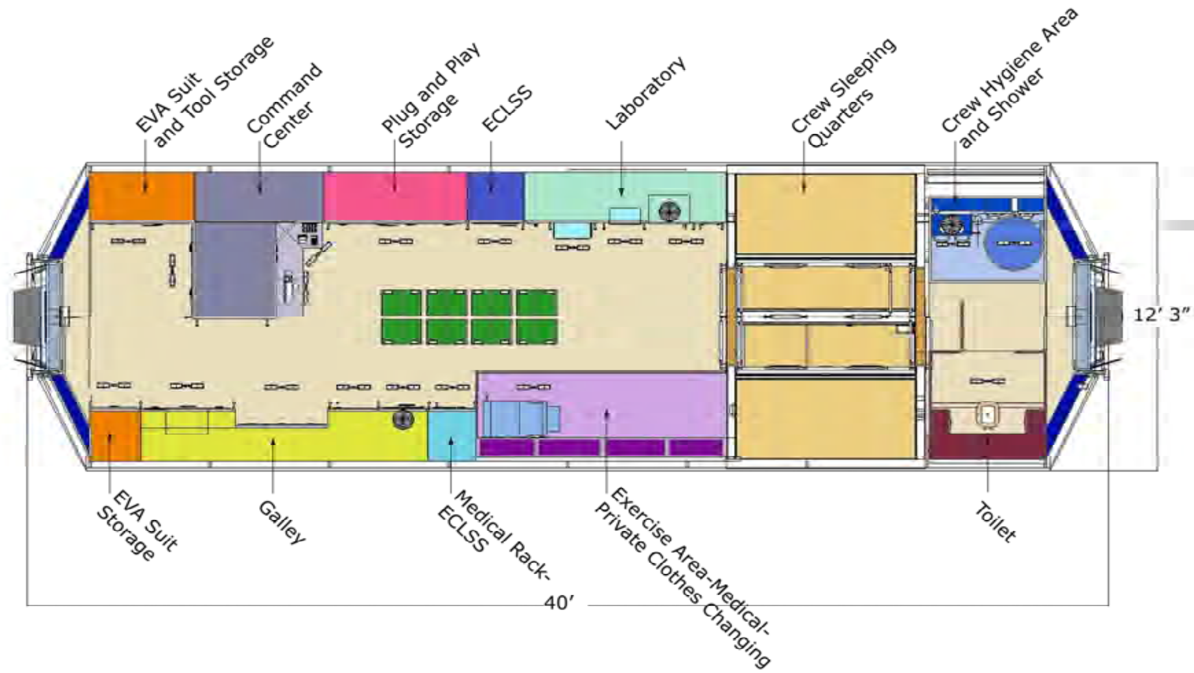
MLV Surface Element Examples





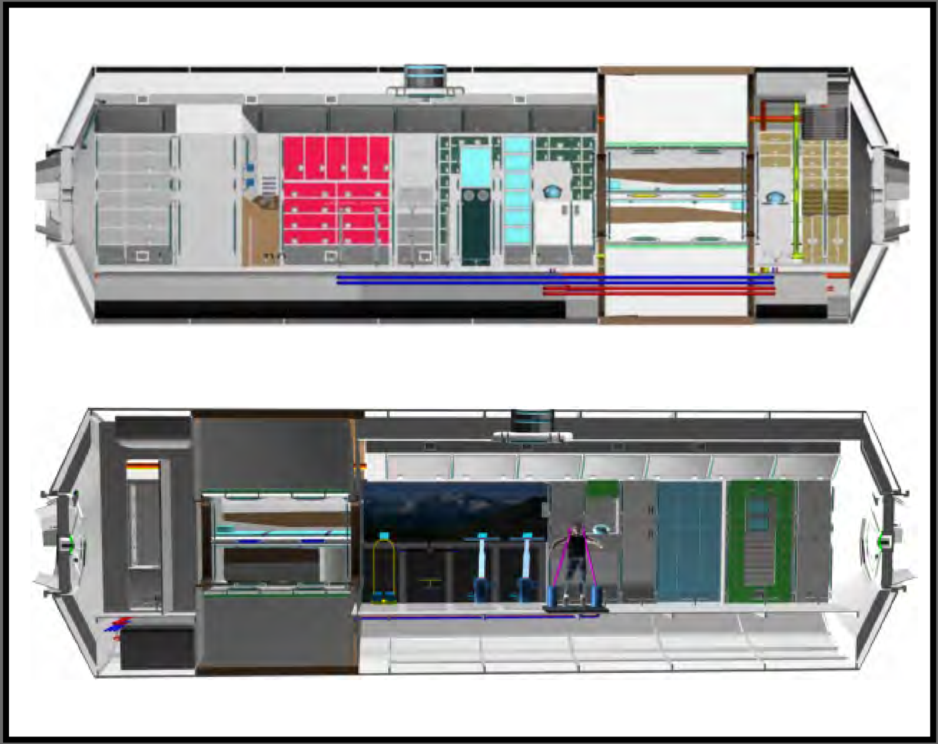


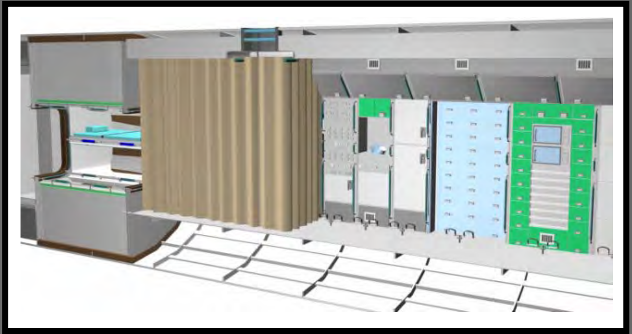
MLV Crew Transfer Mission

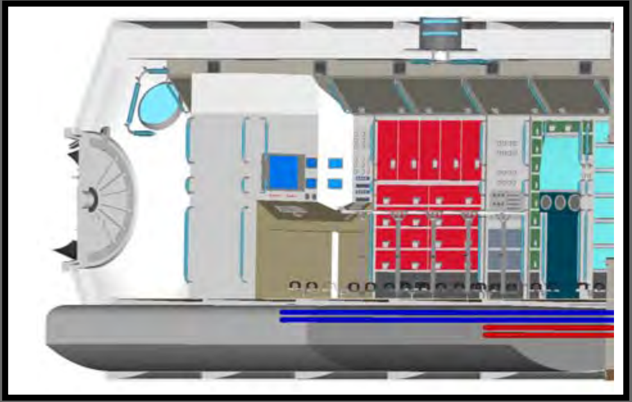


Mars Crew Transfer Module (CTM)

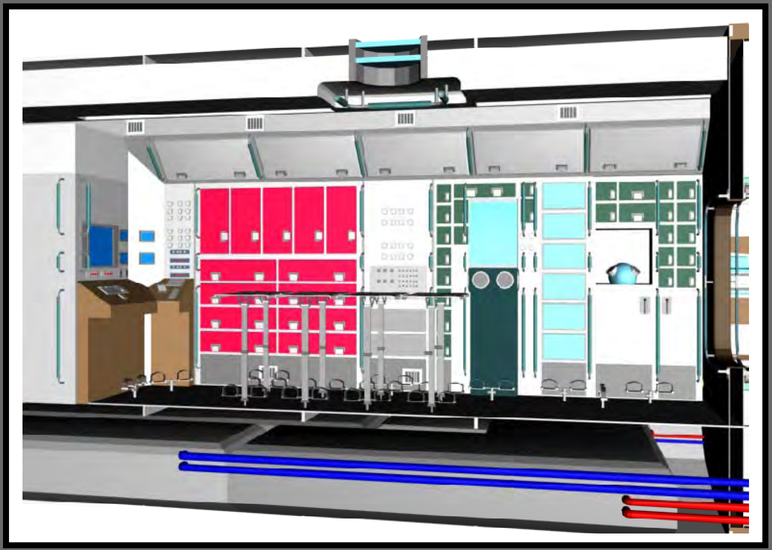
MLV Crew Transfer Module

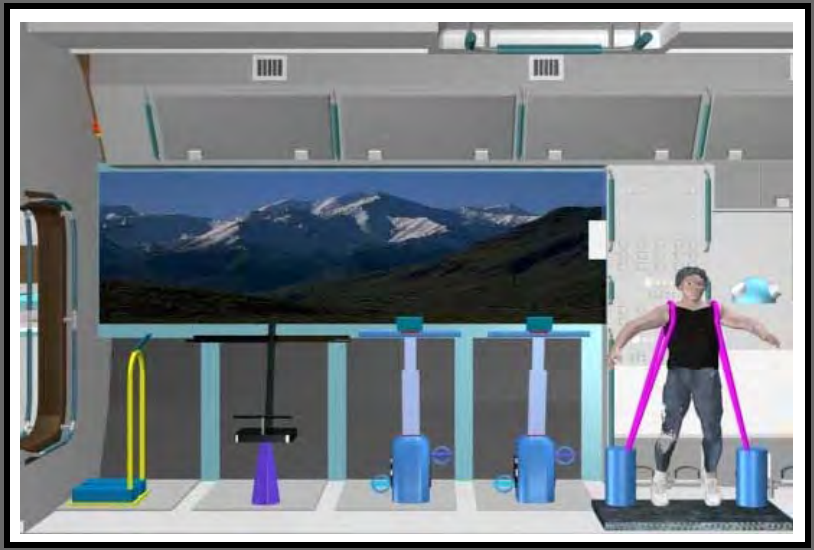






MLV Crew Transfer Module

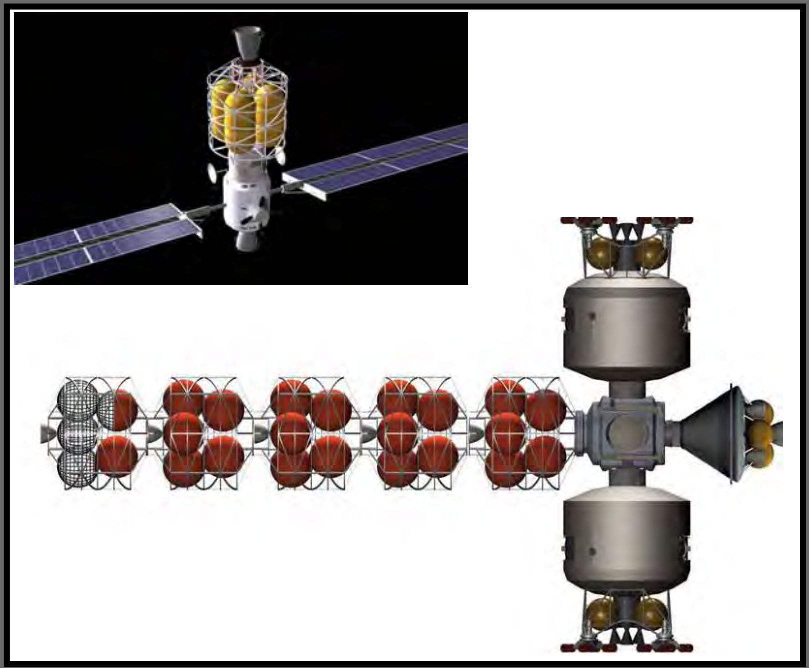




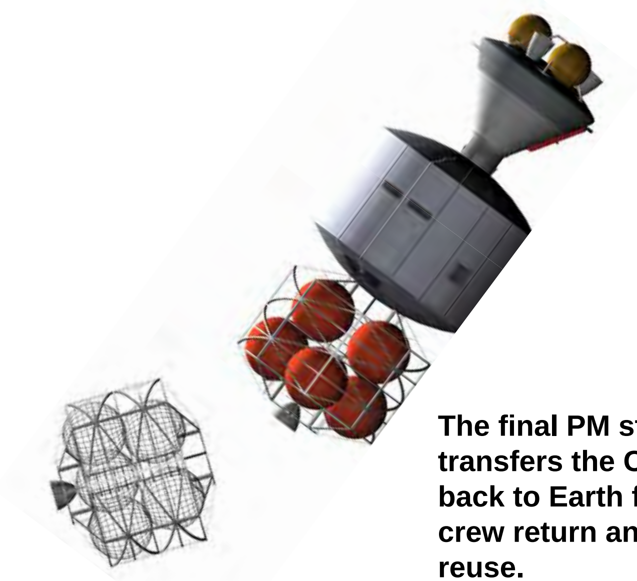
MLV Crew Transfer Module

SICSA has undertaken human exploration mission research and design studies based upon possible availabilities to deliver substantial payloads to LEO:

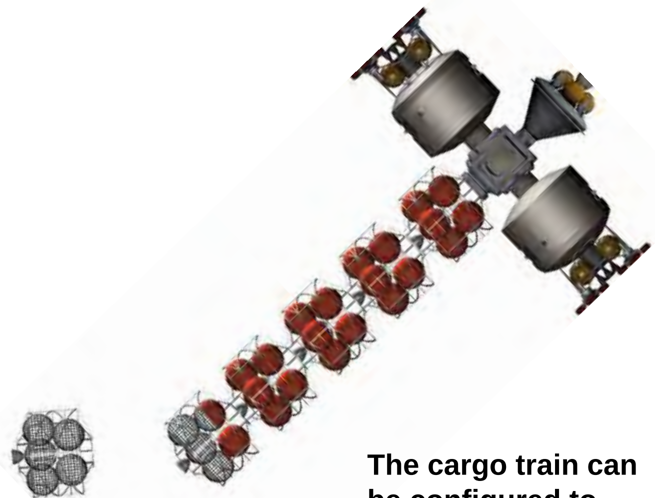
- Important guideline assumptions:
  - Launch capacities will be on the order of 100 MT, with payload shrouds at least 10 m diameter x 12m high.
  - Propulsion Modules (PMs) will be assembled together in a train and attached to a Crew Transfer Module (CTM) and/ or surface payloads in LEO via autonomous maneuvering vehicles.
  - While aerobraking at the Moon is not possible (there is no atmosphere), surface payload aerobraking into the thin atmosphere of Mars is considered.
  - The Crew Transfer Module (CTM) is reused as a Cycler in the Earth-Moon or Earth-Mars system.



HLV Element Assembly in LEO

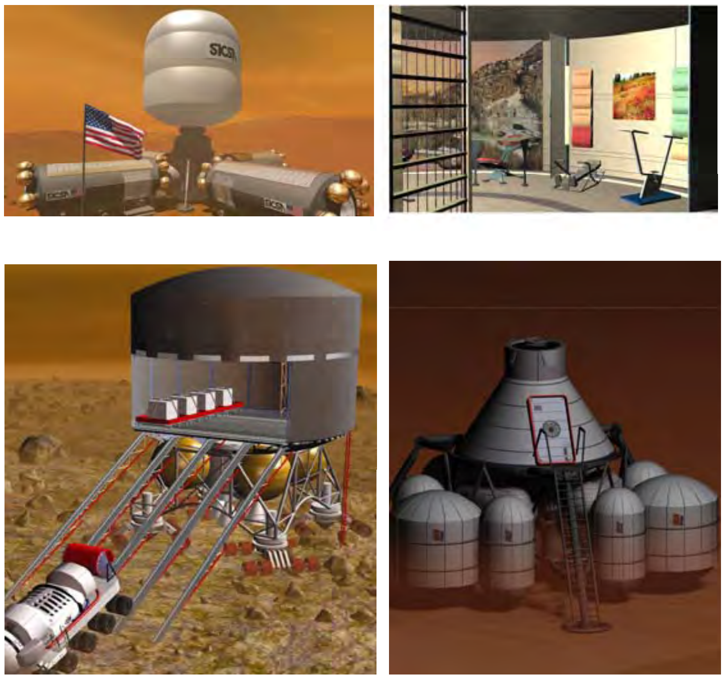


The final PM stage transfers the CTM back to Earth for crew return and reuse.

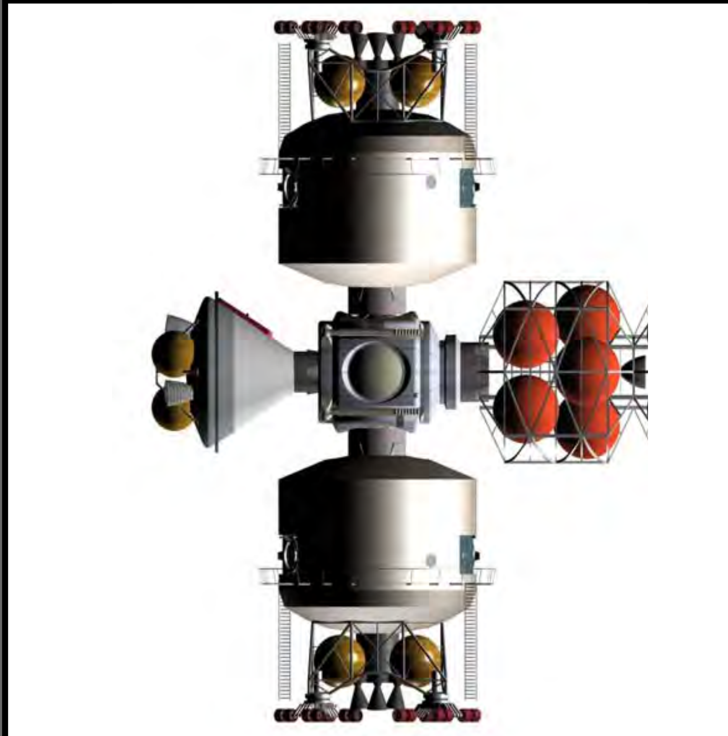


The cargo train can be configured to accommodate a variety of surface elements.

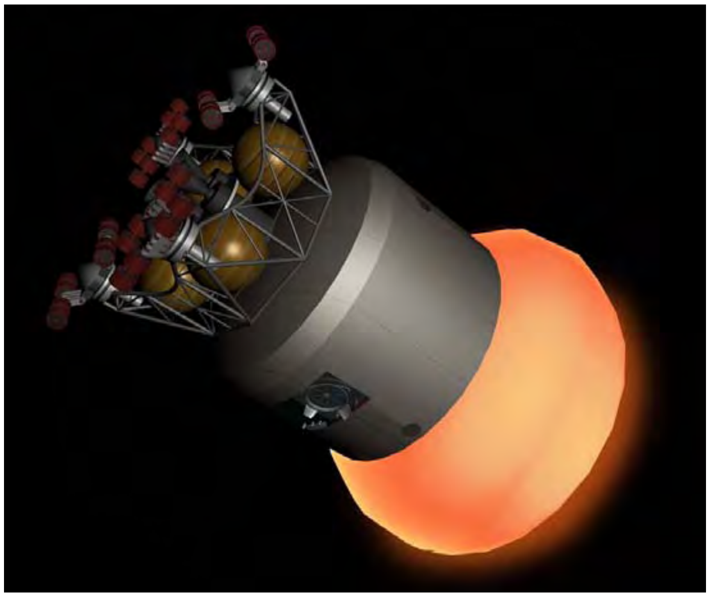
HLV Transfer Missions



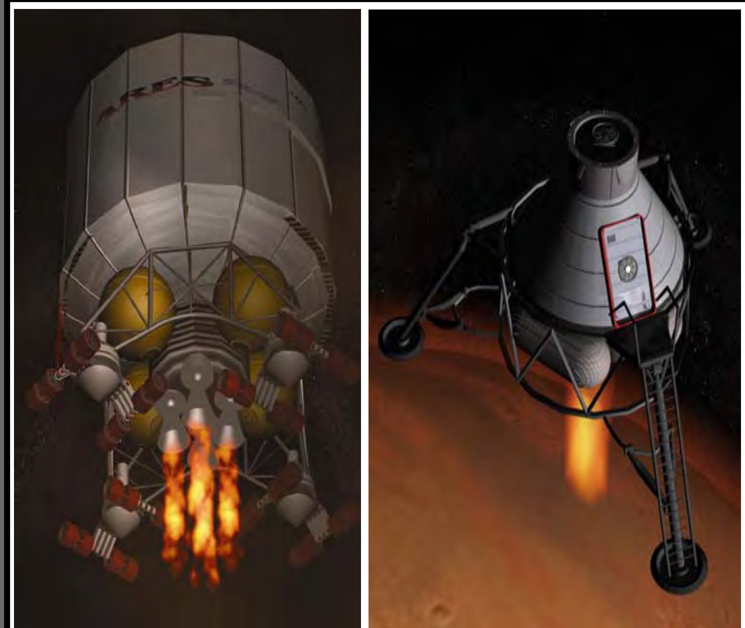
Surface Elements in Deployed Status



HLV Surface Element Examples



Inflatable aerobrakes made of existing high temperature-resistant materials might be used to slow into Mars orbit.

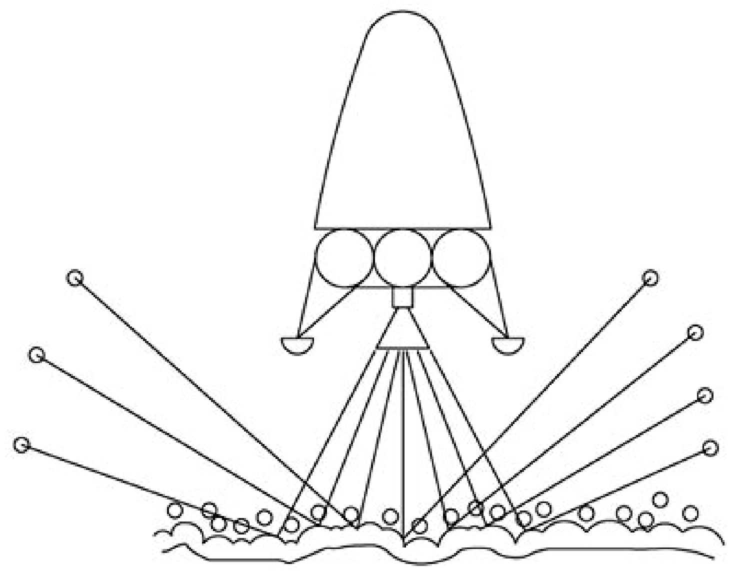


Propulsive Surface Landings

Mars Aerobraking & Landing

Surface conditions on the Moon and Mars pose special challenges for descent/ascent operations: • Global dust storms and local dust devils on Mars can obscure landing visibility.

- Electrostatic dust and extreme temperatures can damage ascent flight systems.
- Landing/ ascent thrusters can hurl surface rocks long distances to present hazards.
- Little or no atmosphere makes parachutes ineffective.
- Rocky and hilly surface terrain can damage or overturn landed payloads.
- Low-gravity conditions and little/ no atmospheric drag will cause rocket plumes to propel surface rocks on long ballistic trajectories.



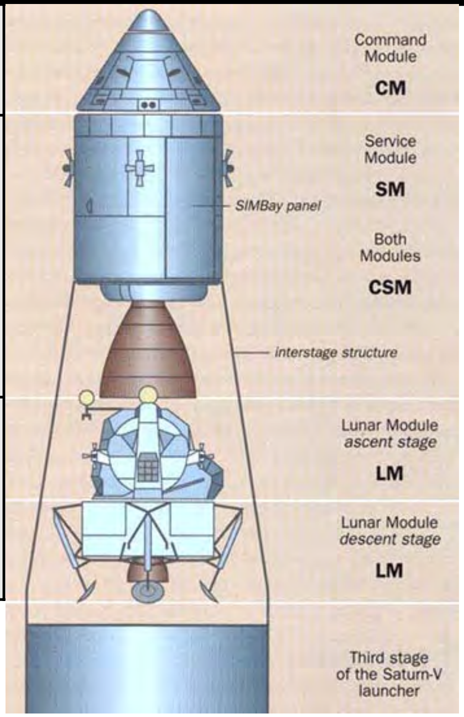
Low-gravity conditions and little/no atmospheric drag will cause rocket plumes to propel surface rocks on long ballistic trajectories.

Special Surface Challenges

TRANSPORTATION SYSTEMS

LUNAR/MARS LANDERS AND ASCENT VEHICLES

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE

The Command Module was slowed down by Earth atmospheric drag latter by parachutes for splash-down at sea.	Crew Transfer & Earth Reentry	
The Service Module provided propulsion & avionics to transfer the entire Command & Service Module Assembly & Lunar Modules to Lunar orbit, and propel the CM on an Earth return trajectory (then was jettisoned).	Trans-Lunar Injection & Lunar Orbit Departure propulsion	
The ascent stage interfaced with the top of the CM for crew transfer for landing & orbital rendezvous return. It was then jettisoned. The descent stage carried an attached surface rover.	Crew Surface Delivery & Orbit Return	
	Landing propulsion & Ascent Platform	

Apollo missions demonstrated capabilities to transport humans and cargo to an extraterrestrial body, land them on the surface, and return crews to Earth.

Apollo Vehicle Elements

The LM was a relatively fragile structure developed by Grumman to support 2 people for 3 days on the surface. About 2/3 of its total approximate 15,000 kg (32,600 lb) mass was landing and ascent engine propellant.

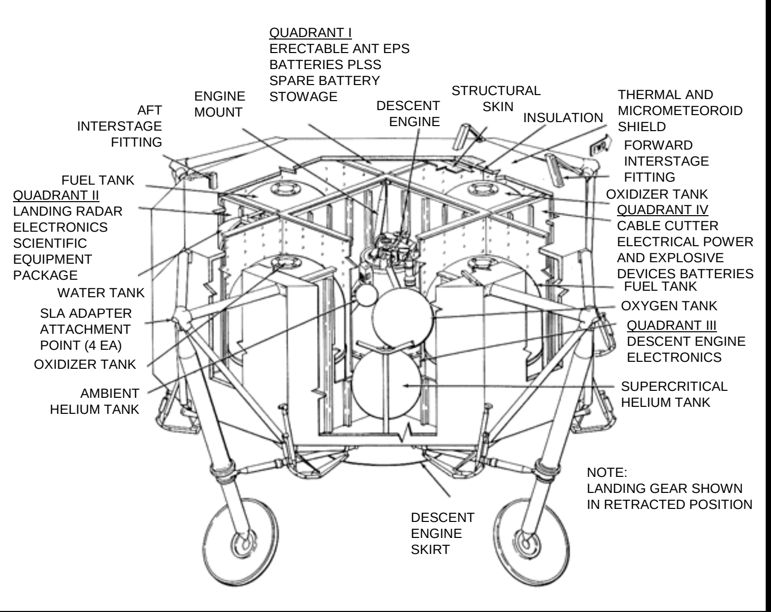
- **Ascent stage:**
  - Crew cabin volume, 6.65 m3 (235 ft3): height 3.76 m (12.34 ft) and diameter 4.2 m (13.78 ft).
  - Total mass, 4,670 kg (10,300 lb): two 19.27 kg water tanks, 11.3 kg of ethylene glycol-water coolant, and 2,353 kg of nitrogen tetroxide-hydrazine propellant.
  - Atmosphere, 100% oxygen at 33 kPa.
  - Power, four 400 Ah silver-zinc batteries.
- **Descent stage:**
  - Height, 3.2 m (10.5 ft) and diameter 4.2 m(13.8 ft).
  - Total mass, 10,334 kg (22,783 lb): one 151 kg water tank and 8,165 kg (18,000 lb) of nitrogen tetroxide-hydrazine propellant.
  - Power, two 296 Ah silver-zinc batteries.

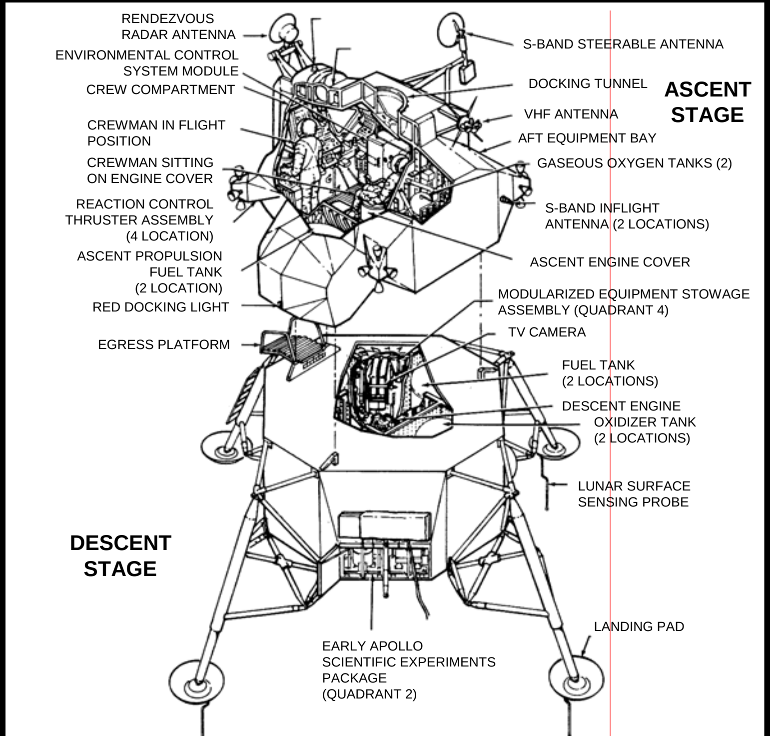
NASA

Apollo Lunar Lander (LM)

NASA

The LM descent stage contained the landing gear, landing radar antenna, descent rocket engine, propellant, and several cargo transfer compartments.



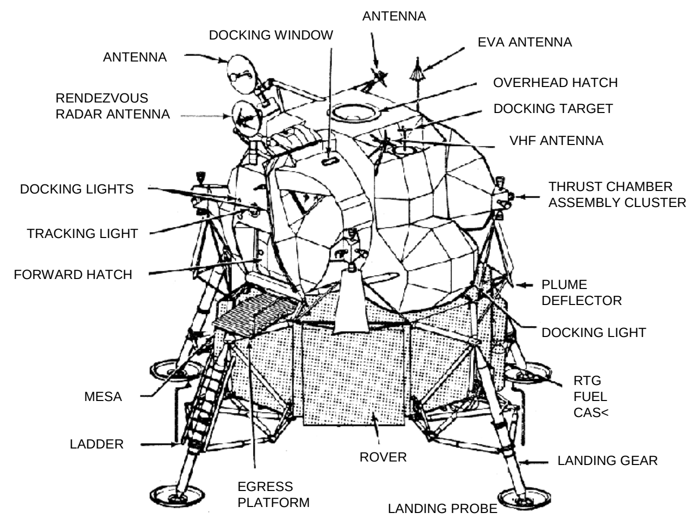


LUNAR MODULE CONFIGURATION FOR INITIAL LUNAR LANDING

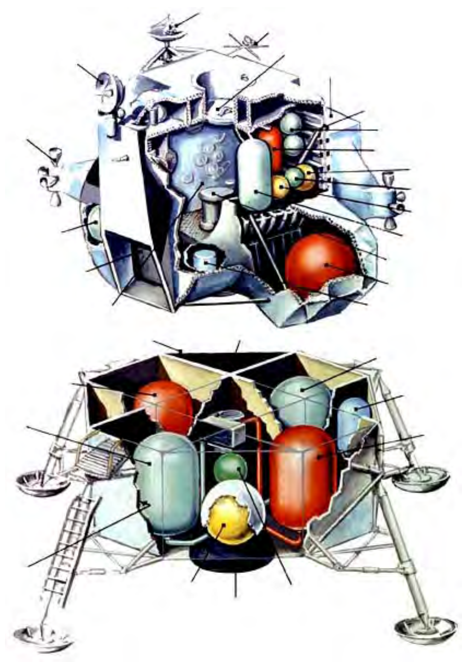
Apollo LM Stages & Components

NASA

The LM ascent stage contained instrument panels, an overhead hatch/ docking port, a forward hatch and orbital return systems.



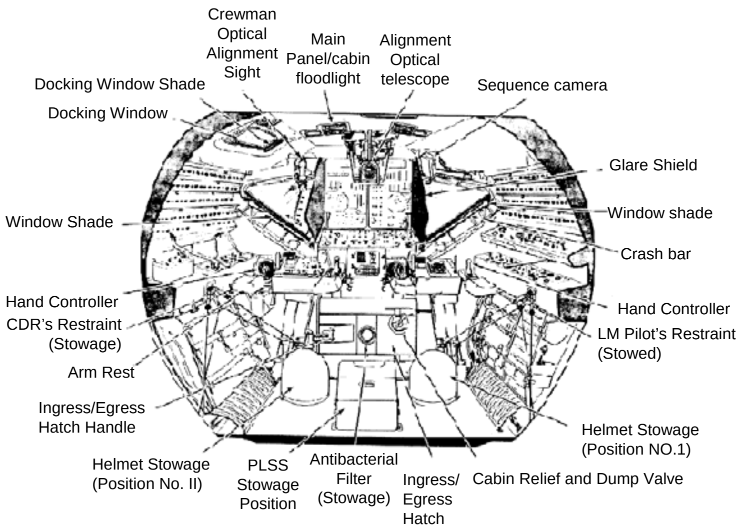
External Systems



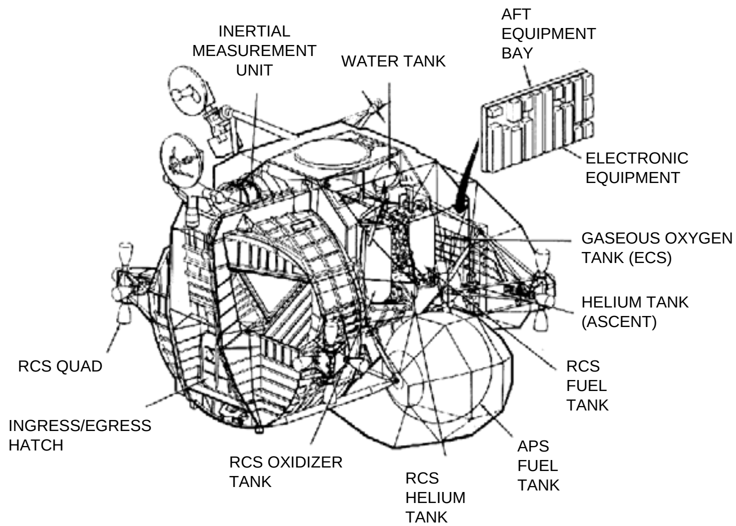
Internal Systems

Apollo LM Ascent & Descent Stages

NASA



Cabin Interior Looking Forward

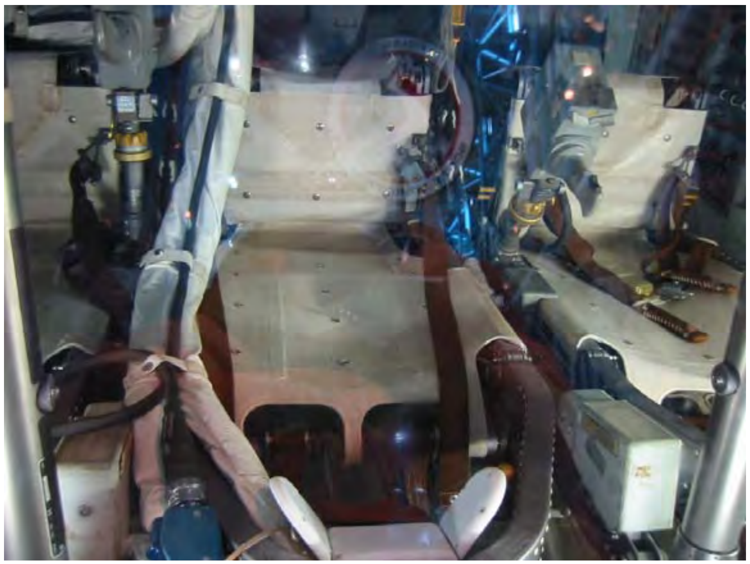


Components

Apollo LM Ascent Stage

TRANSPORTATION SYSTEMS

LUNAR/MARS LANDERS AND ASCENT VEHICLES



Crew Restraint Seat

Crew Area / Instruments

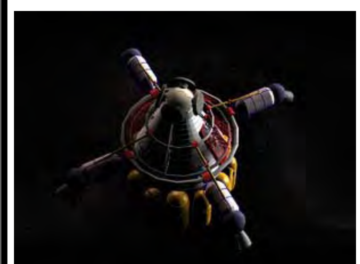
Apollo LM Ascent Cabin

Low-g and reduced (or absent) atmospheric drag conditions on the Moon or Mars can cause landing rockets to send surface rocks on long ballistic trajectories:

- A tethered lander system is proposed to keep thrusters high above the surface to reduce projectile risks to nearby facilities:
  - The tether system would be attached to vertically-oriented payloads prior to launch, and to horizontally-oriented payloads in LEO, to comply with payload shroud constraints.
  - Gimbaled lander engines would pivot down and propulsively slow the entire assembly to a hover position above the surface.
  - Tethers would deploy to soft land the payload and then release it.
  - Relieved of the payload mass, the lander would gain altitude, fly a safe distance from the drop site, and be sacrificed.

Tether System Attached

Tether Deploying



Engines Deployed

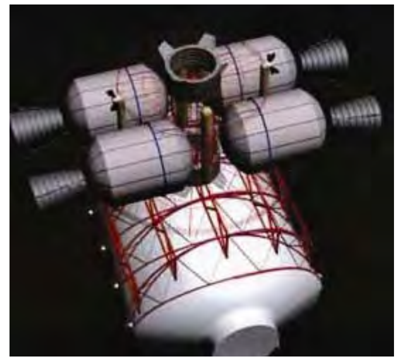
Payload Released

Crew Descent/Ascent Vehicle Landing

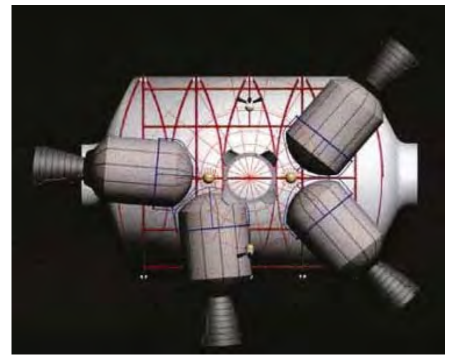
SICSA Tethered Lander Concept

Countermeasures are essential to compensate for loss of a thruster during a landing procedure:

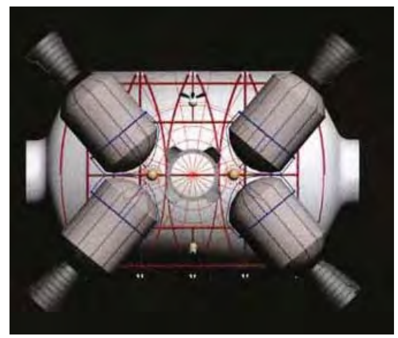
- The tethered system offers an important engine failure contingency advantage :
  - The gimbaled rocket footprint configuration can adapt to provide a better geometry to compensate for loss of any engine.
  - Placement of the engines at corners above the payloads provides a broad footprint to enhance stability under nominal and contingency circumstances.
  - The same general lander design can be applied for vertical and horizontal payloads.
  - Lander positions can be adjusted for varying payload center of gravity locations.



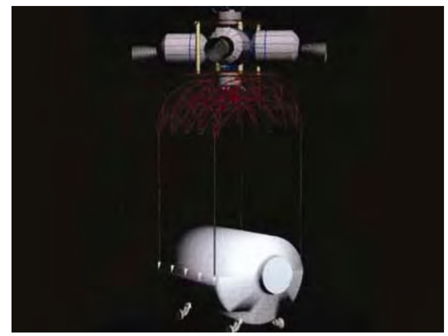
Engines in Closed Position



One Engine Out



Normal Landing Position



Tethered Deployment

Compensation for Engine failure

SICSA Tethered Lander Concept


Engines in Closed Configuration


Engines in Operational Configuration

SICSA Tethered Lander Concept

Tethered Deployment of Horizontal Module

Tethers Released and Lander Sacrificed

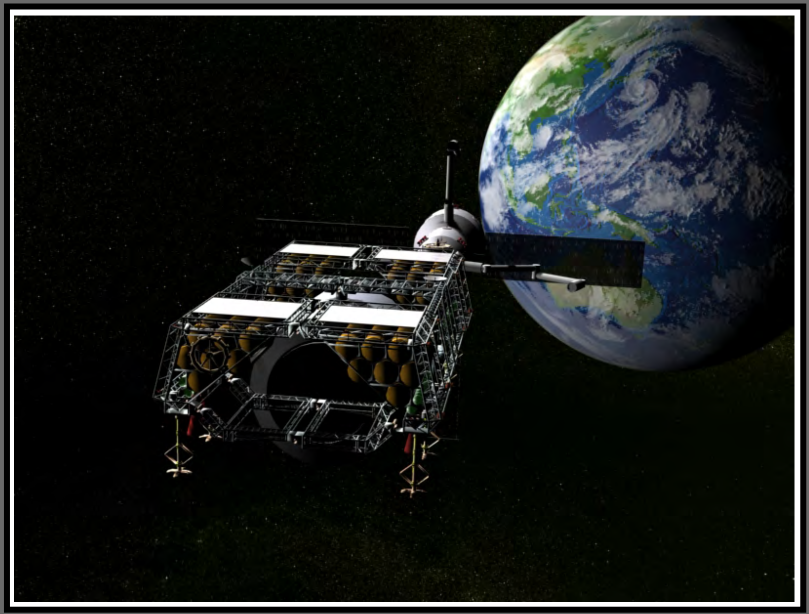
SICSA Tethered Lander Concept

TRANSPORTATION SYSTEMS

LUNAR/MARS LANDERS
AND ASCENT VEHICLES

SICSA's Inter-Lunar Cargo Transfer System (ICLTS) proposed an Earth to lunar surface transportation strategy that utilizes Shuttle-Derived Launch Vehicles, cycling Earth to lunar orbit and lunar orbit to surface vehicles, and multi-purpose propulsion elements:

- The ICLTS approach potentially affords several basic advantages:
  - Utilization of developed and proven vehicle systems and technologies.
  - Opportunities to stage mission segment tests and evaluations prior to full-up operations.
  - Salvaging/ reuse of large system elements to minimize production costs.
  - Reduced launch requirements/ costs due to reusable orbital assets.
  - Propellant economies resulting from efficient orbital mass transfers between vehicles.



SICSA's Inter-Lunar Cargo Transfer System

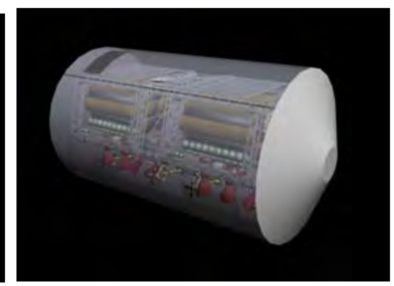
The Inter-Lunar Cargo Transfer System is comprised of four basic types of elements:

- **The Upper Transfer Stage (UTS):**
  - It functions much like the Saturn IV B upper stage that propelled the Apollo Command and Service Modules through Trans-Lunar Injection burns.
- **The Lunar Transfer Vehicle (LTV):**
  - It provides transportation between Earth orbit and lunar orbit on a cyclical round-trip basis.
- **The Lunar Surface Transfer Vehicle (LSTV):**
  - It operates in a manner similar to a harbor tug to cyclically move cargo between the lunar surface and lunar orbit.
- **The Lunar Cargo Element (LCE):**
  - This is a logistics module to carry cargo from Earth to the lunar surface.

Upper Transfer Stage

Lunar Transfer Vehicle

Lunar Surface Transfer Vehicle



Lunar Cargo Element

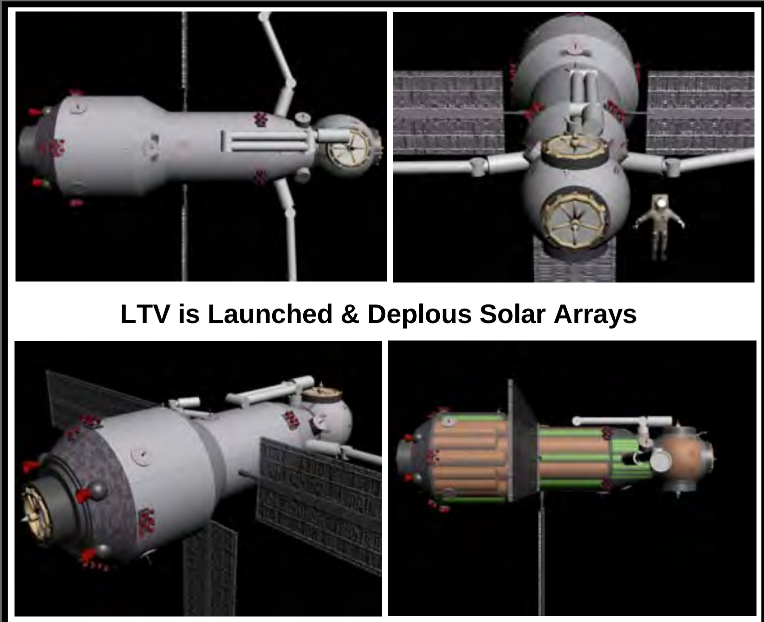
Primary ILCTS Elements

Step 1: LTV Launch to LEO:

- To initiate the mission sequence, a Shuttle-Derived Heavy Lift Vehicle (SDHLV) with a 150 MT capacity launches a Lunar Transfer Vehicle (LTV) to a 200 km circular Earth orbit LEO:
  - The LTV unfurls three 14 m (45.9 ft) long solar arrays.
  - The dry mass of the LTV is approximately 9 MT, and its launch mass is about 111.5 MT (including 101.5 MTof propellant).
  - The LTV has 29 propellant tanks, with a total volume of 74.3 m3.

Step 2: UTS Launch to LEO

- A second HLV launches a fully fueled Upper Transfer Stage (UTS) to power the LTV through a TL1 burn:
  - Fully loaded, the UTS will carry 135 MT of propellant.
  - Future missions (after the first) will carry only as much fuel as payloads require.



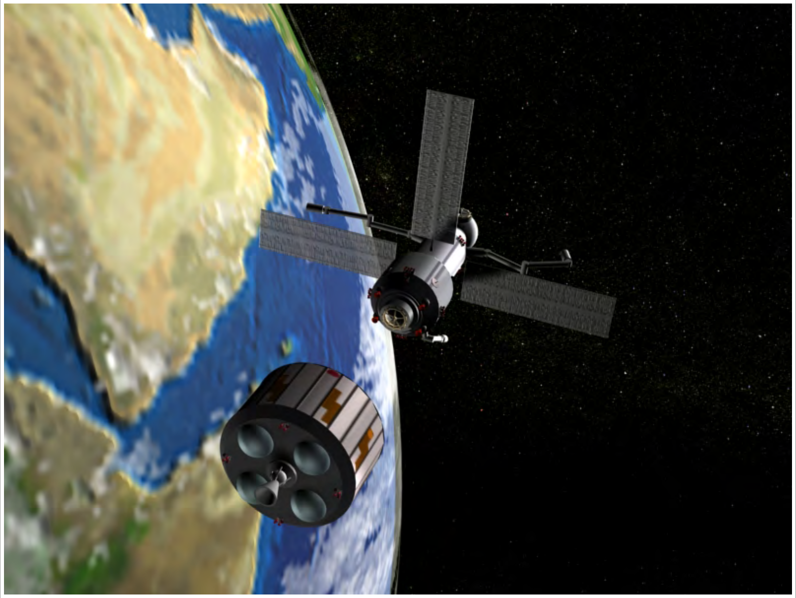
LTV is Launched & Deplous Solar Arrays

LTV in LEO | LTV & UTS Propellant

ICLTS Mission Sequence

Step 3: LTV Rendezvous/ Docking with UTS:

- The LTV undertakes an Automated Rendezvous and Docking (ARAD) maneuver with the UTS, connecting to the LTV's aft docking port:
  - The UTS and LTV now have a combined propellant mass of about 213 MT.
  - The LTV is essentially an internally gutted Russian ISS Service Module that contains one aft and one forward docking mechanism which uses the current ISS APAS design.
  - Rendezvous and docking is controlled by a Russian KURS guidance and control system.
  - Forward and aft docking rings provide a fuel and power transfer capability as well as command and control connectivity between the LTV, UTS and other elements.



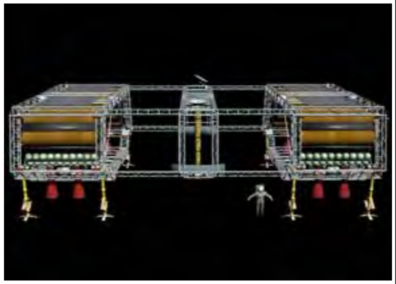
LTV Rendezvous with UTS in LEO

ICLTS Mission Sequence

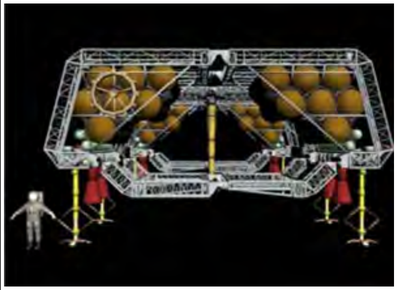
Step 4: LSTV Launch to Rendezvous with LTV/ UTS:

- About one month following launch, orbital insertion and docking of the LTV/ UTS, a third HLV lifts a LSTV into a rendezvous orbit:
  - The LSTV is launched in a "folded" (shortened) configuration which is about 9 m wide and 12.5 m long.
  - Total launch weight is 150 MT, of which about 8 MT is dry vehicle mass and the rest is propellant.
  - The propellant is stored in 28 cylindrical tanks with a total capacity of 88 m3.
- Following LEO insertion, the LSTV extends its H-frame structure and initiates autonomous flight control.
  - The LSTV and LTV/ UTS dock together and undergo power, data and control connectivity checkouts to ensure that all systems are "go" for lunar transfer.

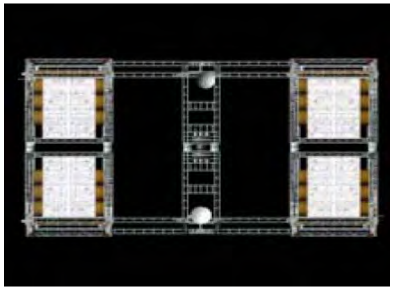
Launch Configuration



Extended Configuration



End View (Closed)

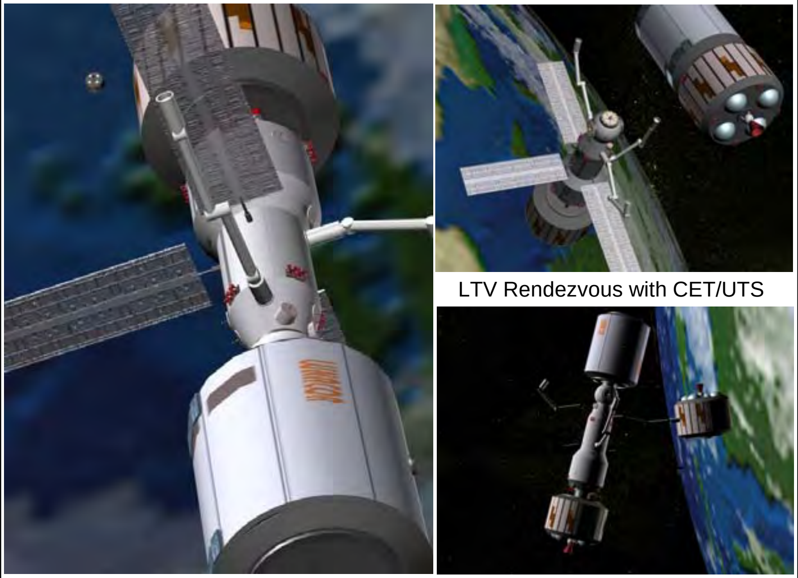


H-Frame Structure

ICLTS Mission Sequence

Step 5: LCE/ UTS Launch and LTE Berthing to LTV:

- A fourth HTV (including an Upper Transfer Stage) (UTS) launch a Lunar Cargo Element (LCE) to LEO within proximity of the co-orbiting LSTV and the LTV/ UTS-LSTV mated stack:
  - The LTV/ UTS then undock from the LSTV and conduct a rendezvous with the LCE.
  - The LTV grapples the LCE with its RMS.
  - The LTV robotically berths the LCE to its forward docking port, and then separates the LCE's UTS.
  - The UTS (now empty of fuel) is jettisoned and deorbits to reenter the Earth's atmosphere.



LTV Rendezvous with CET/UTS

CET Berthea to the LTV After Docking Port

LTV-CTE Attached, UTS Deorbits

CTE Rendezvous and Attachment

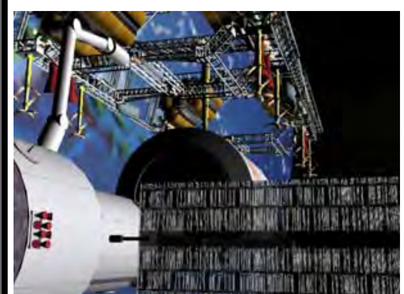
ICLTS Mission Sequence

Step 6: Element Assembly for TLI Departure:

- The LTV/ UTS and berthed CTE rendezvous with the LSTV to commence final assembly is transferred to the LSTV:
  - First, the LTV maneuvers into proximity with the LSTV along its long axis and several meters below.
  - An upper LTV RMS grapples the LSTV frame (which is extended) and secures it for CTE transfer.
  - Side RMS arms on the LTV robotically separate the CTE and center it under the LSTV's open cargo holding fixture.
  - A LSTV winch mechanism deploys a grapple hook that attaches to a CTE Universal Transfer Capture Port (UTCP) that provides power and 2-way data connectivity between the LSTV and LCE.
  - The LTV then docks with the aft LSTV port, and its UTS initiates an LTI burn.

LTV-LSTV Rendezvous

CTE Positioned with LSTV



CTE Grappled by LSTV

Final LTI Configuration

Preparations for LTI Burn

ICLTS Mission Sequence

Step 7: Transfer to Lunar Orbit:

- The complete assembly of 4 elements departs LEO for lunar orbit:
  - The UTS makes a mid-course correction burn, and fires again to slow the assembly for maneuvering descent into a circular Low Lunar Orbit (LLO) of approximately 100 km (62 mile) altitude.
  - Any remaining fuel on the UTS and LTV is transferred to the LSTV, and it separates from the stack and is sacrificed.
  - The LTV transfers its remaining fuel to the LSTV for its powered descent to the surface, and the LSTV with its CTE payload undock from the LTV.
  - The LTV remains in LLO to provide communications and visual relay monitoring support during LSTV surface descent.

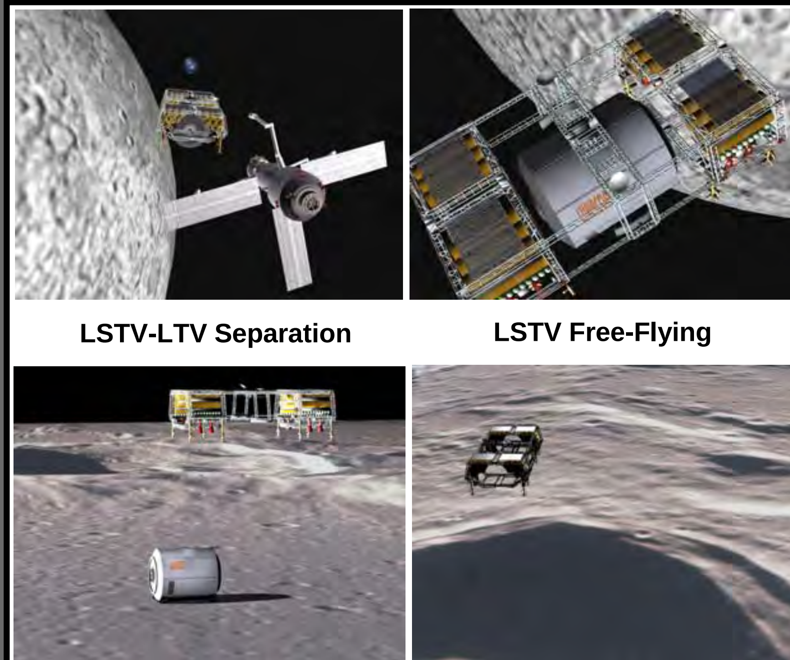
UTS Separation from Stack

Lunar Approach and Orbit Entry

Transfer to Lunar Orbit

Step 8: CTE Surface Placement/ LSTV Reuse

- LSTV thrusters commence a lunar deorbit burn to an elliptical 100 x 17.5 km transfer orbit followed by a 100 m powered descent to the surface:
  - A transponder beacon placed on the surface during a precursor mission will guide the landing.
  - The LSTV will hover at a short distance above the surface while the CTE is released, and can alternatively land nearby, or immediately return to a lunar parking orbit to rendezvous with the LTV.
  - Subsequent cargo transfer missions will deliver UTSs to power LTV cyclers that remain in the Earth-Moon system and resupply LSTV landing/ ascent propellant.



LSTV-LTV Separation | LSTV Free-Flying

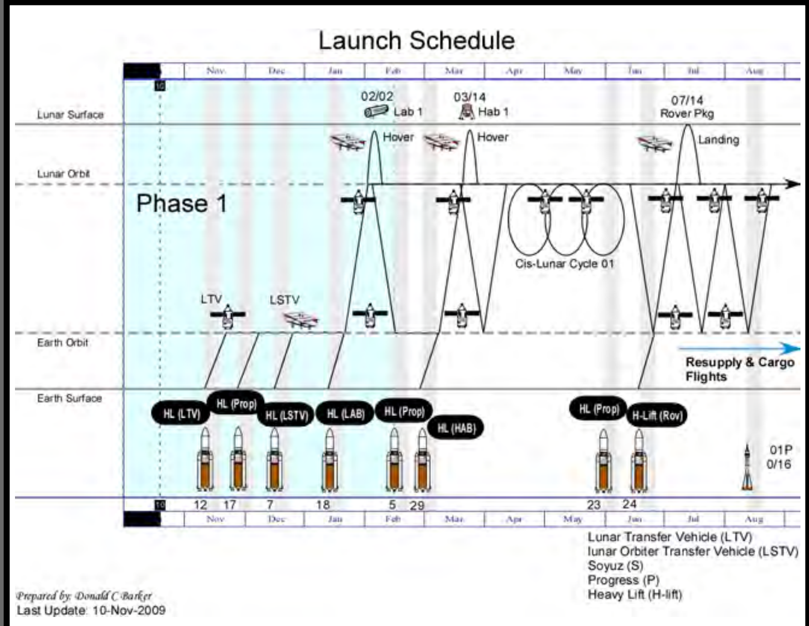
LCE on the Surface | LSTV Return to Orbit

Surface Landing and Orbital Return

ICLTS Mission Sequence

Use of dedicated propellant resupply flights as part of a cargo vehicle staging plan can potentially offer large launch and fuel economies:

- Applying Useful Cargo Mass (UCM) criteria, the ICLTS approach is expected to be capable of delivering much more infrastructure and consumables to the Moon with comparable launches than Apollo missions:
  - The Apollo LM with a total 15 MT mass prior to lunar de-orbit burns delivered about 2 MT of Useful Cargo Mass to the surface (slightly more than 13 percent).
  - A 30 MT ICLTS surface cargo delivery mass would require 8 Apollo Saturn V LM launches.
  - The ICLTS might accomplish this with 4 equivalent launches initially, and with only 2 for subsequent surface delivery missions.



Example Launch/Transfer Schedule

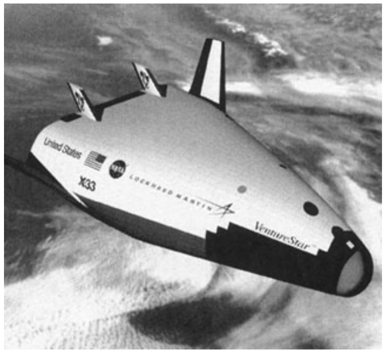
ICLTS Mission Economies

A vehicle's aerodynamic characteristics are of primary importance when entry into an atmosphere is required:

- Different conditions demand different approaches:
  - Aerodynamic designs do not apply for slowing vehicles into lunar orbits or landings on the surface because there is no atmosphere, but are needed for these purposes during LEO reentries and Earth landings.
  - Vehicles approaching Mars can use its thin atmosphere to provide aerodynamic braking that conserves propulsive landing rockets rather than depend upon parachutes or parafoils.
  - Aerobrakes and blunt capsules can be used for Earth atmosphere reentries, and parachutes and parafoils can be used for gliding descents and landings, but parachutes/ parafoils on winged vehicles will impair controlled gliding.

Inflatable Aerocapture/ Aerobrake Concept


Apollo Command Module "Gumdrop" Shape



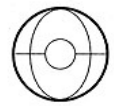
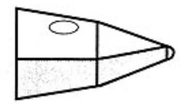
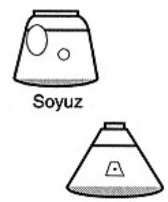
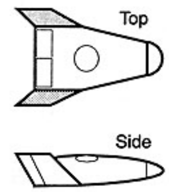
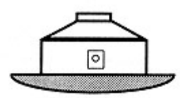
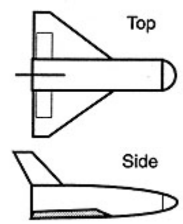
NASA/Lockheed-Martin Proposed X-33 Venture Star


Apollo Command Module Parachute Landing

General Types

A vehicle's Lift/ Drag (L/D) ratio is determined by its shape, center of gravity and ability to mechanically shift the lift vector's direction.

- L/D reflects maneuverability in an atmosphere and indicates deceleration forces and heating that will occur:
  - Vehicles with low L/Ds fly nearly ballistically and experience high deceleration forces and relatively higher peak heating.
  - Vehicles with high L/Ds can maintain better trajectory control to minimize entry deceleration and maneuver left and right of their ground tracks (cross range).
  - While high L/D vehicles can control trajectories and peak heating, their longer flight times in the atmosphere can still produce high total heat loads.
  - High L/D vehicles tend to have leading edges which concentrate heat more than blunter shapes.

Diagram	Description	Diagram	Description
	Sphere • $L/D = 0$ • Not maneuverable • Moderate volumetric efficiency • Example: Russian Vostok		Biconic • $L/D = 0.5–0.7$ • Moderately maneuverable • Excellent volumetric efficiency • Example: Mars lander concept
 Soyuz Apollo	Blunt Cone Capsule • $L/D = 0.2–0.4$ • Slightly maneuverable • Good volumetric efficiency • Examples: Apollo, Gemini, Mercury, Soyuz	 Top Side	Lifting Body • $L/D = 1.0–1.4$ • Highly maneuverable • Moderate volumetric efficiency • Example: X-24, HL-10
	Heatshield with Afterbody • $L/D = 0.2–0.4$ • Slightly maneuverable • Good volumetric efficiency • Example: Viking Mars probe	 Top Side	Winged Body • $L/D = 1.4$ • Highly maneuverable • Good volumetric efficiency • Example: Space Shuttle, Hermes

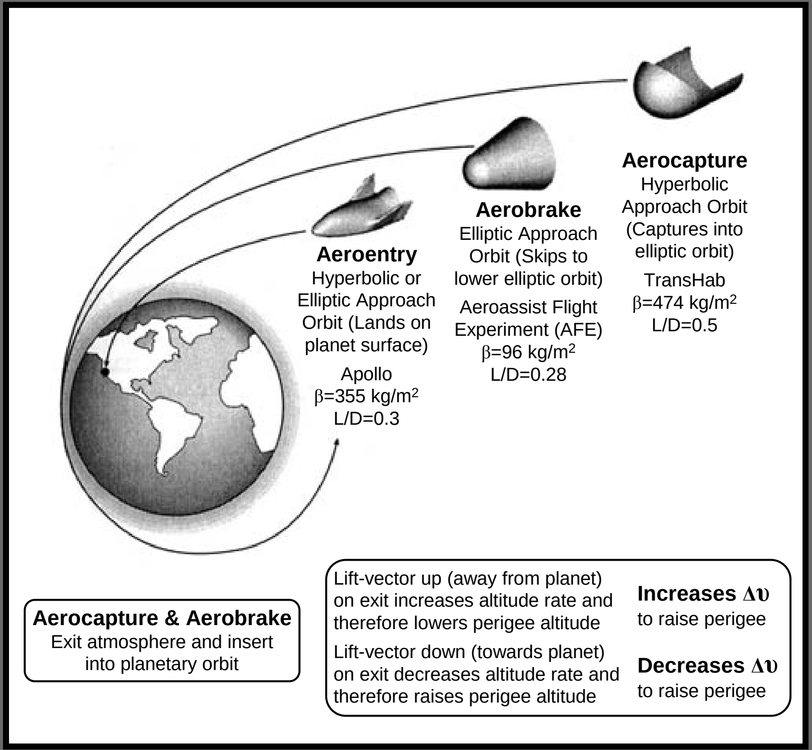
Aerodynamic Vehicle Types

Lift-to-Drag (L/D) Influences

Landing Mode Options	Characteristics	Maneuverability	Suitable Landing Sites	Options for Attenuating Impact
Propulsive	Vertical descent	Moderate	Field or pad	Struts, shock absorbers
Ballistic parachute	Vertical descent with wind drift	None	Field or water	Shock absorbers, air bags, retro-rockets
Lifting parachute	Glide with some directional control	Small	Field or water	Shock absorbers, air bags, retro-rockets
Parafoil	Glide with directional control	Moderate	Field or water	Aerodynamic flare, shock absorbers, air bag, retro-rockets
Deployable wing	Glide with directional control	Moderate to high	Runway	Aerodynamic flare, wheels, shock absorbers
Lifting body	Glide with directional control	High	Runway	Aerodynamic flare, wheels, shock absorbers
Wings	Glide with directional control	High	Runway	Aerodynamic flare, wheels, shock absorbers

Atmosphere can be used to slow vehicles using 3 scenarios:

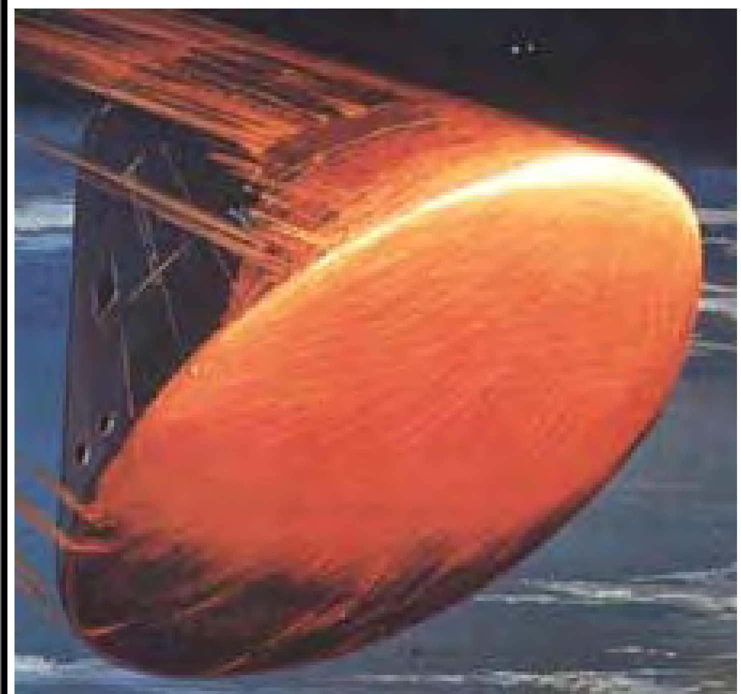
- **Aerocapture-** uses a planet's atmosphere to decelerate into an elliptical parking orbit:
  - Requires a small propulsiveDelta-V for post-aerocapture periapsis recovery.
  - Aeroshell structure may be 15%-20% of Mars spacecraft's pre-TLI mass (compared with 60%-80% for propulsive deceleration system approach).
  - Mass savings of aerocapture over propulsive increase with spacecraft mass increases.
- **Aerobraking**- uses atmosphere to decelerate from an elliptical parking orbit to a lower orbit:
  - Unlike aerocapture which must capture a planetary orbit in a single pass, this maneuver can be accomplished in successive deceleration passes.
- **Aeroentry-**uses a planet's atmosphere to decelerate from a hyperbolic approach or parking orbit to the surface:
  - Is principally used for unmanned robotic missions that don't require landing accuracy.



Atmospheric Deceleration Scenarios

The outer heat shield shell of the Command Module was designed to withstand 3,000° C temperatures during Earth reentry. Constructed of a phenolic/ epoxy resin reinforced with wire mesh, it rejected heat by melting/ boiling away the outer surface through a process called “ablation”.

Apollo 17 Command Module Following Return

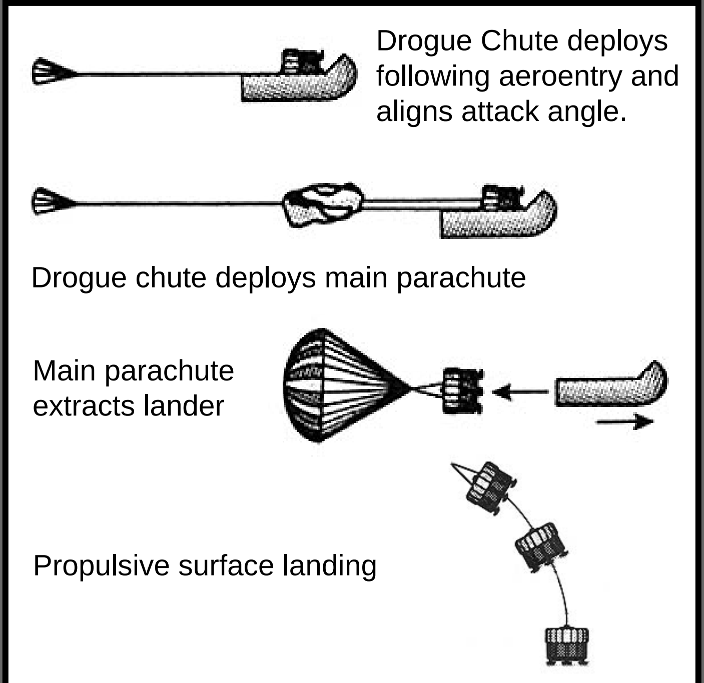


Apollo Command Module

Apollo Command Module

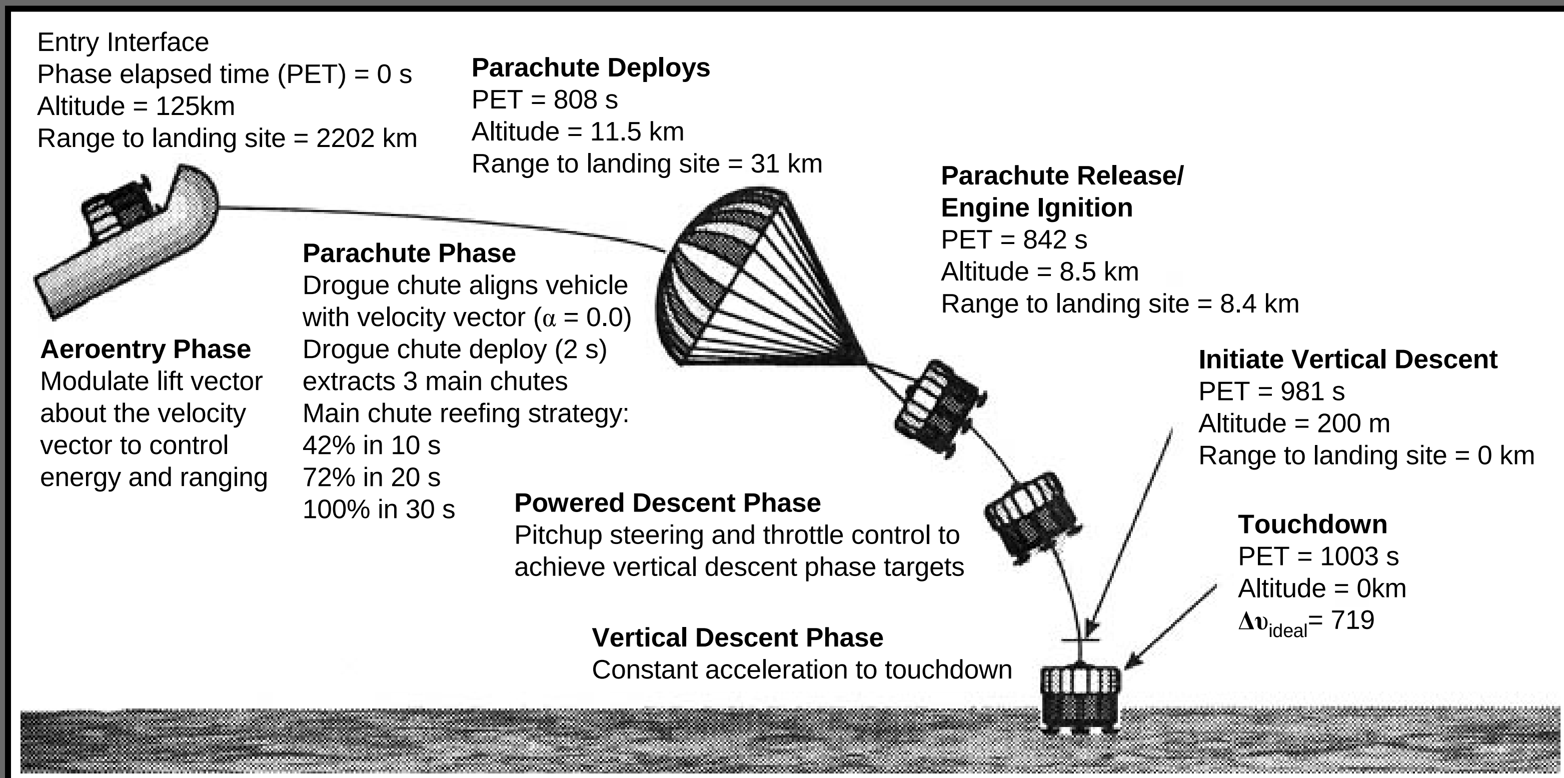
Parachutes might provide benefits on Mars despite the thin atmosphere (1% of Earth's):

- Assuming a typical Mars flight profile, a vehicle might commence aerobraking at an altitude of 125 km and velocity of 3,650 m/s:
  - An aeroshell uses its L/D to reduce the altitude to 11.5 km, at which point a drogue chute is deployed.
  - The drogue chute provides adequate drag to align the vehicle with the velocity vector (zero angle of attack) and to deploy a main chute that extracts a lander from the aeroshell.
  - The main parachute slows the lander to a velocity of about 200 m/s (maximum for engine ignition) and altitude of 8.5 km (minimum required for ample sensor field of view to select a safe landing site and provide navigation alignment distance).
  - After descent engines ignite, the parachute is jettisoned.



Parachute – Assisted Landing

Mars Landing Scenario



Mars Landing Scenario

Parachutes and parafoils offer efficient means to return spacecraft vehicles and crews to the Earth's surface following atmospheric reentry:

- Parachutes have been successfully used on many occasions to land US and Soviet/ Russian capsules:
  - Mercury, Gemini and Apollo vehicles used ballistic parachutes to land in water.
  - The Soyuz capsules: landed on the surface using ballistic parachutes assisted by small retro-rockets that fired just prior to touchdown.
- Parafoils afford a high L/D for hypersonic and cross-range capabilities and decreased entry acceleration and thermal loads:
  - They avoid many complexities associated with winged vehicle landings, and can descend on equilibrium glide trajectories to a site.
  - Apollo-class vehicles on these trajectories wouldn't require water landings/ recoveries or retro-rockets for surface landings.

Parafoil Landing an Experimental Crew Return Vehicle Following a Test Flight

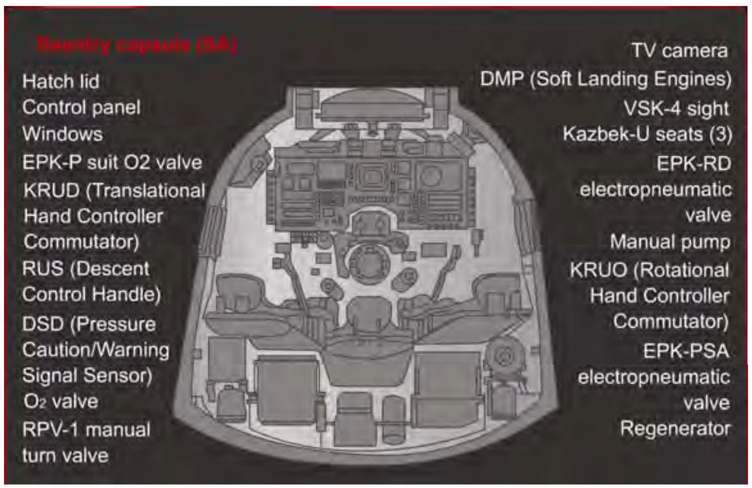
Parachute & Parafoil Earth Landings

NASA

Parachute Descent of Apollo 11 Command Module

Apollo 1 Splashdown in the Pacific SW of Hawaii

Apollo Ballistic/Parachute Return

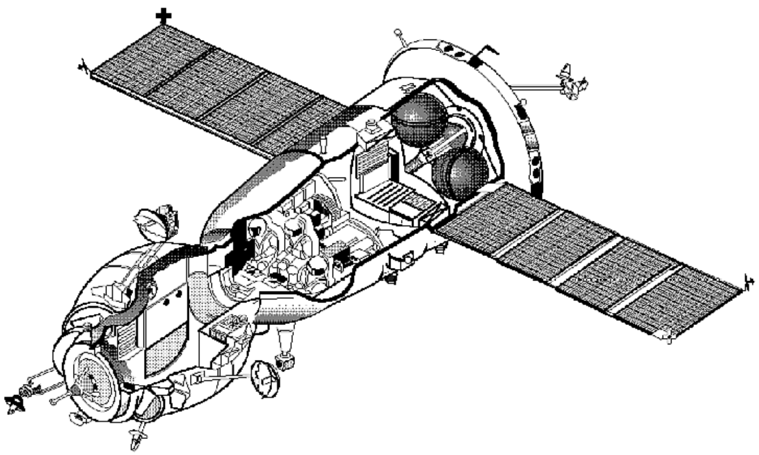


Russian Soyuz Descent Module

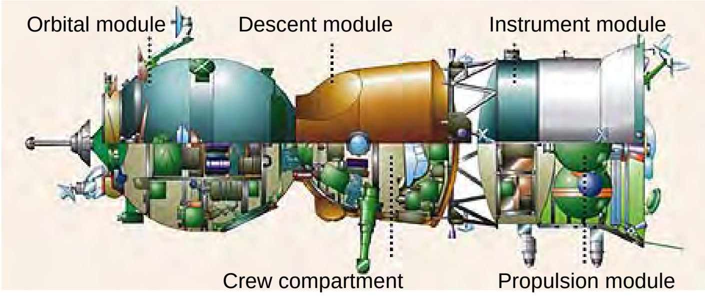
Soyuz Module Surface Landing

Soyuz Ballistic/Parachute Return

THE CAMBRIDGE ENCYCLOPEDIA OF SPACE



Soyuz Spacecraft Layout



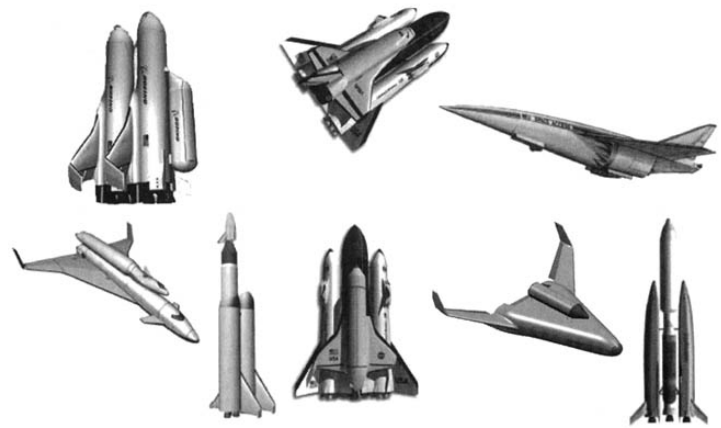
Launch mass (without shroud and launch escape system)	7.1 tons
Descent module	2.9 tons
Orbital module	1.3 tons
Instrumentation/Propulsion module	2.6 tons
Delivered payload (with three crewmembers)	30 kg
Returned payload	50 kg
Length	7 meters
Maximum diameter	2.72 meters
Diameter of habitable modules	2.2 meters
Soyuz TM solar array span	10.7 meters

Soyuz Elements & Statistics

Soyuz Descent Module/Spacecraft

Dramatic launch vehicle improvements will be needed to substantially lower the cost of payload deliveries to LEO:

- A variety of technical possibilities and approaches to accomplish them has been and are being investigated:
  - Advanced propulsion systems with higher specific impulse engines.
  - New lighter, stronger materials to improve fuel/ structure mass fractions to increase propulsion efficiency.
  - Possible Single-Stage-To-Orbit (SSTO) vehicles requiring even more exotic structural/ propulsive technologies to further reduce costs of RLVs and other systems.
  - New generations of RLVs that are more robust and maintainable than the Shuttle with turn-around and re-fly accomplished in days (like airliners) rather than months.

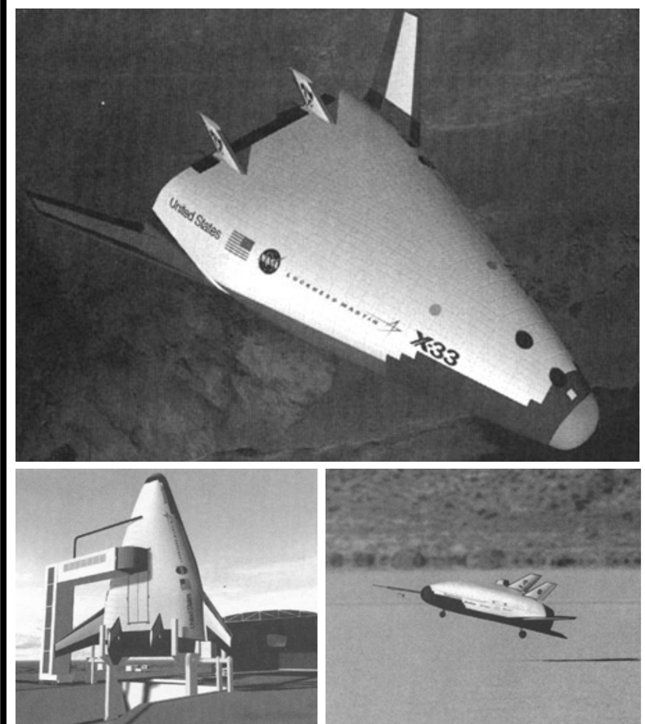


Reusable Launch Vehicle (RLV) Concepts

Technology Advancement Concepts

A Presidential Directive of August 1994 entitled the Space Transportation Policy authorized NASA to investigate a new generation of reusable vehicles.

- One technical approach that was studied was intended to use a single stage vehicle (the X33) to launch people and payloads to LEO:
  - The project used Lockheed Martin's lifting body design to enable the vehicle to launch vertically and land like a glider.
  - The vehicle was to use 2 linear "Aerospace" engines built be Rocketdyne.
  - NASA had planned to spend just under one billion dollars for a series of suborbital test flights, and the first took place early in 1999.
  - Lockheed Martin was expected to contribute 220 million dollars to the project, and desired to commercialize a much larger version as the "Venture Star".
  - In 2001 it appeared that SSTO would not be successful, and NASA dropped the project.
  - Funds were redirected to upgrade the Space Shuttle.



NASA/Lockheed Martin X33

Single Stage-to-Orbit (SSTO) Concept

The X33/ Venture Star project was technically very aggressive (and ultimately too aggressive):

- The goal was to reduce Earth-to-orbit costs sufficiently to provide commercial aircraft-like safety, operations and maintenance:
  - The Aerospike main engine is a revolutionary propulsion system designed for operation at all altitudes, using multiple nozzles to provide directional control for maneuvering.
  - Use of light advanced materials and the lifting body shape were to improve structure mass fractions, thrust-to-weight rocket efficiency and L/ D maneuverability.
  - Unfortunately, key technologies including graphite composite LH2 tanks and the SSTO approach proved unfeasible, and the project cost overruns and delays were unacceptable.

X-33	VENTURESTAR
Length: 69ft	Length: 151ft
Width: 77ft	Width: 159ft
Takeoff weight: 285,000 lbs	Takeoff weight: 2,628,900 lbs
Fuel: LH2/LO2	Fuel: LH2/LO2
Main propulsion: 2J-2S Linear Aerospikes	Main propulsion: 8RS2200 Linear Aerospikes
Take-off thrust: 410,000 lbs	Take-off thrust: 3,537,000 lbs
Maximum speed: Mach 13+	Maximum speed: Mach 25 to orbital

The Proposed Venture Star

Single Stage-to-Orbit (SSTO) Concept

NASA

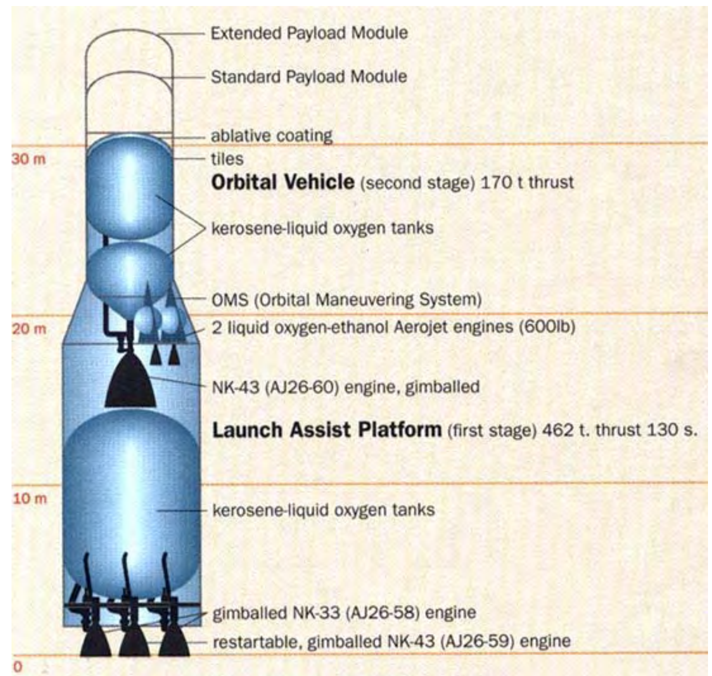
NASA's investments in next-generation reusable launch vehicles were channeled into a new program called the Space Launch Initiative established in 2001.

- One of the companies that received NASA backing is the Kistler Aerospace Corporation which is developing a recoverable K1 launch vehicle:
  - Engineering design and systems tests have been undertaken at a launch site that has been constructed in Woomera, Australia.
  - The K1 is a 2-stage vehicle which will return to Earth intact with the help of parachutes and airbags.
  - The second stage will deliver payloads to a circular LEO trajectory, and reignite one day later to deorbit and return to the vicinity of the launch site.

Kistler K1 Reusable Vehicle

Kistler's K1 vehicle incorporates Russian and US technologies.

- Seven small and large companies are involved:
  - The first stage is built around three NK 33 engines developed by Dvigatel for the Russian lunar programs that are being built under license to Rocketdyne.
  - The second stage uses a single NK 33 engine plus 2 Russian NK 44 engines for orbital maneuvers.
  - Lockheed Martin Space Systems is supplying the LOX-Kerosene propellant tanks.
  - Payload deliveries beyond LEO might use an "active dispenser" third stage which is not recoverable for geostationary satellites.
  - K1 is also being planned for cargo missions to the ISS.

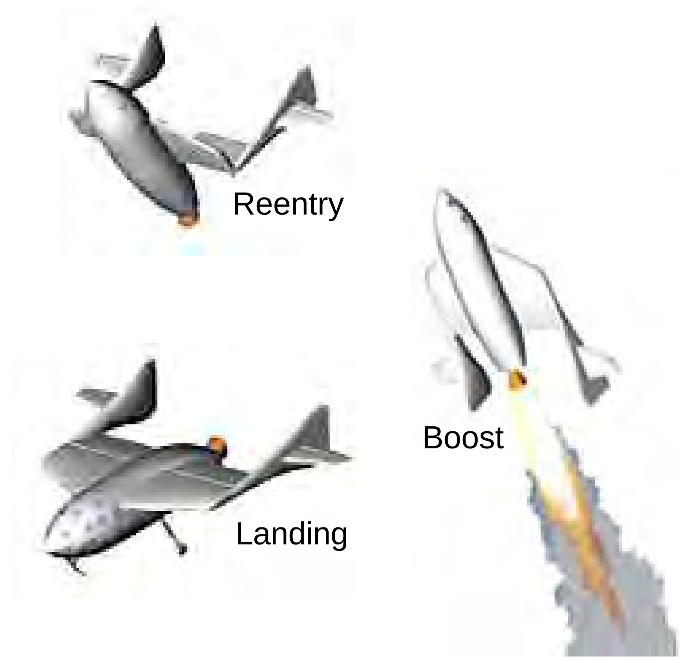


Russian – Derived K1 Launch Vehicle

Kistler K1 Reusable Vehicle

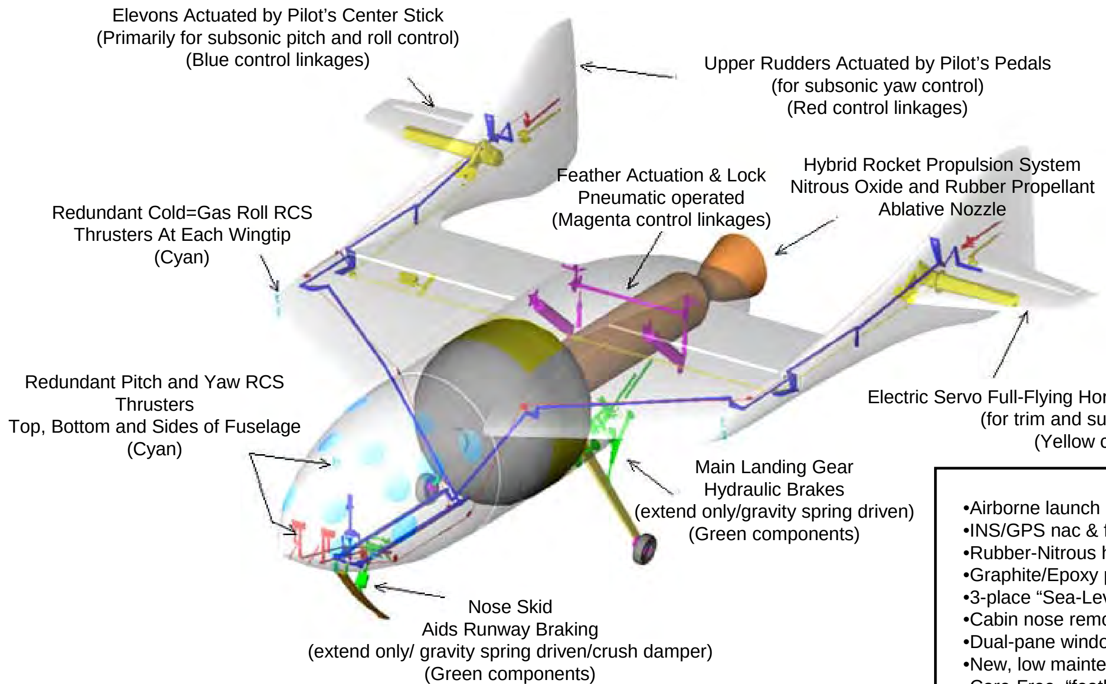
In November of 2004, Space Ship One developed by Burt Rutan became the first private spacecraft to demonstrate high altitude suborbital capabilities to win the coveted $10 million X-Prize:

- Space Ship One is a 3-passenger capacity research vehicle that departs from a conventional runway attached to a specially designed aircraft that delivers it to a 50,000 ft altitude prior to launch:
  - The wing and vertical stabilizer reconfigure in flight from a pneumatic-actuated flat “feather” shape during the boost stage, to a high-drag shape for atmospheric entry.
  - The vehicle uses three flight control systems: manual-subsonic, electric supersonic, and cold-gas RCS.
  - A new non-toxic liquid-nitrous oxide/rubber-fuel hybrid propulsion system is used that includes composite tanks.



Space Ship One Flight Configurations

Space Ship One Commercial Vehicle



Basic Feather

- Airborne launch
- INS/GPS nac & flight director
- Rubber-Nitrous hybrid rocket propulsion system
- Graphite/Epoxy primary structure
- 3-place “Sea-Level, shirt-sleeve” cabin environment
- Cabin nose removes for egress. Side plug door
- Dual-pane windows, dual seals on door & controls
- New, low maintenance Thermal Protection System
- Care-Free, “feather” atmospheric entry

Space Ship One Commercial Vehicle

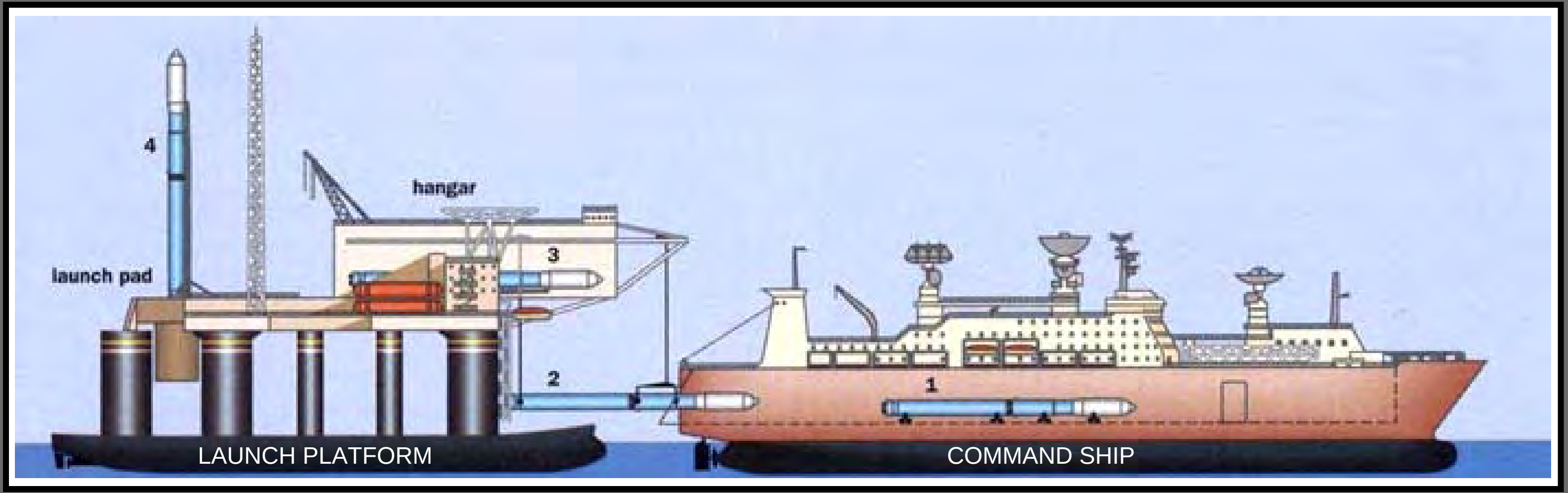
Airborne Space Ship One Lander

TRANSPORTATION SYSTEMS

EXPERIMENTAL INITIATIVES

Ground safety hazards associated with conventional rocket vehicle launches present risks that can be avoided using airborne deployment (such as Space Ship One) and offshore ships.

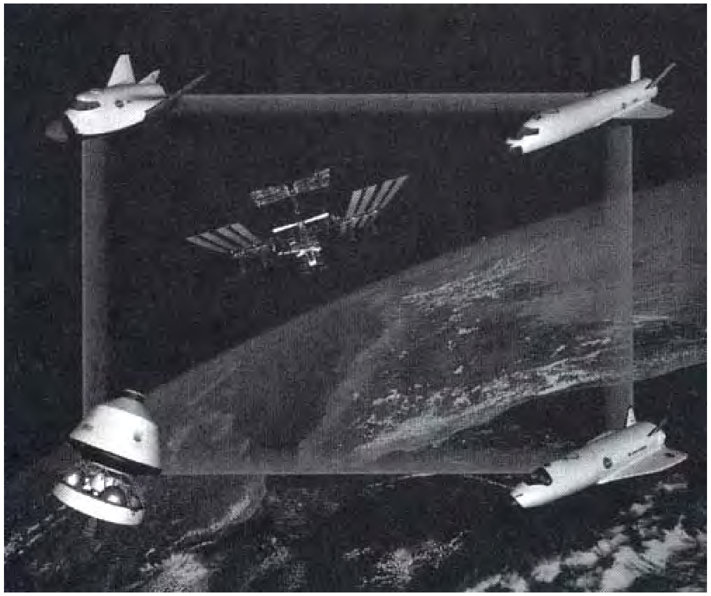
- A “Sea Launch Service” joint venture has demonstrated offshore launch capabilities using a 28,000 MT floating platform called “Odyssey”, and a 30,000 MT command and transportation ship called “Sea Launch Commander”.
- Joint venture partners are Boeing (mission control, sales and payload processing), the Anglo-Norwegian Kvaerner Group, the Ukranian SDO Yuzhnoe/ Po Yushmash and the Russian RSC Energia.



Off-shore International Sea Launch Venture

Development of next-generation crew transport vehicle is a vital space program need.

- Four general types of concepts for a new Orbital Space Plane are being studies:
  - A capsule approach
  - A sharp-winged body
  - A lifting body approach
  - A blunt-winged body
- The new vehicle must meet many important requirements :
  - Risk of crew loss must be lower than the Shuttle.
  - Operational and life-cycle costs must be lower than the Shuttle, and it must require less time to prepare and execute a mission.
  - Launch frequency must be increased with shorter turn-around preparations.
  - It will initially serve as a crew rescue vehicle for ISS, enabling larger, permanent crews to depart rapidly in emergencies.
  - It will also be used to ferry crews and light cargo to the ISS, and later become the foundation for routine crew transfers.

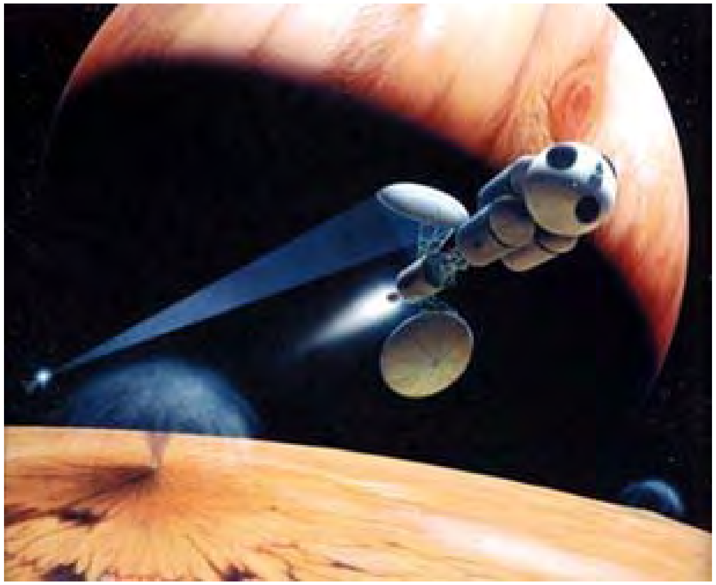


Orbital Space Plane Vehicle Concepts

Next-generation Space Plane

Hypothetical theories sometimes, if rarely, produce breakthrough technological realities. Unfortunately, even theories based upon solid and proven laboratory experiments do not always scale up for practical and substantive applications.

- Next generation space transportation systems must demonstrate capabilities to safely and efficiently move large masses and volumes through Earth's deep gravity well, thick atmosphere and beyond much better than conventional rockets and vehicles:
  - They need to be lighter, stronger and heat-resistant, potentially incorporating new composite, honeycomb and nanotube structures.
  - They must be more robust and require less maintenance, enabling shorter turn-around schedules between flights.
  - Above all, they must utilize much more fuel-efficient engines, and possibly no engines at all, to support routine and affordable commercial applications in LEO, and exploration of the Solar System.



LEO and Beyond

Requirements & Challenges

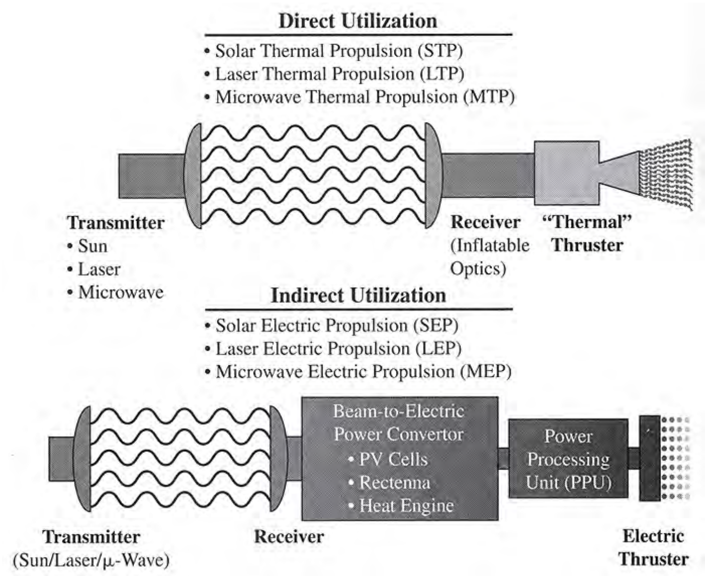
Rocket Systems that Breath Oxygen as
they move through Earth's Atmosphere

Vehicle that are Magnetically Levitated and
Launched at Supersonic Velocities from Tracks

Airbreathers & Meglev Launches

Scientists and engineers are exploring new approaches and technologies to reduce massive fuel requirements imposed by current systems for spacecraft launches and orbital transfers:

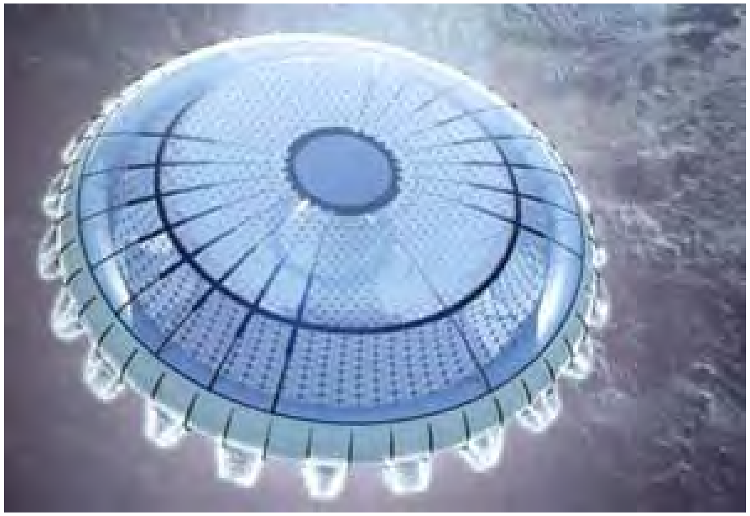
- Beamed-energy propulsion might someday transmit a "beam" of electromagnetic energy using a remote energy source such as the Sun or a ground-based laser to power a propulsion system.
  - This might result in a significant weight reduction over more conventional approaches for improved spacecraft performance, avoiding the need for a heavy power supply and engine system.
  - Solar/ laser/ microwave systems might be used for orbit-to-orbit in-space applications, but only laser or microwave systems might offer sufficient power density for Earth-to-orbit launches.
  - Beam power requirements for laser/ microwave in-space systems are typically 0.1-10MW total, while launch systems will require about 0.1MW of beam power/kg of vehicle mass.



Thermal and Electric Thrusters

Power Beaming Concepts

Future spacecrafts may be powered by microwave energy, and even by high-velocity antimatter engines.



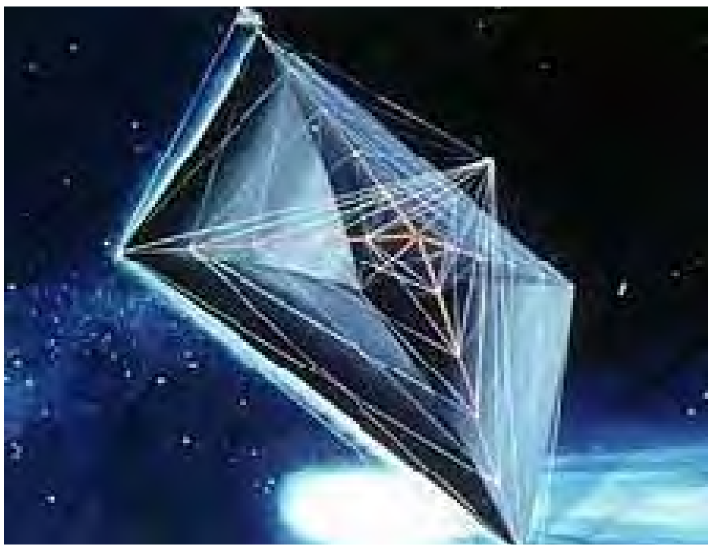
Spacecraft Accelerated by Microwaves Beamed from Earth or Solar Power Satellites

Pulse Propulsion Systems using Anti-matter High-Velocity "Warp Drive"

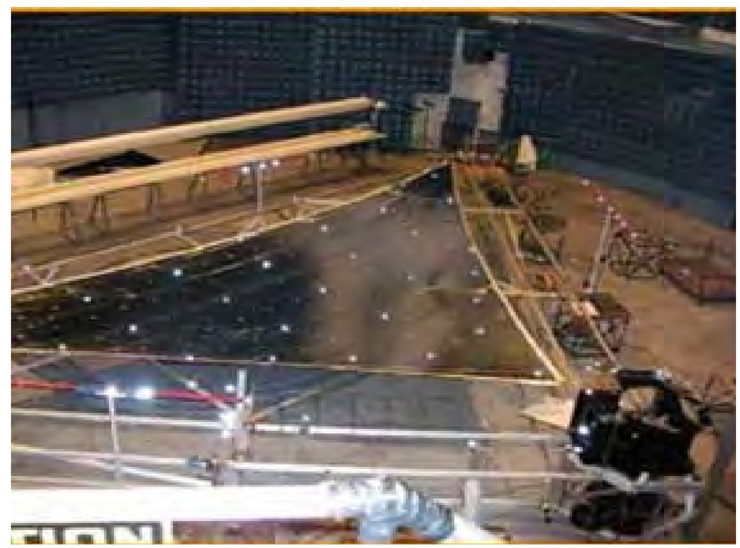
Microwave & Pulsed Propulsion

Future Solar Sails made of very thin reflective materials supported by ultra light-weight deployable structures may accelerate under pressure from solar radiation, avoiding the need for propellants. The thrust-to-weight ratios will be very low, however, resulting in very large structures.

NASA



Applying Solar Protons to Accelerate Giant Ultra-light Sail for Cosmic Voyages

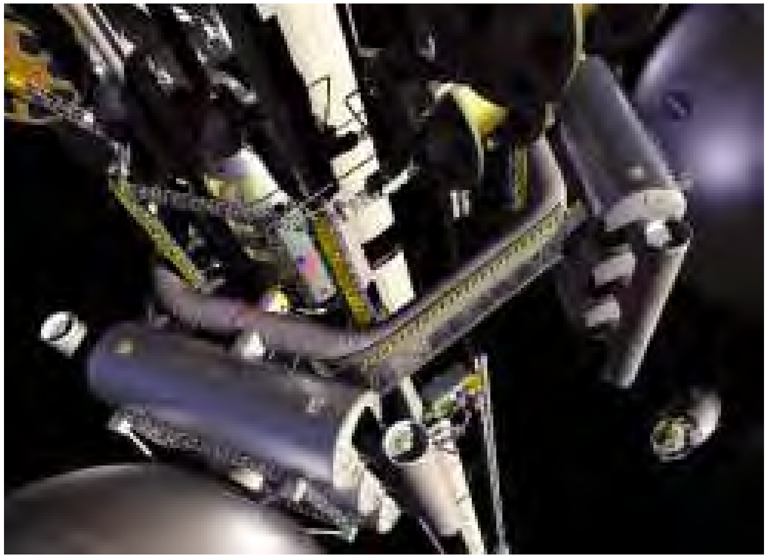


10m Deployed Solar Sail in a 50 ft Diam Vacuum Chamber at NASA Langley Res. Center

Solar Sail-Powered Spacecraft

NASA

Space elevators may someday eliminate the need of some spacecrafts altogether.



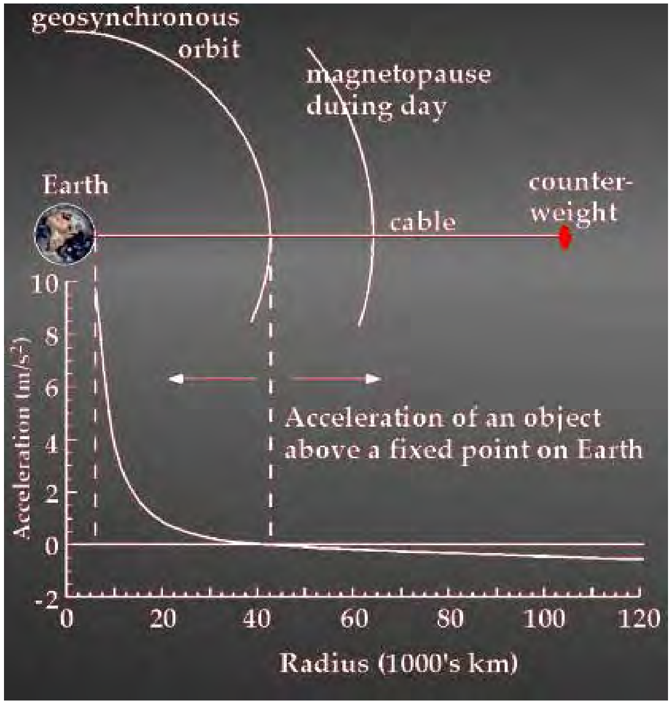
Space Elevators Extending from Earth to Geostationary Orbit might Eliminate the Need for Propulsion Systems

Payload would be Transferred at High Speeds on Cable's Anchored to Earth Platforms

Space Elevators with Climbers

NASA

An operating Facility at the
End of the Cable would Serve as a Counterweight



Centripetal Force of me Counterweight's
Angular Momentum would Balance Earth Gravity

Space Elevators with Climbers

Additional information relevant to this section can be found in Part I of this SICSA Space Architecture Seminar Lecture Series titled Space Structures and Support Systems.

Key reference sources are the Cambridge Encyclopedia of Space (Cambridge University Press), the International Reference Guide to Space Launch Systems - Second Edition (AIAA Publication), Human Spaceflight Analysis and Design (The McGraw Hill Companies Inc.) and Spacecraft System Design and Operations (Kendall/Hunt Publishing Company). These and other relevant publications are listed below:

“ 2000 Reusable Launch Vehicle Programs & Concepts”, Associate Administrator for Commercial Space Transportation (AST), federal Aviation Administration, January, 2000.

Blanchard, B., Fabrycky, *W. Systems & Engineering Analysis*, 2nd Edition, Prentice hall, Englewood Cliffs, New Jersey, 1990.

Boden, D., Larsen, W., *Cost-Effective Space Mission Operations*, McGraw Hill Companies, Inc., 1996.

Chapman, D., 1960, *An Analysis of the Corridor and Guidance Requirements for Super Circular Entry into Planetary Atmospheres*. NASA Technical Report R-55. Washington, DC:National Aeronautics and Space Administration.

Cruz, M.I, 1979. The Aerocapture Vehicle Mission Design Concept- Aerodynamically Controlled Orbit Capture. AIAA Paper 78-0893. Washington, DC. American Institute of Aeronautics and Astronautics.

Cuthbert, P., M. Tigges, and L. Bryant.1990. Lunar/ Mars earth Return Aerocapture Heating Constraint. AIAA Paper 90-2936. Washington, DC. American Institute of Aeronautics and Astronautics.

Damon, T.D. *Introduction To Space: The Science of Spaceflight*, 3rd Edition, Kreiger Publishing Company, 2000.

Duncan, R., 1962. *Dynamics of Atmospheric Entry*. New York, NY: McGraw Hill Book Company, Inc.

Fortescue, P., Stark, J., *Spacecraft Systems Engineering*, 2nd Edition, Wiley Publishing, 1995.

Godwin, R., Rocket and Space Corporation Energia: *The Legacy of S.P Korolev*, Apogee Books, May, 2001.

Hansen, W. et al 1978. The Mars Reference Atmosphere. Innsbruck: Committee on Space Research (COSPAR) and Pasadena, CA:NASA Jet propulsion Library.

Henning, G., “Modeling Launch Vehicle Economies.”, *Aerospace America*, 36-38, March, 1999.

“International Space Station Payloads Operations & Interfaces Manual.”, P/L OPS & IFTM 21109A, August 16, 2002, NASA.

Isakowitz, S., “International Reference Guide to Space Launch Systems”, 2nd Edition, American Institute of Aeronautics and Astronautics, 1995.

Larson, W., Pranke, L., *Human Spaceflight- Mission Analysis & Design*, 1st Edition, McGraw Hill, 1996.

“National Space Policy.” The National Science and Technology Council, September 19, 1996, NASA.

Peters, J., “Spacecraft Systems Design and Operations”, Kendall/ Hunt Publishing Company, Dubuque, Iowa, 2003.

Peterson, W., et al 1992. *Earth-Space-land Landing Analysis for the First Lunar Outpost Mission: Apollo Configuration*. NASA Report JSC-25895. Houston, TX: NASA Johnson Space Center.

“Reusable Launch Vehicle : Technology Development & Test Program”, National Research Council, Washington, D.C.: National Academy Press, 1994.

Rochelle, W.C. et al. 1990 Aerothermodynamic Environments for Mars Entry, Mars return and Lunar Return Aerobraking Missions. Paper presented at the AIAA/ ASME 5th Joint Thermophysics Conference, AIAA 90-1701. Washington, DC: American Institute of Aeronautics and Astronautics.

Shayler, D.J. *Disasters and Accidents in Manned Spaceflights*, Publisher Springer-Praxis, Chichester, UK, 2000.

“The NASA Access-to-Space Study Summary.” Part I, July, 1993, NASA.

“The NASA Access-to-Space Study Summary.” Part II, September 23, 1998, NASA

"The Road from the NASA Access-to-Space Study to a Reusable Launch Vehicle.", 49th International Astronautical Congress, Melbourne, Australia. (1998)

Tigges, M.A., and L. Bryant.1989. *Lunar/Mars Common Vehicle Study*. NASA Report JSC-23894. Houston, TX:NASA Johnson Space Center.

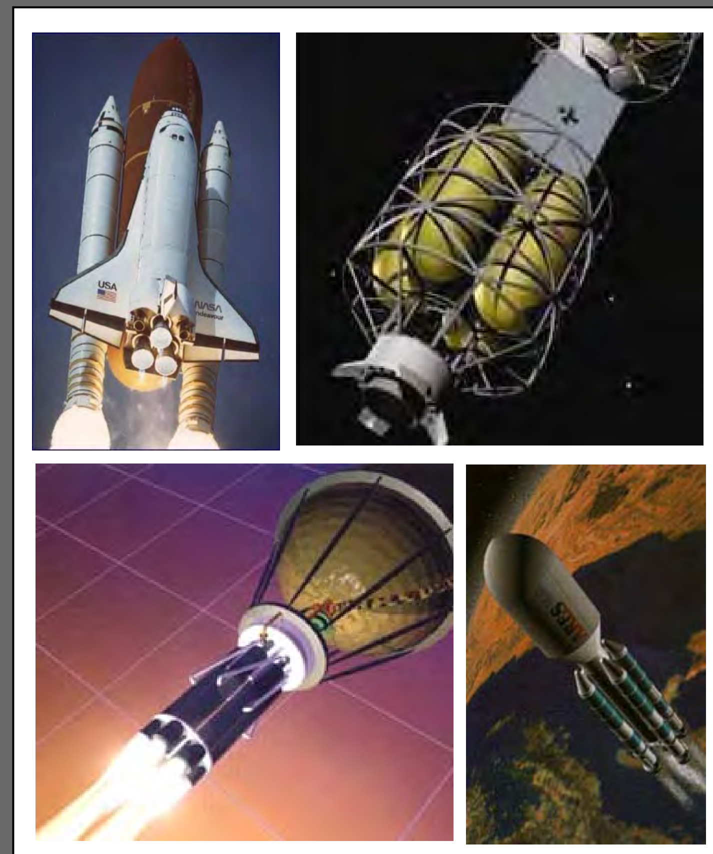
Verger et al "The Cambridge Encyclopedia of Space", Cambridge University Press, 2002.

Vinh, Nguyen X., Adolf Busemann, and Robert D.Culp. 1980. *Hypersonic and Planetary Entry Flight Mechanics*. Ann Arbor, MI: The University of Michigan Press.

Wertz, J.R., Larson, W.J., *Reducing Space Mission Cost.* Microcosm/ Kluwer Publisher, 1999.

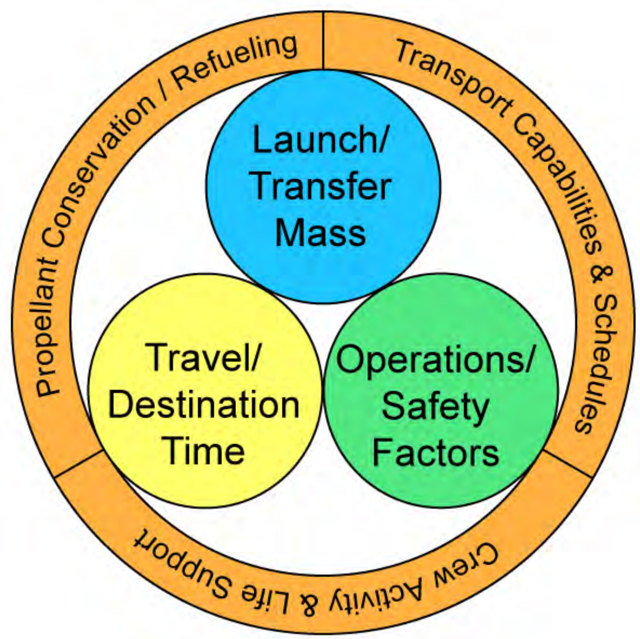
BACK TO THE LIST OF CONTENTS

SECTION B : PROPULSION SYSTEMS



Selection of propulsion systems, in combination with transportation vehicles, has fundamental influences upon broad aspects of space architecture planning:

- Influences upon payload size:
  - How much launch/ delivery mass will be fuel vs. payload?
  - How many launches and orbital assembly procedures will be needed to create large structures from smaller elements?
  - How much mass can be inserted (aerodynamically or propulsively) into the destination orbit and landed on the surface?
- Influences on safety and operations:
  - How rapidly will the transportation vehicle pass through the hazardous Van Allen Belt zone?
  - What transfer orbits are possible, and how long will voyages/ surface times/ returns require?
  - Can engines be re-started for mid-course corrections, propulsive braking and surface descent/ ascent?
- Influences upon economics:
  - How long can fuels be stored in orbit and on the destination surface?
  - Is orbital refueling or replenishment from in-situ propellants possible to conserve mass?

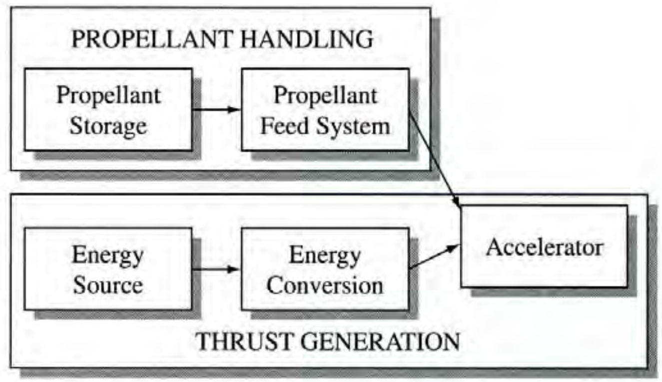


Propulsion Influences on Mission Planning

Space Architecture Influences

Propulsion systems include fuel and engine/ thruster subsystems to launch spacecraft into LEO, maneuver them, transfer them to destination orbits, slow them for insertion into descent orbits, land payloads on the lunar/ planetary surfaces, and return crews to rendezvous orbits for return to Earth.

- Key elements are propellant handling and thrust-producing subsystems which apply different technology approaches:
  - Chemical liquid fuel rockets use super-cooled hypergolic propellants with separately stored compounds that ignite explosively when mixed together in engines or thrusters.
  - Chemical solid fuel rockets use propellants that decompose explosively when brought into contact with an electrically-heated metallic catalyst within engines or thrusters.
  - Nuclear systems use direct heat or electrically-energized magnets to ionize and release plasma to provide low levels of continuous thrust.

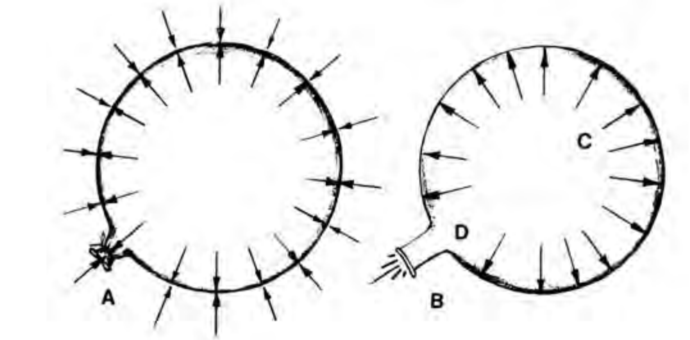


Basic Propulsion System Elements

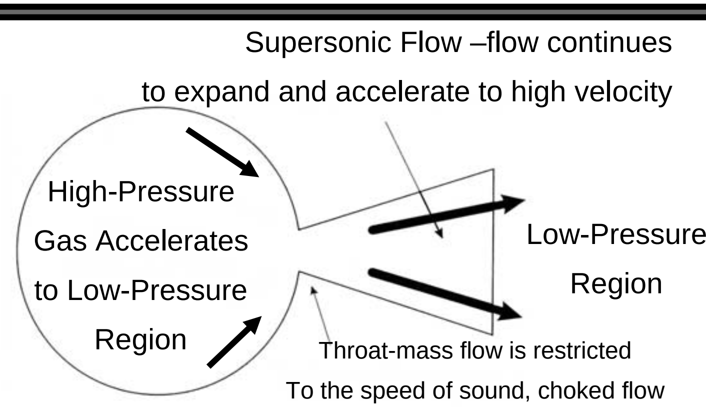
Key types and Elements

Thrust is typically generated by creating a region of high pressure and allowing high-pressure gases or liquids to expand into an area of lower pressure.

- Rocket engines use converging/ diverging nozzles to accelerate the outward flow to supersonic velocities necessary for efficient thrust generation:
  - Higher pressure levels accommodate smaller nozzle throat and engine pressure.
  - Higher pressure levels provide more thrust.
  - Bigger engines produce more thrust.
- Thrust is also influenced by the combustion chemistry of the propellant used which establishes an Oxidizer-to-Fuel ratio (OF):
  - This relationship is governed by flame temperature, molecular mass and isotropic characteristics.
  - These factors also determine the mass-flow rate at which propellants are consumed.



Reaction Engine Principle



Pressure Acceleration Dynamic

Basic Thrust Processes

The most common and simple way to evaluate thrust efficiency is “specific impulse” (Isp) which indicates the amount of force that can be produced by particular combinations of propulsion systems and fuel according to the formula:

$$I_{sp} = \frac{F}{g_o m}$$

Where;
I_{sp}= specific impulse(s)
F= thrust (N)
m= propellant mass-flow rate (kg/s)
g_0= gravity constant = 9.81m/s2 (applies on all planets)

A higher number is better, meaning that more thrust can be delivered for a given amount of propellant.

Note: Monopropellants are typically used for launch or orbit insertion due to low thrust and high system mass characteristics.

Propellant	I_{sp}	Propellant	I_{sp}
Solid: 18% Al, 71% NH_4ClO_4, 11% HTP13	307	Kerosene/N_2O_4	311
Monopropellant: 92% H_2O_2, 8% H_2O	181	Kerosene/92%H_2O_2, 8%H_2O	315
Monopropellant: N_2H_4	218	Kerosene/liquid O_2	337
Mono-methyl-hydrogen/N_2O_4	328	Kerosene/N_2O	299
Liquid H_2/Liquid O_2	448	Nuclear/H_2 propellant	1013

Representative Propellants

Specific Impulse

When discussing capabilities of rocket engines we use the term “thrust” (Isp) rather than force:

- A simple way to think about specific impulse is to regard it is the amount of time required for a propellant of a given mass to produce the same mass equivalence of thrust.

Total thrust capabilities are expressed in Newtons (N):

- A rocket engine rated at a certain N capacity, must able to transport itself, its propellant and its payload to Earth’s escape velocity:
  - As fuel is burned and exhausted the vehicle and rocket combination becomes lighter.
  - If a vehicle accelerates at about 10 mph/ sec, it will travel 10 mph after the 1st second, 20 mph after 2 seconds, 50 mph after 5 seconds, and 3,000 mph after 5 minutes (300 seconds). To go into orbit it must travel at more than 17,000 mph.

Technology	I_{sp} (s)	Thrust (N)
Chemical		0.1-12,000,000
Liquid		
Monopropellant	140-235	
Bipropellant	320-460	
Solid	260-300	
Hybrid	290-350	
Nuclear		up to 12,000,000
Solid core	800-1100	
Liquid core	3000	
Gas core	6000	
Nuclear electric		0.0001-20
Electrothermal	500-1000	
Electromagnetic	1000-7000	
Electrostatic	2000-10,000	

Characteristics of Representative System types

Thrust and Performance

Rockets don't obtain forward motion by pushing against air (which doesn't exist in a vacuum), but operate according to Newton's 3rd Law of Motion which establishes that each force has an opposite and equal reaction (unless unbalanced by an equal and opposite force):

- Rockets are much less efficient inside an atmosphere:
  - Outside air hinders motion by causing frictional drag and retarding the expansion of gases outside nozzles.
  - Rockets typically launch straight up in order to get out of the atmosphere as rapidly as possible.
- As a rocket pitches over to horizontal, its thrust is no longer directed against gravity, but must still overcome atmospheric drag:
  - Drag increases with velocity, but decreases at higher altitudes where the atmosphere is less dense.
  - Gravitational pull decreases as it moves farther away from Earth.

Newton's 2nd Law of Motion is used to calculate the rate of acceleration (a) in relation to net force (F) and the total system/ payload mass according to the formula:

$$a = \frac{F}{m}$$

Where;

Mass= $\frac{\text{weight}}{g}$ = $\frac{\text{total weight (lbs)}}{\text{32 ft/sec2}}$ = "slugs"

(A slug is a unit of mass in the English system of measurement)

- Assuming that a fully loaded Space Shuttle weighs 4.4 million lbs with 3 main engine and 2 SRBs rated at 375,000 lbs each:
  - Downward gravitational force of its weight (Fa)= 4.4 million lbs
  - Upward force (Fb) of its total rockets is about 7.7 million lbs.
  - Net upward force (F) is about 3.3 million lbs.

$$a=\frac{F}{m}=\frac{3{,}300{,}000 \text{ lbs}}{140{,}000 \text{ slugs}} = 24 \text{ ft/sec2}=16 \text{ mph/sec}$$

This means that for each second that passes, the Shuttle will increase its speed by 16 mph.

Appling Newton's 2nd Law of Motion

Force, Resistance & Velocity

A mass fraction (MF) is the proportion of propellant mass to total system mass (MF=Mprop/M tot). This means that as mass fractions become larger, there is less remaining mass available for payloads.

- Conventional Systems:
  - Mass fractions decrease as propellant mass increase (economies of scale).
  - Mass fractions typically increase as the rocket stage numbers increase (because stages usually get smaller as they "go up the vehicle").
  - Solid rockets tend to have somewhat lower mass fractions than liquid, but their mass fractions typically get better as their sizes increase.

- Other systems:
  - Hybrid rockets tend to have mass fractions that are slightly higher than liquids because their fuel packing density is lower, but they waste quite a lot of unburned fuel (about 17%).
  - Nuclear rockets have similar mass fractions to liquids except than the reactor mass can be very substantial (500 kg is typical). Their radiation shields can also add significant mass (about 3,500 kg/m2).
  - Nuclear electric systems can have relatively small propellant and tank masses due to high I_{sp} efficiencies, but their generating equipment may be proportionately larger along with nuclear power sources and shields.

System Mass Fractions

Propulsion system selection processes match application requirements with technology attributes:

- Chemical liquid rockets offer good launch/ orbital insertion thrust and can be restarted for repeated use. Liabilities are mechanical complexity and difficulties in storing cryogenic propellants during long duration space missions.
- Solid fuel rockets are simpler than liquid, and propellants can be stored indefinitely. A big limitation is that once ignited they can't be throttled back, topped off or reused.
- Hybrid rockets use a solid fuel core and often liquid oxygen as an oxidizer. They haven't yet found mainstream and established heritage, but new solid propellants using synthetic polymers are improving burn continuity and thrust.
- Nuclear systems might support long distance/ duration human exploration missions, either to superheat hydrogen has propellants using synthetic polymers are improving burn continuity and thrust.
- Nuclear systems might support long distance/ duration human exploration missions, either to superheat hydrogen gas propellant, or to power electric drive systems. Scale-up challenges, mass and radiation hazards present issues.

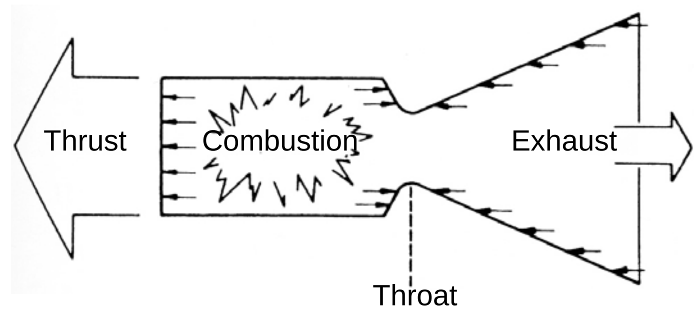
Technology	Applications	Advantages	Disadvantages
Chemical Liquid Monopropellant Bipropellant	Launch Orbit Insertion/ Maneuvers Landing/Ascent	•High thrust •Heritage •Restartable	•Complicated combustion •Fuel storage problem
Solid Hybrid	Launch Orbit Insertion	•High thrust •Heritage	•Not restartable
Nuclear Solid core Liquid core Gas core	Orbit Insertion/ Maneuvers	•High specific impulse	•Unproven •Radiation •Low thrust/weight
Nuclear Electric Electrothermal Electromagnetic Electrostatic	Orbit Insertion/ Maneuvers	•Very high specific impulse	•Radiation •System mass •Low thrust levels •Limited heritage

General Applications and Attributes

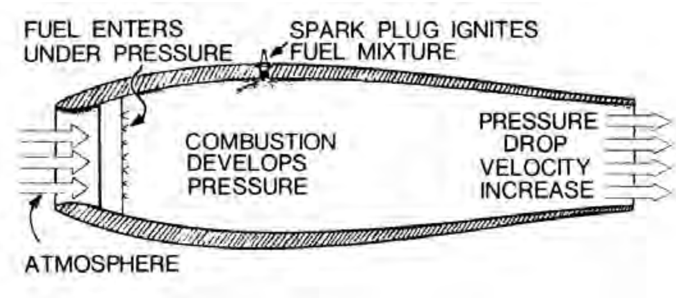
Technology Comparisons

Chemical reaction rockets use a combustion process in which confined fuel gas or gases are oxidized (combined with oxygen) under tremendous pressure:

- The combusted gases expand to increase chamber pressure, pass through a constricted throat which increases the exhaust speed, and expand rapidly in a nozzle to accelerate more and reduce chamber pressure.
- Chamber pressure is then restored by high temperature reactions of successive propellant combustion events to repeat the process.
- Conventional rocket engines carry their own oxygen, enabling them to operate in a vacuum.
- Ramjet engines are reaction rockets that intake oxygen from air in the atmosphere. Since they must move rapidly to compress the air, they cannot take off from the ground or operate at low speeds.



Conventional Reaction Engine Diagram



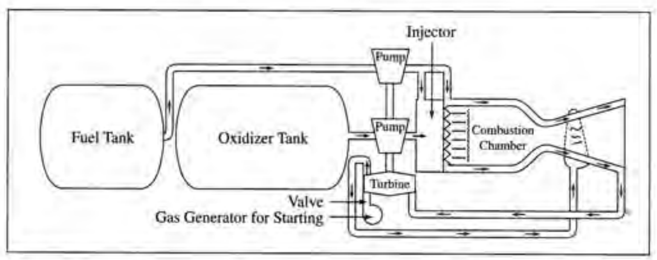
Ramjet Engine Diagram

Chemical Reaction Rockets

Liquid rocket systems have been used extensively in US and Soviet/ Russian manned space missions:

- Most liquid rockets use “hypergolic” storable propellants that ignite spontaneously when mixed together:
  - Although this type of ignition is straightforward, the volatile nature of the materials and complexity of pumps, valves and control systems pose safety hazards.
  - Since fuels are usually denser than oxygen, the propellant tanks are typically placed above the oxidizer tank so that fuel depletion during burns shifts the center-of-gravity forward to improve rocket stability.
  - High-speed turbine pumps drive propellants into a combustion chamber at very fast rates, assisted by pressurized propellant tanks that prevent vacuum as a tank is vacated.
  - Engine burn can be ignited by spark plugs, pyrotechnic charges, heating elements, or even small igniter rockets.

Typical liquid propellant rocket systems include a combustion chamber and nozzle that is led an oxidizer and fuel mixture by turbo pumps. Exhaust gases are forced out of the bell-shaped nozzle. A set of values down stream from the pumps controls propellant pressure.



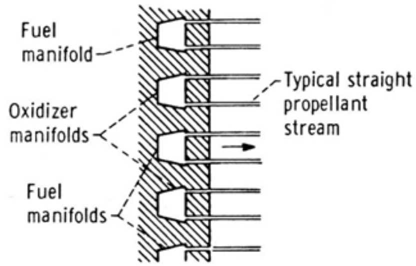
Liquid Rocket Block Diagram

Liquid Fuel Rockets

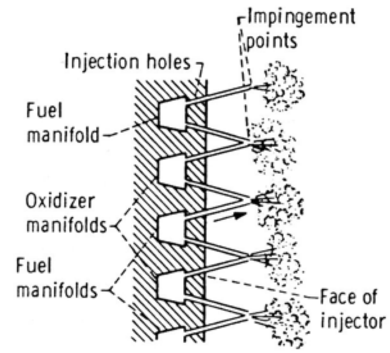
Liquid fuel rockets pose complex technical challenges:

- Extremely cold cryogenic liquid propellants pumped into high temperature combustion chambers present mechanical and safety risks:
  - Liquid oxygen storage tanks must be maintained at temperatures below the boiling point (-297°F). Such low temperatures can cause: contraction and cracking of metal valves, and pumps; freezing of pumps; hardening and shattering of rubber; solidification of lubricants; and congealing of oil.
  - LOX and propellant combustion temperatures reaching 5,560°F and hot gases flowing out can melt steel elements such as the exhaust nozzle throat.
  - LOX is the most widely used oxidizer because it is safer to handle and produce less hazardous exhaust products than other materials such as fluorine that exhausts toxic and corrosive hydrofluoric acid.

Fuel and oxidizer enter the combustion chamber through injectors which break propellant into fire streams or droplets to be ignited and burned at high temperatures reaching more than 5,000˚F. The injectors function much like fuel injectors in automobiles.



Shower Head Stream Type



Double-Impinging Type

Injector Devices

Liquid Fuel Rockets

NASA

Solid propellant rockets are frequently used for military purposes and as boosters:

- They offer important advantaged and disadvantages:
  - Positive features are simplicity of design, ease of construction, high reliability, long-term storability, and rapid flight-ready preparation.
  - Negative features are inabilities to throttle-down , turn off or restart engines, although thrust can be terminated by blowing off top caps to allow gases to escape at both ends to balance exhaust thrust.
- Solid propellants are mixed with a binder into a desired shape along with a stabilizer to prevent decomposition, and sometimes with a catalyst to speed up reactions:
  - Modern propellants include: potassium percolate (oxidizer) and asphalt (fuel); ammonium percolate (oxidizer) and aluminum powder (fuel); and nitrocellulose with notroglycerine (both oxidizers and fuels).

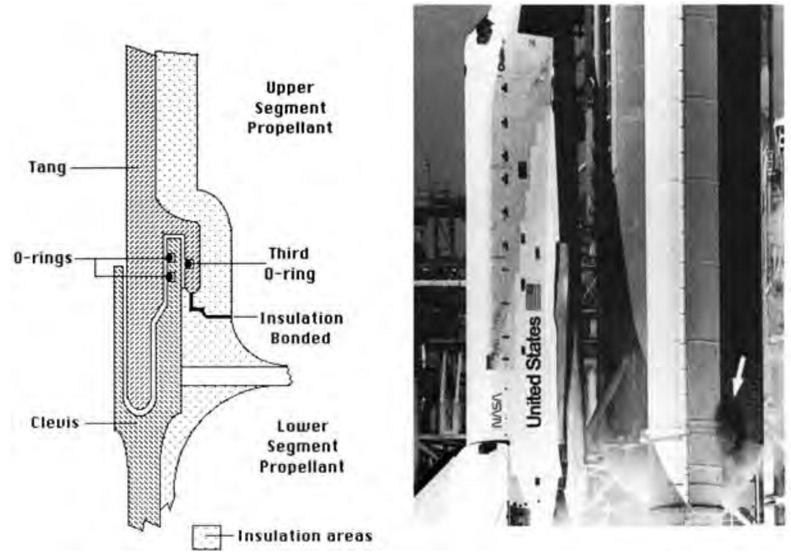
The Space Shuttle uses two SRBs mounted on opposite sides of the external tank. Standing nearly 150 ft tall and more than 12 ft in diameter, they are the largest solid propellant rockets ever built, and the first to be used for manned vehicles.

Space Shuttle Solid Rocket Boosters

Solid Fuel Rockets

Very large solid rockets must be constructed in segments because a one-piece grain casting the size of a Shuttle SRB, for example, would crack while handling and curing, and would be difficult to transport without damage:

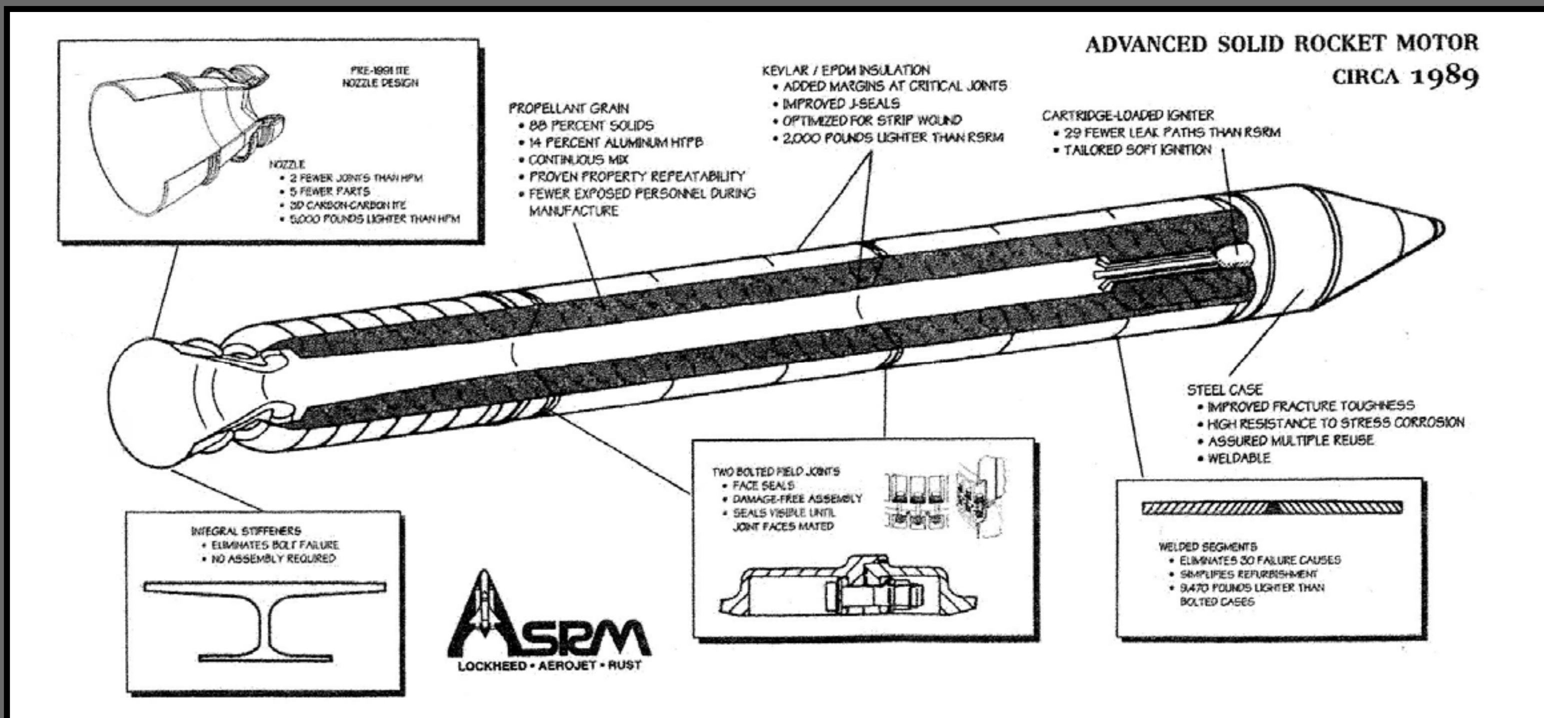
- The joint segments must be properly sealed tightly together to prevent hot internal gases from leaking through the casing, reducing internal pressure/ thrust, and producing unwanted thrust in the direction opposite the leak point (a circumstance that caused the catastrophic Challenger accident in January 1986):
  - Cold weather temperatures caused O-rings between the rocket segments to become stiff and not seal properly.
  - Hot exhaust gases forced zinc chromate putty that filled gaps between connecting metal casing sections to be pushed out.
  - The hot gases escaped, destroying the adjacent External Tank, and ultimately, the Orbiter.



Location of Challenger SRB Failure

Solid Fuel Rockets

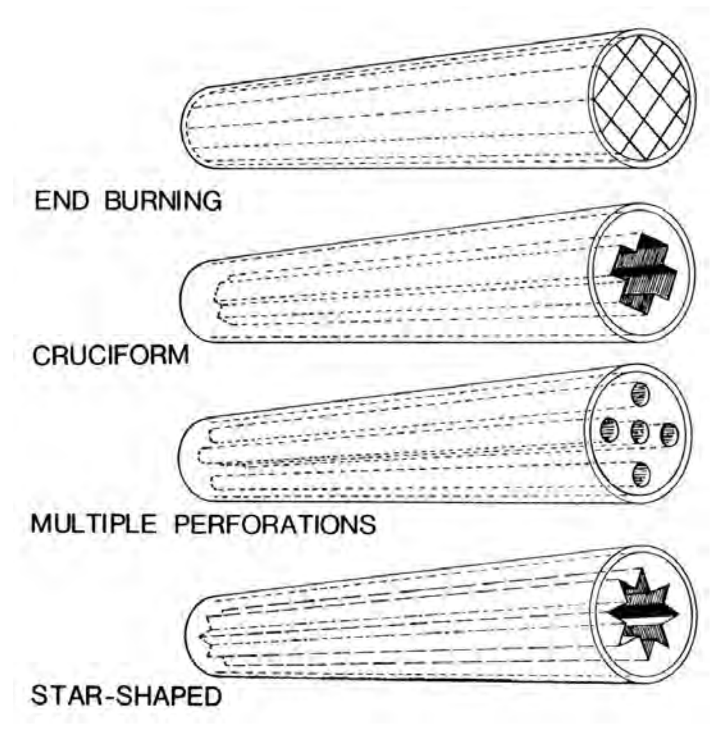
After the propellant is ignited, the expansion of exhaust gas through a nozzle is the same general design used for liquid fuel rockets.



Solid Fuel Rockets

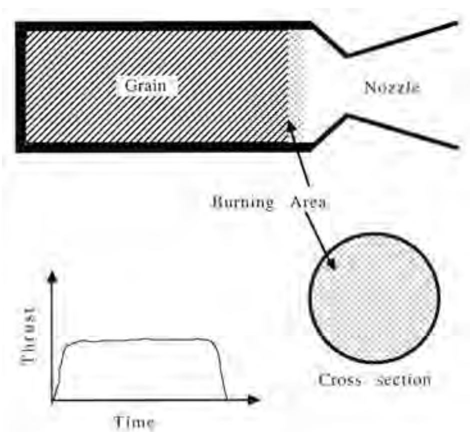
The "grain "of the molded solid propellant burns on its exposed surface to produce hot exhaust gases for thrust.

- Thrust can be controlled by the way the grain is shaped and the rate of burning which are influenced by the exposed surface area:
  - Cylindrical-shaped grains completely fill the casing and burn like a cigarette from one end. Since the surface area is constant, thrust is also constant (a neutral burn), and the rocket becomes top-heavy as the bottom burns away.
  - Some grains are molded with variously-shaped holes ("perforations") running down the center. As burn progresses and holes become larger, thrust increases (progressive burns).
  - Other grains are smaller in diameter than the casings and are centered in place by metal "spiders". As the burning surface area decreases, thrust also decreases (regressive burns).

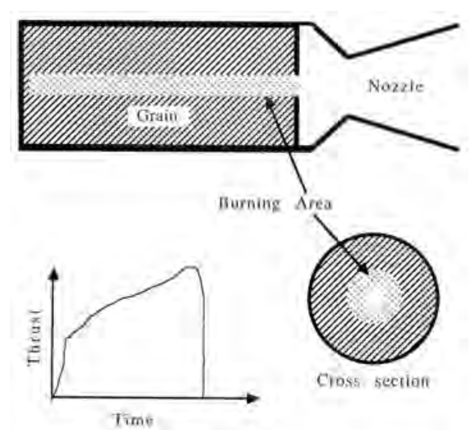


Solid Propellant Grain Types

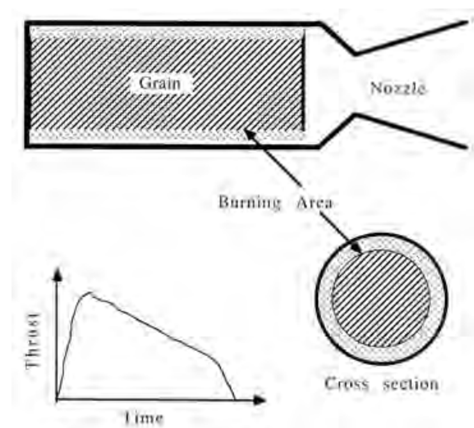
Solid Fuel Rockets



Neutral Burn – Cylindrical Grain



Progressive Burn – Bored Grain



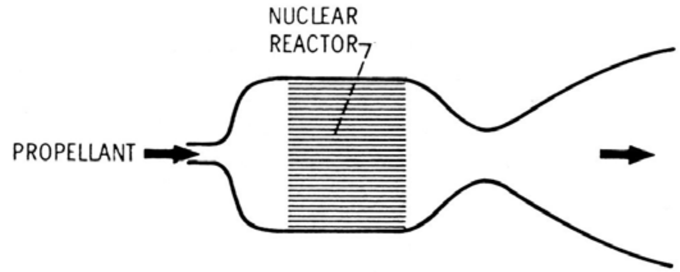
Regressive Burn – Supported Grain

Solid Rocket Burn Patterns

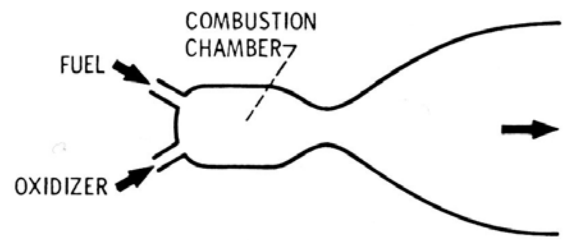
Solid Fuel Rockets

High-efficiency nuclear propulsion systems might support human and cargo missions over long distances and time spans:

- Comparisons with liquid fuel rockets:
  - Conventional liquid rockets require that a fuel and oxidizer be pumped from separate storage tanks and then combined to produce hot, high-velocity gases.
  - Nuclear systems eliminate the weight of one tank, one fluid, one pump, and the need for fuel-oxidizer combustion required for liquid fuel engines.
  - Energy available from nuclear reactions (fission, fusion and matter-antimatter annihilation) can range from 107-109 times more than from chemical reactions (greatly increased I_{sp}).
  - Energy available from a unit mass of fissionable material is approximately 107 times greater than the most energetic chemical reactions.



Nuclear Rocket Engine



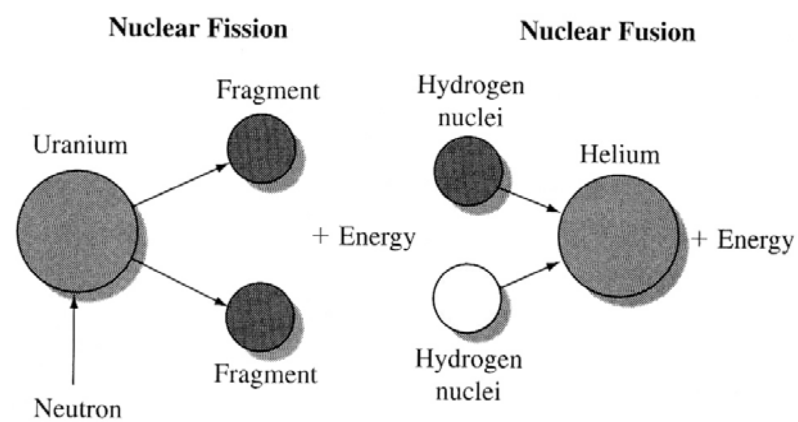
Chemical Rocket Engine

Block Diagram Comparisons

Nuclear vs. Chemical Engines

Fission and fusion processes can be used to heat a reactor core that superheats a hydrogen gas propellant which exits a nozzle at a very high velocity:

- The engines produce low, continuous thrust levels which can efficiently power spacecraft at high interplanetary speeds:
  - Specific impulses of 900 seconds and higher are possible (more than twice the Space Shuttle).
  - Although low thrust/mass ratios are inadequate for launches, the engines can be used for economical propulsive braking at a destination orbit (such as the Moon and Mars) by reversing the vehicle's direction near mid-course trajectory to gradually slow the spacecraft.
  - Vehicle crews must be protected from fusion radiation hazards by reactor shielding and/ or a distance between the vehicle and nuclear reaction processes.



Nuclear fission is a proven technology, while fusion presents radiation safety advantages but is technically uncertain.

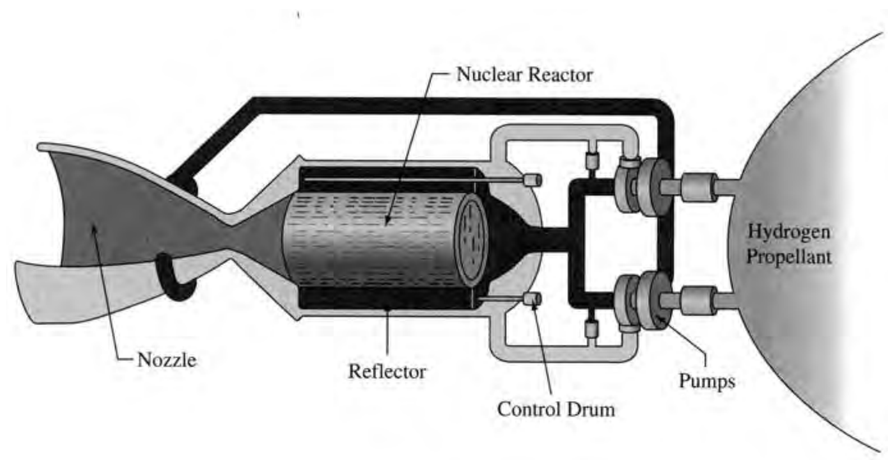
Comparison of Reactions

Nuclear Fission and Future Fusion

SPACECRAFT SYSTEMS DESIGN & OPERATIONS

Solid-core nuclear engines use reactors to heat a propellant to high exhaust temperatures that produce nozzle thrust.

- As with all reactor-based technologies, capabilities are ultimately constrained by temperature limits of materials and construction:
  - Propellant is heated as it passes through a solid fuel reactor core at temperatures which may reach 5,000°F or more.
  - Control drums located around the reactor core control the amount of reactivity.
  - The maximum operating temperature must be less than the melting point of the fuel, the moderator, and the cure structural materials.
  - Temperature limitations correspond with I_{sp} values of about 800-900 seconds.
  - Hydrogen is typically used as a propellant, and storage to prevent boil-off for long-duration space mission applications will present a technical challenge.



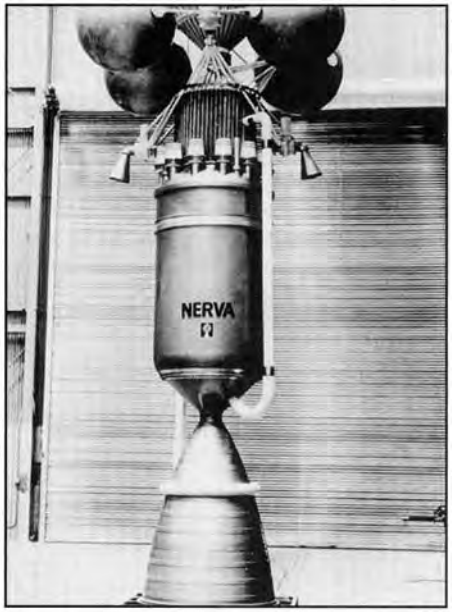
Block Diagram

Solid-Core Nuclear Engines

Extensive solid-core nuclear engine design, development and test activities were undertaken at the Los Alamos Scientific Laboratory from about 1956-1971:

- A “Nuclear Engine for Rocket Vehicle Applications” (NERVA) flight test engine that was being developed at the end of this period was cancelled to release funding for the Space Shuttle:
  - The NERVA engine used liquid hydrogen pumped through a reactor which heated it to about 4,000°F prior to being ejected through a nozzle.
  - The hydrogen acted as a moderator to slow neutrons from nuclear reactions.
  - The pump was driven by hydrogen gas that was heated in the reactor.

The NERVA engine was designed to operate at 1,500 kW provide 333 kN of thrust at Isp 825, weighed 10.4 MT, and was engineered for a 10 hour life at 60 operating cycles.



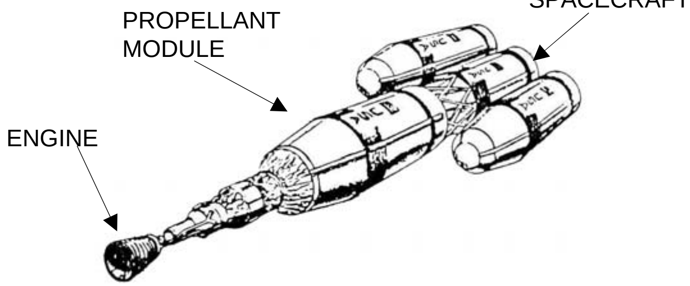
The NERVA Engine

Solid-Core Nuclear Engines

US nuclear solid-core rocket development during the 1956-1971 period demonstrated encouraging progress:

- 5,500°F exhaust temps.
- I_{sp} 850 seconds
- 250,000 lbs of thrust
- 90 min. burn time

Early Nuclear Rocket Performance Characteristics



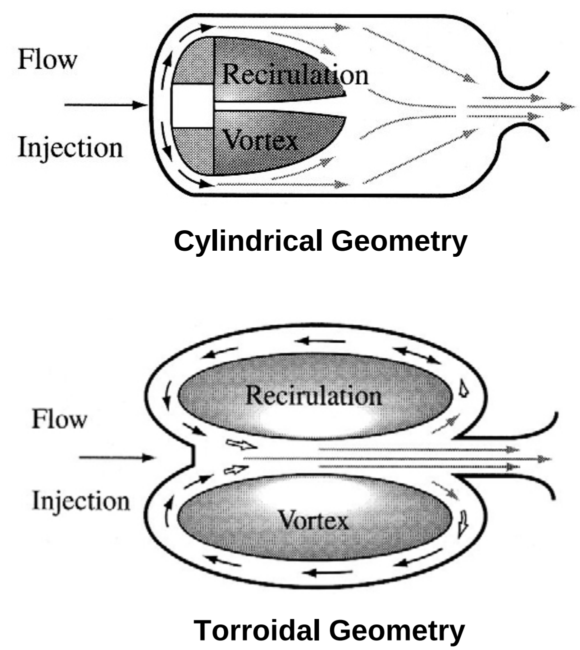
Spacecraft Concept with NERVA Engine

Phoebus Nuclear Rocket Engine

Solid-Core Nuclear Engines

Gas-core nuclear engine concepts were investigated during the 1960s and indicated large potential benefits including reasonable mass fractions for LEO applications, relatively high thrust and I_{sp} in the 2,000 second range:

- Radiant energy transferred from a high-temperature fission plasma to a hydrogen propellant drives the system:
  - The propellant temperatures can be significantly higher than the engine structural temperature.
  - In some designs the propellant stream is seeded with sub-micron particles (up to 20%) to enhance heat transfer.
  - Radioactive fuel loss reducing performance can be a major problem, particularly with open-cycle design concepts.
  - Closed-cycle approaches minimize or avoid fuel loss be containing the plasma in a quartz capsule which allows the radiation to pass through and be absorbed by the hydrogen.
  - The quartz wall and nozzle would be regeneratively cooled by the hydrogen propellant.

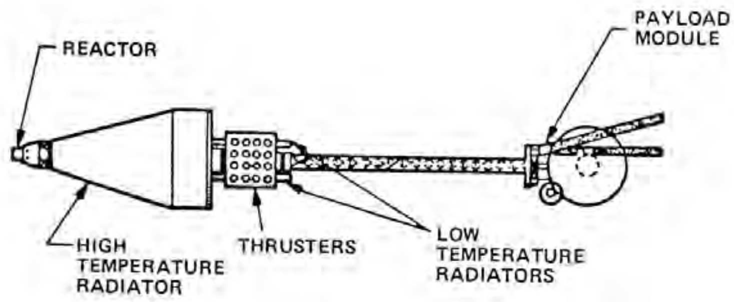


Closed- Cycle Designs

Gas-Core Nuclear Engines

Very small solar-powered electric propulsion systems are frequently used for orbital satellite maneuvering and station keeping. Large nuclear-powered versions may prove highly beneficial for human lunar/ Mars spacecraft applications:

- New nuclear-electric technologies are being developed to overcome fuel efficiency limitations of chemical liquid and solid fuel rockets:
  - Nuclear reactors are used to generate large amounts of electricity to ionize and accelerate a vaporized fuel (possibly cesium or hydrogen) to high speeds that are unobtainable by chemical combustion.
  - Thrust can be created by ejecting heat-induced high temperature, high velocity gas streams; beams of positively-charged ions accelerated by electric fields; or charged plasmas accelerated by powerful electro-magnetic fields.
  - Continuous high I_{sp}, low-thrust features can offer substantial fuel economies over "blast and coast" high-thrust chemical systems, but require more time to reach internplanetary velocities.



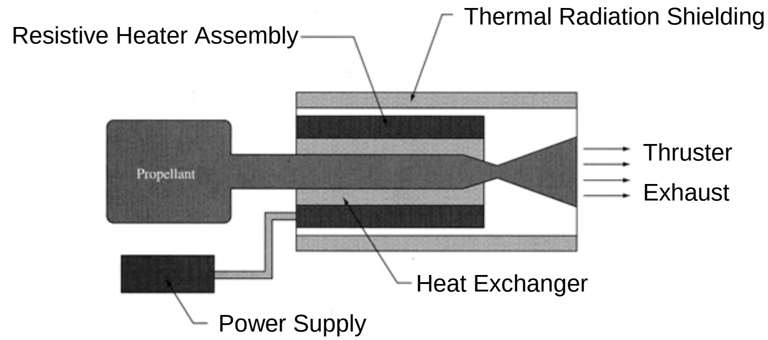
High temperature radiators reject heat from the reactor system that powers electrical energy turbines, and low temperature radiators remove exhaust heat from multiple thrusters. Payloads are mounted at a safe distance from the reactor to mitigate radiation hazards to crews and/ or electronic systems.

Schematic Spacecraft Concept

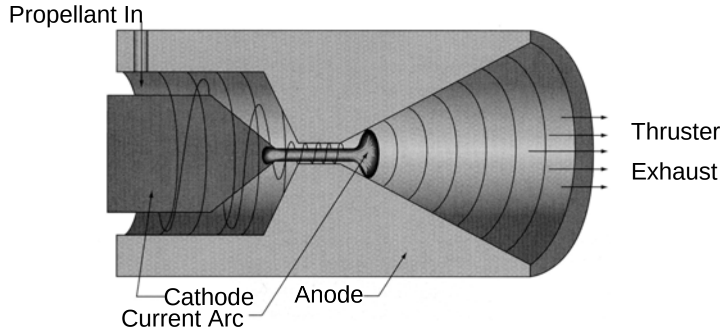
Nuclear Electric Systems

Electrothermal thrusters use resistive, arcing or microwave techniques to heat propellant electrically and then isentropically expand it through a convergent/ divergent nozzle:

- Resistojets use a resistive heater surrounded by a propellant heat exchanger to superheat the gas:
  - Thrust is created by ejecting a high temperature, high-velocity gas stream from the exhaust nozzle.
  - Many configurations have been developed, including some that are used routinely in space.
  - Propellant gases used include ammonia, biowastes, hydrazine and hydrogen (I_{sp} values around 300 seconds).
- Arcjets produce thrust by heating propellant with an electric arc and expanding the gas in an exhaust nozzle:
  - Of several configurations, the DC arcjet is most highly developed and is used in GEO satellites for north-south station keeping.
  - Ammonia, hydrogen and hydrazine are common propellants which are selected according to a variety of use-specific requirements.



Resistojet Thruster Concept

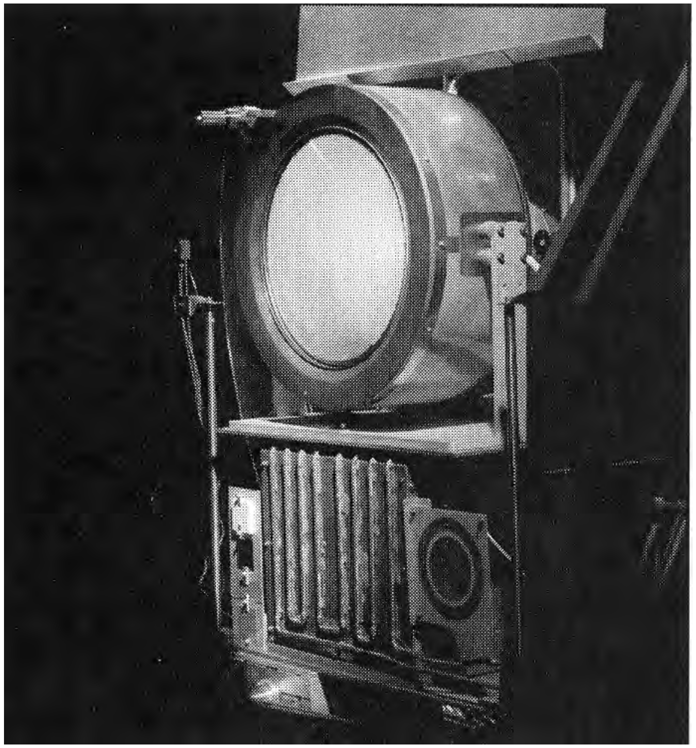


DC Archjet Thruster Concept

Electrothermal Thrusters/ Nuclear Electrical Candidates

Electrostatic ion thrusters use an ionized propellant that is accelerated through direct application of electric fields:

- Thrust is produced by accelerating a beam of positive ions through an electrostatic field to a high velocity:
  - Positive ions are produced by electron bombardment of neutral propellant atoms in a discharge chamber (typically a cylindrical anode with a central axial hollow cathode).
  - Thermionic emissions of electrons occurs when the cathode is heated.
  - A magnetic field in the discharge chamber increases the electron path length and residence time in the chamber, increasing the probabilities of energetic collisions between propellant atoms (such as xenon) and electrons.
  - The collisions remove electrons from the atoms creating positive ions.
  - A series of 2-3 perforated electrodes (called grids) attract, accelerate and focus the positive ions into a beam.

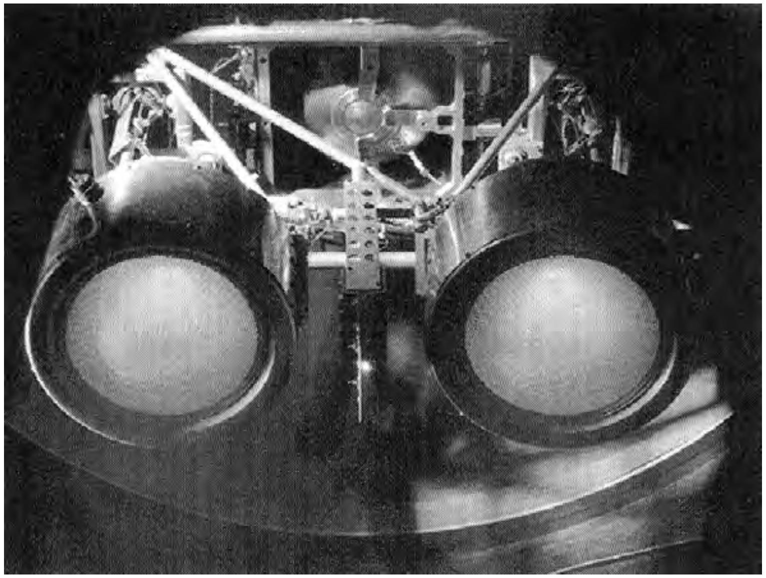


NSTAR Ion Thruster Test at NASA-JPL

Electrostatic Ion Thrusters/ Nuclear Electrical Candidates

Ion propulsion (as with all electric propulsion) is only feasible for large spacecraft applications that can provide large amounts of electrical power:

- The thrust of an ion engine is proportional to the square of the grid diameter:
  - A 30 cm diameter, 10 kW propulsion module developed and tested at the NASA Jet Propulsion Laboratory operating at 5 kW produced thrust on the order of 0.2 Newtons with an I_{sp} of 3,800 seconds using xenon.
  - Neutralizers are needed to eject electrons into the ion beam in the same numbers as ions to prevent spacecraft from developing a large negative potential charge.
  - Key life-limiting factors of ion engines are cathode life, grid erosion and spalling of spattered-deposited material that degrades the discharge chamber.
  - These limitations pose serious reliability risks for long-duration space exploration applications.

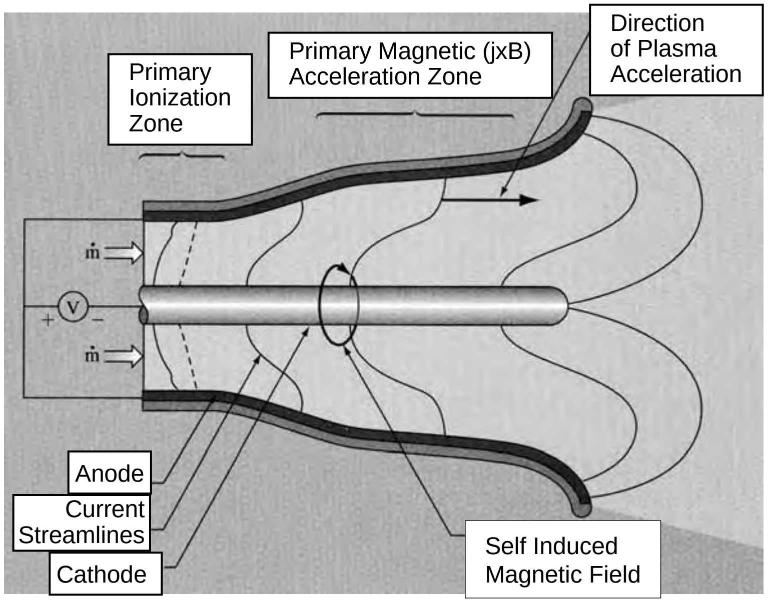


Ion Propulsion Module Test at NASA-JPL

Electrostatic Ion Thrusters/ Nuclear Electrical Candidates

Electrodynamic thrusters produce thrust using electric and magnetic body forces interacting with highly-charged plasmas, and include several different types.

- Magnetoplasmadynamic (MPD) thrusters are favored for large exploration-class applications involving high total power levels (IMW electric and greater) because of their substantial power-per-thruster rate (0.1-10 MW each) requiring fewer thrusters.
  - MPD thrusters have an axiosymmetric geometry (annular anode surrounding a central cathode) which generates a "Lorentz" body force to eject a high-velocity plasma stream.
  - A large current (1000s of Amps) flows between the coaxial electrodes to ionize and accelerate the propellant gas in either a steady-state or pulse mode.
  - The current induces a significant azimuthal magnetic field that axially accelerates the plasma-producing thrust.



MPP Thruster Elements and Operations

Electrodynamic Thrusters/ Nuclear Electrical Candidates

Additional information relevant to this section can be found in Part I of this SICSA Space Architecture Seminar Lecture Series titled Space Structures and Support Systems.

Key reference sources are the Cambridge Encyclopedia of Space (Cambridge University Press), the International Reference Guide to Space Launch Systems - Second Edition (AIAA Publication), Human Spaceflight Analysis and Design (The McGraw Hill Companies Inc.) and Spacecraft System Design and Operations (Kendall/Hunt Publishing Company). These and other relevant publications are listed below:

Astore, W., Giffen, R., Larsen, W., *Understanding Space: An Introduction To Astronautics*. 2nd Edition. The McGraw-Hill Companies, Inc., 2000

“Auxiliary Power Unit Manual:, AUX 2102, NASA, Johnson Space center, Texas, March 31, 2000

“Auxiliary Power Unit/ Hydraulic/ Water Spray Boiler Manual”, APU/HYD 2102, NASA, Johnson Space Center, Texas, April 20, 2001.

Brown, C.D. “Spacecraft Propulsion”, AIAA, 1996.

Damon, T.D. *Introduction To Space : The Science of Spaceflight*, 3rd Edition, Krieger Publishing Co., 2000

Fortescue, P., Stark, J., *Spacecraft Systems Engineering*. 2nd Edition, Wiley Publishing, 1995

Hill P.G, Peterson, C.R, *Mechanics and Thermodynamics of Propulsion*, Addison-Wesley Publishing, 1992

Houston, A., Rycroft, M., *Keys to Space- An Interdisciplinary Approach to Space Studies*, McGraw Hill, 1999

Humble, R.W, henry, G.N.,Larson, W.J, *Space propulsion Analysis and Design*, McGraw hill, 1995

“International Space Station Familiarization” ISS FAM C 21109, Rev B, October 18, 2001, NASA

“International Space Station Russian Segment Crew Reference Guide”, TD9901, August, 2001, NASA

Jahn, R.G., *Physics of Electric Propulsion*, McGraw Hill, 1968

Kuo, K.K., Sommerfield, M., eds. “Fundamentals of Solid propellant Combustion”, AIAA, 1984

Larson, W. Pranke, L., *Human Spaceflight- Mission, Analysis and Design.* 1st Edition, McGraw Hill, 1996

“Main Propulsion System Overview”, MPS OV 21002, NASA, Johnson Space Center, Texas, May 26, 1995

“Orbital Maneuvering System Manual”, OMS 2102, NASA, Johnson Space Center, Texas, March 13, 1996

“Reaction Control System Manual”, RCS 2102, NASA, Johnson Space Center, Texas, May 26, 1995

“Space Shuttle Vehicle Familiarization Manual”, SSV FAM 1107, NASA, Johnson Space Center, Texas, May 26, 1995

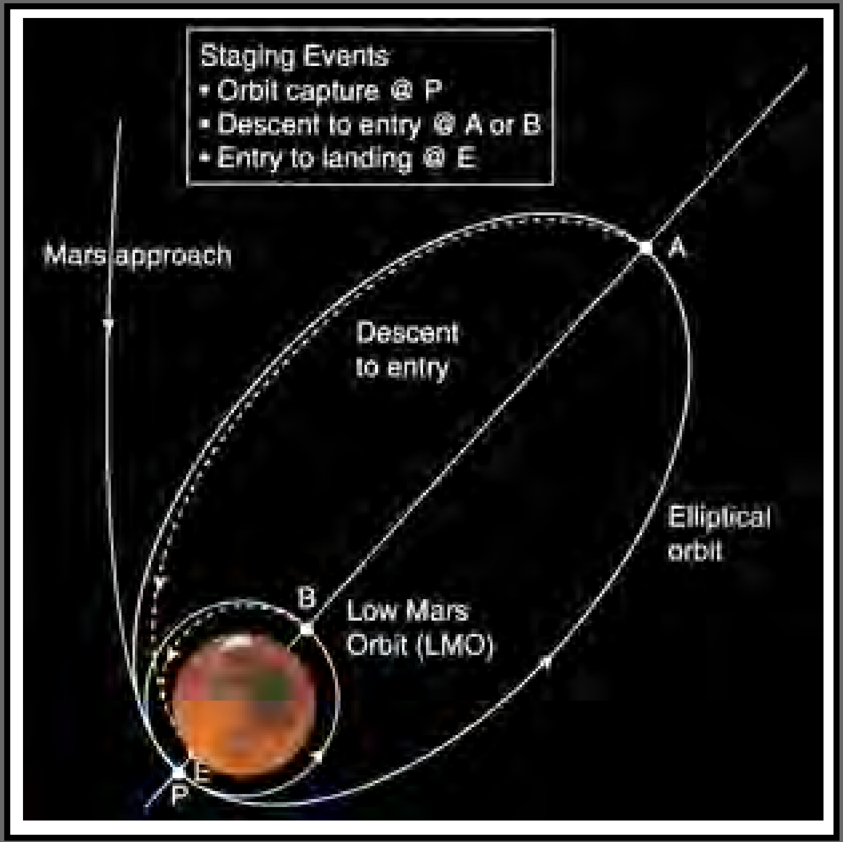
Sutton, G.P., *Rocket Propulsion Elements*, John Wiley and Sons, 1986

Timnat, Y.M., *Advanced Chemical Rocket Propulsion*, Academic press, 1987

Woodcock, G., *Space Stations and Platforms*, Orbit Books Company, Malabar, Florida, 1986

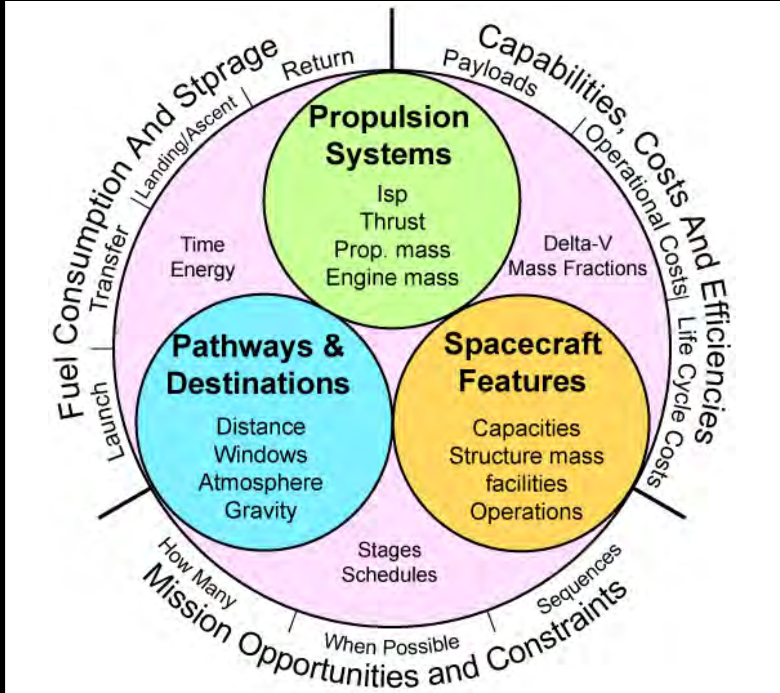
BACK TO THE LIST OF CONTENTS

SECTION C : PATHWAYS AND DESTINATIONS



Having discussed the vehicles and propulsion devices needed to access space, it is now appropriate to consider pathways and destinations that will govern what transport systems we might use, how much we can carry, how long the trips will require, and when we can leave and return:

- Pathways options are governed by natural laws, and are mapped in 4 dimensions of space and time:
  - Laws of physics establish the rules of the road that apply to behaviors of all natural elements and their manmade derivatives.
  - Principles of orbital mechanics determine ways to maneuver steep hills and valleys in the most rapid and efficient manner.
  - Solar System dynamics influence the distance and timing of departures, arrivals and returns so that we plan our schedules and know much to pack for the trips.

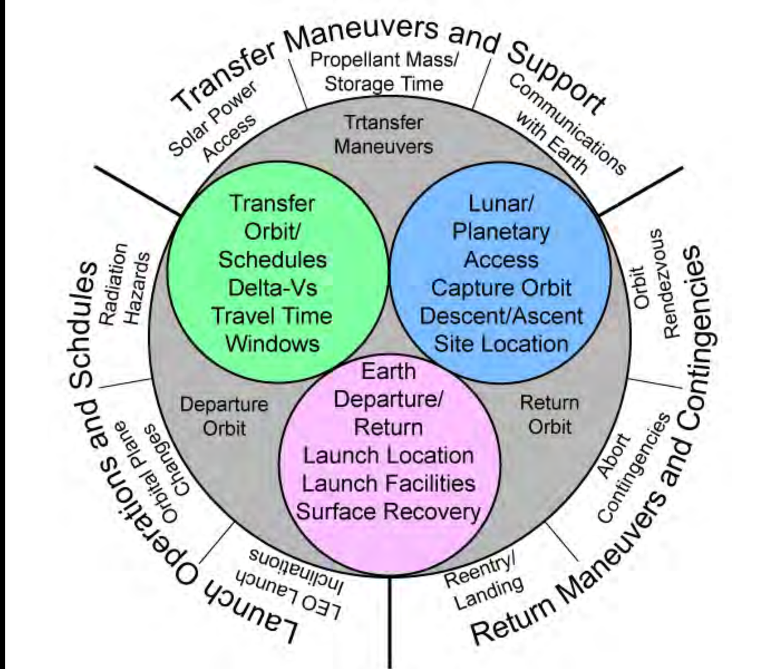


Interdependent Planning Factors

Mapping in Four Dimensions

Returning crews safely back to Earth following successful missions is always the goal, and returning them safety after disruptive emergencies is vital:

- Each pathway segment presents a variety of planning considerations:
  - Corrections of launch latitudes with necessary Earth and transfer orbit plane changes influencing fuel-costly maneuvers.
  - Transit periods through the Earth's trapped radiation belts and transfer trajectory exposures to potential solar proton storms that present hazards to crews and electronic systems.
  - Transfer and surface periods without line-of-site connections to photovoltaic solar power and Earth for communications.
  - Extended orbital pre-departure, transfer, surface operations and return periods that require means to prevent boil-off of stored gaseous propellants.
  - Contingency plans to ensure a way back in the event of a missed orbital rendezvous or any critical system failures.

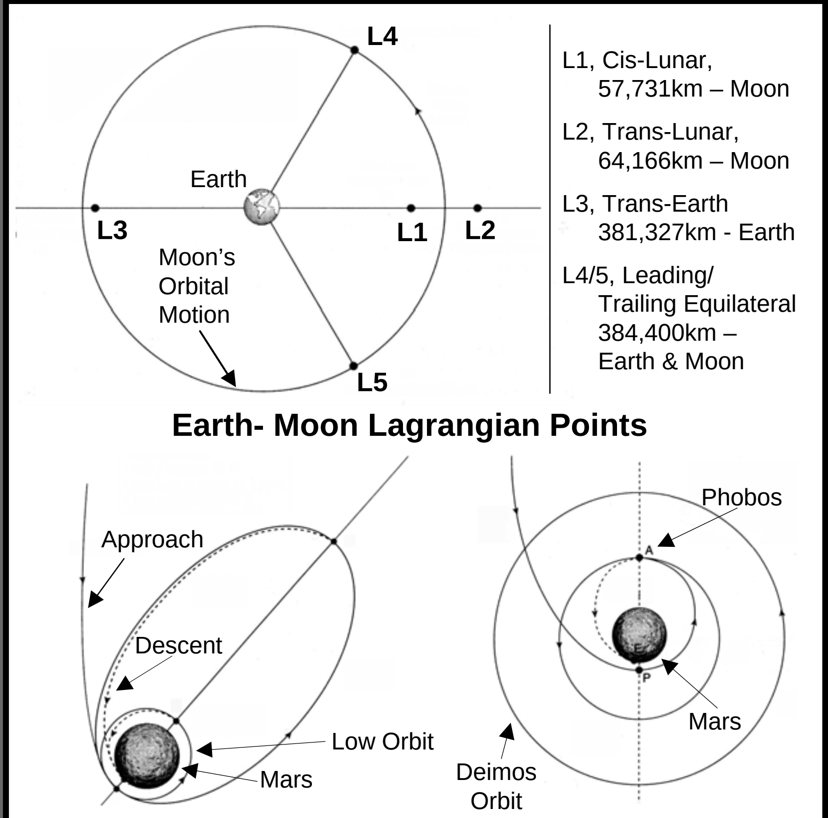


Origin and Destination Influences on Pathway Options

Earth: The Ultimate Destination

In space, the shortest distance between two points is never a straight line, and from an energy standpoint, often not even between closest points:

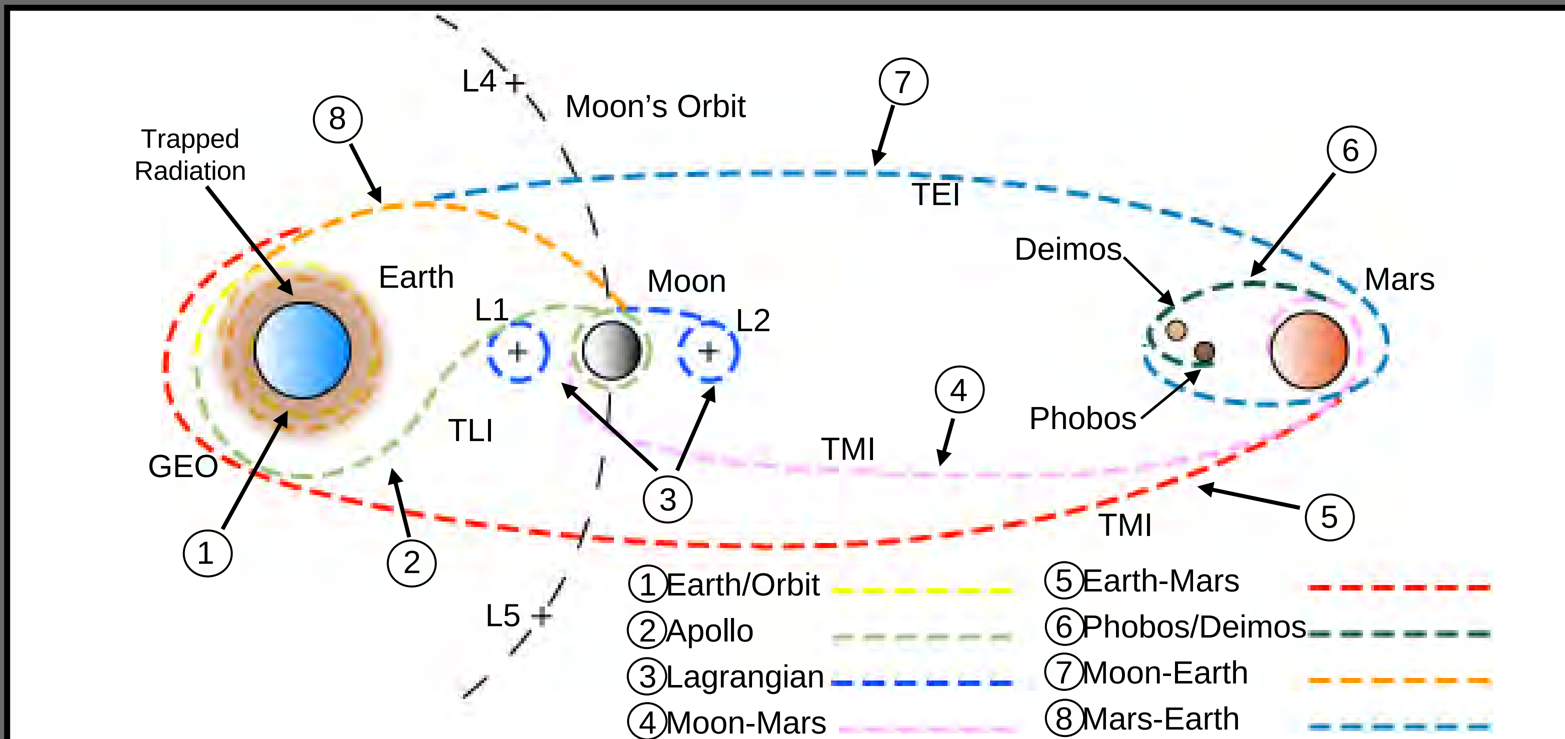
- Many different approaches for accessing important destinations have been proposed, each presenting particular priorities regarding where to go, what vehicles to use, and best ways to get there and back:
  - Some advocate the Moon as a place to establish permanent bases where resources can be harvested, used and exported.
  - Some primarily view the Moon as a technological and operational stepping stone for Mars missions.
  - Some believe that certain Earth-Moon Lagrangian points (or "libration" points) are ideal places for permanent space stations and/ or Earth-Moon and Earth-Mars transfers.
  - Others advocate direct Earth-Mars transits/ returns, or use of the Mars' moons Phobos and Diemos as Mars staging areas and material sources.



Earth- Moon Lagrangian Points

Earth- Mars and Earth- Phobos/ Mars

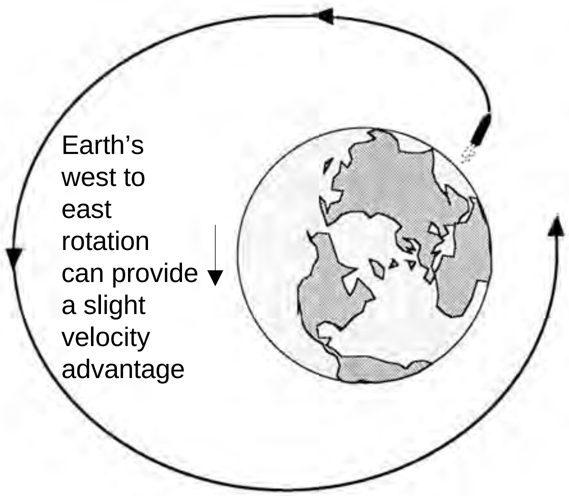
Destinations & Transfer Points



Mission Segments

Newton's 1st Law of Motion states that an object in motion will continue to move in s straight line unless some external force acts upon it:

- Above the surface of any planetary body, the external force of gravity will bend an object's trajectory towards the largest influencing mass (which may be the planet itself):
  - Since the Earth is spherical, a cannonball fired from the surface would strike the ground after traveling a relatively short distance.
  - If the projectile had sufficient speed it would never hit the ground, but would continue to fall in a curved path and never quite get there.
  - The speed necessary for a projectile's path to never fall back to Earth is 17,500 mph, a velocity that would cause it to vaporize from friction in Earth's dense near-surface atmosphere.
  - Spacecraft don't approach such speeds until they are above the denser part of the atmosphere and pitch over horizontally to accelerate more rapidly.



Spacecraft launch vertically to get through Earth's dense atmosphere as quickly as possible, then pitch over to horizontal to rapidly accelerate to an orbital velocity of 17,500 mph.

Earth Ascent and Acceleration

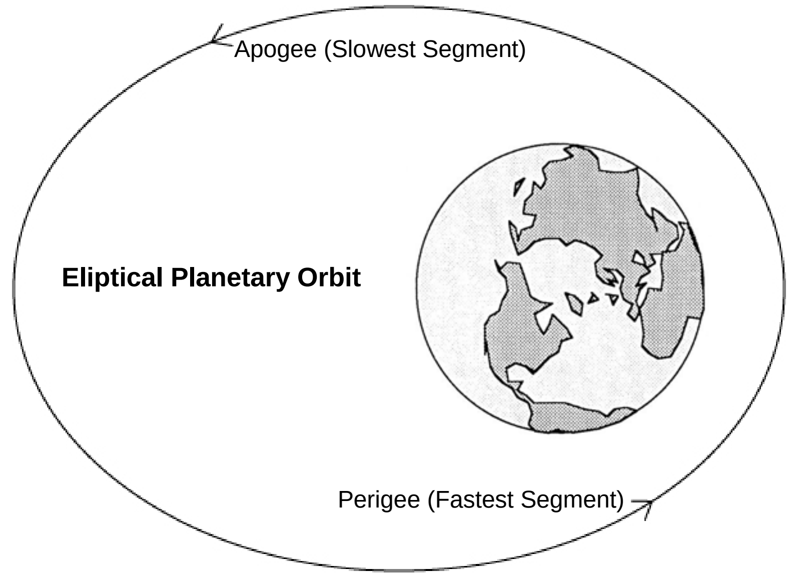
Achieving Orbital Velocity

Conservation of energy and momentum and basic laws governing orbital mechanics apply to the orbital motions of spacecraft and planets, where total energy and total momentum are constants:

- Components of energy are potential energy and kinetic energy:
  - Potential energy is a function of an object's position (or altitude) and mass.
  - Kinetic energy is a function of an object's mass and velocity.

Total Energy= Potential Energy + Kinetic Energy= Hw + ½ mV2
Where: H= altitude, w= weight, m=mass and V=velocity

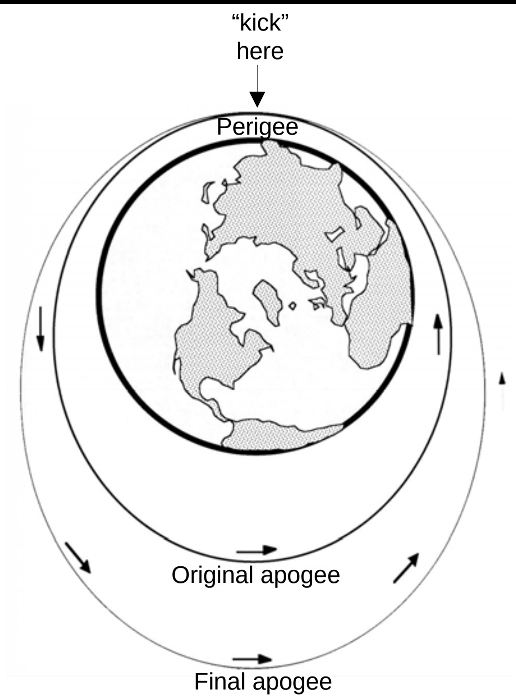
- Based upon exchanges between energy and momentum, a spacecraft's orbital altitude can be raised or lowered by changing its kinetic energy:
  - Altitude at perigee can be increased by boosting kinetic energy (velocity) at apogee, which increases the orbit's eccentricity.
  - Decreasing velocity at perigee, or boosting velocity at apogee will decrease the orbit's eccentricity (reducing the prerigee).



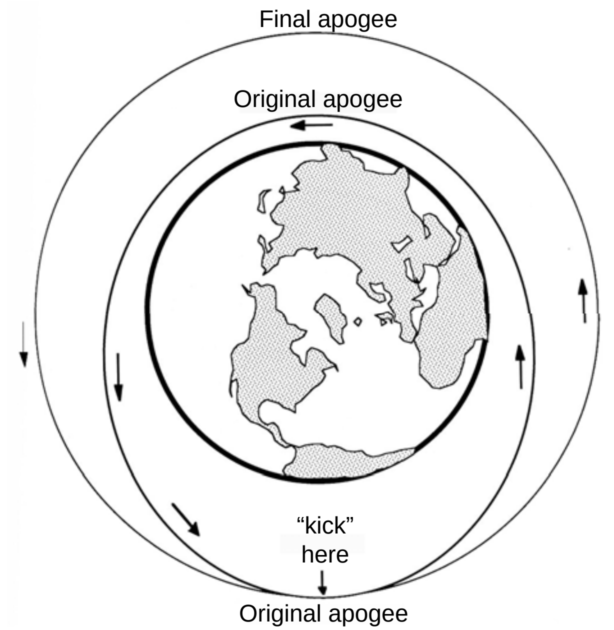
Like a ball tossed up, gravity slows it's velocity as it rises and accelerates it as it falls back.

Earth Ascent and Acceleration

Conservation of Energy and Momentum



Perigree Kick to Increase Apogee

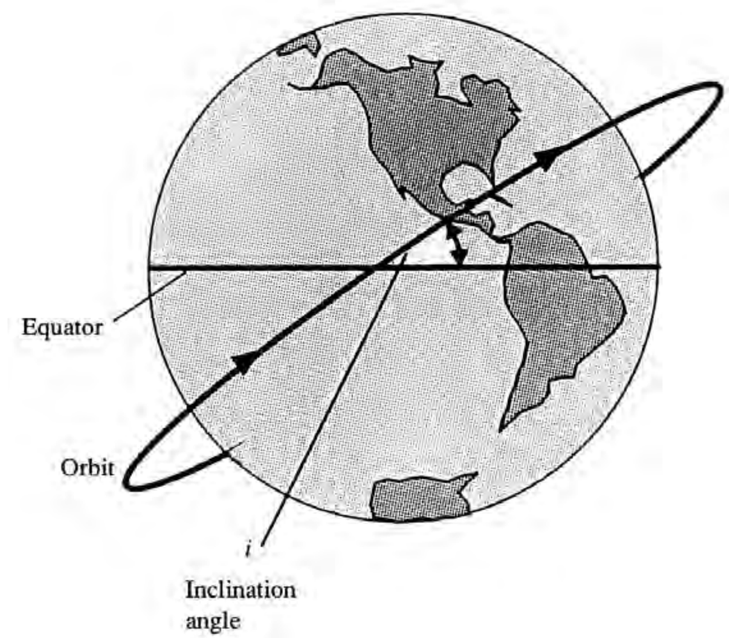


Apogee Kick to Decrease Perigree
(or Reverse Perigree Kick to Reduce Apogee)

Velocity Relationships to Orbit Shapes

An orbit's inclination is defined as the angle between the Earth's equatorial plane and the spacecraft's orbital plane:

- The angle is measured clockwise from the equator at a place where the spacecraft crosses the equator heading north.
  - The inclination is 0° at the equator, and 90° over the poles.
  - Since inclinations are additive and progressive from the equatorial plane, they can exceed 90°.
- A spacecraft's launch site latitude position influences which orbit inclinations are most readily accessible:
  - Near-equatorial latitudes benefit most to take advantage of the Earth's west-to-east rotation rate of about 1,035 mph for a jump start.
  - The Kennedy Space Center's easterly velocity provides about a 5.3% advantage over total velocity required to reach low-inclination LEO.

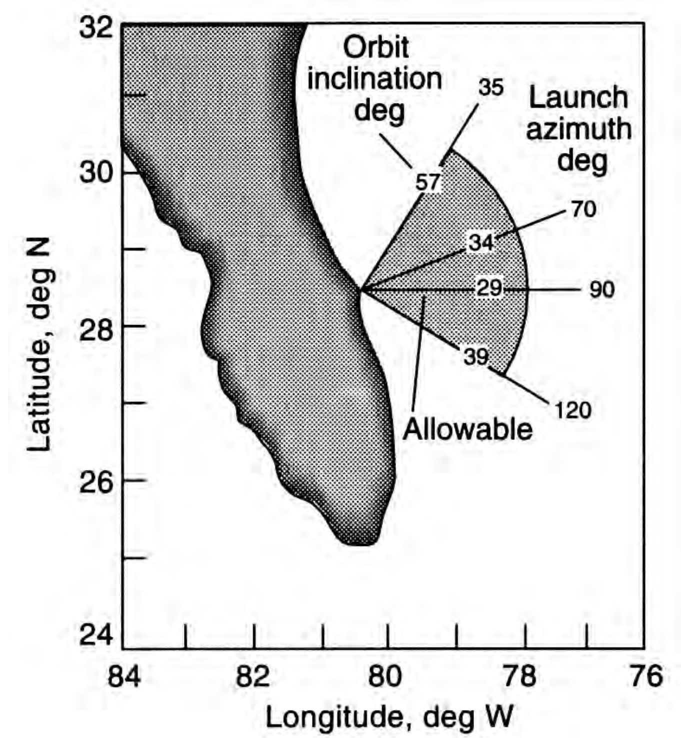


Angle Inclination Measured From the Equator Clockwise heading North

Orbital Plane Inclination Angles

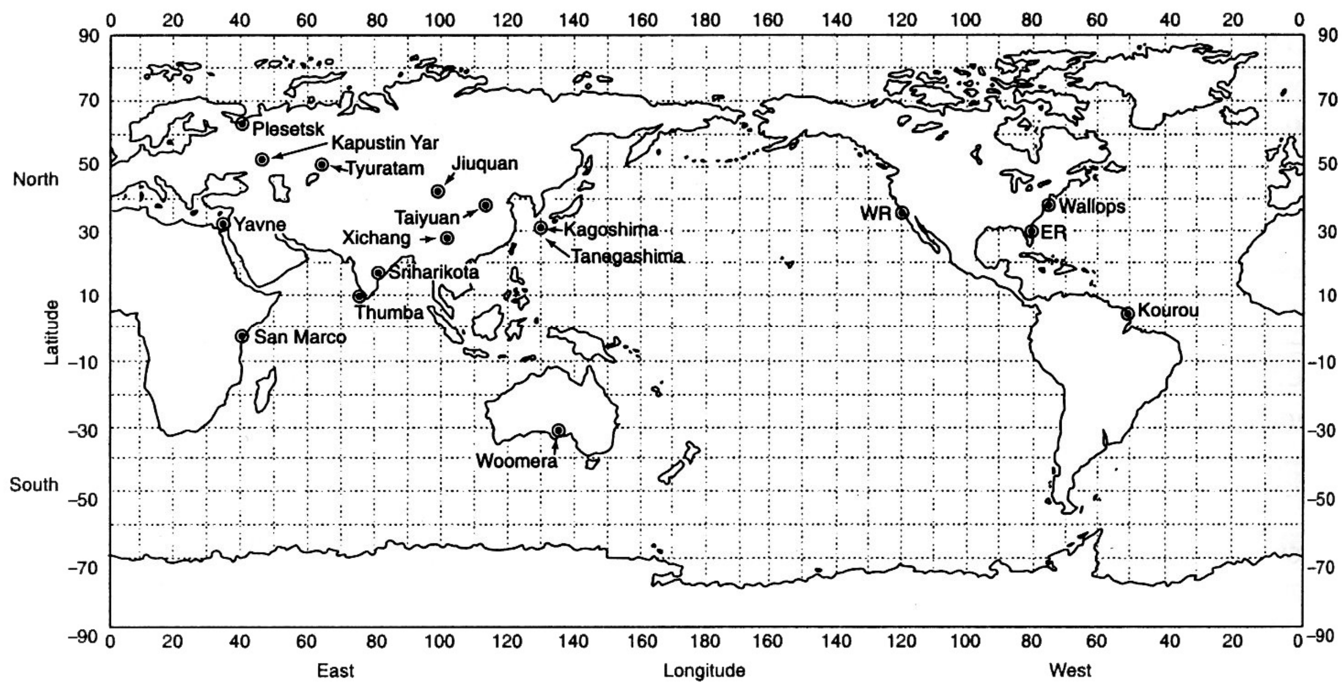
Launch sites influence vehicle options and accessible inclination orbits:

- All launch sites must have clear down-range corridors:
  - Launches from the US East Coast (Kennedy Space Center) are suitable only for low-inclination launches with azimuth trajectories that avoid emergency overflight hazards to population centers.
  - Shuttle launches also require a suitable landing strip with acceptable wind/ weather conditions and special vehicle/ payload processing facilities.
  - US high-inclination launches use the West Coast Vandenberg Air Force Base to avoid population overflights.
- Site latitudes influence the amount of propulsion required to deliver vehicles/ payloads to different inclination orbits:
  - Near-equatorial latitudes require less energy to access low-inclination orbits, or can deliver larger payloads with a given amount of propellant.
  - High-inclination launches lack the advantage of using the Earth's rotation to gain a velocity boost.



Orbital Inclinations Available from KSC

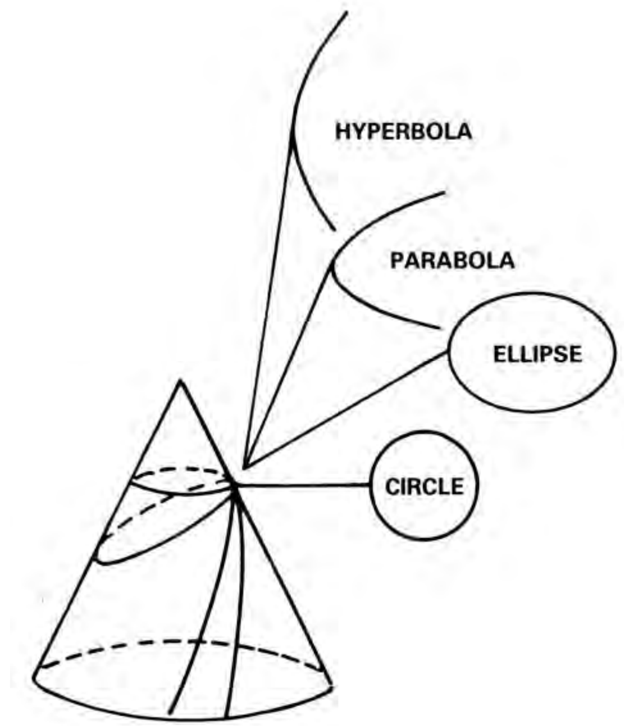
Launch Locations & Trajectories



Global Launch Sites Global Launch Sites

Kepler's 1st Law of Planetary Motion defines characteristics of orbit geometries that conform with gravitational influences:

- If two bodies interact gravitationally, each will describe an orbit that is a conic section about the common mass of the pair:
  - Sections at less than the conic half-angle are ellipses.
  - Orbits of planets are ellipses with the Sun as the foci, and orbits of planetary satellites are elliptical (or circular), with the planetary bodies as the foci.
- If two bodies are permanently associated with each other, their orbits will be hyperbolas or parabolas (open curves):
  - Those exactly at the half-angle are parabolic, representing flight paths that are exactly at the orbital escape energy.
  - Those that are steeper than the half-angle represent energies greater than escape energy.

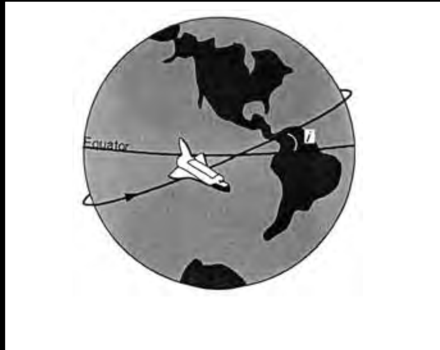


Conic Sections

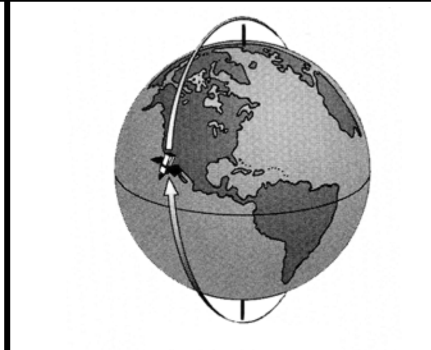
Gravitational Geometric Influences

Earth orbits range in altitude from about 175 miles (circular LEO) to 25,000 miles (high-eccentricity apogee):

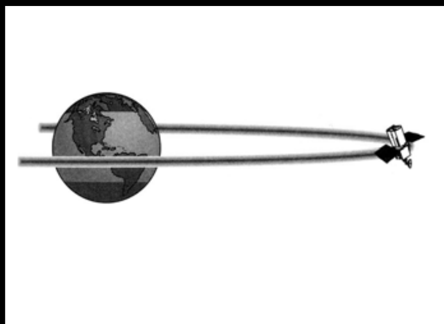
- LEO low-inclination are easiest to reach and have many useful applications:
  - Microgravity conditions enable unique materials and life sciences research not possible on Earth.
  - Higher resolution Earth sensing is possible from LEO.
  - LEO can be used to assemble and/ or launch spacecraft to higher and interplanetary orbits.
  - Problems in LEO include orbital debris hazards and periodic reboost requirements to prevent orbital decay caused by drag.
- LEO polar, geosynchronous and high-eccentricity orbits are principally used for unmanned satellites:
  - Geosynchronous orbits place satellites in geostationary positions over points on Earth indefinitely.
  - High-eccentricity orbits place satellites over northern regions not covered by GEO where they can hover during long apogee segments of each orbital period.



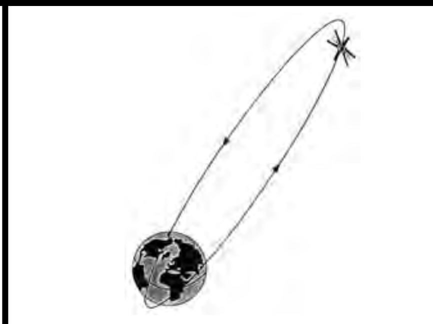
LEO Low-Inclination



LEO Polar



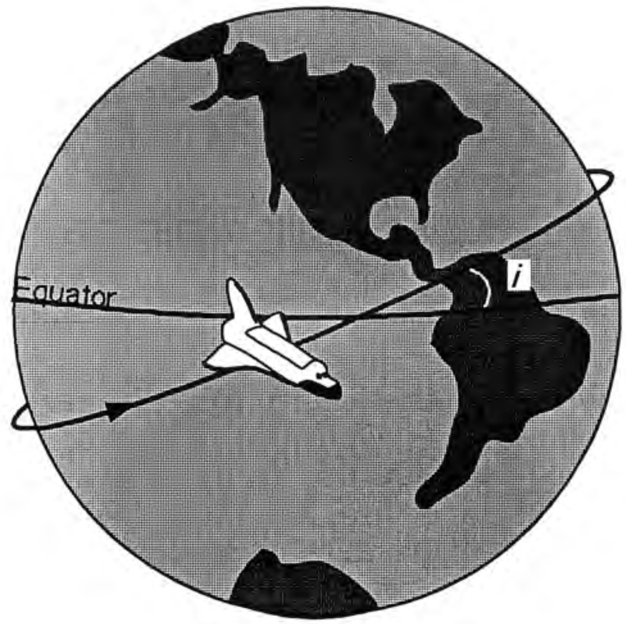
Geosynchronous



High-Eccentricity

Low Earth Orbits (LEO) ranges from about 175-600 miles above the surface, and are nearly circular, with a period (orbital completion time) of about 90 minutes:

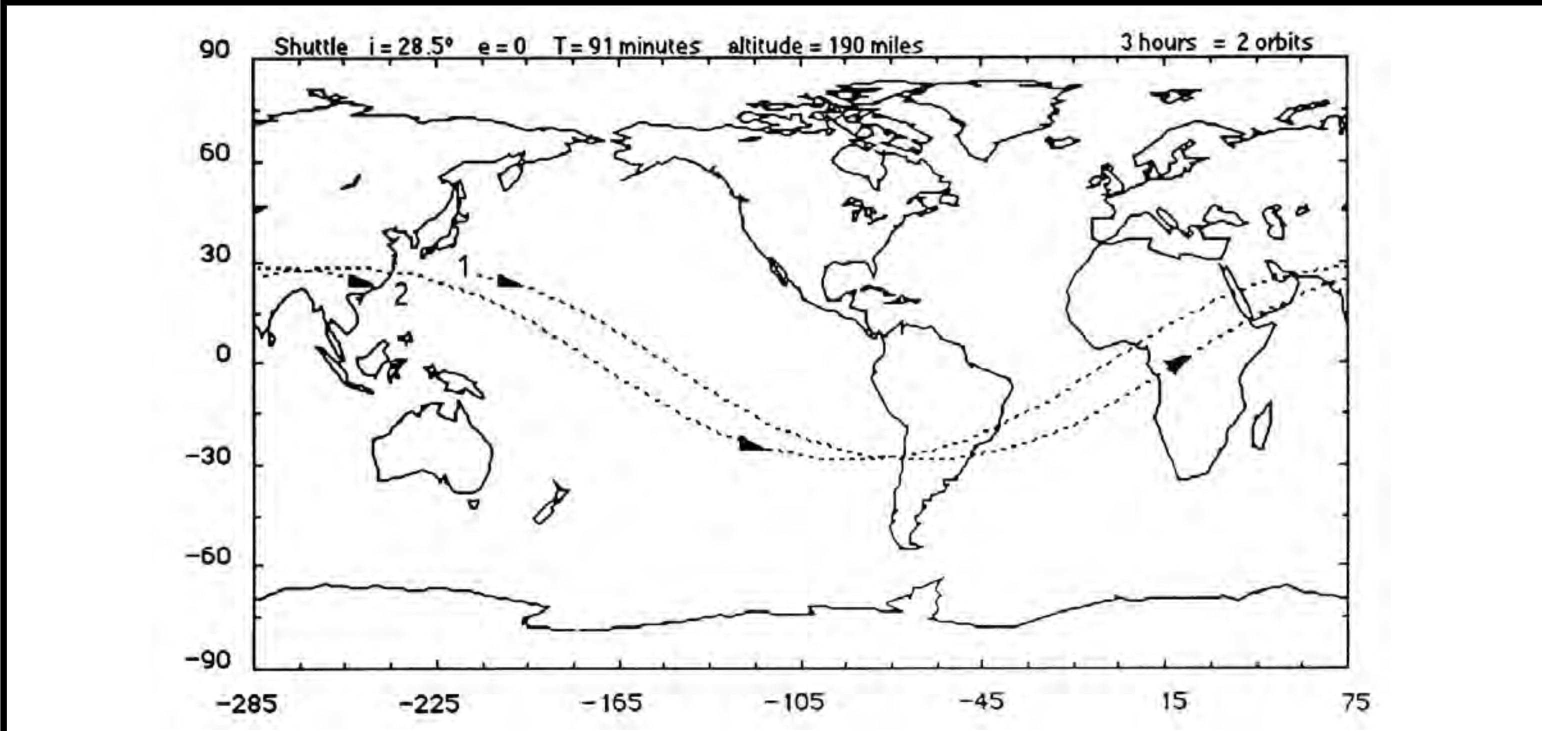
- LEO affords certain advantages:
  - Located above most of Earth's atmosphere, it presents little drag that must be compensated.
  - Nearly weightless conditions are desired for unique natural and material sciences.
  - Low-inclination orbits are accessible from many international sites for space station, astronomical and Earth research, and platforms for higher orbit launches.
  - Polar, high-inclination orbits pass over the entire Earth's surface in a few days at altitudes of about 500 miles and periods of about 100 minutes (90° inclination) which is useful for Earth observation. At 98° a satellite becomes "sun synchronous", passing over the Earth at the same local time on each westward pass.



A typical Shuttle orbit at 190 mile attitude inclined at 28.5° with a period of 91 minutes.

Low-Inclination Orbit

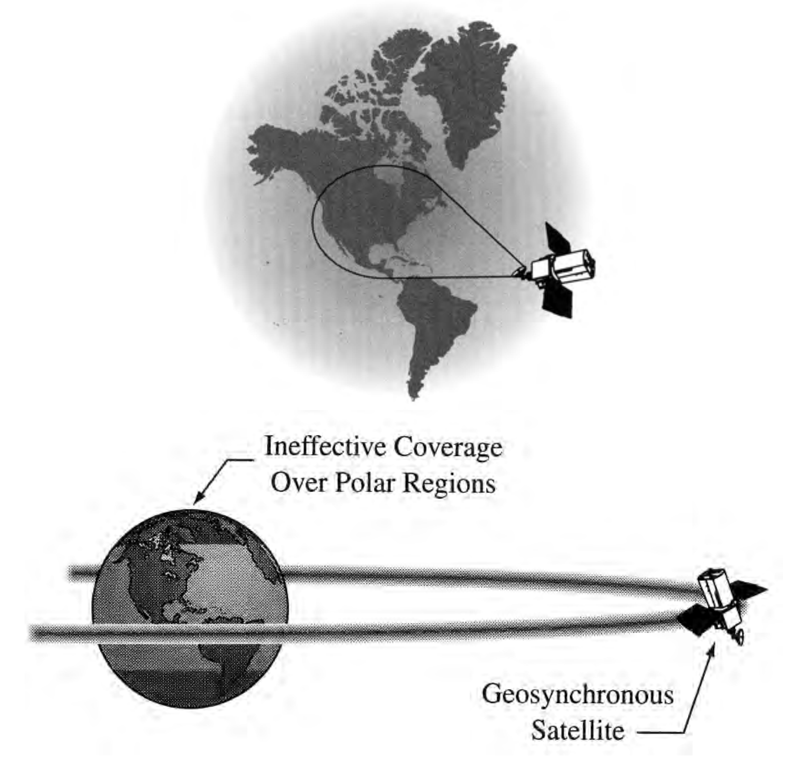
Low Earth Orbits



28.5° LEO Ground Track

Geosynchronous (or "geostationary") orbit (GEO) is a circular, low-inclination orbit that has the same Earth rotational period of 23 hours, 56 minutes, 4 seconds which occurs at a distance of 23,400 miles (35,800 km, or 5.6 Earth radii):

- Satellites placed in GEO remain in a fixed position over a given place (a geostationary position on Earth):
  - They are principally used for communications satellites, including TDRSS, satellite television, large-scale weather satellites, and missile attack detection and warnings.
  - A GEO satellite transmitter can cover nearly one-third of the Earth's surface, and can remain in place indefinitely with little station keeping so long as its systems function.
  - A limitation is an inability to cover populated northern regions of the globe.

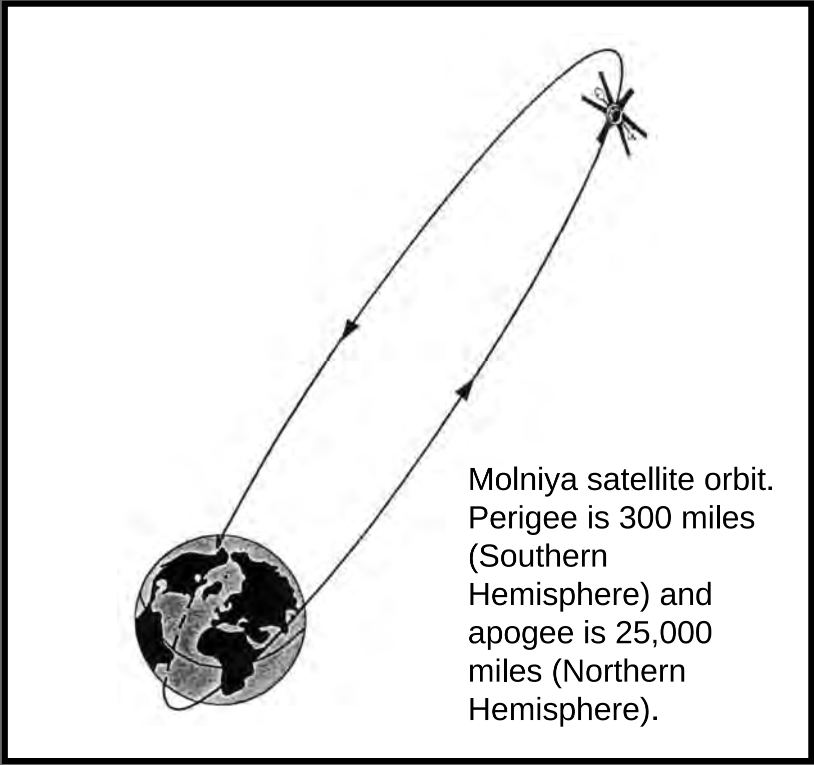


Coverage Over the USA

Geosynchronous Orbit

Elliptical orbits can be designed for a wide variety of applications including planetary rendezvous:

- The eccentricity, size and inclination can be tailored to special mission requirements:
  - A useful feature is to enable the apogee of a communications satellite to hover over a far north area on Earth which cannot be seen by an equatorial GEO satellite.
  - A 300 mile perigee, 25,000 mile apogee Russian Molniya communications satellite remains nearly stationary near the GEO altitude for more than 8 hours over Siberia.
  - It then speeds up, drops down past perigee and returns to apogee in less than 4 hours.
  - Continuous 100 percent coverage is provided using 2 satellites operating in a constellation.

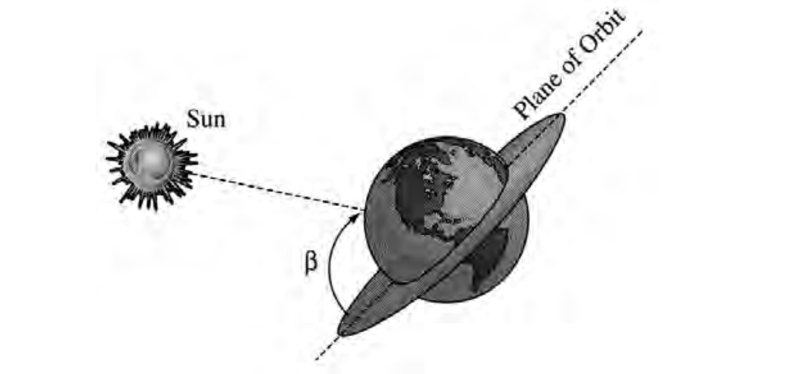


High-Eccentricity Orbits

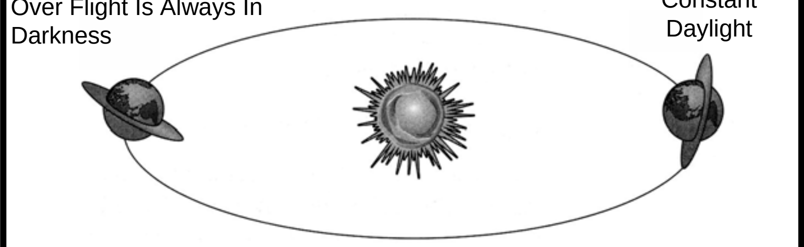
The inclination plane of an orbit relative to the Sun effects spacecraft thermal loads and power options:

- The solar beta angle (B) effects the spacecraft's thermal environment and photovoltaic solar-cell generation ability, and influences responsive vehicle altitude orientation maneuvers to optimize these conditions.

- Flights over the Northern Hemisphere are always dark at inclinations corresponding to a retrograde orbit (plane aligned with the Sun), and experience constant daylight at posigrade orbital inclinations.



Solar Beta Angle



Retrograde and Posigrade Inclinations

Beta Angles & Inclinations

Spacecraft altitude orientation to the orbital plane also has many application influences:

Earth orientation constantly maintains a spacecraft altitude pointing in the direction of Earth.

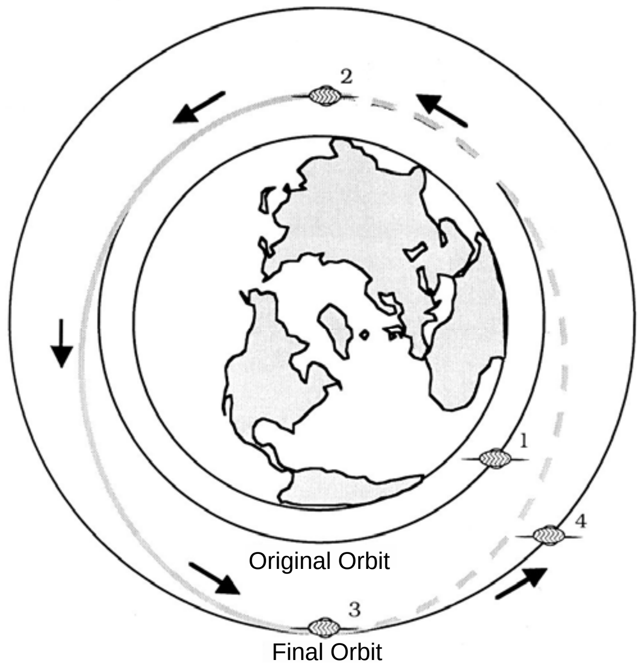
Inertial orientation maintains a fixed attitude with respect to a space object such as the Sun or a star.

Earth Oriented	Inertial
+ Favorable for Earth observation and communications	+ Favorable of Astronomy
+ Allows to use gravity gradient for altitude control	+ Simplified collectors and radiators
+ More flexibility for microgravity experimentation	+ Constant light conditions for EVA
+ Earth is a reference for crew orientation (EVA)	+ Constant thermal control conditions
+ Easier rendezvous and docking operations	– Gravity gradient is always a perturbation
+ More mass distribution flexibility for growth	– Difficult to keep optimal mass distribution
– Need solar array and radiator tracking	
– Variable lighting conditions (EVA)	

Spacecraft Orientation

Maneuvers to raise spacecraft altitudes and velocities are applied in the Earth orbit and lunar/planetary transfer orbit applications:

- The most energy-efficient approach to change from a lower orbit to a higher one in the same plane is to use "Hohmann" transfers named after a German who conceived the idea in 1925:
  - First, spacecraft engines are ignited to change a circular orbit to a longer elliptical shape.
  - At apogee, engines fire again to circularize the orbit.
  - Hohmann transfers can place spacecraft on a co-planar trajectory to the Moon, maneuver which was used for US Apollo missions.
  - The maneuver can also be applied for some Mars missions.

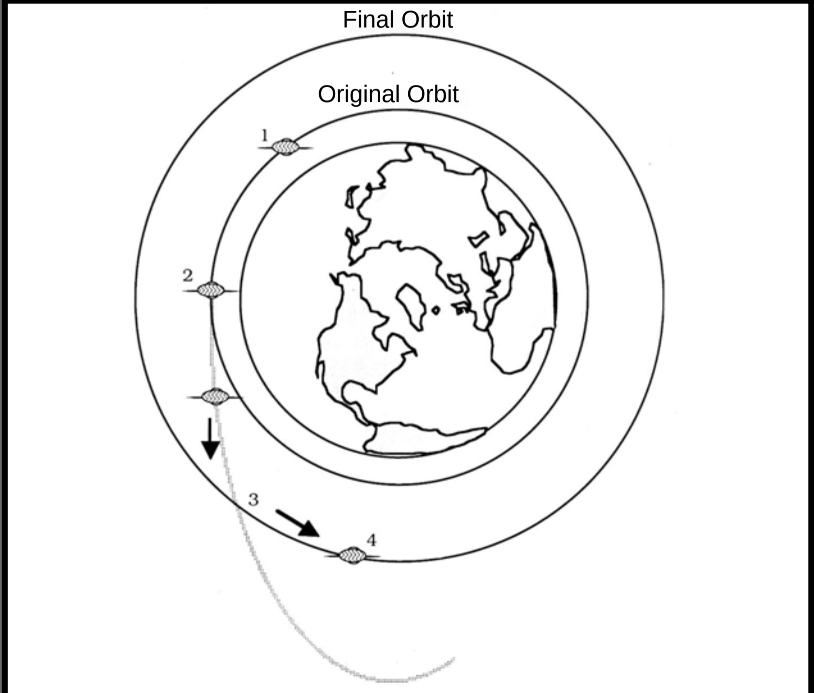


Hohmann Transfer to a Higher Orbit

Transfers to Higher Orbits

An alternate “fast transfer” approach to raise orbital altitudes is more rapid than the Hohmann maneuver, but also is much less efficient:

- The maneuver requires two separate engine ignitions:
  - The first stage involves a more energetic engine firing than Hohmann in the direction of the velocity vector to create a larger elliptical orbit that intersects the desired circular transfer orbit.
  - The second stage fires engines again with even greater energy to alter both the vehicle’s speed and direction to “turn the corner” into the transfer orbit vector.
  - Fast transfers can be used in emergency situations which require very rapid orbit changes that are not possible using the Hohmann maneuver.

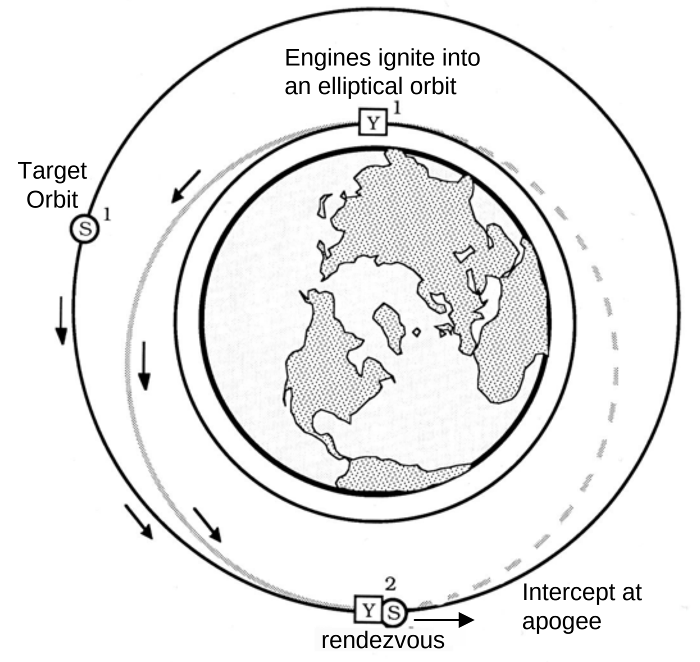


Fast Transfer to a Higher Orbit

Transfers to Higher Orbits

Orbital rendezvous techniques are basically Hohmann transfers that take advantages of faster lower orbits and slower higher orbits to adjust phasing:

- Adjustments require careful alignments of inertial orbit planes, with interceptor spacecraft launched when Earth's rotation is at the right point relative to the target's plane (windows that are sometimes only minutes long):
  - If the target is positioned ahead of an interceptor in the same orbit, the interceptor is put into a smaller phasing orbit with a shorter period to catch up.
  - If the target is behind the interceptor, the interceptor increases velocity to enter a higher, slower orbit that allows the target to catch up.
  - "R-Bar" approaches intercept the target along an Earth radial vector.
  - "V-Bar" approaches intercept the target along the velocity vector.



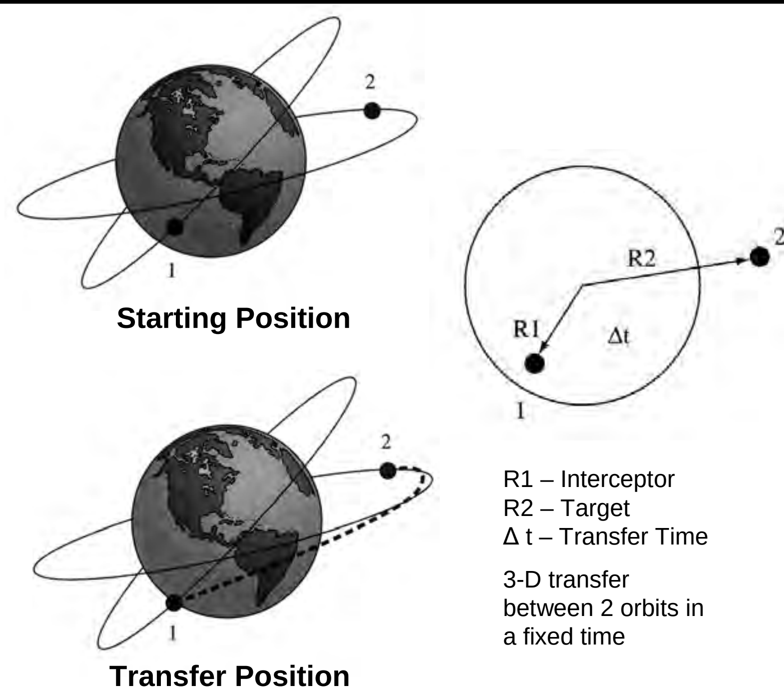
Rendezvous Using a Hohmann Transfer

Rendezvous Maneuvers

Rendezvous is more difficult for elliptical and non-coplanar orbits, but procedures are essentially the same:

- A common technique often used by the Space Shuttle permits 3-dimensional transfers between two non-coplanar orbits in a fixed time which is computed according to "Lambert's Theorem":
  - Step 1 determines the delta-V needed to accomplish the intercept.
  - Step 2 determines the delta-V required for the interceptor to achieve the target orbit.
  - Step 3 determines the delta-V required for the interceptor to approach and meet the target.

Course burns are used for large orbital changes, and several small burns are used for intercept timing.

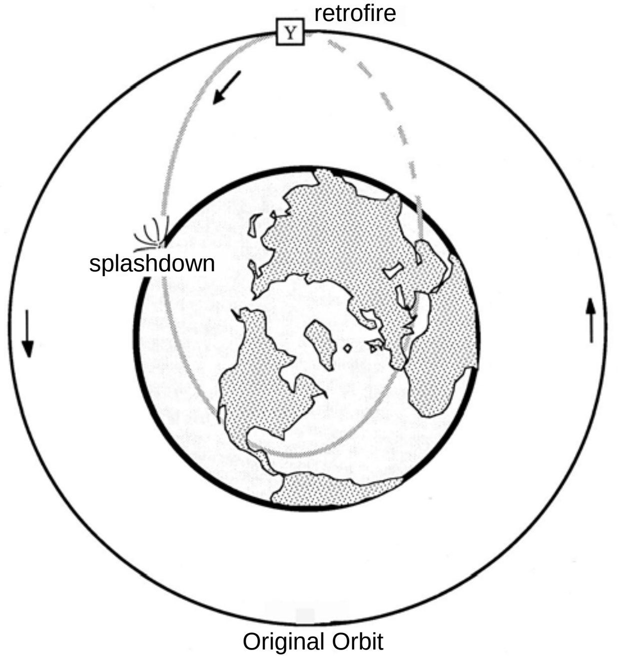


Lambert's Transfer Technique

Non-Coplanar Intercepts

Earth crew returns must provide for all possible emergency contingencies:

- The Space Shuttle Orbiter was designed with a more than 1,000 mile cross-range flight capability to enable Abort Once Around (AOA) return from polar orbit launches (which correlates with the Earth's rotational velocity):
  - Reentry and landing back on Earth are accomplished by turning the vehicle around and firing engines to apply thrust in the opposite direction.
  - Retrofiring reduces the spacecraft's energy, transferring it to a lower elliptical orbit which intersects Earth, increasing the vehicle's speed as it falls.
  - Timing and length of burn must be precise so that the descent orbit intersects Earth at the desired landing spot.
  - Atmospheric drag that distorts the reentry orbit from a true ellipse must be taken into account in the calculations.

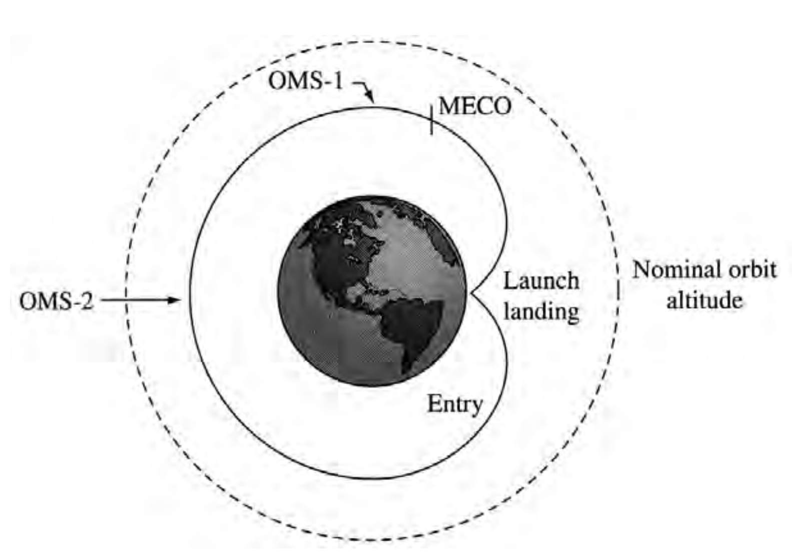


Typical Propulsive Deorbit

Deorbit Maneuvers

The Abort Once Around (AOA) is used in cases where a major system problem following launch makes it necessary to land the vehicle rapidly:

- The AOA enables the spacecraft to circle the Earth once for a period of about 90 minutes and make a normal reentry and landing:
  - In the AOA abort mode an OMS thrusting sequence is made to adjust the post-Main Engine Cutoff (MECO) orbit so that a second Orbital Maneuvering System thrusting sequence will enable the vehicle to land.
  - Shuttle Orbiter landing options are White Sands, New Mexico, Edwards Air Force Base and the Kennedy Space Center.
  - Examples of possible abort emergencies include loss of cabin pressure, electrical power failures, and when insufficient OMS propellant is available to achieve orbit.

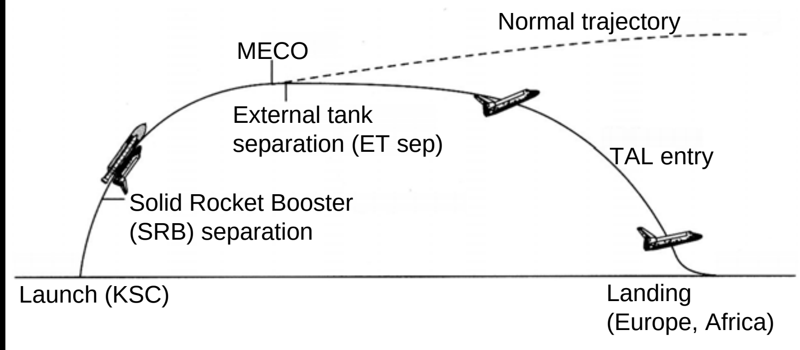


Orbiter Abort Once Around Profile

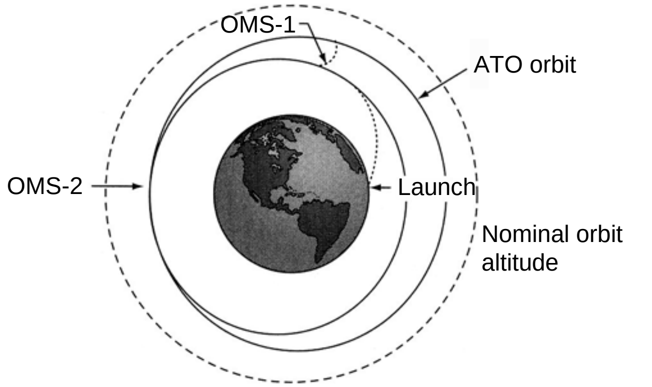
Abort Once Around (AOA)

Emergency abort modes also enable the Shuttle Orbiter to land at sites in Europe and Africa, or to boost to a safer orbital altitude:

- A Transoceanic Abort Landing (TAL) can occur if the vehicle lacks sufficient speed to achieve a nominal orbital trajectory following MECO:
  - The TAL landing site is selected based upon the trajectory inclination, and lasts about 35 minutes following liftoff.
  - This mode involves a ballistic trajectory which does not require an OMS maneuver.
- An Abort to Orbit (ATO) mode is used to achieve a safe orbital altitude if a vehicle cannot reach the planned orbit:
  - An example would be if adequate orbital insertion speed is not achieved in the region of MECO.
  - OMS engines would be used to circularize the lower orbit.



Orbiter Transoceanic Abort Landing

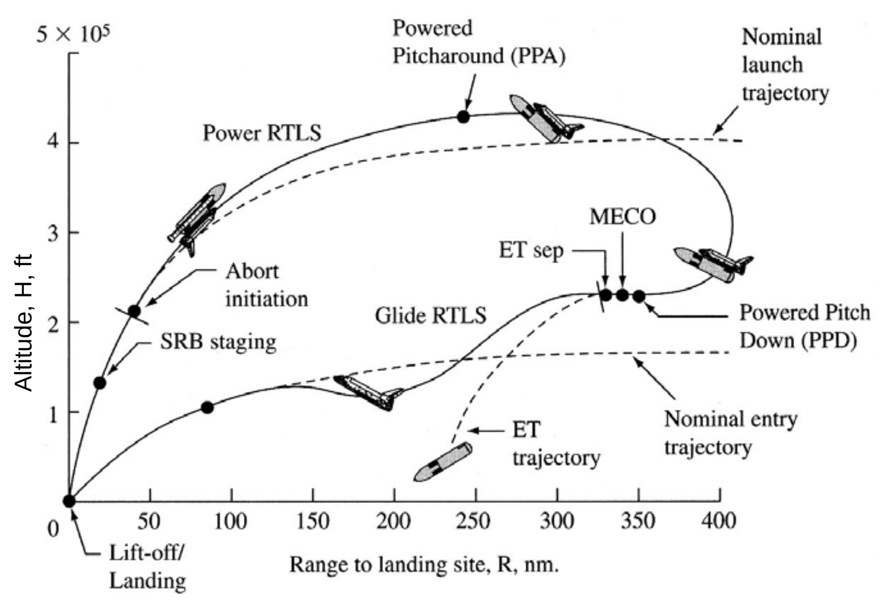


Orbiter Abort to Orbit Profile

TAL & AOA Abort Maneuvers

A Return to Launch Site (RTLS) abort option would be executed in the event an engine failure or other major system malfunction occurred within about 4 minutes and 20 seconds into a launch:

- This must take place while there is still sufficient Main Engine propellant remaining to enable the return, and consists of 3 stages:
  - A powered stage takes place when the Main Engines are still thrusting to a Powered Pitcharound (PPA) point above the nominal launch trajectory.
  - An External Tank (ET) separation stage occurs immediately following MECO and a preceding Powered Pitch Down (PPD) maneuver about 350 nautical miles from the launch/ landing site.
  - A glide phase enters its trajectory at an altitude below the nominal entry trajectory about 150 nautical miles from touchdown.



Shuttle Orbiter Return to Launch Site Profile

Return to Launch Site Abort

Velocities necessary to put spacecraft on hyperbolic escape velocities that transfer them to interplanetary space are determined by the “depth” of the celestial body’s “gravity well” which must be overcome:

The velocity can be calculated by the formula

$$V = \sqrt{\frac{2Gm}{r}}$$

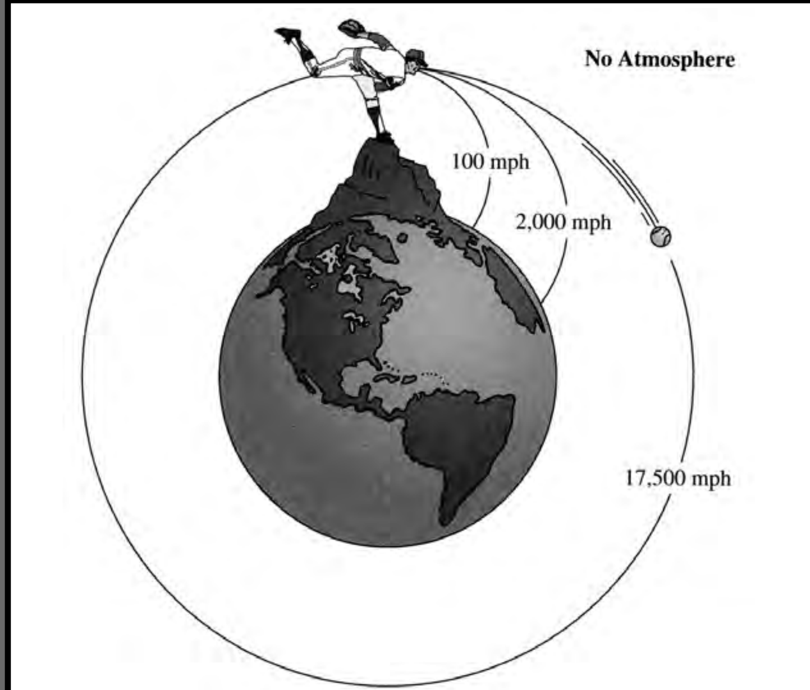
Where r = the distance from the center of the body and Gm = the gravitational constant.

- Applying this formula, the escape speed from the surface of the Earth (r) = about 4,000 miles and v is calculated in miles per hour, so that

$$V(Earth) = \sqrt{\frac{(2)(1.\ \times 10^{12})}{24000}} = 24,900 mph$$

(7mi/se 11km/sec)

- Applying the formula to lunar and Moon returns:
  - -Escape from the Moon (2,172 miles diameter) requires an escape velocity of 5,355 mph (2.38 km/sec).
  - -Escape from Mars (4,246 miles diameter) requires an escape velocity of 11,250 mph (5km/sec).

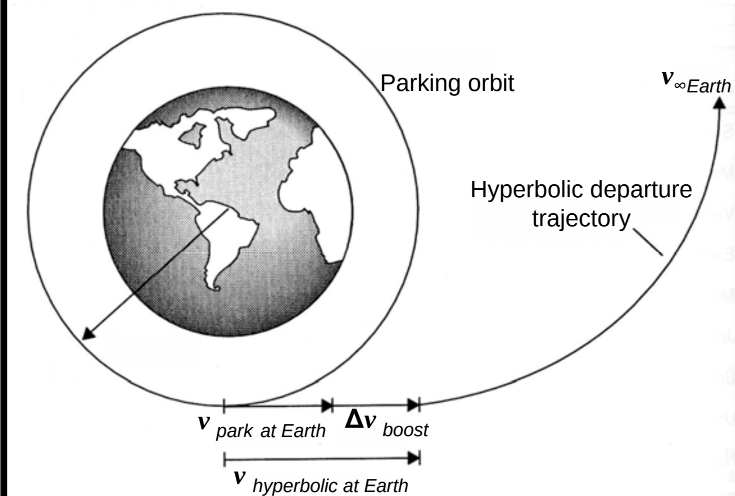


Earth Escape Assuming No Atmosphere

Hyperbolic Escape Velocities

Planet/Sun	Mean Radius (km)	Circular Velocity (km/s)	Gravitational Parameter (km^3/s^2)
Sun	--	--	1.327×10^{11}
Mercury	57.9×10^6	47.87	2.232×10^4
Venus	9.281×10^7	35.04	3.257×10^5
Earth	14.95×10^7	29.79	3.986×10^5
Mars	22.78×10^7	24.14	4.305×10^4
Jupiter	77.8×10^7	13.06	1.268×10^8
Saturn	14.26×10^8	9.65	3.795×10^7
Uranus	28.68×10^8	6.80	5.820×10^6
Neptune	44.94×10^8	5.49	6.896×10^6
Pluto	58.96×10^8	4.74	3.587×10^5

Parameters from Various Departure Origins



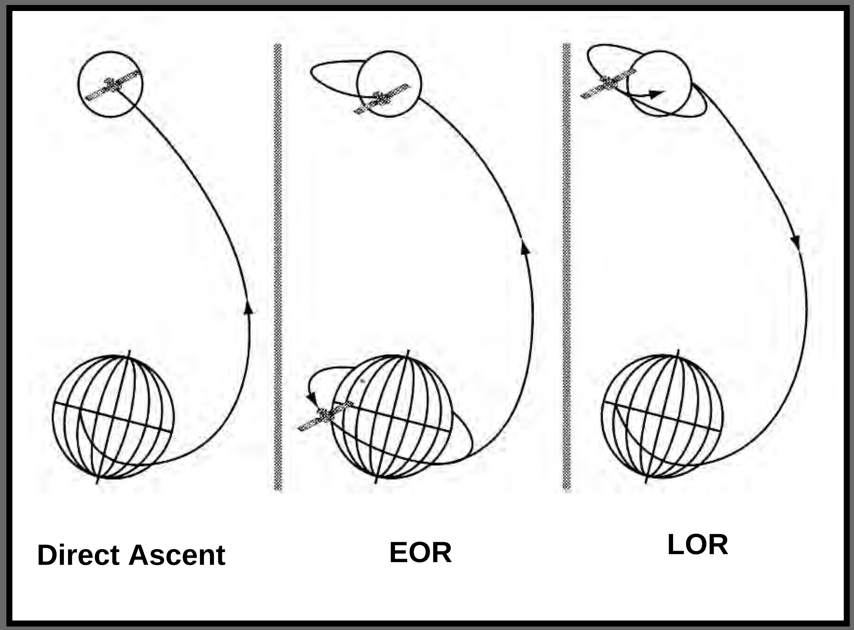
From an Earth-entered perspective, delta-V at park must be increased by delta-v"boost"

Hyperbolic Escape from Earth

Hyperbolic Escape Velocities

NASA considered 3 different trajectory options for lunar missions:

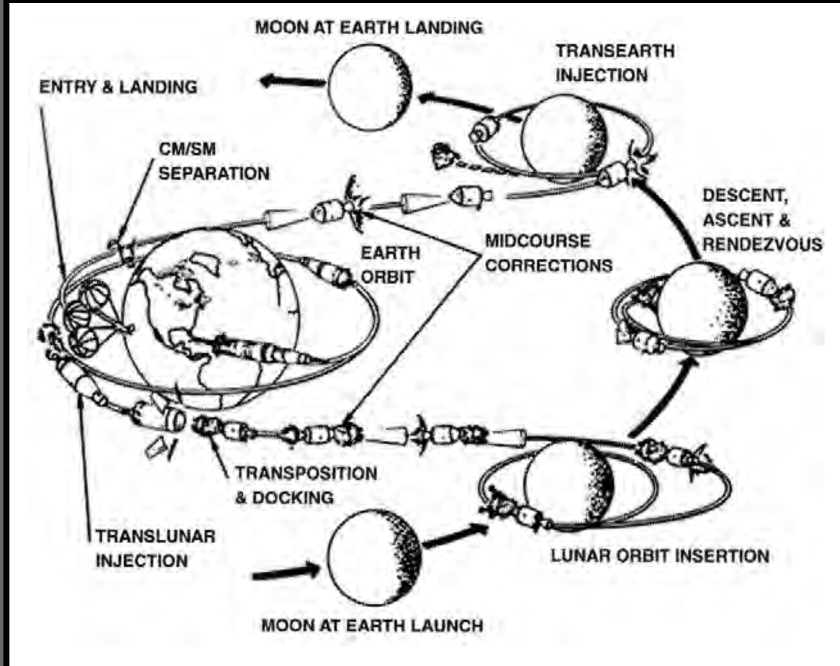
- A direct ascent approach would have sent a single rocket from Earth to the Moon, landed it, and returned:
  - This was rejected because a rocket large enough to carry all fuel required to leave Earth, brake at the Moon, return, and brake again at Earth was deemed unrealistically massive.
- An Earth-Orbit Rendezvous (EOR) approach would separately launch a lunar mission vehicle and transfer rocket to LEO using available Saturn boosters:
  - The lunar mission vehicle would proceed to orbit the Moon, and then return to LEO.
- A Lunar-Orbit Rendezvous (LOR) approach was selected:
  - Everything would be launched from Earth to lunar orbit in a single rocket and dispatch a small light weight module that would land with a crew, ascend, and then rendezvous with the spacecraft in lunar orbit for crew Earth return.



Apollo Mission Option Considerations

The Apollo Program's Lunar-Orbit Rendezvous approach used a Hohmann transfer maneuver that was timed with the Moon's orbit around the Earth for optimum Lunar Orbit Insertion and TransEarth Injection:

- The trajectory concept afforded several advantages over the other options:
  - It utilized existing Saturn boosters, saving development time and money, particularly in comparison with a massive new vehicle that would be required for direct ascent.
  - It afforded means to place people and equipment on the lunar surface.
  - Since only a small lightweight module would land (rather than an entire spacecraft) large fuel savings would result.
- Apollo trip times ranged from 2.75-3.75 days, depending upon the Moon's position in orbit and the landing site location:
  - Minimum-energy Hohmann transfers require about 5 days.
  - Slightly higher delta-Vs can reduce transit time by a day or more, but require more fuel.



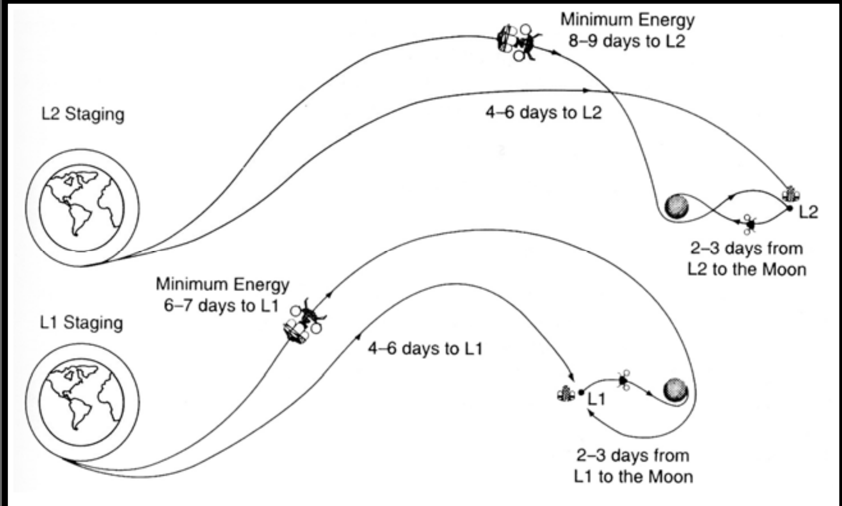
Mission Transfer Sequences

Apollo Program LOR Approach

HUMAN
SPACEFLIGHT

The Moon can also be reached from Earth via Lagrangian (or "liberation") points L1 or L2 using either direct ballistic or minimum-energy trajectories:

- Mission staging from L1 or L2 allows access to any point on the lunar surface at any time without limits on phasing or orbital geometry associated with other trajectory types:
  - Orbit maintenance is low because a spacecraft naturally remains in the Lagrange point orbit with little or no station keeping.
  - Disadvantages are extra flight time needed to reach the Moon and return to Earth, and added propellant required to enter/ depart.
- The minimum-energy way to reach L1 or L2 is to fly near the Moon where its gravity reshapes the trajectory and lowers the necessary delta-V:
  - This approach can reduce delta-V costs by as much as 100 m/ s.
  - A minimum-energy disadvantage is added flight time (3-4 days for L2 staging, and 1-2 days extra for L1 staging).

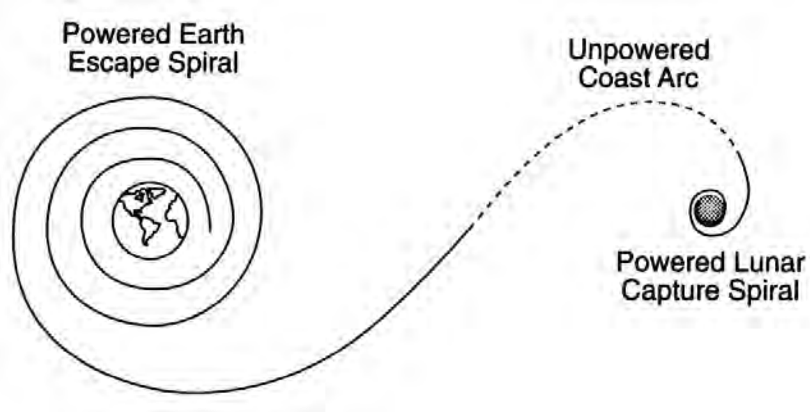


L1 and L2 Staging Comparisons

Mission staging through Lagrangian Points

Low-thrust trajectories use smaller thrust levels over longer periods of time to accelerate away from a planet (such as Earth) on a spiral course:

- Once out of the planet's sphere of influence or after velocity is built up sufficiently, the trajectory appears ballistic:
  - The principal advantage of low-thrust trajectories is associated with their highly efficient propulsion systems.
  - Efficiency benefits are offset for some applications (such as crew missions) by greatly extended travel times required during the spiral-out and spiral-in trip segments.
  - Spiral-out segments from Earth are particularly troublesome due to added time in the hazardous Van Allen radiation belts.
  - Fuel efficiencies of low-thrust systems may be most attractive for cargo transport and extended time/ distance missions to Mars where spiral-out time is small relative to total transit time and fuel requirements.



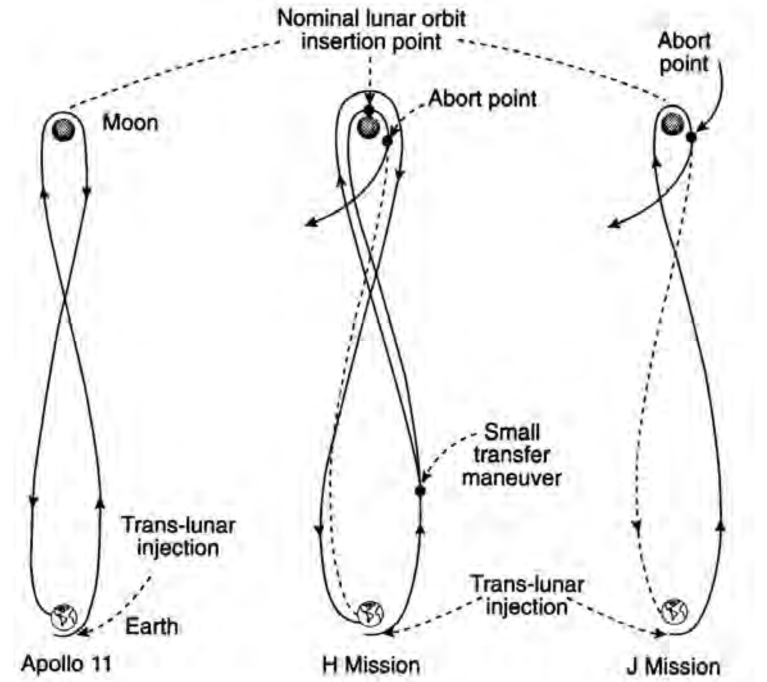
Low-thrust systems can thrust continuously, or can reach ballistic trajectories and coast to the destination orbit. They can also reverse vector orientation near mid-course and decelerate for gradual propulsion braking.

Low-thrust Options

Low-thrust Trajectories

Apollo missions used various free return and powered return trajectories with abort options that took advantage of the Moon's substantial mass for a swingby (or "slingshot") boost:

- A free return trajectory was used for Apollo 8, 10 and 11 missions:
  - The spacecraft flew by the Moon and used the lunar gravitational field to bend the orbit and enable a return directly back to Earth without any intermediate propulsive maneuvers.
- A modified free return called "H Mission" was used for Apollo 13 and 14:
  - The early trajectory followed Apollo 13 until successful docking with the Lunar Module, then used the LM's engine to alter the trajectory.
- A powered return was used for later Apollo 15, 16 and 17 missions:
  - The trajectory was similar to the others, but required intermediate propulsive maneuvers to keep the Moon's gravity from bending the trajectory too far from Earth.



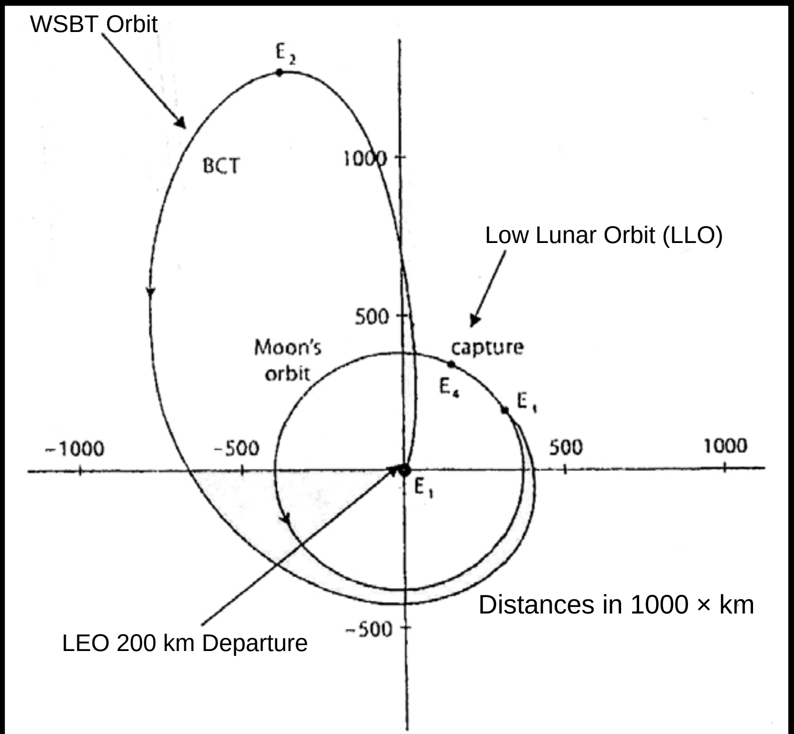
Apollo missions used 3 different trajectory approaches that incorporated abort opportunities.

Apollo Trajectories with Aborts

Free and Powered Returns/ Aborts

Weak Stability Boundary (WSB) orbits exploit low-gravity fields in “chaotic dynamic” regions located where the gravitational “pulls” of large bodies cancel out to benefit spacecraft velocities/ orbits for one-way cargo missions:

- WSB conditions in a boundary region about the Moon are similar in some respects to Lagrange points 1 and 2, where small maneuvers can have large effects upon spacecraft motion that offer important advantages:
  - Departing a 200 km LEO altitude and crossing Sun-Earth WSB at about 1.5 km from Earth can substantially increase Translunar orbit energy.
  - Upon arrival at the Moon, the Earth-Moon WSB can be used to reduce the spacecraft delta-V to a level that it is captured by the Moon’s gravity (potentially saving about 25% of propulsive requirements).
  - WSB potentially offers access to all lunar surface sites, an exclusive advantage over all other trajectories, and with no Earth launch window constraints.

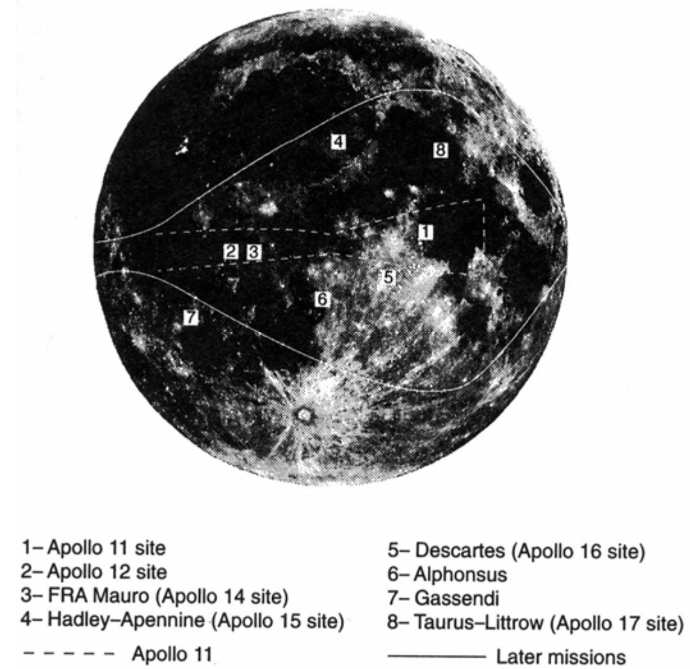


Weak Stability Boundary Transfer Orbit

Applying Weak Stability Boundaries

Apollo missions used surface sites within ± 40° longitude to enable landing and ascent operations to be tracked from Earth:

- Each of the Apollo free and powered trajectories presented severe restrictions regarding sites that were accessible:
  - Free return trajectories were limited to lunar orbit inclinations of ± 5° about the Moon's equator (assuming that the abort option isn't used and the spacecraft enters into lunar orbit).
  - H and J missions allowed trajectories at higher latitudes, but could not reach polar sites without giving up the option of free or powered abort returns to Earth.
  - Polar sites have received recent interest due to the possibility that surface water may exist in these areas as a source of hydrogen for propellant and other uses.



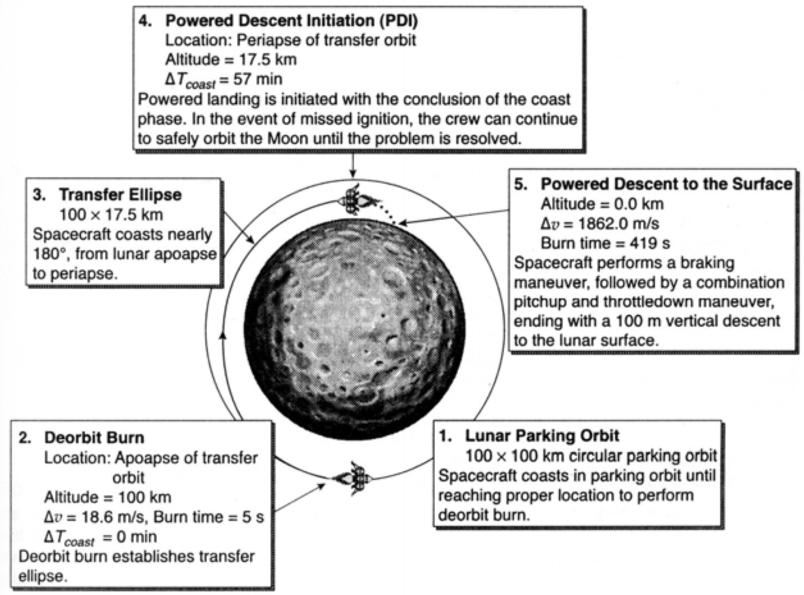
Dotted line areas near the equator are accessible using free return, and solid lines bound areas accessible by H and J mission trajectories.

Apollo Landing Sites

Accessible Landing Sites

Powered lunar descent strategies must provide for abort contingencies, trajectory shaping to clear terrain, pilot or sensor viewing, and piloted or automated detection and avoidance of landing hazards:

- The process begins with a small propulsive burn in the lunar parking orbit to put the vehicle into a transfer descent orbit:
  - The transfer orbit has an apoasis (similar to apogee on Earth) of about 100 km, and a periapsis (perigee on Earth) of about 17.5 km.
  - The periapsis altitude is high enough above the surface to keep the vehicle orbiting in the event of descent engine failure.
  - The direction of landing can be retrograde (opposite the Moon's rotation) because the rotation is slow, costing only about 10 m/s more delta-V than for a landing.
  - Following deorbit from parking, the vehicle coasts from apoasis to periapsis, fires its engines, and lands softly on the surface.



A deorbit burn lowers the spacecraft from a circular 100km parking orbit to a 100m X 17.5km transfer orbit. It then coasts from apoasis to periapsis to begin powered descent to the surface.

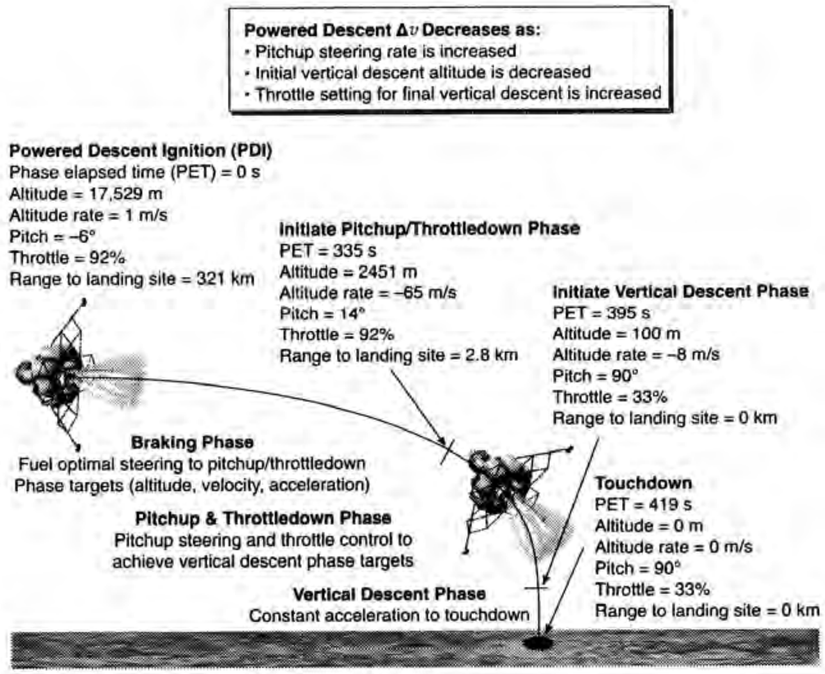
Deorbit, Coast and Descent to the Surface

Powered Descent & landing

HUMAN SPACEFLIGHT

Powered lunar descent consists of 3 basic landing phases:

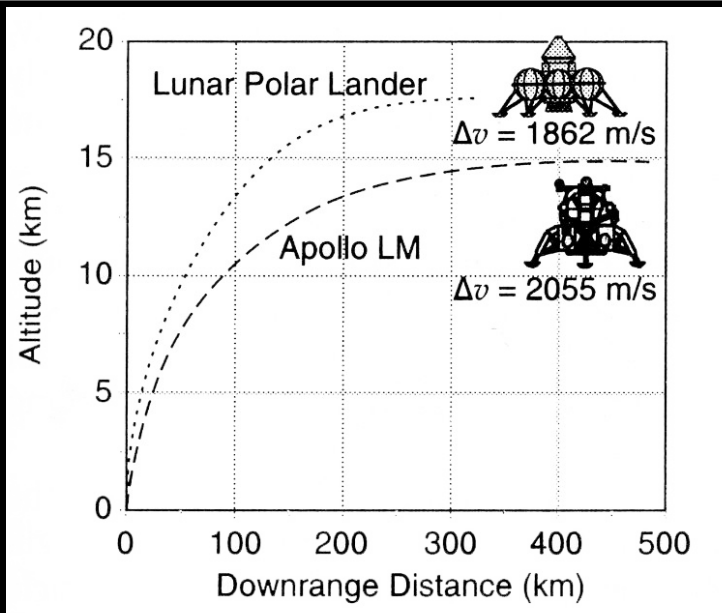
- A braking phase removes most of the vehicle's horizontal velocity and altitude:
  - Engines thrust at the nominal maximum level (but may be less than their maximum thrust to allow for a possible abort).
- A pitching up-throttling down phase achieves a vertical altitude and reduces thrust acceleration for landing:
  - A reduced delta-V improves engine efficiency and slows the approach speed to enable crew or survey sensors (visual, infrared optical imaging, laser, radar) to evaluate the landing site.
- A final vertical descent phase decreases the vertically-oriented vehicle's delta-V for landing:
  - This begins at a 30 m –100 m altitude for a landing at about 1.2 lunar g's.



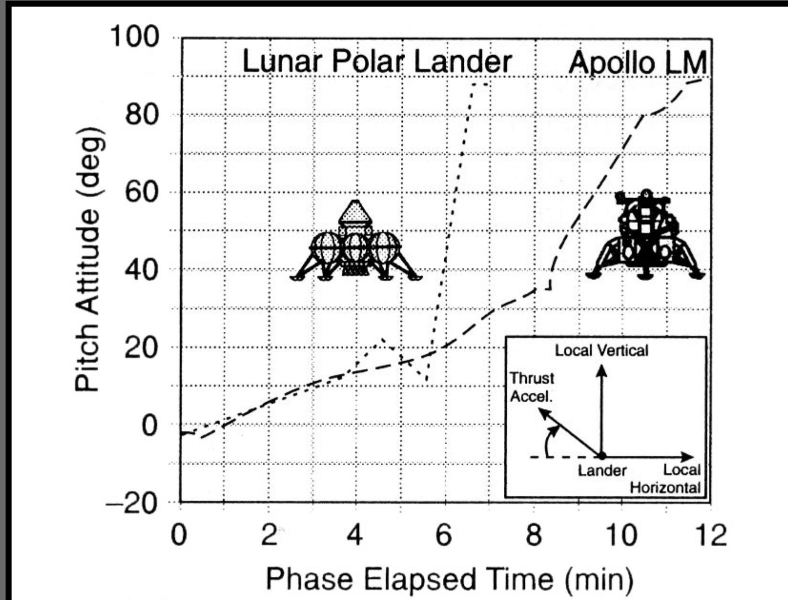
Final Phases of Powered Descent

Powered Descent & landing

HUMAN SPACEFLIGHT



Altitude vs. Downrange Distance
Lunar polar Lander & Apollo Lunar Module



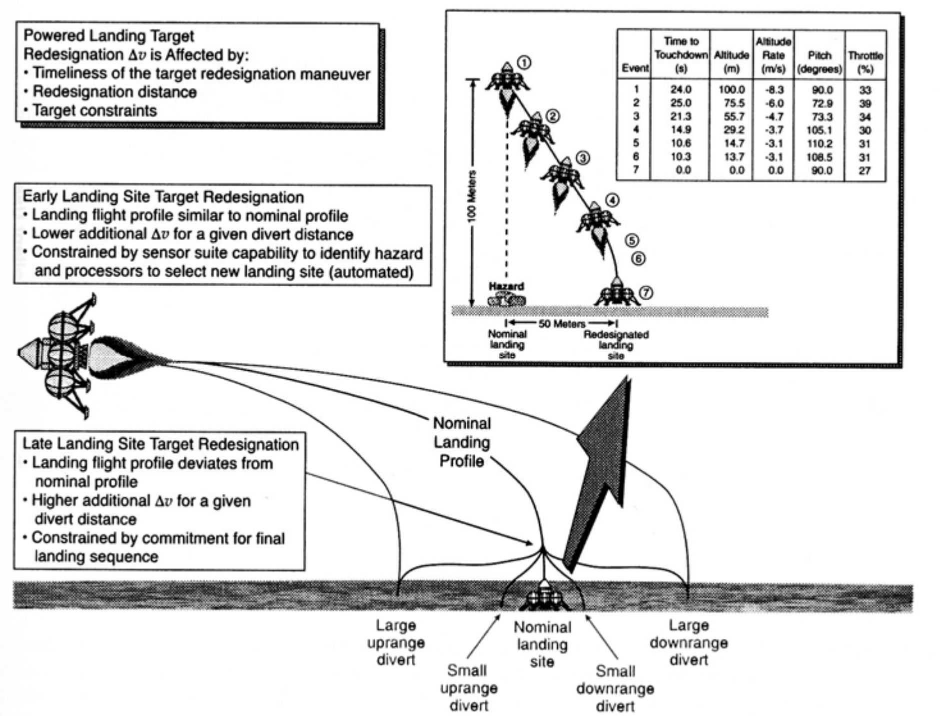
Pitch Altitude vs. Phase Elapsed Time
Lunar Polar Lander & Apollo Lunar Module

Powered Descent & landing

HUMAN SPACEFLIGHT

If a crew or sensors detect site hazards they must have opportunities to redesignate an alternative location, applying 3 key considerations:

- Timelines of the maneuver to redesignate:
  - By diverting early, the vehicle can stay on a landing trajectory that conserves fuel.
- Distance to the landing site:
  - If a vehicle is close to the surface it will require substantial fuel/ delta-V to offset gravitational acceleration without reducing velocity or altitude.
- Target constraints:
  - A final vertical descent altitude of 100 m affords the crew or sensors a good viewing perspective, and a slow approach provides time for navigation corrections.



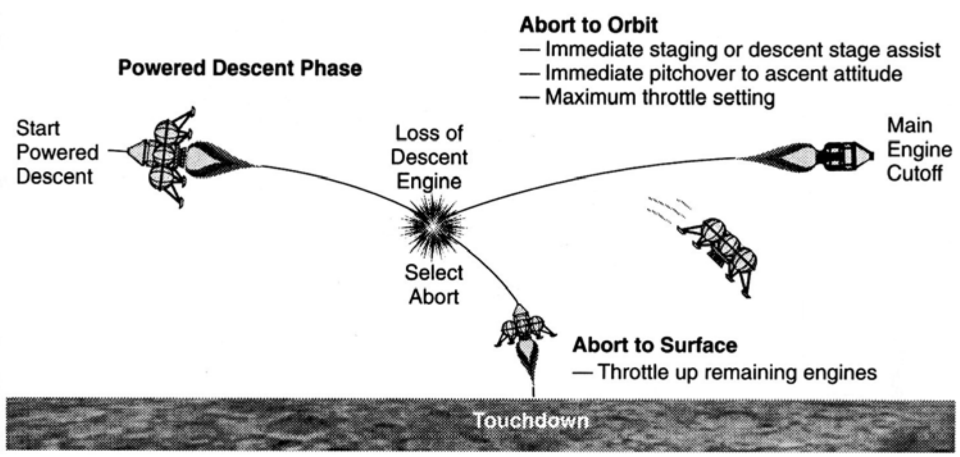
Event	Time to Touchdown (s)	Altitude (m)	Altitude Rate (m/s)	Pitch (degrees)	Throttle (%)
1	24.0	100.0	-8.3	90.0	33
2	25.0	75.5	-6.0	72.9	39
3	21.3	55.7	-4.7	73.3	34
4	14.9	29.2	-3.7	105.1	30
5	10.6	14.7	-3.1	110.2	31
6	10.3	13.7	-3.1	108.5	31
7	0.0	0.0	0.0	90.0	27

Redesignating a Landing Site Target

Powered Descent & landing

Abort to Orbit (ATO) decisions must be made early in the braking phase:

- The delta-V required to abort increases as phase elapsed time progresses:
  - During early descent a vehicle has greater downrange velocity, requiring less delta-V for abort.
  - Later in the descent profile when a vehicle is rapidly descending with relatively little down range velocity, abort can require more delta-V than is needed for surface liftoff.
  - A vehicle can be descending so fast that an ascent stage can't stop it before it hits the surface.



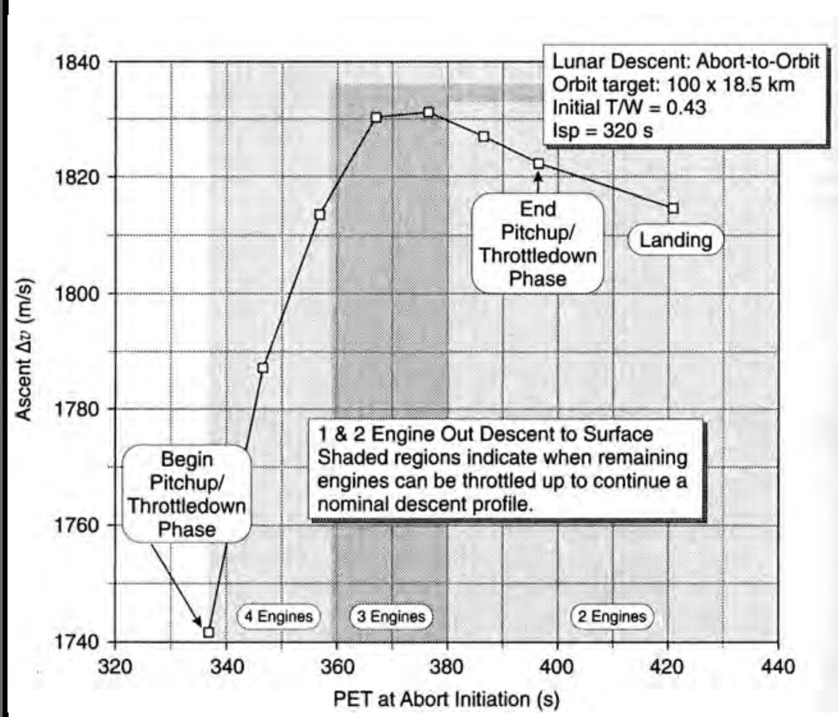
Depending upon how far a multi-engine stage (4 engines) has descended, a lander can do 2 possible aborts if an engine fails: 1) jettison the failed descent stage and ignite the ascent stage and return to orbit; 2) throttle up the remaining descent stage engines and land.

Abort to Orbit or Surface with Engine failure

Aborts During Landing

If a 4-engine lander loses 1 or more engines it can abort either back to orbit or to the surface:

- The design of the descent flight profile allows an Abort to Orbit (ATO) at any time during the descent:
  - Assuming the descent stage is discarded and the ascent stage is used for the maneuver, more propellant may be required than would be needed for nominal ascent from the surface.
- For an Abort to Surface (ATS), a lander can recover if 1 or 2 engines fail:
  - Depending upon when the descent engine(s) fail, it may be possible to throttle up the remaining engines so that they have the same thrust as 4 engines do for nominal descent.



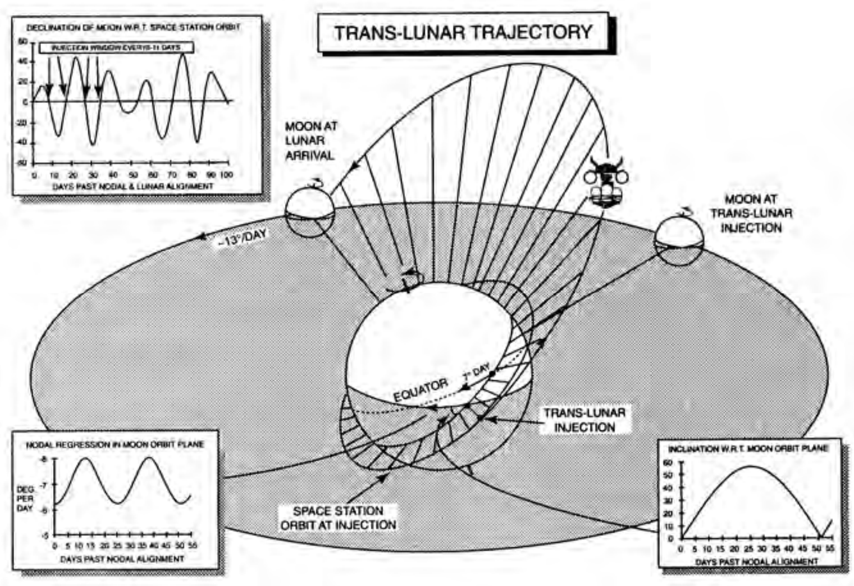
Abort to Orbit or Surface with Engine failure

Aborts During Landing

HUMAN SPACEFLIGHT

Hohmann transfer trajectories for Trans-Lunar Injection (TLI) can potentially be launched either from the Earth's surface or from LEO (such as the ISS orbit):

- Missions departing from Earth typically have 2 launch opportunities/ day:
  - Launches place the vehicle in an appropriate parking orbit that intersects the desired transfer orbit to the Moon.
- From an ISS LEO orbit the vehicle must wait until the orbit is tangentially aligned with the TLI plane to avoid costly out-of-plane thrusting penalties:
  - The Moon moves around Earth at approximately 13°/ day in a prograde direction.
  - The ISS orbital plane inclination regresses about 5°/ day.
  - Combining these 2 motions yields a TLI launch opportunity averaging once each 9 days, but can vary from 6-11 days.
  - Windows for each launch typically last less than 1 day.



The Spacecraft must lead the Moon as it departs, but the trajectory does not need to be in the same plane as the Moon's Earth orbit.

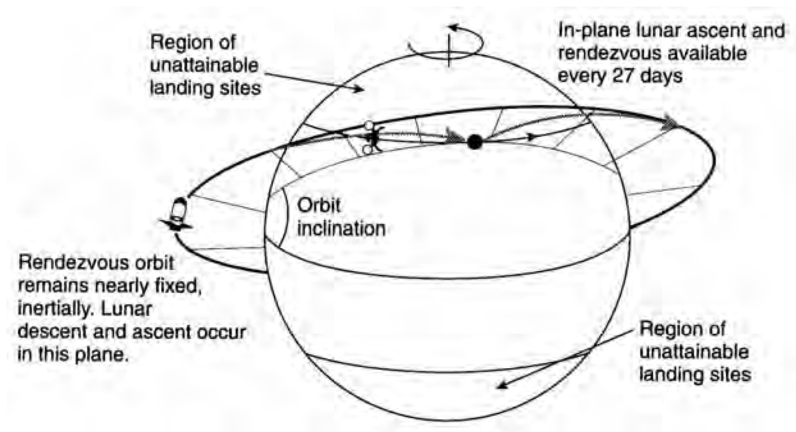
Departure Conditions from Earth to Moon

Earth & LEO TLI Launch Windows

HUMAN SPACEFLIGHT

If a mission requires a lunar orbit, the ground track requirements or landing site accessibility affect the lunar orbit's desired inclination:

- For a mission driven by a particular landing site, the orbital inclination should be greater, than or equal to the landing site's latitude:
  - Once in lunar orbit, the spacecraft can descend to the surface or ascend from it only during certain windows of opportunity.
  - The plane of the orbiting vehicle remains essentially fixed in internal space, while the surface site rotates at the Moon's rate of about 13°/day (360° in 27 days).
  - For the special case of a spacecraft in an equatorial orbit, the vehicle has nearly continuous access to all near-equatorial sites (± 5° latitude), with window opportunities separated only by the orbital period.
  - In-plane lunar ascent/ rendezvous opportunities for other orbits have 27 days.



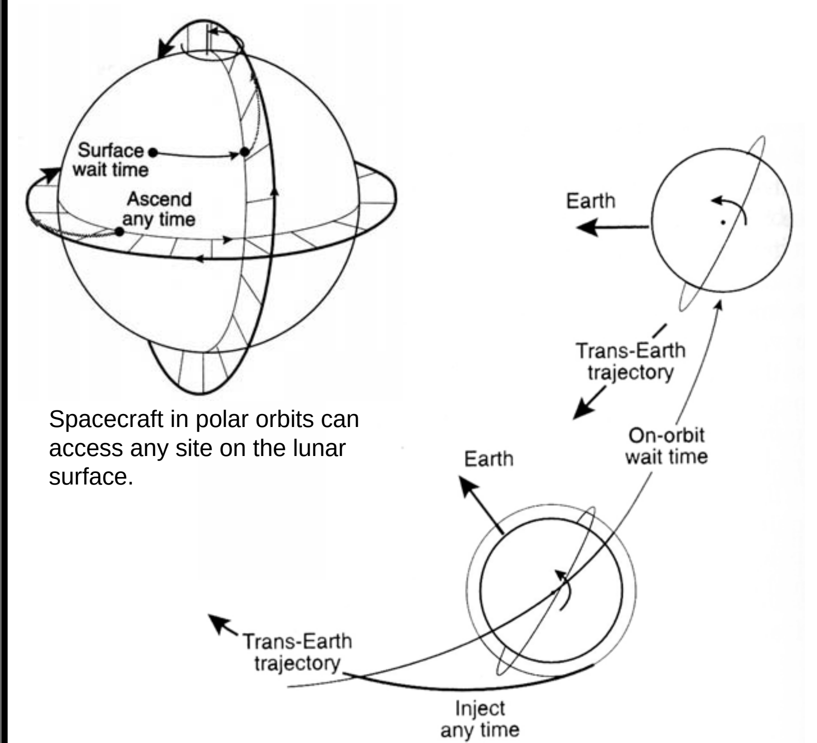
A spacecraft's orbital inclination dictates site accessibility, and the Moon's rotation rate dictates intervals between ascent/ rendezvous opportunities.

Relationships Between Orbits and Landing Sites

Orbital Planes & Site Access/ Windows

The phasing required for returning to Earth is comparable to leaving for Earth:

- The lunar orbit must be correctly oriented with respect to the Trans-Earth Insertion (TEI) trajectory:
  - Spacecraft in lunar orbits near the equator (inclinations less than about 15°) can depart for Earth once every orbit.
  - Spacecraft in higher inclination orbits can have Earth return windows lasting about 3 days and occurring about every 14 days.
  - Spacecraft departing from inclinations between about 15° and polar orbits can also have opportunities to leave every 14 days to land on the Earth's surface or enter Earth orbit, but not necessarily to the ISS orbit.
  - The worst-case scenario for phasing from the lunar surface to orbit, and from lunar orbit to Earth orbit, would require 27 days.

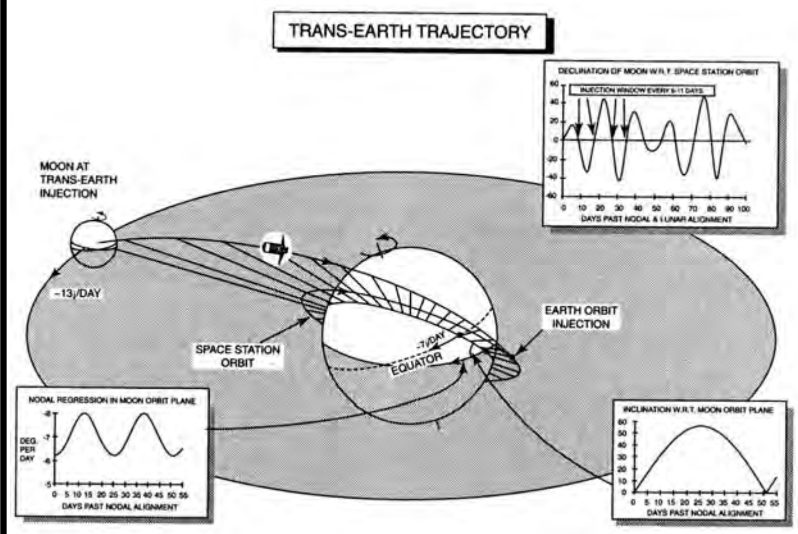


Earth Return: Launch Window Opportunities

Phasing of Earth Returns

Phasing of transfers from Earth to lunar orbits, surface landings and returns is also influenced by operation and safety factors:

- Apollo missions required a specific solar lighting angle with respect to the lunar surface to enable crews to detect and avoid obstacles such as rocks and craters:
  - Sometimes a specific landing site had necessary conditions only once each month.
- Apollo missions also required Earth landing in daylight to assist safe recovery:
  - Since crews landed in open oceans, recovery forces needed to be able to rapidly locate them and relocate to the landing area.
- Future missions may require returns to the ISS orbit which impose unique access maneuvers and phasing, or to surface sites with night landing accommodations:
  - Abort contingencies must account for orbital safe haven provisions, and for landing at alternate locations to avoid unacceptable weather conditions at a planned site.



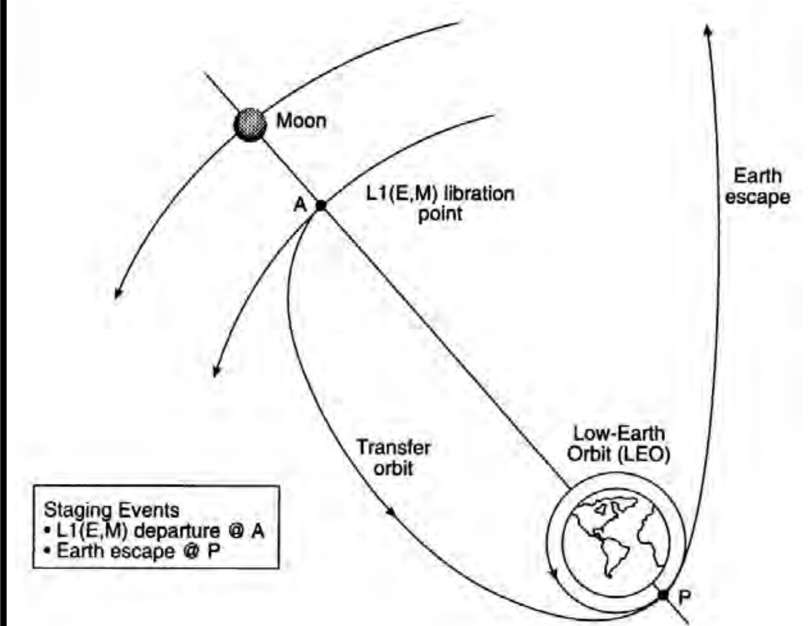
Returns from the Moon to the ISS Earth orbit impose more restrictive constraints than Earth surface landings.

ISS Orbit Return Constraints

Special Phasing Restrictions

LEO and L2 are logical staging points for missions to Mars:

- Circular LEO can be used as a place to assemble and supply transfer vehicles and payloads:
  - This will enable huge mass requirements for Mars to be delivered to the departure orbit in parts using commercial-class launches, avoiding a need to develop enormous new boosters.
  - Staging might take advantage of ISS support, but the ISS orbit inclination may not align with the Mars departure vector when transfers are desired.
- Staging from L1 might be beneficial in the event that hydrogen and/ or oxygen from the Moon could be used for propellant:
  - This might split the fuel supplies between Earth and lunar sources.
  - L1 staging to Mars would require less energy than from LEO, potentially reducing Mars transfer vehicle size and mass.



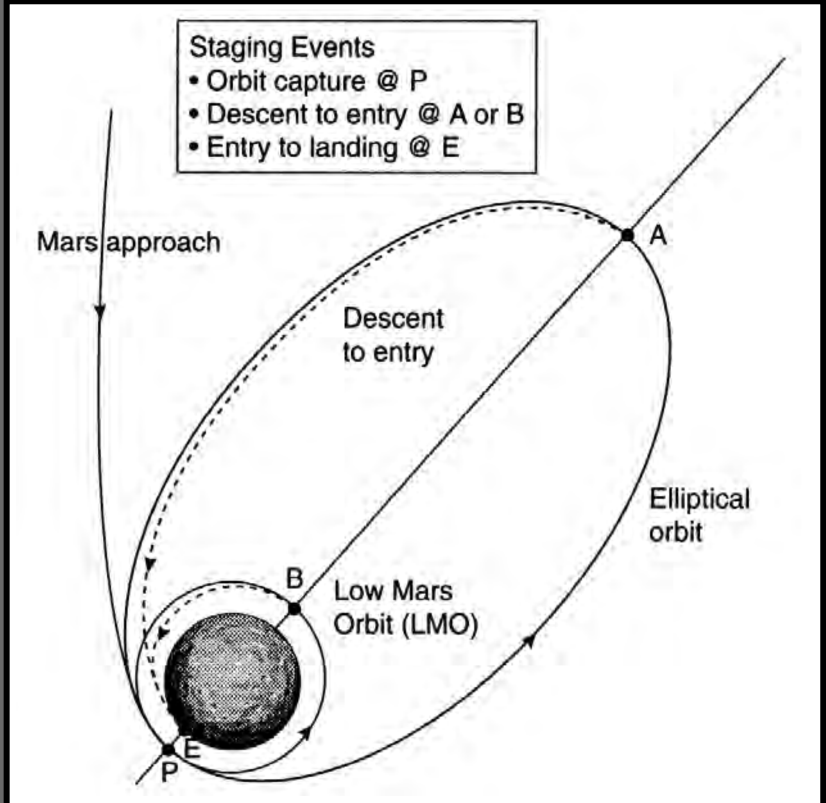
Staging from L1 can be beneficial if lunar propellant is made available. Less escape energy is needed than from LEO, and transfer mass could be reduced.

Staging from LEO and L1

Staging Points

Most Mars mission scenarios assume an Apollo-like strategy that retains an Earth return vehicle in a Martian orbit while a smaller descent/ ascent vehicle is dispatched to the surface.

- Two types of Martian parking orbits might be used:
  - Low Mars orbits offer advantages of reduced entry heating and less delta-V required for ascent from the surface, but need more Delta-V to access from Earth and depart from.
  - Highly elliptical orbits (similar to the Molniya orbits at Earth) place greater delta-V costs on vehicles ascending from the surface, but relieve some delta-V burdens for Mars departures.
  - Properly positioned highly elliptical orbits can be used for communication relay between Earth-return vehicles and Earth throughout the parking periods.



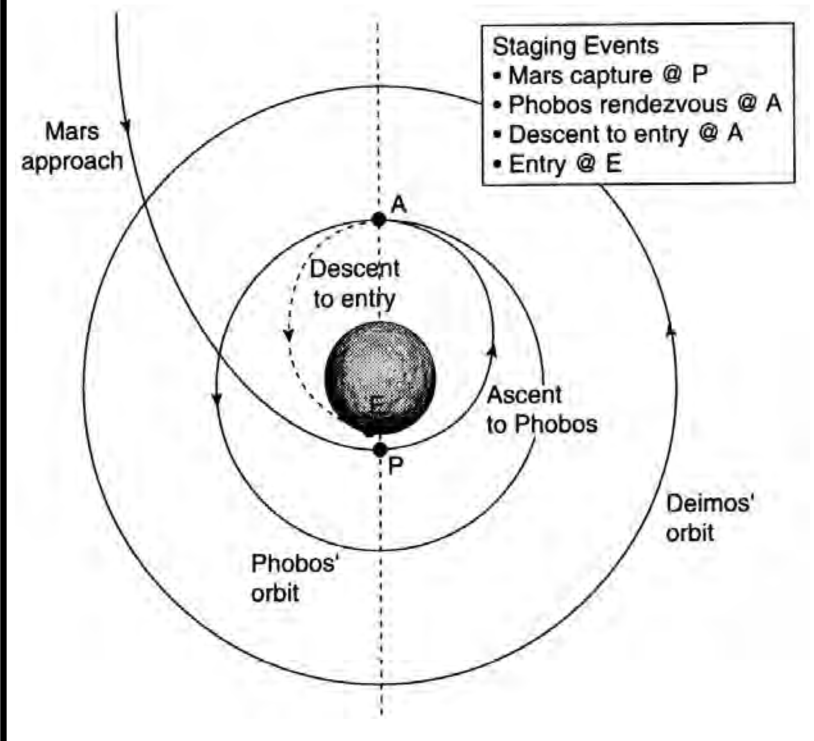
Low-altitude and Elliptical Orbits

Mars Destination Orbits

Staging for Mars missions might also be conducted from the Martian moon Phobos:

• This approach anticipates the possibility that Phobos might be a source of useful materials, including propellants:

- Phobos is relatively high in the Martian gravity well, making it less costly to reach and depart from on a round-trip mission than from the surface.
- A disadvantage of Phobos as a staging point is that a vehicle needed to land on Mars and return will be massive due to the required propellant.
- To access Phobos it will be logical to achieve capture at Mars' periapsis, followed by a rise to a circular orbit that intersects a desired parking orbit around Phobos.
- A second burn would circularize the parking orbit around Phobos and prepare for possible descent to its surface.

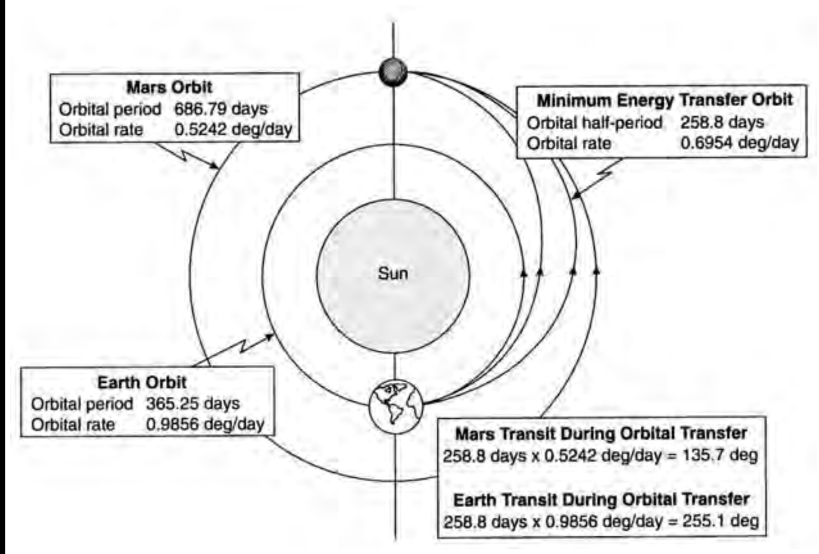


Staging from the Martian Moon Phobos

Phobos as a Staging Point & Destination

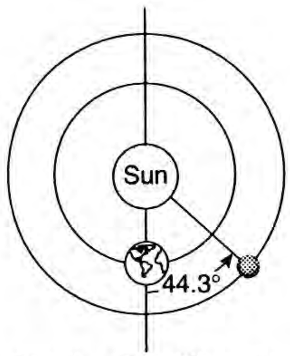
Orbit alignment issues for lunar missions also apply to Mars missions, but a larger phasing problem results from movements of the Earth and Mars around the Sun which influence Earth departure and return opportunities:

- Relative locations of the Earth and Mars determine outbound/ return trajectory options and flight times:
  - While Earth completes 1 rotation around the Sun in 365.25 days (average angular rate of 0.9856°/ day), Mars completes 1 rotation in 686.79 Earth days (average angular rate of 0.5242°/ day).
  - A Hohmann transfer between Earth and Mars requires 258.8 days to complete the 180° transit traveling at an average rate of 0.6954°/ day.
  - Using these averaged angular dates, Mars must be 44.3° ahead of Earth at time of launch in order for rendezvous to occur 258.8 days later.

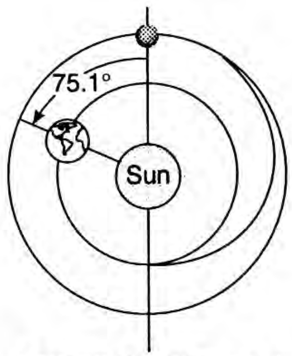


Minimum-energy Transfer to Mars

Influences of Planetary Motions



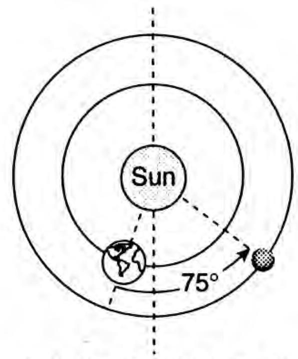
Relative Conditions at Earth Departure
Phase Angle = 44.3°



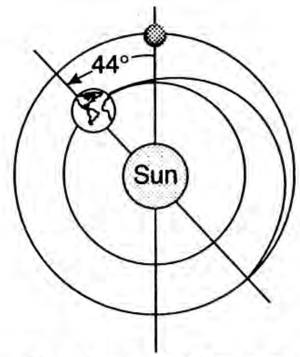
Relative Conditions at Mars Arrival
Phase Angle = –75°
Cumulative Mission Duration = 259 days

Using a minimum-energy transfer, Mars must be 44.3° ahead of Earth at the time of departure. The outbound trip will require about 259 days.

Earth-Mars Travel Time



Relative Conditions at Mars Departure
Phase Angle = 44.3°
Cumulative Mission Duration = 714 days



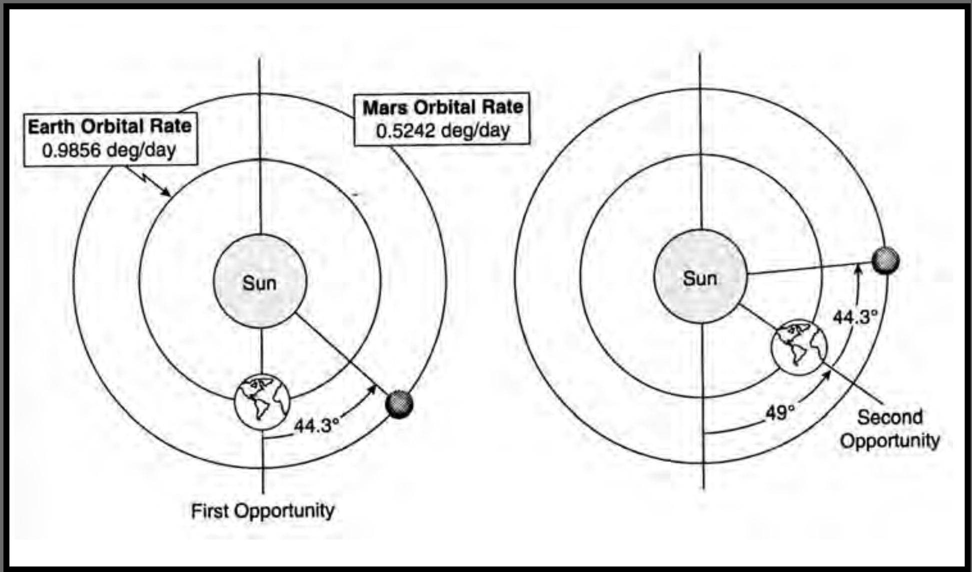
Relative Conditions at Earth Arrival
Phase Angle = –44°
Cumulative Mission Duration = 972 days

For return, Mars must be 75° ahead of Earth at departure. Correct alignment will require 44 days of waiting plus 259 days of return travel.

Surface and Return Time

Minimum-energy Mission Time

HUMAN SPACEFLIGHT



$$\text{Repetition rate for identical phasing} = \frac{360\ \text{deg}}{0.9856\ \text{deg/day} - 0.5242\ \text{deg/day}} = 780\ \text{days}$$

$$\text{Number of opportunities for full progression around Sun} = \frac{360\ \text{deg}}{49\ \text{deg per opportunity}} \approx 7$$

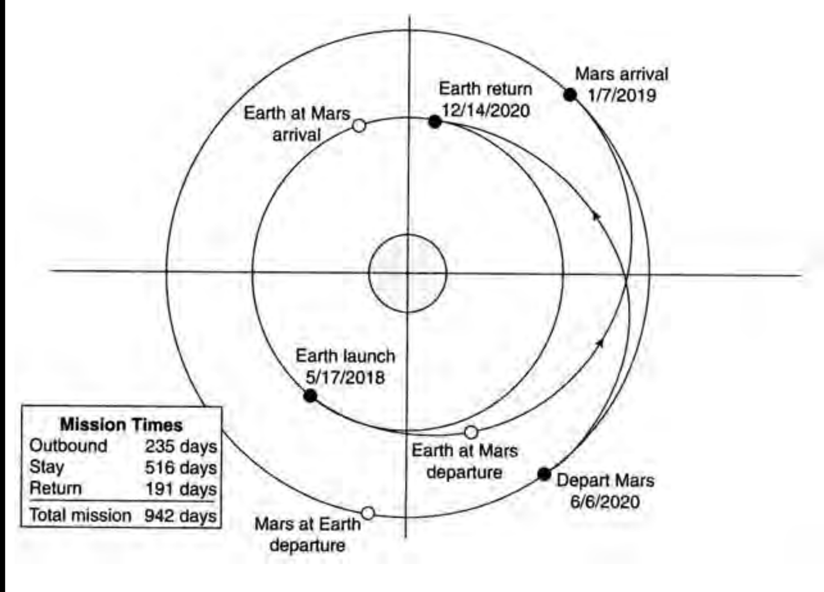
The Earth and Mars return to their original initial positions on the 8th sequence.

The time required between comparable alignments of 2 orbiting bodies is called the “synodic period”. This period for the Moon and Mars is 780 days (26 months), which is the time between successive opportunities to launch spacecraft to Mars. On the 8th opportunity in this sequence, the Earth and Mars will return to the same inertial space they occupied on the 1st opportunity 15 years earlier.

Repetitive Phasing of Earth & Mars

Three general categories of trajectories proposed for Earth-Mars missions are “conjunction- class”, “opposition-class” and “Venus flybys”.

- Conjunction-class refers to Hohmann-type minimum-energy trajectories which are the most traditional and slowest approach, providing long stay times at Mars:
  - At Mars arrival, Earth is moving into conjunction with Mars (the Sun and Mars are on the same side of Earth).
  - The 2 planets are out of phase for return until 1.5 years (516 days) later.
  - When planetary phasing is correct, the return trajectory concludes with a relatively short flight (191 days return vs. 235 days outbound).
  - Long stay times in combination with travel times require 2.9 years for lowest-energy cases.
  - Long outbound travel times make these low delta-V trajectories particularly attractive for cargo missions.



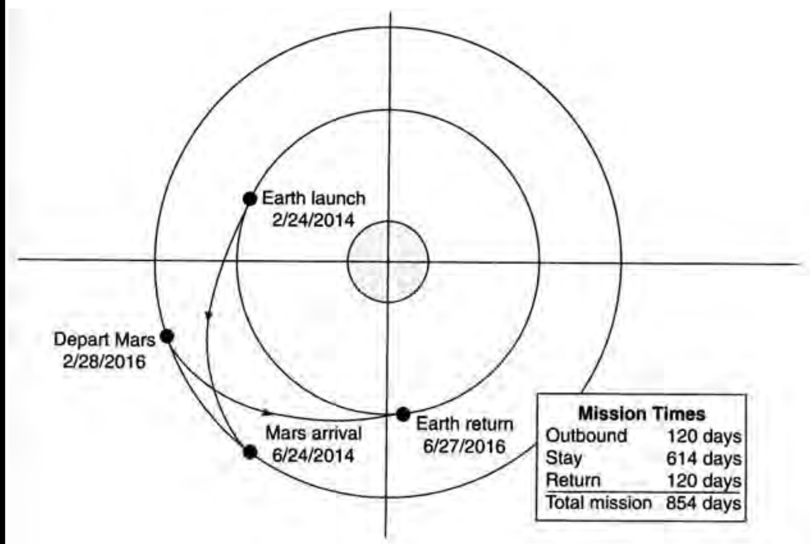
Typical Conjunction-class Category

Conjunction-class Missions

HUMAN SPACEFLIGHT

A conjunction-class variation is a "fast-transit" type which offers relatively short transit times to and from Mars at the cost of higher delta-V requirements:

- The spacecraft overtakes Mars following a 120 day outbound leg, and returns to Earth in the same amount of time about 1.7 years later:
  - Shortened travel times may offer health advantages for crews by reducing deconditioning effects of weightlessness in addition to lessened exposures to space radiation.
  - Travel time advantages may be offset to some extent by extended surface time (more than 3 months longer than conventional conjunction-class), although total mission time will be reduced by nearly that much.
  - Larger delta-V requirements translate into more fuel mass that must be launched from Earth, adding substantial mission expense.



Fast-transit Conjunction Trajectory

Conjunction-class Missions

Launch Date	Δv TMI*	Outbound (days)	Δv MOI†	Mars Stay-Time (days)	Δv TEI	Return (days)	Total Mission Duration (days)	Δv Total**
4/03/01	3639	200	2532	545	2108	205	950	8278
6/08/03	3574	204	2095	547	2647	192	943	8316
8/20/05	3963	217	2038	492	2703	214	923	8704
10/06/07	4199	248	2032	437	2278	262	947	8509
11/08/09	4035	278	1988	374	2064	270	922	8087
11/28/11	3672	252	2532	418	1989	259	929	8193
1/17/14	3832	224	2794	458	1941	237	919	8567
3/11/16	3739	204	2677	529	1983	212	945	8399
5/11/18	3530	204	2230	553	2466	190	946	8227
7/27/20	3807	207	2031	517	2746	203	927	8584

TMI Trans-Mars Injection
MOI Mars Orbit Insertion
TEI Trans-Earth Injection

* Launch from ISS altitude orbit
† 500 km circular orbit at Mars
** Assumes direct entry upon Earth return (all velocities in m/s)

This data assumes that aerocapture will be used to make propellant delta-V for Earth braking unnecessary.

Earth-Mars Travel Time

Launch Date	Δv TMI*	Outbound (days)	Δv MOI†	Mars Stay-Time (days)	Δv TEI	Return (days)	Total Mission Duration (days)	Δv Total**
09/12/05	4850	120	4130	624	5010	120	864	13,990
11/01/07	5706	120	5749	622	6392	120	862	17,848
12/10/09	6141	120	6882	621	6641	120	861	19,664
01/16/12	6213	120	6998	618	5593	120	858	18,804
02/24/14	5817	120	6157	614	3844	120	854	15,818
04/11/16	4996	120	4649	624	2888	120	864	12,533
06/12/18	4224	120	3345	639	3373	120	879	10,942
08/20/20	4525	120	3559	624	4431	120	864	12,515
10/14/22	5346	120	5192	624	5946	120	864	16,485
11/25/24	6055	120	6509	621	6706	120	861	19,270

TMI Trans-Mars Injection
MOI Mars Orbit Insertion
TEI Trans-Earth Injection

* Launch from ISS altitude orbit
† 500 km circular orbit at Mars
** Assumes direct entry upon Earth return (all velocities in m/s)

A flight time of 120 days was selected to be within current US mission experience. Earth aerocapture is assumed.

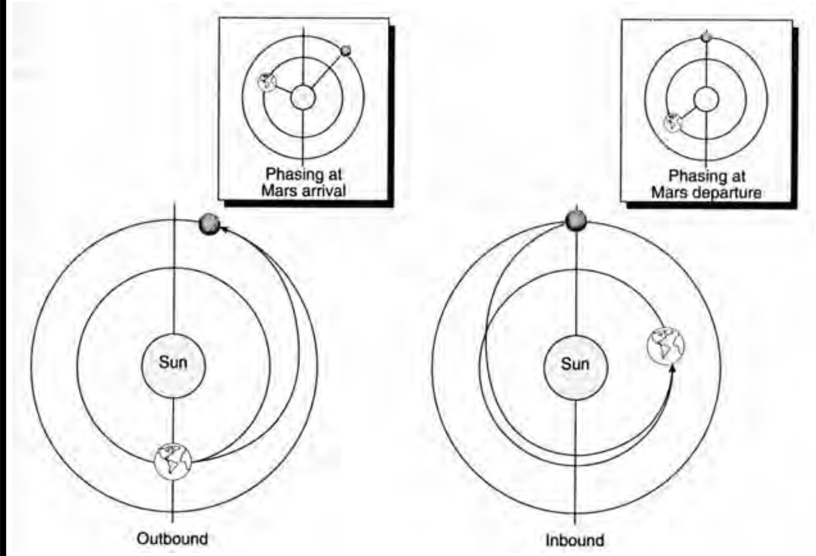
Surface and Return Time

Conventional & fast-transit Conjunction-class Parameters

HUMAN SPACEFLIGHT

Opposition-class (high-energy) trajectories can reduce total mission durations by half over all conjunction-class missions, but at the cost of more fuel:

- The spacecraft arrives at Mars as Earth is leaving opposition with Mars (the Sun and Mars are on opposite side of the Earth):
  - Surface stay time is relatively short (20-40 days), after which the spacecraft must get back on a return trajectory to catch up with Earth which is moving out of phase.
  - The vehicle must move inside Earth's orbit (closer to the Sun) in order to achieve the high velocity needed to catch up, adding substantial fuel requirements.
  - Approximate 3-6 week surface times may be adequate for most human Mars missions, and combined with reasonably short travel times may afford significant crew health/ safety advantages.



These trajectories have 2 unequal transfer "legs" which afford options of placing the longer leg on either the outbound or inbound mission segment as desired.

Phasing of Arrival and Departure

Opposition-class Missions

Launch Date	Δv TMI*	Outbound (days)	Δv MOI†	Mars Stay-Time (days)	Δv TEI	Return (days)	Δv RET	Total Mission Duration (days)	Δv Total**
08/27/00	7692	262	4437	40	5401	168	0	470	17,531
10/05/02	7276	251	4595	40	4209	169	0	460	16,079
11/06/04	7889	236	4899	40	3404	174	0	450	16,192
12/13/06	9421	219	5352	40	3133	201	0	460	17,907
01/19/09	11790	207	5795	40	2972	233	0	480	20,557
03/27/11	13053	221	5217	40	4812	209	0	470	23,081
06/06/13	11142	247	4757	40	6244	183	0	470	22,143
07/31/15	8844	264	4467	40	5085	176	0	480	18,396
09/15/17	7488	260	4454	40	4556	170	0	470	16,498
10/24/19	7576	242	4781	40	3585	168	0	450	15,943

TMI Trans-Mars Injection
MOI Mars Orbit Insertion
TEI Trans-Earth Injection
RET Δv Earth Return

* Launch from ISS altitude orbit
† 500 km circular orbit at Mars
** Assumes direct entry upon Earth return (all velocities in m/s)

This data assumes that aerocapture will be used to make propellant delta-V for Earth braking unnecessary.

Opposition-class Missions

Launch Date	Δv TMI*	Venus Swingby	Outbound (days)	Δv MOI†	Mars Stay-Time (days)	Δv TEI	Return (days)	Total Mission Duration (days)	Δv Total**
04/01/01	3635	Inbound	201	2538	40	4248	345	586	10,422
08/22/02	3820	Outbound	302	4744	40	3134	261	603	11,704
06/09/04	4131	Outbound	344	4429	40	2639	271	655	11,198
08/27/07	4600	Inbound	188	4341	40	4030	340	568	12,972
01/17/09	4208	In & Out	330	3339	40	3367	367	737	11,342
11/28/10	4426	Outbound	330	3502	40	2494	303	673	10,422
11/21/13	3692	Inbound	281	2464	40	4419	311	632	10,575
10/26/15	4865	Inbound	279	3136	40	4810	261	580	12,811
04/06/17	4181	Outbound	359	3780	40	2531	245	645	10,502
06/09/20	4164	Inbound	190	2707	40	3961	364	594	10,832

TMI Trans-Mars Injection
MOI Mars Orbit Insertion
TEI Trans-Earth Injection

* Launch from ISS altitude orbit
† 500 km circular orbit at Mars
** Assumes direct entry upon Earth return (all velocities in m/s)

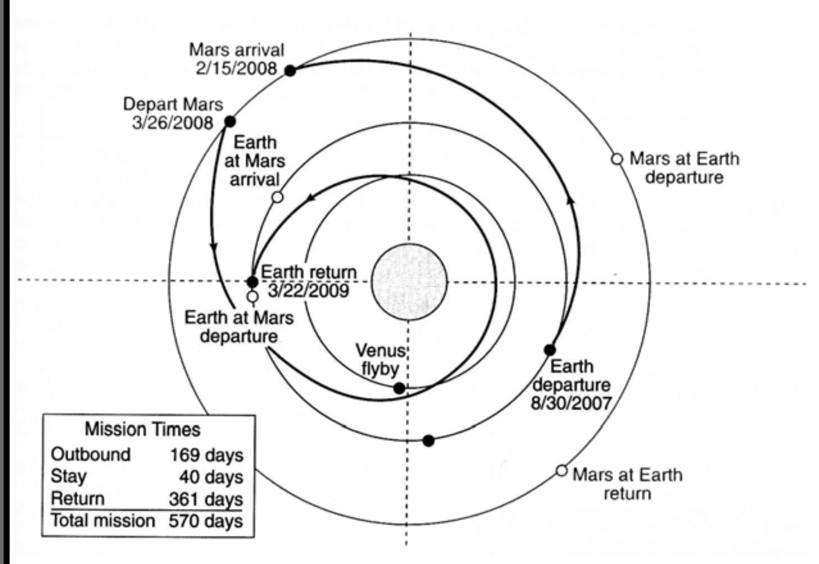
This data assumes that aerocapture will be used to make propellant delta-V for Earth braking unnecessary.

Venus Flyby Missions

Opposition-class & Venus Flyby Parameters

Opposition-class trajectories can be made more efficient by applying a deep-space “flyby” maneuver that uses another planet such as Venus as a “slingshot” to reduce propellant requirements:

- This trajectory applies the same opposition-class outbound transfer, using the swingby to add velocity on the return leg:
  - Flight crews will now stay on the Mars surface for roughly 60 days (compared with 3 weeks) before the Earth catch-up maneuver on the return.
  - Venus would reshape the return trajectory to provide a gravity-assist for a tangential approach to Earth.
  - Total trip time increases slightly to about 1.9 years (0.4 year more than non-swingby), but requires much less fuel with a more reasonable stay time.
  - Use of this approach requires proper phasing with Venus as well as with Mars.



One leg of this trajectory passes well inside the Earth’s orbit relative to the Sun.

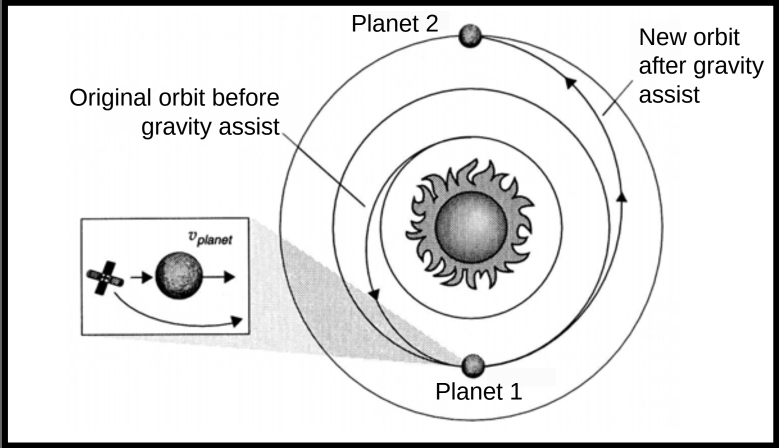
Venus Flyby Between Earth and Mars

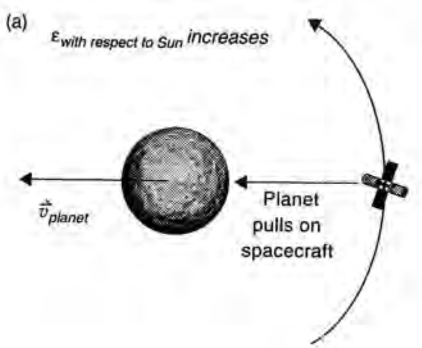
Opposition Mission with Flyby

During a gravity-assist maneuver, a planet pulls the spacecraft, changing its velocity with respect to the Sun and altering its orbit.

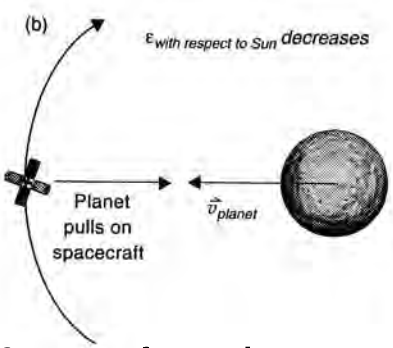
If a spacecraft passes behind a planet in the planet's orbit around the Sun, the spacecraft's energy/ velocity will increase (a).

If a spacecraft passes in front of a planet in the planet's orbit around the Sun, the spacecraft's energy/ velocity will decrease (b).





Spacecraft Passing Behind a Planet in its Orbit of the Sun

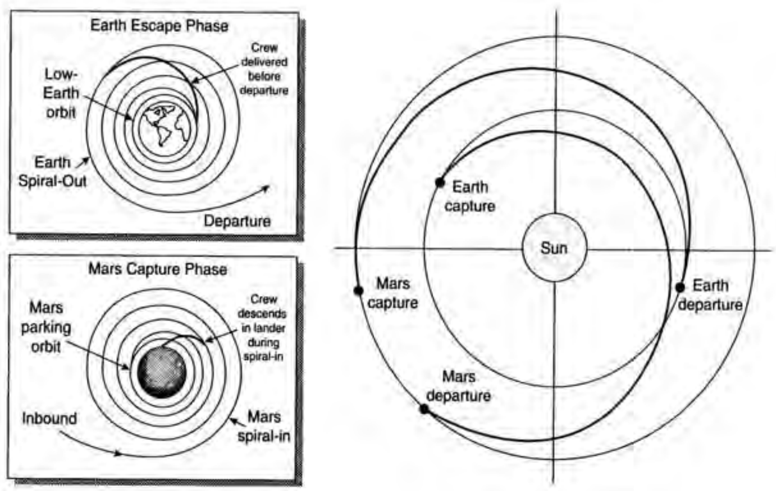


Spacecraft Passing Between a Planet and the Sun

Flyby Orbit Gravity- assists

Low-thrust trajectories for Mars are similar in fundamental aspects to those formerly discussed for the Moon:

- Beginning with a slow outward spiral to gain velocity necessary to escape Earth's gravity well, they continue to accelerate to interplanetary speed, then coast, and later reverse orientation to thrust for deceleration to the destination orbit:
  - If used also for Earth return, the procedure is reversed, although they may be most useful for efficient one-way cargo delivery missions.
  - Surface stay times prior to Earth return opportunities may range from 100-200 days, with total missions requiring about 2.5 years (depending upon the particular system's thrust characteristics).
  - Flight times can actually be faster than low-energy ballistic trajectories can achieve, but prolonged periods of spiral transitions through the Van Allen radiation belts present crew hazards.



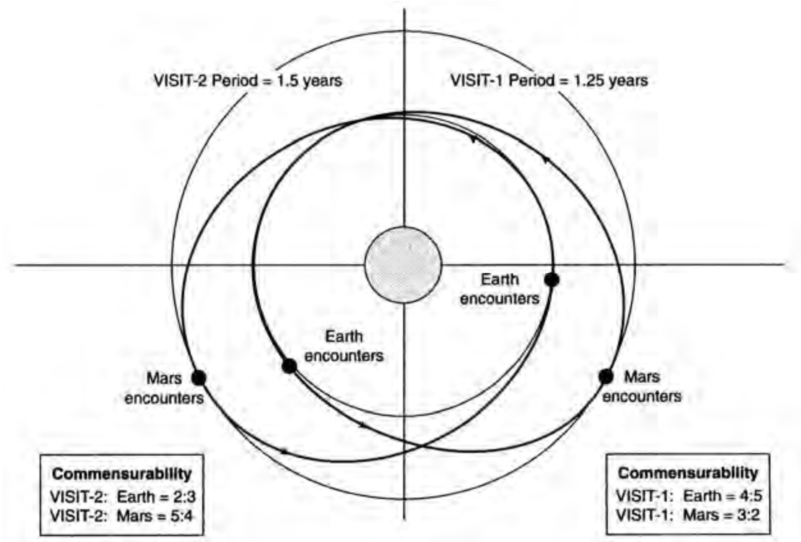
Low-thrust Trajectory Phases

Low-thrust Missions

HUMAN
SPACEFLIGHT

Cycler orbits are proposed to enable spacecraft to repeatedly re-encounter Earth and Mars to transfer crews and supplies using shuttle-type “taxis” to rendezvous between the surface and orbit at both planets.

- A “Versatile International Station for Interplanetary Transports” (VISIT) concept would accomplish each round-trip voyage in a 1.25 year period:
  - The VISIT orbit is commensurable with Earth in a 4:5 ratio (as the spacecraft travels around the Sun 4 times, the Earth would go around 5 times).
  - The spacecraft would re-encounter Earth every 5 years.
  - With Mars, the same orbit has a commensurability of 3:2 (the spacecraft completes 3 orbits while Mars orbits the Sun twice), and the spacecraft re-encounters Mars every 3.75 years.
  - To use these orbits most effectively, 2-3 cycler spacecraft would operate simultaneously in different Earth-Mars orientations.

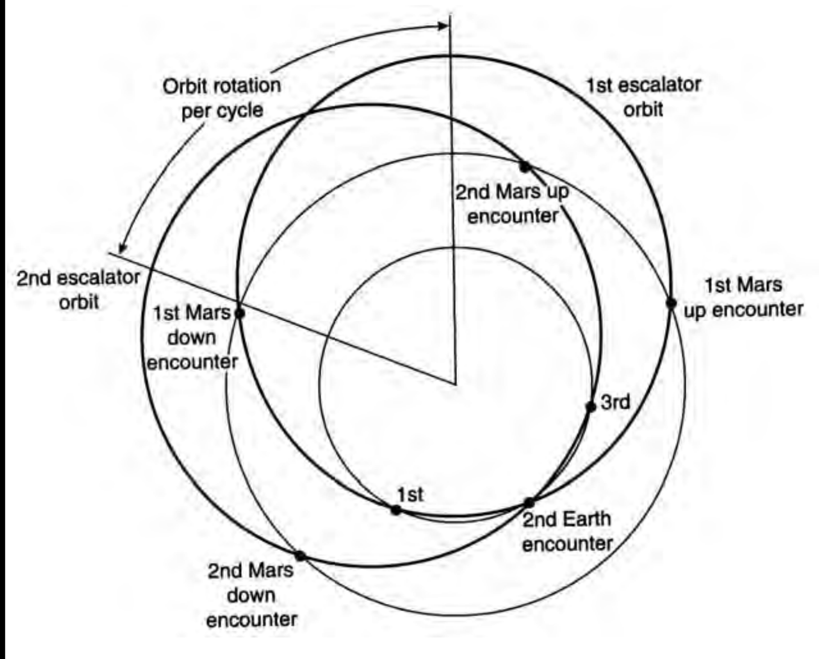


The VISIT Concept

Low-thrust Cycler Orbits

Apollo Astronaut, Dr. Buzz Aldrin has proposed a cycler concept known as the “UP/ DOWN-Escalator” orbit:

- The UP part of the orbit originates with a short transfer “up” past Mars for a longer transfer “down” through an Earth orbit:
  - An Earth gravity-assist maneuver around the orbit’s major axis sets the spacecraft on a path for the next Mars encounter, followed by another long trip back for another Earth swingby.
  - The swingbys adjust the spacecraft course to enable it to keep up with the progressive process of Earth-Mars phasing orientation (rotating the semi-major axis about 50° counterclockwise between successive phases around the Sun).
  - The Escalator orbits have relatively higher encounter velocities at Earth and Mars than the VISIT concept, but offer much more frequent encounters (every 2 years compared with 3.75), potentially reducing the number of cyclers needed.



UP/ DOWN- Escalator Concept

Additional information relevant to this section can be found in Part I of this SICSA Space Architecture Seminar Lecture Series titled Space Structures and Support Systems.

Key reference sources are the Cambridge Encyclopedia of Space (Cambridge University Press), the International Reference Guide to Space Launch Systems - Second Edition (AIAA Publication), Human Spaceflight Analysis and Design (The McGraw Hill Companies Inc.) and Spacecraft System Design and Operations (Kendall/Hunt Publishing Company). These and other relevant publications are listed below:

American Astronautical Society, *The Journal of Astronautical Sciences*, quarterly publication of the American Astronautical Society

Astore, W., Giffen, R., Larsen, W., *Understanding Space: An Introduction To Astronautics*. 2nd Edition. The McGraw-Hill Companies, Inc., 2000

Bate, Roger, R., et al. *Fundamentals of Astrodynamics*, Dover Publications, 1971. Calculus level mathematics

Battin, Richard H., 1987, *An Introduction to Mathematics and Methods of Astrodynamics*, New York, NY: AIAA Education Series

Belbruno, E., and Carrico, “Calculation of Weak Stability Boundary Ballistic Lunar Transfer trajectories”, AIAA/AAS Astrodynamics Specialist Conference, AIAA 2000-4142, Denver, Colorado, August 14-17, 2000

Capellari, Jr., J.O, ed. 1972, Where on the Moon? An Apollo Systems Engineering Problem. *The Bell Systems Technical Journal*, 51:995-1127

Cortright, Edgar, M., editor, *Apollo Expeditions to the Moon*, NASa SP-350, Government Printing Office, 1975

Damon, T.D. *Introduction To Space : The Science of Spaceflight*, 3rd Edition, Krieger Publishing Co., 2000

Melbourne, William, G., “Navigation Between the Planets”, *Scientific American*, June, 1976

Herman, A.L and B.A Conway, 1998, optimal, Low-thrust, Earth-Moon Orbit Transfer, *Journal of Guidance, Control, and Dynamics*, 21:141-147

Hoffman, S.J., et al. 1984, “Concepts for the Early Realization of Manned Mars Missions”, The Case for Mars II Conference, AAS Science & technology Series, Vol. 62, July 10-14, 1984, Boulder, co.

Houston, A., Rycroft, M., *Keys to Space- An Interdisciplinary Approach to Space Studies*, McGraw Hill, 1999

Jursa, A.S., ed. “Handbook of Geophysics and the Space Environment”, Air Force Geophysics Laboratory, Hansfcom, AFB, 1985

Kivelson, M., Russel, C.T., eds. *Introduction to Space Physics*, Cambridge University Press, 1995

Larson, W., Pranke, L., *Human Spaceflight: Mission Analysis and Design*, 1st edition, McGraw hill, 1996

Little, A.D., Inc. 1987, *Final Report of the Advanced Space Transportation Systems* (ASTS) Study, ADL Reference 54104.

Muolo, Michael, J., et al. Space handbook, Vol. 1, *A Warfighter’s Guide to Space* and Vol.2, *An Analyst’s Guide*, Air University Press, December, 1993. Mostly descriptive with some algebra

Neihoff, J.C., and S.J. Hoffman.1996, “Strategies for Mars: A Guide to Human Exploration”, Science & technology Series, A Supplement to Advances in the Astronautical Sciences, Vol. 86, AAS 95-478, ed. Carol Stoker and Carter Emmert, San Diego, CA: Univelt, Inc.

Nicogossian, A., Huntoon, C., Pool, S., Lea and Febiger *Space Physiology and Medicine*, Lea & Febiger, Philadelphia, PA, 1989

Peters, J., *Spacecraft Systems Design and Operations*, Kendall/ Hunt Publishing Co., Dubuque, Iowa, 2002

Prussing, John E., and Bruce A. Conway, *Orbital Mechanics*, Oxford University Press, 1993. Vector calculus levels mathematics

Sellers, Jerry.Jon, 1994, *Understanding Space: An Introduction to Astronautics*, New York, McGraw Hill.
Tascione, T., *Introduction to the Space Environment*, Orbit Book Company, 1988.
Thomson, William T., *Introduction to Space Dynamics*, Dover Publications, 1986. calculus level mathematics
Tribble, A., The Space Environment, Princeton University Press, 1995
Vallado, David A., 1997, *Fundamentals of Astrodynamics & Applications*, New York, McGraw Hill.
Zombeck, M.V, Handbook of Space Astronomy and Astrophysics, Cambridge University Press, 1990

BACK TO THE LIST OF CONTENTS

ACRONYMS

AFSCN	Air Force Satellite Control Network
AOA	Abort Once Around
ASAT	Anti-Satellite
ATCS	Active Thermal Control System
ATV	Autonomous Transfer Vehicle
ATU	Audio Thermal Unit
ATLO	Assembly, Test & Launch Operations
AV	Avionics
C&C	Command and Control
C&C MDM	Command and Control Multiplexer/ Demultiplexer
C&T	Communication and Tracking
CAS	Common Attach System
CB	Control Bus
CBM	Common Berthing Mechanism
CELSS	Controlled Ecological Life Support System
CLCS	Checkout & Launch Complex System
CMG	Control Moment Gyroscope
CNTL	Control
COF	Columbus Orbital Facility
CRV	Crew Rescue Vehicle
EACP	EVA Audio Control Panel
ECLSS	Environmental Control and Life Support System
EE	End Effectors
ELV	Expandable Launch Vehicle
ELOC	Extended Loss of Communication
EMU	Extravehicular Mobility Unit
EOR	Earth orbit Rendezvous
ESSMDM	Enhanced MDM
ETA	Environmental Test Article
ETCS	External Thermal Control System
ETO	Earth to Orbit
EVA	Extravehicular Activity
FDS	Fire Detection and Suppression
FF	Free Flyer
FRCI	Fibrous Refraction Composite Insulation
FRGF	Flight Releasable Grapple Fixture
GEO	Geosynchronous Earth Orbit
GF	Grapple Fixture
GLONASS	Global Navigation Satellite System
GN_2	Gaseous Nitrogen
GSE	Ground Support Equipment Government Support Equipment
GSTDN	Ground Station Tracking and Data Network
GTO	Geosynchronous Transfer Orbit
Hab	Habitation Module

IFHX	Interface Heat Exchanger
IMMI	IVA Man-machine Interface
IOCU	Input/ Output Controller Unit
IV	Intravenous
IVA	Intravehicular Activity
JEM	Japanese Experiment Module
JEMPM	Japanese Experiment Module Pressurized Module
JPL	Jet Propulsion Lab
JSC	Johnson Space Center
KSC	Kennedy Space Center
Lab	Laboratory Module
LEO	Low Earth Orbit
LEM	Lunar Excursion Module
LGA	Low Gain Antenna
LLA	Low Load Analog
LOR	Lunar orbit Rendezvous
LOS	Loss-of-Signal
LOX	Liquid Oxygen
LSS	Life Support System
LVLH	Local Vertical/ Local Horizontal
MA	Main Arm
MBM	Manual Berthing Mechanism
MBS	Mobile Remote Servicer Base System
MCC-H	Mission Control Center-Houston
MCC-M	Mission Control Center-Moscow
MM/ OD	Micrometeroroid/ Orbital Debris
MON	Monitoring System
MPD	Magnetoplasmadynamics
MLP	Mobile Launch Platform
MO&DA	Mission Operations and Data Analysis
MPLM	Multi-Purpose Logistics Module
MPS	Main Propulsion System
MSFC	Marshall Space Flight Center
MT	Mobile Transporter
NASA	National Aeronautics and Space Administration
NASDA	National Space and Development Agency (Japan)
ORU	Orbital Replacement Unit
OSP	Orbital Space Plane
PDGF	Power and Data Grapple Fixture
PLL	Phase Lock Loop
PLSS	Primary Life Support System
PM	Propulsion Module
PMA	Pressurized Mating Adapters
PRSD	Power Reactant Storage and Distribution

ACRONYMS

PTCS	Passive Thermal Control System
PV	Photovoltaic
PVA	Photovoltaic Array
PVR	Photovoltaic Radiator
PVTCS	Photovoltaic Thermal Control System
PYR	Pitch, Yaw, and Roll
QD	Quick Disconnect
QF	Quality Factor
RA	Radar Altimeter
RACU	Russian-to-American Converter Unit
RAD	Radiation Dose
RCS	Reaction Control System
RGA	Rate Gyro Assembly
RIP	Reusable Interface Panel
RLV	Reusable Launch Vehicle
RMS	Remote Manipulator System
RPM	Rotations per minute
RPY	Roll, Pitch, Yaw
RT	Remote terminal
RTAS	Rocketdyne Truss Attach System
RTG	Radioisotope Thermoelectronic Generator
SAA	South Atlantic Anomaly
SAFER	Simplified Aid for EVA Rescue
S&M	Structures and Mechanisms
SAS	Space Adaptation Syndrome
SAW	Solar Array Wing
SPDA	Secondary Power Distribution Assembly
SPG	Single Point Ground
SPP	Science Power Platform
SRB	Solid Rocket Booster
SSA	Space Suit Assembly
SSCS	Space to Space Communication System
SSMDM	Standard MDM
SSPF	Space Station Processing Facility
SSSR	Space-to-Space Station Radio
SSU	Sequential Shunt Unit
STDN	Spaceflight Tracking and Data Network
STS	Space Transportation System
TDRS	Tracking and Data Relay Satellite
TPS	Thermal Protection System
TRK	Tracking System
T/W	Thrust/ Weight
WFPC	Wide Field Planetary Camera
WLP	Wallops Flight Facility
WM	Waste Management
WSC	White Sands Center
XPOP	X-axis Pointing Out of Plane
YPR	Yaw, Pitch and Roll

ACRONYMS