Handbook of Space Access & Expendable Space Launch Systems

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Table of Contents

Chapter 1

Beam-powered Propulsion

Beam-powered propulsion is a class of aircraft or spacecraft propulsion mechanisms that use energy beamed to the spacecraft from a remote power plant to provide energy. Most designs are rocket engines where the energy is provided by the beam, and is used to superheat propellant that then provides propulsion, although some obtain propulsion directly from light pressure acting on a light sail structure, and at low altitude heating air gives extra thrust.

The beam would typically either be a beam of microwaves or a laser. Lasers are subdivided into either pulsed or continuous beamed.

The rule of thumb that is usually quoted is that it takes a megawatt of power beamed to a vehicle per kg of payload while it is being accelerated to permit it to reach low earth orbit.

Other than launching to orbit, applications for moving around the world quickly have also been proposed.

Background

Rockets are momentum machines; they use mass ejected from the rocket to provide momentum to the rocket. Momentum is the product of mass and velocity, so rockets generally attempt to put as much velocity into their working mass as possible, thereby minimizing the amount of working mass that is needed. In order to accelerate the working mass, energy is required. In a conventional rocket, the fuel is chemically combined to provide the energy, and the resulting fuel products, the ash or exhaust, are used as the working mass.

There is no particular reason why the same fuel has to be used for both energy and momentum. In the jet engine, for instance, the fuel is used only to produce energy, the working mass is provided from the air that the jet aircraft flies through. In modern jet engines, the amount of air propelled is much greater than the amount of air used for energy, bypass ratios of 10 to 1 or greater are typical. This is not a solution for the rocket, however, as they quickly climb to altitudes where the air is too thin to be useful as a source of working mass.

Rockets can, however, carry their working mass and use some other source of energy. The problem is finding an energy source with a power-to-weight ratio that competes with chemical fuels. Small nuclear reactors can compete in this regard, and considerable work on nuclear thermal propulsion was carried out in the 1960s, but environmental concerns and rising costs led to the ending of most of these programs.

A further improvement can be made by removing the energy creation off of the spacecraft entirely. If the nuclear reactor is left on the ground and its energy transmitted to the spacecraft, the weight of the reactor is removed as well. The issue then is to get the energy into the spacecraft. This is the idea behind beamed power.

Thermal propulsion

With beamed propulsion one can leave the power-source stationary on the ground, and directly (or via a heat exchanger) heat propellant on the spacecraft with a maser or a laser beam from a fixed installation. This permits the spacecraft to leave its power-source at home, saving significant amounts of mass, greatly improving performance.

Since a laser can heat propellant to extremely high temperatures, this potentially greatly improves the efficiency of a rocket, as exhaust velocity is proportional to the square root of the temperature. Normal chemical rockets have an exhaust speed limited by the fixed amount of energy in the propellants, but beamed propulsion systems have no particular theoretical limit (although in practice there are temperature limits).

Ablative Laser Propulsion is a form of laser propulsion that uses a laser to create a plasma plume from a metal propellant, thus producing thrust.

In addition, microwaves can be used to heat a suitable heat exchanger, which in turn heats a propellant (very typically hydrogen). This can give a combination of high specific impulse (700–900 seconds) as well as good thrust/weight ratio (50-150)..

A variation, developed by brothers James Benford and Gregory Benford, is to use thermal desorption of propellant trapped in the material of a very large microwave-sail. This produces a very high acceleration compared to microwave pushed sails alone.

Electric propulsion

Some proposed spacecraft propulsion mechanisms use power in the form of electricity. Usually these schemes assume either solar panels, or an on-board reactor. However, both power sources are heavy.

Beamed propulsion in the form of laser can be used to send power to a photovoltaic panel, for *Laser electric propulsion.* In this system, careful design of the panels is necessary as the extra power tends to cause a fall-off of the conversion efficiency due to heating effects.

A microwave beam could be used to send power to a rectenna, for *microwave electric propulsion*. Microwave broadcast power has been practically demonstrated several times (e.g. Goldstone, California in 1974), rectennas are potentially lightweight and can handle high power at high conversion efficiency. However, rectennas tend to need to be very large for a significant amount of power to be captured.

Direct impulse

A beam could also be used to provide impulse by directly "pushing" on the sail.

One example of this would be using a solar sail to reflect a laser beam. This concept, called a *laser-pushed lightsail,* was analyzed by physicist Robert L. Forward in 1989 as a method of Interstellar travel that would avoid extremely high mass ratios by not carrying fuel. His work elaborated on a proposal initially made by Marx. Further analysis of the concept was done by Landis , Mallove and Matloff , Andrews and others.

In a later paper, Forward proposed pushing a sail with a microwave beam . This has the advantage that the sail need not be a continuous surface. Forward tagged his proposal for an ultralight sail "Starwisp". A later analysis by Landis suggested that the Starwisp concept as originally proposed by Forward would not work, but variations on the proposal continue to be proposed.

The beam has to have a large diameter so that only a small portion of the beam misses the sail due to diffraction and the laser or microwave antenna has to have a good pointing stability so that the craft can tilt its sails fast enough to follow the center of the beam. This gets more important when going from interplanetary travel to interstellar travel, and when going from a fly-by mission, to a landing mission, to a return mission. The laser or the microwave sender would probably be a large phased array of small devices, which get their energy directly from solar radiation. The size of the array obsoletes any lens or mirror.

Another beam-pushed concept would be to use a magnetic sail or MMPP sail to divert a beam of charged particles from a particle accelerator or plasma jet . Jordin Kare has proposed a variant to this whereby a "beam" of small laser accelerated light sails would transfer momentum to a magsail vehicle.

Another beam-pushed concept uses ordinary matter and works in vacuum. The matter from a stationary mass-driver is "reflected" by the spacecraft, cf. mass driver. The spacecraft neither needs energy nor reaction mass for propulsion of its own.

Proposed systems

Lightcraft

A **lightcraft** is a vehicle currently under development that uses an external pulsed source of laser or maser energy to provide power for producing thrust.

The laser shines on a parabolic reflector on the underside of the vehicle that concentrates the light to produce a region of extremely high temperature. The air in this region is heated and expands violently, producing thrust with each pulse of laser light. In space, a lightcraft would need to provide this gas itself from onboard tanks or from an ablative solid. By leaving the vehicle's power source on the ground and by using ambient atmosphere as reaction mass for much of its ascent, a lightcraft would be capable of delivering a very large percentage of its launch mass to orbit. It could also potentially be very cheap to manufacture.

Testing

Early in the morning of 2 October 2000 at the High Energy Laser Systems Test Facility (HELSTF), Lightcraft Technologies, Inc. (LTI) with the help of Franklin B. Mead of the U.S. Air Force Research Laboratory and Leik Myrabo set a new world's altitude record of 233 feet (71 m) for its 4.8 inch (12.2 cm) diameter, 1.8 ounce, laser-boosted rocket in a flight lasting 12.7 seconds. Although much of the 8:35 am flight was spent hovering at 230+ feet, the Lightcraft earned a world record for the longest ever laser-powered free flight and the greatest "air time" (i.e., launch-to-landing/recovery) from a light-propelled object. This is comparable to Robert Goddard's first test flight of his rocket design. Increasing the laser power to 100 kilowatts will enable flights up to a 30-kilometer altitude. Their goal is to accelerate a one-kilogram microsatellite into low Earth orbit using a custom-built, one megawatt ground-based laser. Such a system would use just about 20 dollars' worth of electricity, placing launch costs per kilogram to many times less than current launch costs (which are measured in thousands of dollars).

Myrabo's "lightcraft" design is a reflective funnel-shaped craft that channels heat from the laser, towards the center, using a reflective parabolic surface causing the laser to literally explode the air underneath it, generating lift. Reflective surfaces in the craft focus the beam into a ring, where it heats air to a temperature nearly five times hotter than the surface of the sun, causing the air to expand explosively for thrust.

Jordin Kare's heat exchanger system

Jordin Kare has proposed a simpler, nearer term concept which has a rocket containing liquid hydrogen and water. The propellant is heated in a heat exchanger that the laser beam shines on before leaving the vehicle via a conventional nozzle. This concept can use continuous beam lasers, and the semiconductor lasers are now cost effective for this application.

Non-spacecraft applications

In 1964 William C. Brown demonstrated a miniature helicopter equipped with a combination antenna and rectifier device called a rectenna. The rectenna converted microwave power into electricity, allowing the helicopter to fly.

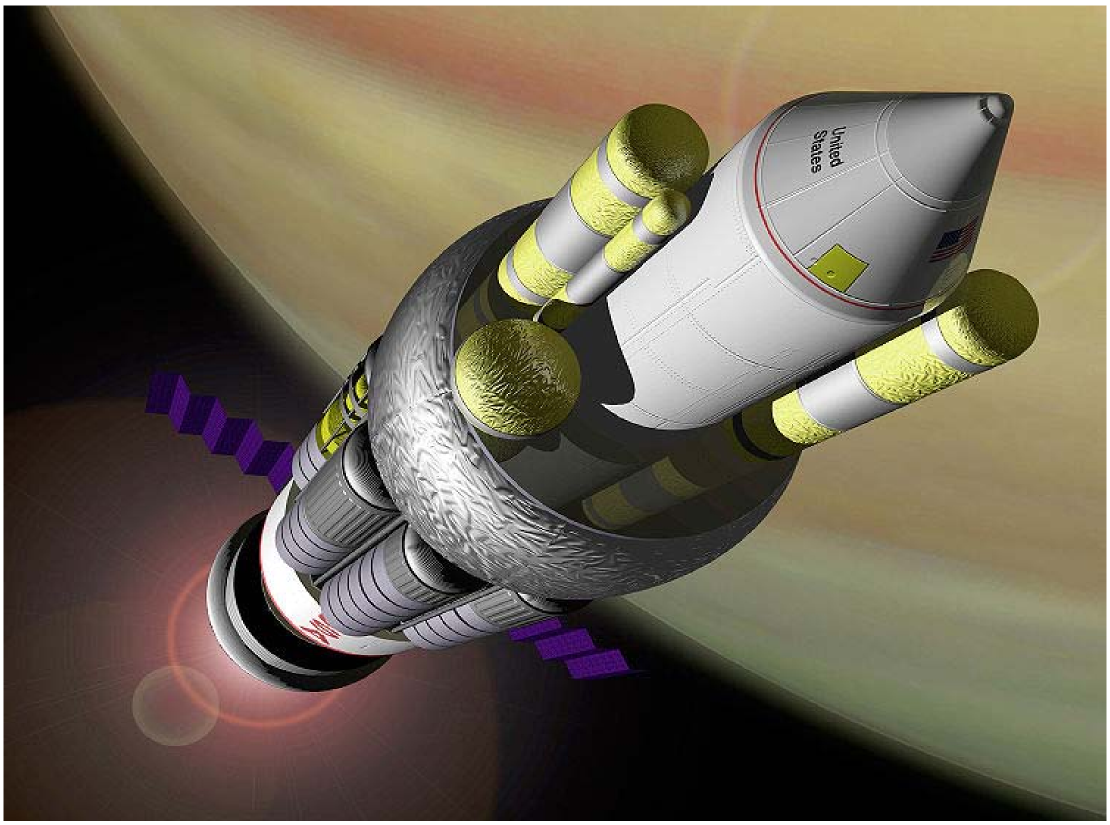
In 2002 a Japanese group propelled a tiny aluminium airplane by using a laser to vaporize a water droplet clinging to it, and in 2003 NASA researchers flew an 11 ounce (312 g) model airplane with a propeller powered with solar panels illuminated by a laser. It is possible that such beam-powered propulsion could be useful for long-duration high altitude unmanned aircraft or balloons, perhaps designed to serve as communication relays or surveillance platforms.

A "laser broom" has been proposed to sweep space debris from Earth orbit. This is another proposed use of beam-powered propulsion, used on objects that were not designed to be propelled by it, for example small pieces of scrap knocked off ("spalled") satellites. The technique works since the laser power ablates one side of the object, giving an impulse that changes the eccentricity of the object's orbit. The orbit would then intersect the atmosphere and burn up.

"Lasermotive" demonstrated laser powerbeaming at one kilometer during NASA's 2009 powerbeaming contest. Also "Lighthouse DEV" (a spin off of NASA Power Beaming Team) along with "University of Maryland" is developing an eye safe laser system to power a small UAV.

Chapter 2

Project Orion



An artist's conception of the NASA reference design for the Project Orion spacecraft powered by nuclear propulsion.

Project Orion was a study of a spacecraft intended to be directly propelled by a series of explosions of atomic bombs behind the craft (Nuclear pulse propulsion). Early versions of this vehicle were proposed to have taken off from the ground with significant associated nuclear fallout; later versions were presented for use only in space.

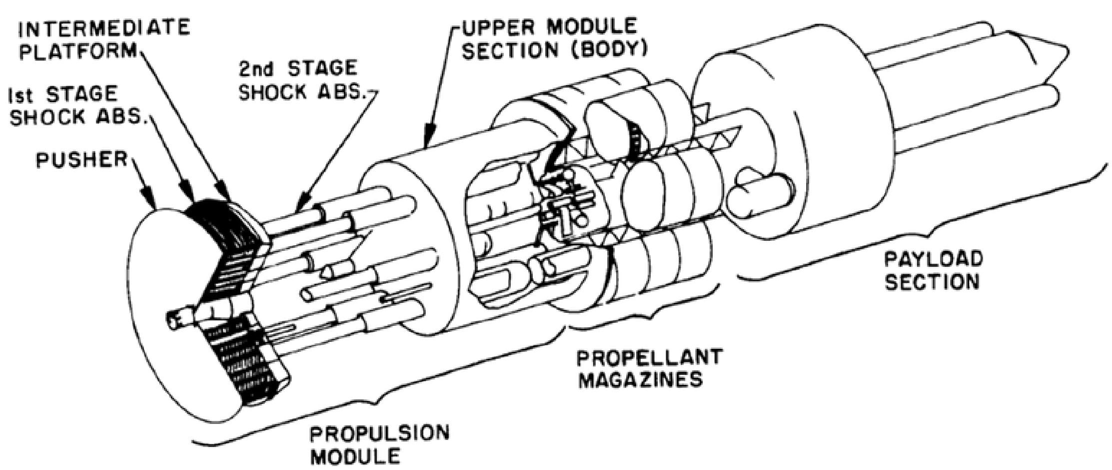
A 1955 Los Alamos Laboratory document states (without offering references) that general proposals were first made by Stanislaw Ulam in 1946, and that preliminary calculations were made by F. Reines and Ulam in a Los Alamos memorandum dated 1947. The actual project, initiated in 1958, was led by Ted Taylor at General Atomics and physicist Freeman Dyson, who at Taylor's request took a year away from the Institute for Advanced Study in Princeton to work on the project.

By using energetic nuclear power, the Orion concept offered high thrust and high specific impulse, or propellant efficiency, at the same time. As a qualitative comparison, traditional chemical rockets—such as the Saturn V that took the Apollo program to the Moon—produce high thrust with low specific impulse, whereas electric ion engines produce a small amount of thrust very efficiently. Orion would have offered performance greater than the most advanced conventional or nuclear rocket engines then under consideration. Supporters of Project Orion felt that it had potential for cheap interplanetary travel, but it lost political approval over concerns with fallout from its propulsion. The Partial Test Ban Treaty of 1963 is generally acknowledged to have ended the project.

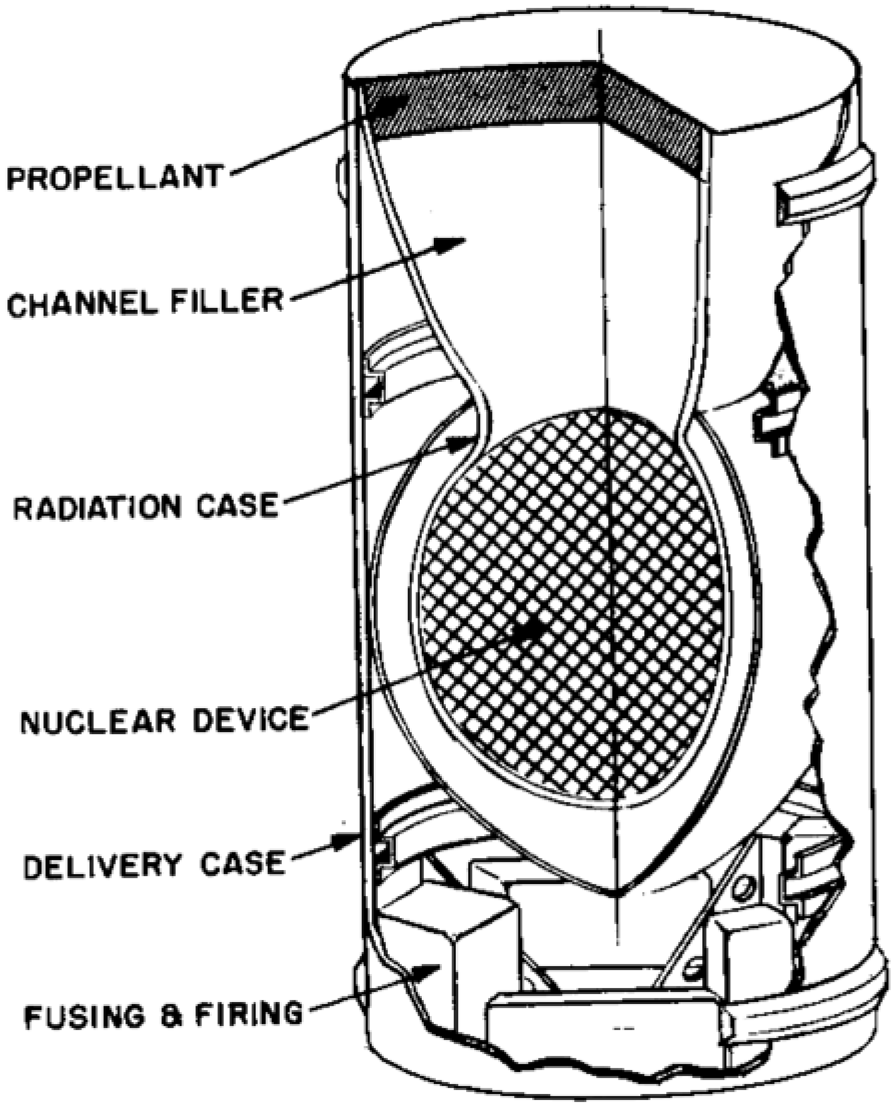
Nuclear power

During the late 1940s, Stanisław Ulam realized that nuclear explosions could *not yet* be realistically contained in a combustion chamber. Such a project did briefly exist, named Helios, but while its theoretical performance was similar to that of what would become the Orion, the lack of materials that could withstand the propulsion generating process meant that Helios never got beyond the drawing board.

Instead, the Orion design would have worked by dropping small shaped charge fission or thermonuclear explosives (referred to as *pulse units*) out the rear of a vehicle, detonating them 200 feet (60 m) out, and catching the blast with a thick steel or aluminum pusher plate.



The Orion Spacecraft – key components



A design for a pulse unit

Large multi-level high shock absorbers (pneumatic springs) were to have absorbed the impulse from the plasma wave as it hit the pusher plate, spreading the millisecond shock wave over several seconds and thus giving an acceptable acceleration speed. The long arm pistons proved one of the most difficult design features. Low pressure gas bags were also proposed as a primary shock absorber. The two sets of shock absorption systems were tuned to different frequencies to avoid resonance.

One aspect of the proposed vessel seems counter-intuitive today: because of the force involved in the thermonuclear detonations and the need to absorb the energy without harm, massive vessel designs were needed. Early designs had crew compartments and storage areas that were several stories tall, as opposed to contemporary chemical rockets whose height was almost all multi-stage fuel tanks with relatively little payload.

Reaction mass for Orion would have been built into the bombs or dropped between 'pulses' to provide thrust. Polyethylene masses, garbage and sewage were all considered for use as reaction mass.

The 'base design' consisted of a 4,000 ton model planned for ground launch from Jackass Flats, Nevada. Each 0.15 kt of TNT (600 MJ) (sea-level yield) blast would add 30 mph (50 km/h, 13 m/s) to the craft's velocity. A graphite based oil would be sprayed on the pusher plate before each explosion to prevent ablation of the surface. To reach low Earth orbit (300 mi), this sequence would have to be repeated about 800 times, like an atomic pogo stick.

Jerry Pournelle, who is acquainted with the project and its former team leader Freeman Dyson, has been quoted as saying that a single mission could have provided us with a large permanent moon base. Alternatively, an Orion could reach Pluto and return to Earth inside of a year. In the early 1960s a NASA study proposed a rocket launched hybrid for a Mars round trip.

Background

In the 1954 Operation Castle nuclear test series at Bikini Atoll, a crucial experiment by Lew Allen proved that nuclear explosives could be used for propulsion. Two graphite-covered steel spheres were suspended near the test article for the Castle Bravo shot. After the explosion, they were found intact some distance away, proving that engineered structures could survive a nuclear fireball.

Performance

The Orion nuclear pulse drive combines a very high exhaust velocity, from 20 to 30 km/s, with meganewtons of thrust. Many spacecraft propulsion drives can achieve one of these or the other, but nuclear pulse rockets are the only proposed technology that could potentially deliver both. Specific impulse measures how much thrust can be derived from a given mass of fuel, and is the standard figure of merit for rocketry.

Since weight is no limitation, an Orion craft can be extremely robust. An unmanned craft could tolerate very large accelerations, perhaps 100 *g*. A human-crewed Orion, however, must use some sort of damping system behind the pusher plate to smooth the instantaneous acceleration to a level that humans can comfortably withstand – typically about 2 to 4 *g*.

The high performance depends on the high exhaust velocity, in order to maximize the rocket's force for a given mass of propellant. The velocity of the plasma debris is proportional to the square root of the change in the temperature (T_c) of the nuclear fireball. Since fireballs routinely achieve ten million degrees Celsius or more in less than a millisecond, they create very high velocities. However, a practical design must also limit the destructive radius of the fireball. The diameter of the nuclear fireball is proportional to the square root of the bomb's explosive yield.

The shape of the bomb's reaction mass is critical to efficiency. The original project designed bombs with a reaction mass made of tungsten. The bomb's geometry and materials focused the X-rays and plasma from the core of nuclear explosive to hit the reaction mass. In effect each bomb would be a nuclear shaped charge.

A bomb with a cylinder of reaction mass expands into a flat, disk-shaped wave of plasma when it explodes. A bomb with a disk-shaped reaction mass expands into a far more efficient cigar-shaped wave of plasma debris. The cigar shape focuses much of the plasma to impinge onto the pusher-plate.

For example, a 10 kiloton of TNT equivalent atomic explosion will achieve a plasma debris velocity of about 100 km/s, and the destructive plasma fireball is only about 100 meters in diameter. A 1 megaton TNT explosion will have a plasma debris velocity of about 10,000 km/s but the diameter of the plasma fireball will be about 1000 m..

The maximum effective specific impulse, I_{sp}, of an Orion nuclear pulse drive generally is equal to:

$$I_{sp} = \frac{C_0 \cdot V_e}{g_n}$$

where C_0 is the collimation factor (what fraction of the explosion plasma debris will actually hit the impulse absorber plate when a pulse unit explodes), V_e is the nuclear pulse unit plasma debris velocity, and g_n is the standard acceleration of gravity (9.81 m/s^2; this factor is not necessary if I_{sp} is measured in N·s/kg or m/s). A collimation factor of nearly 0.5 can be achieved by matching the diameter of the pusher plate to the diameter of the nuclear fireball created by the explosion of a nuclear pulse unit.

The smaller the bomb, the smaller each impulse will be, so the higher the rate of impulses and more than will be needed to achieve orbit. Smaller impulses also mean less *g* shock on the pusher plate and less need for damping to smooth out the acceleration.

The optimal Orion drive bomblet yield (for the human crewed 4,000 ton reference design) was calculated to be in the region of 0.15 KT, with approx 800 bombs needed to orbit and a bomb rate of approx 1 per second.

Sizes of Orion vehicles

The following can be found in George Dyson's book pg. 55 published in 2002. The figures for the comparison with Saturn V are taken from this section and converted from metric (kg) to US short tons.

	Orbital Test	Interplanetary	Advanced interplanetary	Saturn V
Ship mass	880 t	4,000 t	10,000 t	3,350 t
Ship diameter	25 m	40 m	56 m	10 m
Ship height	36 m	60 m	85 m	110 m
Bomb yield (sea level)	0.03 kt	0.14 kt	0.35 kt	n/a
Bombs (to 300 mi Low Earth Orbit)	800	800	800	n/a
Payload (to 300 mi LEO)	300 t	1,600 t	6,100 t	130 t
Payload (to Moon soft landing)	170 t	1,200 t	5,700 t	52 t
Payload (Mars orbit return)	80 t	800 t	5,300 t	–
Payload (3yr Saturn return)	–	–	1,300 t	–

In late 1958 / early 1959 it was realized that the smallest practical vehicle would be determined by the smallest achievable bomb yield. The use of 0.03 kT (sea-level yield) bombs would give vehicle mass of 880 tons. However this was regarded as too small for anything other than an orbital test vehicle and the team soon focused on a 4,000 ton 'base design'.

At that time, the details of small bomb designs were shrouded in secrecy. Many Orion design reports had all details of bombs removed before release. Contrast the above details with the 1959 report by General Atomics, which explored the parameters of three different sizes of hypothetical Orion spacecraft:

	"Satellite" Orion	"Midrange" Orion	"Super" Orion
Ship diameter	17–20 m	40 m	400 m
Ship mass	300 t	1000–2000 t	8,000,000 t
Number of bombs	540	1080	1080
Individual bomb mass	0.22 t	0.37–0.75 t	3000 t

The biggest design above is the "super" Orion design; at 8 million tonnes, it could easily be a city. In interviews, the designers contemplated the large ship as a possible interstellar ark. This extreme design could be built with materials and techniques that could be obtained in 1958 or were anticipated to be available shortly after. The practical upper limit is likely to be higher with modern materials.

Most of the three thousand tonnes of each of the "super" Orion's propulsion units would be inert material such as polyethylene, or boron salts, used to transmit the force of the propulsion units detonation to the Orion's pusher plate, and absorb neutrons to minimize fallout. One design proposed by Freeman Dyson for the "Super Orion" called for the pusher plate to be composed primarily of uranium or a transuranic element so that upon reaching a nearby star system the plate could be converted to nuclear fuel.

Interplanetary applications

The Orion nuclear pulse rocket design has extremely high performance. Orion nuclear pulse rockets using nuclear fission type pulse units were originally intended for use on interplanetary space flights.

Missions that were designed for an Orion vehicle in the original project included single stage (i.e., directly from Earth's surface) to Mars and back, and a trip to one of the moons of Saturn.

One possible modern mission for this near-term technology would be to deflect an asteroid that could collide with Earth. The extremely high performance would permit even a late launch to succeed, and the vehicle could effectively transfer a large amount of kinetic energy to the asteroid by simple impact. Also, such an unmanned mission would eliminate the need for shock absorbers, the most problematic issue of the design.

Nuclear fission pulse unit powered Orions could provide fast and economical interplanetary transportation with useful human crewed payloads of several thousand tonnes.

Interstellar missions

Freeman Dyson performed the first analysis of what kinds of Orion missions were possible to reach Alpha Centauri, the nearest star system to the Sun . His 1968 paper "Interstellar Transport" (Physics Today, October 1968, p. 41–45) retained the concept of large nuclear explosions but Dyson moved away from the use of fission bombs and considered the use of one megaton deuterium fusion explosions instead. His conclusions were simple: the debris velocity of fusion explosions was probably in the 3000–30,000 km/s range and the reflecting geometry of Orion's hemispherical pusher plate would reduce that range to 750–15,000 km/s.

To estimate the upper and lower limits of what could be done using contemporary technology (in 1968), Dyson considered two starship designs. The more conservative *energy limited* pusher plate design simply had to absorb all the thermal energy of each impinging explosion (4×10^{15} joules, half of which would be absorbed by the pusher plate) without melting. Dyson estimated that if the exposed surface consisted of copper with a thickness of 1 mm, then the diameter and mass of the hemispherical pusher plate would have to be 20 kilometers and 5 million metric tons, respectively. 100 seconds would be required to allow the copper to radiatively cool before the next explosion. It would then take on the order of 1000 years for the energy-limited heat sink Orion design to reach Alpha Centauri.

In order to improve on this performance while reducing size and cost, Dyson also considered an alternative *momentum limited* pusher plate design where an ablation coating of the exposed surface is substituted to get rid of the excess heat. The limitation is then set by the capacity of shock absorbers to transfer momentum from the impulsively accelerated pusher plate to the smoothly accelerated vehicle. Dyson calculated that the properties of available materials limited the velocity transferred by each explosion to ~30 meters per second independent of the size and nature of the explosion. If the vehicle is to be accelerated at 1 Earth gravity (9.81 m/s) with this velocity transfer, then the pulse rate is one explosion every three seconds. The dimensions and performance of Dyson's vehicles are given in the table below

	"Energy Limited" Orion	**"Momentum Limited" Orion**
Ship diameter (meters)	20,000 m	100 m
Mass of empty ship (metric tons)	10,000,000 t (incl.5,000,000 t copper hemisphere)	100,000 t (incl.50,000 t structure+payload)
+Number of bombs = total bomb mass (each 1MT bomb weighs 1 metric ton)	30,000,000	300,000
=Departure mass (metric tons)	40,000,000 t	400,000 t
Maximum velocity (kilometers per second)	1000 km/s (=0.33% of the speed of light)	10,000 km/s (=3.3% of the speed of light)
Mean acceleration (Earth gravities)	0.00003 g (accelerate for 100 years)	1 g (accelerate for 10 days)
Estimated cost	1 year of U.S. GNP (1968)	0.1 year of U.S. GNP

Later studies indicate that the top cruise velocity that can theoretically be achieved by a thermonuclear Orion starship is about 8% to 10% of the speed of light (0.08-0.1c). An atomic (fission) Orion can achieve perhaps 3%-5% of the speed of light. A nuclear pulse drive starship powered by matter-antimatter pulse units would be theoretically capable of obtaining a velocity between 50% to 80% of the speed of light.

At 0.1*c*, Orion thermonuclear starships would require a flight time of at least 44 years to reach Alpha Centauri, not counting time needed to reach that speed (about 36 days at constant acceleration of 1*g* or 9.8 m/s^2). At 0.1*c*, an Orion starship would require 100 years to travel 10 light years. The late astronomer Carl Sagan suggested that this would be an excellent use for current stockpiles of nuclear weapons.

Later developments

A concept similar to Orion was designed by the British Interplanetary Society (B.I.S.) in the years 1973–1974. Project Daedalus was to be a robotic interstellar probe to Barnard's Star that would travel at 12% of the speed of light. In 1989, a similar concept was studied by the U.S. Navy and NASA in Project Longshot. Both of these concepts require significant advances in fusion technology, and therefore cannot be built at present, unlike Orion.

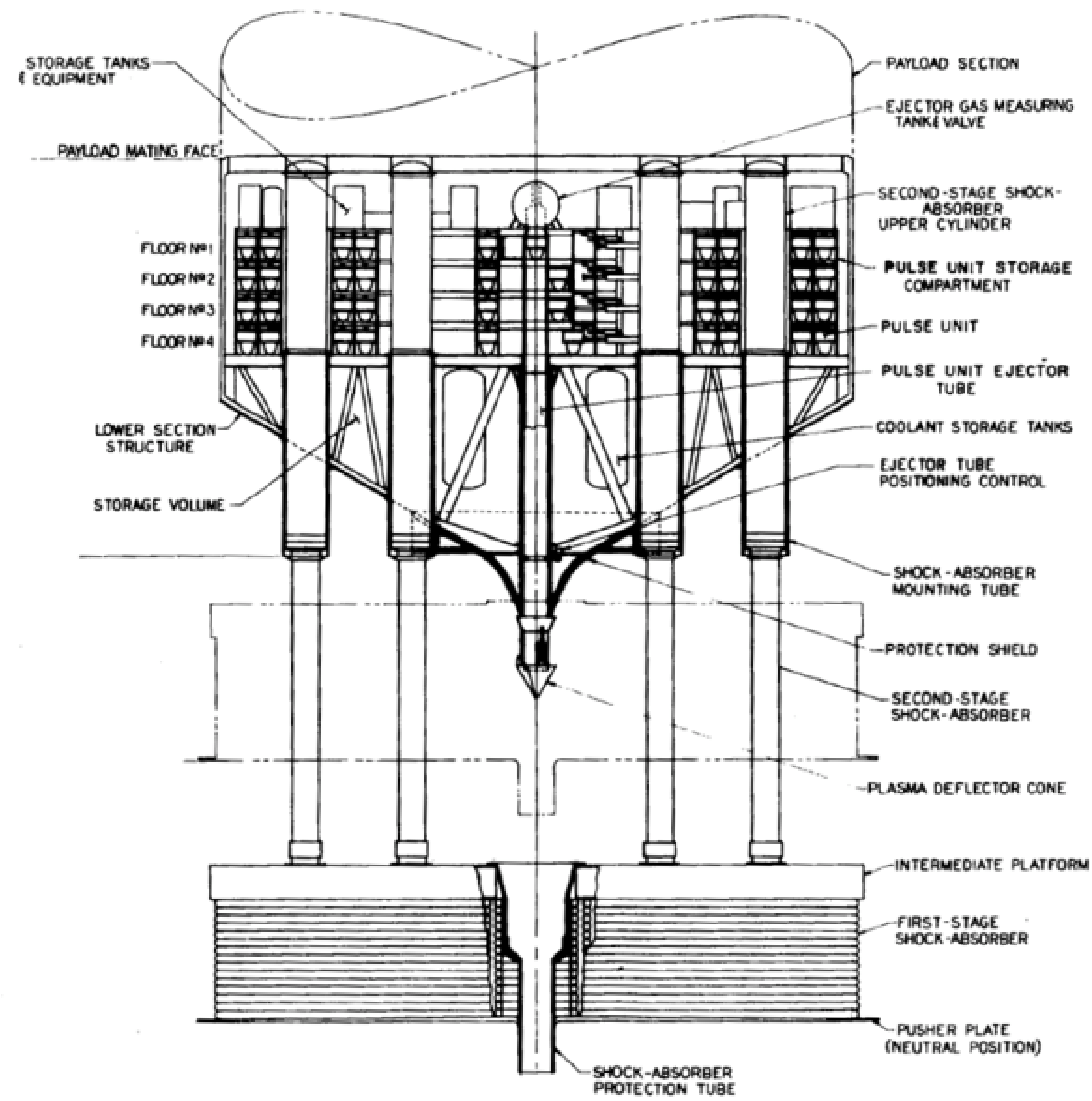
From 1998 to the present, the nuclear engineering department at Pennsylvania State University has been developing two improved versions of the Daedalus design known as Project Ican and Project Aimstar.

Economics

The expense of the fissionable materials required was thought high, until the physicist Ted Taylor showed that with the right designs for explosives, the amount of fissionables used on launch was close to constant for every size of Orion from 2,000 tons to 8,000,000 tons. The larger bombs used more explosives to super-compress the fissionables, reducing fallout. The extra debris from the explosives also serves as additional propulsion mass.

Project Daedalus later proposed fusion explosives (deuterium or tritium pellets) detonated by electron beam inertial confinement. This is the same principle behind inertial confinement fusion. However, theoretically, it might be scaled down to far smaller explosions, and require small shock absorbers.

Vehicle architecture



A design for the Orion propulsion module

From 1957 until 1964 this information was used to design a spacecraft propulsion system called "Orion," in which nuclear explosives would be thrown behind a pusher-plate mounted on the bottom of a spacecraft and exploded. The shock wave and radiation from the detonation would impact against the underside of the pusher plate, giving it a powerful "kick." The pusher plate would be mounted on large two-stage shock absorbers that would smoothly transmit acceleration to the rest of the spacecraft.

During take-off, there were concerns of danger from fluidic shrapnel being reflected from the ground. One proposed solution was to use a flat plate of conventional explosives spread over the pusher plate, and detonate this to lift the ship from the ground before going nuclear. This would lift the ship far enough into the air that the first focused nuclear blast would not create debris capable of harming the ship.

A preliminary design for the explosives was produced. It used a shaped-charge fusion-boosted fission explosive. The explosive was wrapped in a beryllium oxide "channel filler," which was surrounded by a uranium radiation mirror. The mirror and channel filler were open ended, and in this open end a flat plate of tungsten propellant was placed. The whole thing was built into a can with a diameter no larger than 6 inches (15 cm) and weighed just over 300 lb (140 kg) so it could be handled by machinery scaled-up from a soft-drink vending machine (indeed, Coca-Cola was consulted on the design).

At 1 microsecond after ignition, the gamma bomb plasma and neutrons would heat the channel filler, and be somewhat contained by the uranium shell. At 2–3 microseconds, the channel filler would transmit some of the energy to the propellant, which vaporized. The flat plate of propellant formed a cigar-shaped explosion aimed at the pusher plate.

The plasma would cool to 14,000 °C, as it traversed the 25 m distance to the pusher plate, and then reheat to 67,000 °C, as (at about 300 microseconds) it hit the pusher plate and recompressed. This temperature emits ultraviolet, which is poorly transmitted through most plasmas. This helps keep the pusher plate cool. The cigar shaped distribution profile and low density of the plasma reduces the instantaneous shock to the pusher plate.

The pusher plate's thickness would decrease by about a factor of 6 from the center to the edge, so that the net velocity of the inner and outer parts of the plate are the same, even though the momentum transferred by the plasma increases from the center outwards.

At low altitudes where the surrounding air is dense, gamma scattering could potentially harm the crew and a radiation refuge would be necessary anyway on long missions to survive solar flares. Radiation shielding effectiveness increases exponentially with shield thickness, so on ships with mass greater than a thousand tons, the structural bulk of the ship, its stores, and the mass of the bombs and propellant would provide more than adequate shielding for the crew.

Stability was initially thought to be a problem due to inaccuracies in the placement of the bombs, but it was later shown that the effects would tend to cancel out.

Numerous model flight tests (using conventional explosives) were conducted at Point Loma in 1959. On November 14, the one-meter model, called "Hot Rod" (or "putt-putt"), first flew using RDX (chemical explosives) in a controlled flight for 23 seconds to a height of 56 meters. Film of the tests has been transcribed to video shown on the BBC TV program "To Mars by A-Bomb" in 2003 with comments by Freeman Dyson and Arthur C. Clarke. The model landed by parachute undamaged and is in the collection of the Smithsonian National Air and Space Museum.

The first proposed shock absorber was merely a ring-shaped airbag. However, it was soon realized that, should an explosion fail, the 500 to 1000 ton pusher plate would tear away the airbag on the rebound. So a two-stage, detuned spring/piston shock absorber design was developed. On the reference design, the first stage mechanical absorber was tuned 4.5 times the pulse frequency whilst the second stage gas piston was tuned to 1/2 times the pulse frequency. This permitted timing tolerances of 10 ms in each explosion.

The final design coped with bomb failure by overshooting and rebounding into a 'center' position. Thus, following a failure (and on initial ground launch) it would be necessary to start (or restart) the sequence with a lower yield device. In the 1950s methods of adjusting bomb yield were in their infancy and considerable thought was given to providing a means of 'swapping out' a standard yield bomb for a smaller yield one in a 2 or 3 second time frame (or to provide an alternative means of firing low yield bombs). These days the yield of a standard device would be 'tuned down', as needed, 'on the fly'.

The bombs had to be launched behind the pusher plate fast enough to explode 20 to 30 m beyond it every 1.1 seconds or so. Numerous proposals were investigated, from multiple guns poking over the edge of the pusher plate to rocket propelled bombs launched from 'roller coaster' tracks, however the final reference design used a simple gas gun to shoot the devices through a hole in the center of the pusher plate.

Potential problems

Exposure to repeated nuclear blasts raises the problem of *ablation* (erosion) of the pusher plate. However, calculations and experiments indicate that a steel pusher plate would ablate less than 1 mm if unprotected. If sprayed with an oil, it need not ablate at all (this was discovered by accident; a test plate had oily fingerprints on it, and the fingerprints suffered no ablation). The absorption spectra of carbon and hydrogen minimize heating. The design temperature of the shockwave, 67,000 °C, emits ultraviolet. Most materials and elements are opaque to ultraviolet, especially at the 340 MPa pressures the plate experiences. This prevents the plate from melting or ablating.

One issue that remained unresolved at the conclusion of the project was whether or not the turbulence created by the combination of the propellant and ablated pusher plate would dramatically increase the total ablation of the pusher plate. According to Freeman Dyson, during the 1960s they would have had to actually perform a test with a real nuclear explosive to determine this; with modern simulation technology, this could be determined fairly accurately without such empirical investigation.

Another potential problem with the pusher plate is that of spalling—shards of metal—potentially flying off the top of the plate. The shockwave from the impacting plasma on the bottom of the plate passes through the plate and reaches the top surface. At that point spalling may occur, damaging the pusher plate. For that reason, alternative substances (e.g., plywood and fiberglass) were investigated for the surface layer of the pusher plate, and thought to be acceptable.

If the conventional explosives in the nuclear bomb detonate, but a nuclear explosion does not ignite (a dud), shrapnel could strike and potentially critically damage the pusher plate.

True engineering tests of the vehicle systems were said to be impossible because several thousand nuclear explosions could not be performed in any one place. However, experiments were designed to test pusher plates in nuclear fireballs. Long-term tests of pusher plates could occur in space. Several of these tests almost flew. The shock-absorber designs could be tested at full-scale on Earth using chemical explosives.

But the main unsolved problem for a launch from the surface of the Earth was thought to be nuclear fallout. Any explosions within the magnetosphere would carry fissionables back to earth unless the spaceship were launched from a polar region such as a barge in the higher regions of the Arctic, with the initial launching explosion to be a large mass of conventional high explosive only to significantly reduce fallout; subsequent detonations would be in the air and therefore much cleaner. Antarctica is not viable, as this would require enormous legal changes as the continent is presently an international wildlife preserve. Freeman Dyson, group leader on the project, estimated back in the 1960s that with conventional nuclear weapons, each launch would cause on average between 0.1 and 1 fatal cancers from the fallout. Danger to human life was not a reason given for shelving the project – those included lack of mission requirement (no-one in the US Government could think of any reason to put thousands of tons of payload into orbit), the decision to focus on rockets (for the Moon mission) and, ultimately, the signature of the Partial Test Ban Treaty in 1963. The danger to electronic systems on the ground (from electromagnetic pulse) is insignificant from the sub-kiloton blasts proposed.

Orion-style nuclear pulse rockets can be launched from above the magnetosphere so that charged ions of fallout in its exhaust plasma are not trapped by the Earth's magnetic field and are not returned to Earth.

The fallout for the entire launch of a 6,000 short ton (5,500 metric ton) Orion is only equal to a ten-megaton (40 petajoule) blast, assuming the use of pure fission weapon-type nuclear explosives.

With special designs of the nuclear explosive, Ted Taylor estimated that it could be reduced tenfold, or even to zero if a pure fusion explosive could be constructed; however, a pure fusion explosive has yet to be successfully developed.

The vehicle and its test program would violate the Partial Test Ban Treaty of 1963 as currently written, which prohibited all nuclear detonations except those conducted underground, both as an attempt to slow the arms race and to limit the amount of radiation in the atmosphere caused by nuclear detonations. There was an effort by the US government to put an exception into the 1963 treaty to allow for the use of nuclear propulsion for spaceflight, but Soviet fears about military applications kept the exception out of the treaty.

One way around the restrictions of the treaty would be to use a form of the Project Daedalus fusion microexplosion rocket. Daedalus class systems use pellets of one gram or less ignited by particle or laser beams to produce very small fusion explosions with a maximum explosive yield of only 10–20 tons of TNT equivalent.

The launch of such an Orion nuclear bomb rocket from the ground or from low Earth orbit would generate an electromagnetic pulse that could cause significant damage to computers and satellites, as well as flooding the van Allen belts with high-energy radiation. This problem might be solved by launching from very remote areas, because the EMP footprint would be only a few hundred miles wide. The Earth is well shielded by the Van Allen belts. In addition, a few relatively small space-based electrodynamic tethers could be deployed to quickly eject the energetic particles from the capture angles of the Van Allen belts.

Assembling a pulse drive spacecraft in orbit by more conventional means and only activating its main drive at a safer distance would be a less destructive approach. The Lofstrom launch loop or a space elevator hypothetically provides an excellent solution, although in the case of the space elevator existing carbon nanotubes composites do not yet have sufficient tensile strength. All chemical rocket designs are extremely inefficient (and expensive) when launching mass into orbit, however could be employed if the result was seen as worth the cost (for example, the alternative being the impact of an asteroid of size similar to that of the Cretaceous-Tertiary extinction event).

Adverse public reaction to any use of nuclear explosives of any type is likely to remain a stumbling block even if practical and legal difficulties are overcome.

Operation Plumbbob

A test similar to the test of a pusher plate occurred as an accidental side effect of a nuclear containment test called "Pascal B" conducted on 27 August 1957. The test's experimental designer Dr. Brownlee performed a highly approximate calculation that suggested that the low-yield nuclear explosive would accelerate the massive (900 kg) steel capping plate to six times escape velocity. The plate was never found, and Dr. Brownlee believes that the plate never left the atmosphere (for example it could have been vaporized by compression heating of the atmosphere due to its high speed). The calculated velocity was sufficiently interesting that the crew trained a high-speed camera on the plate, which unfortunately only appeared in one frame, but this nevertheless gave a very high lower bound for the speed.

Chapter 3

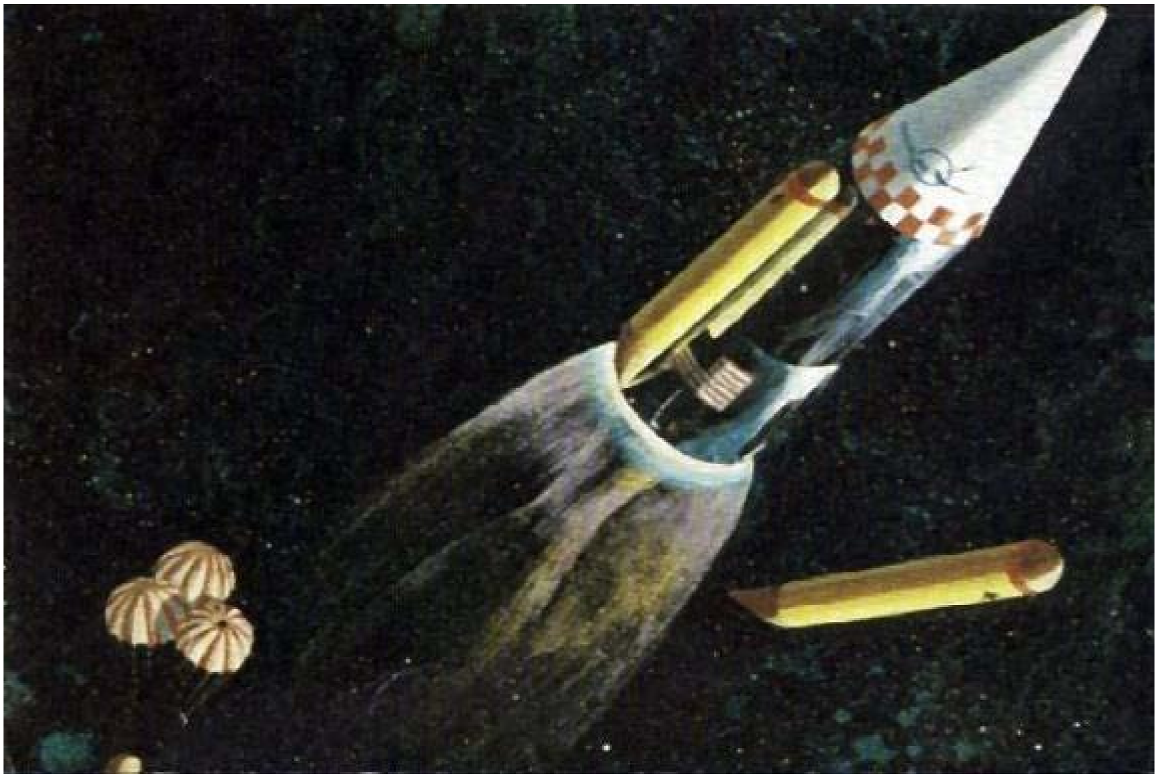
Reusable Launch System

A **reusable launch system** (or **reusable launch vehicle**, RLV) is a launch system which is capable of launching a launch vehicle into space more than once. This contrasts with expendable launch systems, where each launch vehicle is launched once and then discarded.

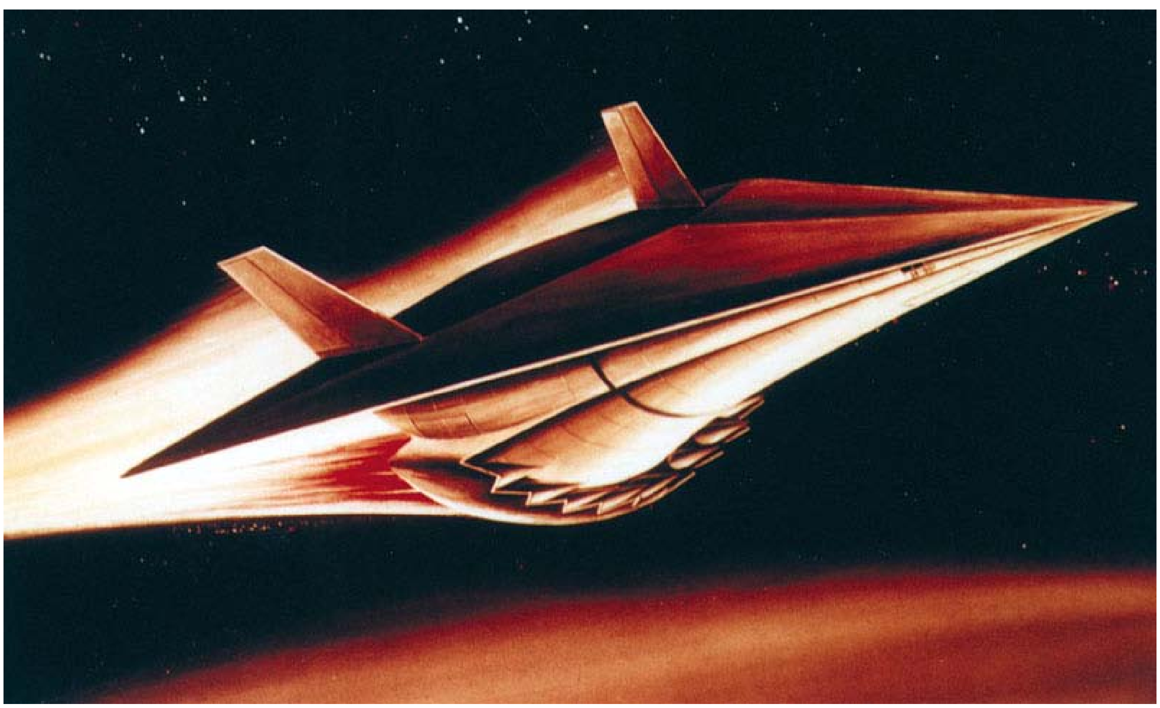
No true orbital reusable launch system is currently in use. The closest example is the partially reusable Space Shuttle. The orbiter, which includes the main engines, and the two solid rocket boosters, are reused after several months of refitting work for each launch. The external fuel drop tank is typically discarded, but it is possible for it be re-used in space for various applications.

Orbital RLVs are thought to provide the possibility of low cost and highly reliable access to space. However, reusability implies weight penalties such as non-ablative reentry shielding and possibly a stronger structure to survive multiple uses, and given the lack of experience with these vehicles, the actual costs and reliability are yet to be seen.

History



ROMBUS



Aerospaceplane 1

As usual, science fiction preceded science fact in this area. In the early 1950s popular science fiction often depicted space launch vehicles as either single-stage reusable rocketships which could launch and land vertically (SSTO VTVL), or single-stage reusable rocketplanes which could launch and land horizontally (SSTO HTHL).

The realities of early engine technology with low specific impulse or insufficient thrust-to-weight ratio to escape our gravity well, compounded by construction materials without adequate performance (strength, stiffness, heat resistance) and low weight seemingly rendered that original single-stage reusable vehicle vision impossible.

However advances in materials and engine technology have rendered this concept potentially feasible.

Before VTVL SSTO designs came the partially reusable multi-stage NEXUS launcher by Krafft Ehricke. The pioneer in the field of VTVL SSTO, Philip Bono, worked at Douglas. Bono proposed several launch vehicles including: ROOST, ROMBUS, Ithacus, Pegasus and SASSTO. Most of his vehicles combined similar innovations to achieve SSTO capability. Bono proposed:

- Plug nozzle engines to retain high specific impulse at all altitudes.
- Base first reentry which allowed the reuse of the engine as a heat shield, lowering required heat shield mass.
- Use of spherical tanks and stubby shape to reduce vehicle structural mass further.
- Use of drop tanks to increase range.
- Use of in-orbit refueling to increase range.

Bono also proposed the use of his vehicles for space launch, rapid intercontinental military transport (Ithacus), rapid intercontinental civilian transport (Pegasus), even Moon and Mars missions (Project Selena, Project Deimos).

In Europe, Dietrich Koelle, inspired by Bono's SASSTO design, proposed his own VTVL vehicle named BETA.

Before HTHL SSTO designs came Eugen Sänger and his Silbervogel ("Silverbird") suborbital skip bomber. HTHL vehicles which can reach orbital velocity are harder to design than VTVL due to their higher vehicle structural weight. This led to several multi-stage prototypes such as a suborbital X-15. Aerospaceplane being one of the first HTHL SSTO concepts. Proposals have been made to make such a vehicle more viable including:

- Rail boost.
- Use of lifting body designs to reduce vehicle structural mass.
- Use of in-flight refueling.

Other launch system configuration designs are possible such as horizontal launch with vertical landing (HTVL) and vertical launch with horizontal landing (VTHL). One of the few HTVL vehicles is the 1960s concept spacecraft Hyperion SSTO, designed by by

Philip Bono. X-20 Dyna-Soar is an early example of a VTHL design, while the HL-20 and X-34 are examples from the 1990s. As of February 2010, the VTHL X-37 has completed initial development and flown an initial classified orbital mission of over seven months duration. Currently proposed VTHL manned spaceplanes include the Dream Chaser and Prometheus, both circa 2010 concept spaceplanes proposed to NASA under the CCDev program.

The late 1960s saw the start of the Space Shuttle design process. From an initial multitude of ideas a two-stage reusable VTHL design was pushed forward. That eventually ended up as a reusable orbiter with an expendable drop tank and reusable solid rocket boosters to reduce design expenses.

During the 1970s further VTVL and HTHL SSTO designs were proposed for solar power satellite and military applications. There was a VTVL SSTO study by Boeing. HTHL SSTO designs included the Rockwell Star-Raker and the Boeing HTHL SSTO study. However the focus of all space launch funding in the United States on the Shuttle killed off these prospects. The Soviet Union followed suit with Buran. Others preferred expendables for their lower design risk, and lower design cost.

Eventually the Shuttle was found to be expensive to maintain, even more expensive than an expendable launch system would have been. The cancellation of a Shuttle-Centaur rocket after the loss of Challenger also caused an hiatus that would make it necessary for the United States military to scramble back towards expendables to launch their payloads. Many commercial satellite customers had switched to expendables even before that, due to unresponsiveness to customer concerns by the Shuttle launch system.

In 1986 President Ronald Reagan called for an airbreathing scramjet plane to be built by the year 2000, called NASP/X-30 that would be capable of SSTO. Based on the research project **copper canyon** the project failed due to severe technical issues and was cancelled in 1993.

This research may have inspired the British HOTOL program, which rather than airbreathing to high hypersonic speeds as with NASP, proposed to use a precooler up to Mach 5.5. The program's funding was canceled by the British government when the research identified some technical risks as well as indicating that that particular vehicle architecture would only be able to deliver a relatively small payload size to orbit.

When the Soviet Union imploded in the early nineties, the cost of Buran became untenable. Russia has only used pure expendables for space launch since.

The 1990s saw interest in developing new reusable vehicles. The military Strategic Defense Initiative ("Star Wars") program "Brilliant Pebbles" required low cost, rapid turnaround space launch. From this requirement came the McDonnell Douglas Delta Clipper VTVL SSTO proposal. The DC-X prototype for Delta Clipper demonstrated rapid turnaround time and that automatic computer control of such a vehicle was possible. It also demonstrated it was possible to make a reusable space launch vehicle which did not require a large standing army to maintain like the Shuttle.

In mid-1990, further British research and major reengineering to avoid deficiencies of the HOTOL design led to the far more promising Skylon design, with much greater payload.

From the commercial side, large satellite constellations such as Iridium satellite constellation were proposed which also had low cost space access demands. This fueled a private launch industry, including partially reusable vehicle players, such as Kistler, and reusable vehicle players such as Rotary Rocket.

The end of that decade saw the implosion of the satellite constellation market with the bankruptcy of Iridium. In turn the nascent private launch industry collapsed. The fall of the Soviet Union eventually had political ripples which led to a scaling down of ballistic missile defense, including the demise of the "Brilliant Pebbles" program. The military decided to replace their aging expendable launcher workhorses, evolved from ballistic missile technology, with the EELV program. NASA proposed riskier reusable concepts to replace Shuttle, to be demonstrated under the X-33 and X-34 programs.

The 21st century saw rising costs and teething problems lead to the cancellation of both X-33 and X-34. Then the Space Shuttle Columbia disaster and another grounding of the fleet. The Shuttle design was now over 20 years old and in need of replacement. Meanwhile the military EELV program churned out a new generation of better expendables. The commercial satellite market is depressed due to a glut of cheap expendable rockets and there is a dearth of satellite payloads.

Against this dire backdrop came the Ansari X Prize contest, inspired by the aviation contests made in the early 20th century. Many private companies competed for the Ansari X Prize, the winner being Scaled Composites with their reusable HTHL SpaceShipOne. It won the ten million dollars, by reaching 100 kilometers in altitude twice in a two week period with the equivalent of three people on board, with no more than ten percent of the non-fuel weight of the spacecraft replaced between flights. While SpaceShipOne is suborbital like the X-15, some hope the private sector can eventually develop reusable orbital vehicles given enough incentive. SpaceX is a recent player in the private launch market which has partially reusable vehicles.

Reusability concepts

Single stage

There are two approaches to Single stage to orbit or SSTO. The rocket equation says that an SSTO vehicle needs a high mass ratio. Mass ratio is defined as the mass of the fully fueled vehicle divided by the mass of the vehicle when empty (zero fuel weight, ZFW).

One way to increase the mass ratio is to reduce the mass of the empty vehicle by using very lightweight structures and high efficiency engines. This tends to push up maintenance costs as component reliability can be impaired, and makes reuse more expensive to achieve. The margins are so small with this approach that there is uncertainty whether such a vehicle would be able to carry any payload into orbit. Also, lightweight implies small vehicles, which in turn implies small payloads, increasing the cost per kilogram of the payload.

Two or more stages to orbit

Two stage to orbit requires designing and building two independent vehicles and dealing with the interactions between them at launch. Usually the second stage in launch vehicle is 5-10 times smaller than the first stage, although in **biamese** and **triamese** approaches each vehicle is the same size.

In addition, the first stage needs to be returned to the launch site for it to be reused. This is usually proposed to be done by flying a compromise trajectory that keeps the first stage above or close to the launch site at all times, or by using small airbreathing engines to fly the vehicle back, or by recovering the first stage downrange and returning it some other way (often landing in the sea, and returning it by ship.) Most techniques involve some performance penalty; these can require the first stage to be several times larger for the same payload, although for recovery from downrange these penalties may be small.

The second stage is normally returned after flying one or more orbits and reentering.

Horizontal landing

Scaled Composites SpaceShipOne used horizontal landing after being launched from a carrier airplane

In this case the vehicle requires wings and undercarriage (unless landing at sea). This typically requires about 9-12% of the landing vehicle to be wings; which in turn implies that the takeoff weight is higher and/or the payload smaller.

Concepts such as lifting bodies attempt to deal with the somewhat conflicting issues of reentry, hypersonic and subsonic flight; as does the delta wing shape of the Space Shuttle.

Vertical landing

McDonnell Douglas DC-X used vertical takeoff and vertical landing

Parachutes could be used to land vertically, either at sea, or with the use of small landing rockets, on land (as with Soyuz).

Alternatively rockets could be used to softland the vehicle on the ground from the subsonic speeds reached at low altitude. This typically requires about 10% of the landing weight of the vehicle to be propellant.

A slightly different approach to vertical landing is to use an autogyro or helicopter rotor. This requires perhaps 2-3% of the landing weight for the rotor.

Horizontal takeoff

XCOR Aerospace EZ-Rocket used horizontal takeoff and landing using a standard airport runway

The vehicle needs wings to take off. For reaching orbit, a 'wet wing' would often need to be used where the wing contains propellant. Around 9-12% of the vehicle takeoff weight is perhaps tied up in the wings.

Vertical takeoff

This is the traditional takeoff regime for pure rocket vehicles. Rockets are good for this regime, since they have a very high thrust/weight ratio (~100).

Airbreathing

Airbreathing approaches use the air for propulsion during ascent. The most commonly proposed approach is the scramjet, but turborocket, Liquid Air Cycle Engine (LACE) and precooled jet engines are also proposed to be used.

In all cases the highest speed that airbreathing can reach is far short of orbital speed (about Mach 15 for Scramjets and Mach 5-6 for the other engine designs) and rockets would be used for the remaining 10-20 Mach for orbit.

The thermal situation for airbreathers (particularly scramjets) can be awkward; normal rockets fly steep initial trajectories to avoid drag, whereas scramjets would deliberately fly through relatively thick atmosphere at high speed generating enormous heating of the airframe. The thermal situation for the other airbreathing approaches is much more benign, although is not without its challenges.

Propellant

Hydrogen fuel

Hydrogen is often proposed since it has the highest exhaust velocity. However tankage and pump weights are high due to insulation and low propellant density; and this wipes out much of the advantage.

Still, the 'wet mass' of a hydrogen fuelled stage is lighter than an equivalent dense stage with the same payload and this can permit usage of wings, and is good for second stages.

Dense fuel

Dense fuel is sometimes proposed since, although it implies a heavier vehicle, the specific tankage and pump mass is much improved over hydrogen. Dense fuel is usually suggested for vertical takeoff vehicles, and is compatible with horizontal landing vehicles, since the vehicle is lighter than an equivalent hydrogen vehicle when empty of propellant. Non-cryogenic dense fuels also permit the storage of fuel in wing structures. Projects have been underway to densify existing fuel types through various techniques. These include slush technologies for cryogenics like hydrogen and propane. Another densifying method has been studied that would also increase the specific impulse of fuels. Adding finely powdered carbon, aluminum, titanium, and boron to hydrogen and kerosene have been studied. These additives increase the specific impulse (Isp) but also the density of the fuel. For instance, the French ONERA missile program tested boron with kerosene in gelled slurries, as well as embedded in paraffin, and demonstrated increases in volumetric specific impulse of between 20-100%.

Tripropellant

Dense fuel is optimal early on in a flight, since the thrust to weight of the engines is better due to higher density; this means the vehicle accelerates more quickly and reaches orbit sooner, reducing gravity losses.

However, for reaching orbital speed, hydrogen is a better fuel, since the high exhaust velocity and hence lower propellant mass reduces the take off weight.

Therefore tripropellant vehicles start off burning with dense fuel and transition to hydrogen. (In a sense the Space Shuttle does this with its combination of solid rockets and main engines, but tripropellant vehicles usually carry their engines to orbit.)

Propellant costs

As with all current launch vehicles propellant costs for a rocket are much lower than the costs of the hardware. However, for reusable vehicles if the vehicles are successful, then the hardware is reused many times and this would bring the costs of the hardware down. In addition, reusable vehicles are frequently heavier and hence less propellant efficient, so the propellant costs could start to multiply up to the point where they become significant.

Launch assistance

Since rocket delta-v has a non linear relationship to mass fraction due to the rocket equation, any small reduction in delta-v gives a relatively large reduction in the required mass fraction; and starting a mission at higher altitude also helps.

Many systems have proposed the use of aircraft to gain some initial velocity and altitude; either by towing, carrying or even simply refueling a vehicle at altitude.

Various other launch assists have been proposed, such as ground based sleds, or maglev systems, high altitude (80 km) maglev systems such as launch loops, to more exotic systems such as tether propulsion systems to catch the vehicle at high altitude, or even Space Elevators.

Reentry heat shields

Robert Zubrin has said that as a rough rule of thumb, 15% of the landed weight of a vehicle needs to be aerobraking reentry shielding.

Reentry heat shields on these vehicles are often proposed to be some sort of ceramic and/or carbon-carbon heat shields, or occasionally metallic heat shields (possibly using water cooling or some sort of relatively exotic rare earth metal.)

Some shields would be single use ablatives and would be discarded after reentry.

A newer Thermal Protection System (TPS) technology was first developed for use in steering fins on ICBM MIRVs. Given the need for such warheads to reenter the atmosphere swiftly and retain hypersonic velocities to sea level, researchers developed what are known as SHARP materials, typically hafnium diboride and zirconium diboride, whose thermal tolerance exceeds 3600 C. SHARP equipped vehicles can fly at Mach 11 at 30 km altitude and Mach 7 at sea level. The sharp-edged geometries permitted with these materials also eliminates plasma shock wave interference in radio communications during reentry. SHARP materials are very robust and would not require constant maintenance, as is the case with technologies like silica tiles, used on the Space Shuttle, which account for over half of that vehicles maintenance costs and turnaround time. The maintenance savings alone are thus a major factor in favor of using these materials for a reusable launch vehicle, whose raison d'etre is high flight rates for economical launch costs.

Weight penalty

The weight of a reusable vehicle is almost invariably higher than an expendable that was made with the same materials, for a given payload.

R&D

The R&D costs of reusable vehicle are expected to be higher, because making a vehicle reusable implies making it robust enough to survive more than one use, which adds to the testing required. Increasing robustness is most easily done by adding weight; but this reduces performance and puts further pressure on the R&D to recoup this in some other way.

These extra costs must be recouped; and this pushes up the average cost of the vehicle.

Maintenance

Reusable launch systems require maintenance, which is often substantial. The Space Shuttle system requires extensive refurbishing between flights, primarily dealing with the silica tile TPS and the high performance LH2/LOX burning main engines. Both systems require a significant amount of detailed inspection, rebuilding and parts replacement between flights, and account for over 75% of the maintenance costs of the Shuttle system. These costs, far in excess of what had been anticipated when the system was constructed, have cut the maximum flight rate of Shuttle to 1/4 of that planned. This has also quadrupled the cost per pound of payload to orbit, making Shuttle economically infeasible in today's launch market for any but the largest payloads, for which there is no competition.

For any RLV technology to be successful, it must learn from the failings of Shuttle and overcome those failings with new technologies in the TPS and propulsion areas.

Manpower & Logistics

The Space Shuttle program requires a standing army of over 9,000 employees to maintain, refurbish, and relaunch the shuttle fleet, irrespective of flight rates. That manpower budget must be divided by the total number of flights per year. The fewer flights means the cost per flight goes up significantly. Streamlining the manpower requirements of any launch system is an essential part of making an RLV economical. Projects that have attempted to develop this ethic include the DC-X Delta Clipper project, as well as the current SpaceX Falcon 1 and Falcon 9 programs.

One issue mitigating against this drive for labor savings is government regulation. Given that NASA and USAF (as well as government programs in other countries) are the primary customers and sources of development capital, government regulatory requirements for oversight, parwork, quality, safety, and other documentation tend to inflate the operational costs of any such system.

Orbital reusable launchers

Currently in use

- Space Shuttle (partially reusable)

Planned

- PlanetSpace Silver Dart (partly reusable spaceplane, based on hypersonic glider design)
- SpaceX Falcon 1 (announced as partially reusable; 28 September 2008 test flight reached orbit, but vehicle recovery not yet demonstrated)
- SpaceX Falcon 9 (announced as partially reusable; maiden flight achieved orbit, 4 June 2010)
- Reaction Engines Skylon (proposed airbreathing SSTO spaceplane)
- Avatar RLV (proposed reusable Indian launch system for small payloads)

Historical

- Soviet Union Energia-Buran system (partially reusable, now cancelled)

Cancelled

- Hopper (proposed reusable European launch system)
- HOTOL British SSTO
- Hyperion SSTO — 1960s concept HTVL spacecraft
- Kliper Russian-European partially reusable spacecraft that was to be launched around 2011 for the first time. It has been reported, however, that Energia is still working on the craft, and the Russian space program plans to review the decision to cancel the Kliper once they produce their now-planned modernized version of the Soyuz spacecraft. Thus, the Russian space program may decide to use it after all, and even so Energia may be able to market it to other space programs if they finish it.
- Phoenix SSTO
- Roton Commercial launch vehicle project, cancelled in 2000 due to lack of funds.

Suborbital reusable launchers

Planned

- ARCASPACE: Orizont
- Armadillo Aerospace: Black Armadillo

- Blue Origin: New Shepard
- Canadian Arrow
- The da Vinci Project: Wild Fire MK VI
- Masten Space Systems: XA 1.0
- Pablo de Leon & Associates: Gauchito
- Rocketplane Limited, Inc.: Rocketplane XP
- Space Adventures: Explorer
- Starchaser Industries: Starchaser V - Thunderstar
- TGV Rockets
- The Spaceship Company: SpaceShipTwo
- Reusable Vehicle Testing project of the Japanese Space Exploration Agency (JAXA)

Historical

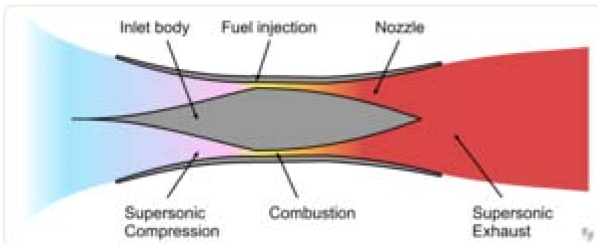
- North American X-15
- Scaled Composites: SpaceShipOne

Regulations

In 2006, the US Federal Aviation Administration issued a new regulation regarding commercial reusable launch vehicles, both suborbital and orbital, as Part 431. The text can be found under the US Federal Code at 14 CFR Part 431. The new regulation was made in anticipation of planned commercial reusable launch operations including the American companies listed above. FAA regulations only have jurisdiction within the United States and its territories, and to aircraft and spacecraft registered in the United States.

Chapter 4

Scramjet



A **scramjet** (*supersonic* ***c****ombustion* ***ramjet***) is a variant of a ramjet airbreathing combustion jet engine in which the combustion process takes place in supersonic airflow. As in ramjets, a scramjet relies on high vehicle speed to forcefully compress and decelerate the incoming air before combustion (hence *ram*jet), but whereas a ramjet decelerates the air to subsonic velocities before combustion, airflow in a scramjet is supersonic throughout the entire engine. This allows the scramjet to efficiently operate at extremely high speeds: theoretical projections place the top speed of a scramjet between Mach 12 and Mach 24, which is near orbital velocity. The fastest air-breathing plane is a SCRAM jet design, the NASA X-43a which reached Mach 9.8. For comparison, the second fastest manned airbreathing aircraft, the SR-71 Blackbird, has a cruising speed of Mach 3.2.

The scramjet is composed of three basic components: a converging inlet, where incoming air is compressed and decelerated; a combustor, where gaseous fuel is burned with atmospheric oxygen to produce heat; and a diverging nozzle, where the heated air is accelerated to produce thrust. Unlike a typical jet engine, such as a turbojet or turbofan engine, a scramjet does not use rotating, fan-like components to compress the air; rather, the achievable speed of the aircraft moving through the atmosphere causes the air to compress within the nozzle. As such, no moving parts are needed in a scramjet, which greatly simplifies both the design and operation of the engine. In comparison, typical turbojet engines require inlet fans, multiple stages of rotating compressor fans, and multiple rotating turbine stages, all of which add weight, complexity, and a greater number of failure points to the engine. It is this simplicity that allows scramjets to operate at such high velocities, as the conditions encountered in hypersonic flight severely hamper the operation of conventional turbomachinery.

Due to the nature of their design, scramjet operation is limited to near-hypersonic velocities. As they lack mechanical compressors, scramjets require the high kinetic energy of a hypersonic flow to compress the incoming air to operational conditions. Thus, a scramjet-powered vehicle must be accelerated to the required velocity by some other means of propulsion, such as turbojet, railgun, or rocket engines. In the flight of the experimental scramjet-powered Boeing X-51A, the test craft was lifted to flight altitude by a turbofan powered B-52 before being released and accelerated by a detachable rocket to near Mach 4.5.

While scramjets are conceptually simple, actual implementation is limited by extreme technical challenges. Hypersonic flight within the atmosphere generates immense drag, and temperatures found on the aircraft and within the engine can be nearly six-times greater than that of the surrounding air. Maintaining combustion in the supersonic flow presents additional challenges, as the fuel must be injected, mixed, ignited, and burned within milliseconds. While scramjet technology has been under development since the 1950s, only very recently have scramjets successfully achieved powered flight.

History

During and after World War II, tremendous amounts of time and effort were put into researching high-speed jet- and rocket-powered aircraft. The Bell X-1 attained supersonic flight in 1947 and, by the early 1960s, rapid progress towards faster aircraft suggested that operational aircraft would be flying at "hypersonic" speeds within a few years. Except for specialized rocket research vehicles like the North American X-15 and other rocket-powered spacecraft, aircraft top speeds have remained level, generally in the range of Mach 1 to Mach 3.

In the 1950s and 1960s a variety of experimental scramjets engines were built and ground tested in US and the UK. In 1964, Dr. Frederick S. Billig and Dr. Gordon L. Dugger submitted a patent application for a supersonic combustion ramjet based on Billig's Ph.D. thesis. This patent was issued in 1981 following the removal of an order of secrecy.

In 1981 tests were made in Australia under the guidance of Professor Ray Stalker in the T3 ground test facility at ANU.

First successfull flight test of Scramjet was performed by Russia in 1991. It was axisymmetric hydrogen-fueled dual-mode scramjet developed by Central Institute of Aviation Motors (CIAM), Moscow from late 1970th. The scramjet flight was flown captive-carry atop the SA-5 surface-to-air missile that included an experiment flight support unit known as the Hypersonic Flying Laboratory (HFL), "Kholod" . Then during the period since 1992 up to 1998 additional 6 flight tests of the axisymmetric high-speed scramjet-demonstrator have been conducted by CIAM together with France and then with NASA, USA . Maximum flight velocity greater than Mach 6.4 was achived and Scramjet operation during 77 seconds was demonstrated. These flight test series also provided insight into autonomous hypersonic flight controls.

The main goal of civilian air transport has been reducing operating cost, rather than increasing flight speeds. Because supersonic flight using conventional jet engines requires significant amounts of fuel, airlines have favored subsonic jumbo jets rather than supersonic transports. The production supersonic airliners, Concorde and the Tupolev Tu-144, operated with little profit for the French and Russian airlines, but British Airways flew Concorde at a 60% profit margin over its commercial life (though this does not include government-subsidized initial costs). Military combat aircraft design has focused on maneuverability, more recently combined with stealth.

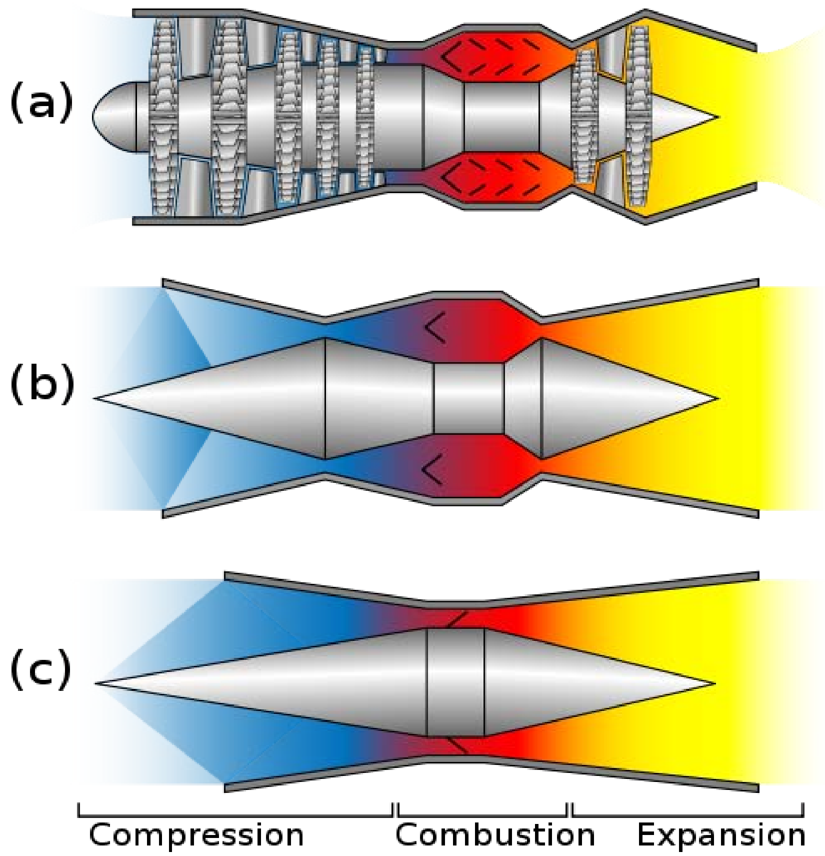
Hypersonic flight concepts haven't gone away, however, and low-level investigations have continued over the past few decades. Presently, the US military and NASA have formulated a "National Hypersonics Strategy" to investigate a range of options for hypersonic flight.

Different U.S. organizations have accepted hypersonic flight as a common goal. The U.S. Army desires hypersonic missiles that can attack mobile missile launchers quickly. NASA believes hypersonics could help develop economical, reusable launch vehicles. The Air Force is interested in a range of hypersonic systems, from air-launched cruise missiles to orbital spaceplanes, that the service believes could bring about a true "aerospace force."

The problem is complicated by the release of previously classified material and by partial publication, where claims are made, but specific parts of an experiment are kept secret. Additionally experimental difficulties in verifying that supersonic combustion actually occurred, or that actual net thrust was produced mean that at least four consortia have legitimate claims to "firsts", with several nations and institutions involved in each consortium. On June 15, 2007, the US Defense Advanced Research Project Agency (DARPA) and the Australian Defence Science and Technology Organization (DSTO), announced a successful scramjet flight at Mach 10 using rocket engines to boost the test vehicle to hypersonic speeds, at the Woomera Rocket Range in Central Australia.

No scramjet powered vehicle has yet been produced outside an experimental program.

Design principles



The compression, combustion, and expansion regions of: (a) turbojet, (b) ramjet, and (c) scramjet engines.

Scramjet engines are a type of jet engine, and rely on the combustion of fuel and an oxidizer to produce thrust. Similar to conventional jet engines, scramjet-powered aircraft carry the fuel on board, and obtain the oxidizer by the ingestion of atmospheric oxygen (as compared to rockets, which carry both fuel and an oxidizing agent). This requirement limits scramjets to suborbital atmospheric flight, where the oxygen content of the air is sufficient to maintain combustion.

Basic principles

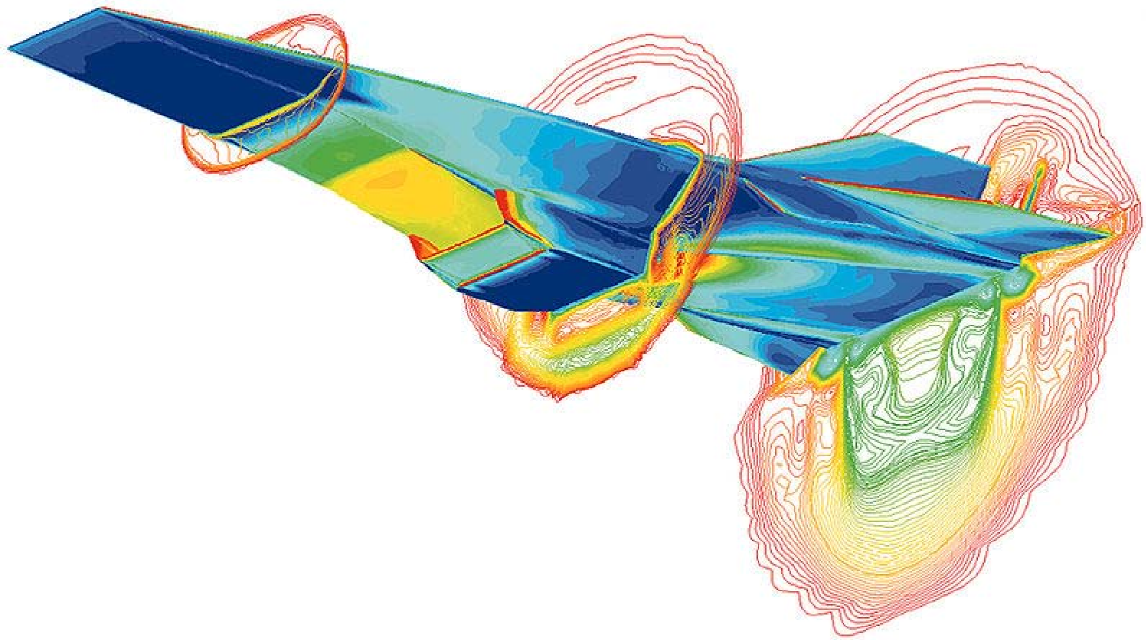
Scramjets are designed to operate in the hypersonic flight regime, beyond the reach of turbojet engines, and, along with ramjets, fill the gap between the high efficiency of turbojets and the high speed of rocket engines. Turbomachinery-based engines, while highly efficient at subsonic speeds, become increasingly inefficient at transonic speeds, as the compressor fans found in turbojet engines require subsonic speeds to operate. While the flow from transonic to low supersonic speeds can be decelerated to these conditions, doing so at supersonic speeds results in a tremendous increase in temperature and a loss in the total enthalpy of the flow. Around Mach 3–4, turbomachinery is no longer useful, and ram-style compression becomes the preferred method.

Ramjets utilize high-speed characteristics of air to literally 'ram' air through an inlet nozzle into the combustor. At transonic and supersonic flight speeds, the air upstream of the nozzle is not able to move out of the way quickly enough, and is compressed within the nozzle before being diffused into the combustor. Combustion in a ramjet takes place at subsonic velocities, similar to turbojets, but the combustion products are then accelerated through a convergent-divergent nozzle to supersonic speeds. As they have no mechanical means of compression, ramjets cannot start from a standstill, and generally do not achieve sufficient compression until supersonic flight. The lack of intricate turbomachinery allows ramjets to deal with the temperature rise associated with decelerating a supersonic flow to subsonic speeds, but this only goes so far: at near-hypersonic velocities, the temperature rise and inefficiencies discourage decelerating the flow to the magnitude found in ramjet engines.

Scramjet engines operate on the same principles as ramjets do, but do not decelerate the flow to subsonic velocities. Rather, a scramjet combustor is supersonic: the inlet decelerates the flow to a lower Mach number for combustion, after which it is accelerated to an even higher Mach number through the nozzle. By limiting the amount of deceleration, temperatures within the engine are kept at a tolerable level, from both a material and combustive standpoint. Even so, current scramjet technology requires the use of high-energy fuels and active cooling schemes to maintain sustained operation, often using hydrogen and regenerative cooling techniques.

Theory

All scramjet engines have fuel injectors, a combustion chamber, a thrust nozzle and an intake, which compresses the incoming air. Sometimes engines also include a region which acts as a flame holder, although the high stagnation temperatures mean that an area of focused waves may be used, rather than a discrete engine part as seen in turbine engines. Other engines use pyrophoric fuel additives, such as silane, to avoid such issues. An isolator between the inlet and combustion chamber is often included to improve the homogeneity of the flow in the combustor and to extend the operating range of the engine.



Computational fluid dynamics (CFD) image of the NASA X-43A with scramjet attached to the underside at Mach 7

A scramjet is reminiscent of a ramjet. In a typical ramjet, the supersonic inflow of the engine is decelerated at the inlet to subsonic speeds and then reaccelerated through a nozzle to supersonic speeds to produce thrust. This deceleration, which is produced by a normal shock, creates a total pressure loss which limits the upper operating point of a ramjet engine.

For a scramjet, the kinetic energy of the freestream air entering the scramjet engine is large comparable to the energy released by the reaction of the oxygen content of the air with a fuel (say hydrogen). Thus the heat released from combustion at Mach 25 is around 10% of the total enthalpy of the working fluid. Depending on the fuel, the kinetic energy of the air and the potential combustion heat release will be equal at around Mach 8. Thus the design of a scramjet engine is as much about minimizing drag as maximizing thrust.

This high speed makes the control of the flow within the combustion chamber more difficult. Since the flow is supersonic, no upstream influence propagates within the freestream of the combustion chamber. Thus throttling of the entrance to the thrust nozzle is not a usable control technique. In effect, a block of gas entering the combustion chamber must mix with fuel and have sufficient time for initiation and reaction, all the while traveling supersonically through the combustion chamber, before the burned gas is expanded through the thrust nozzle. This places stringent requirements on the pressure and temperature of the flow, and requires that the fuel injection and mixing be extremely efficient. Usable dynamic pressures lie in the range 20 to 200 kPa (0.2-2 bar), where

$$q = \frac{1}{2}\rho v^2$$

where

q is the dynamic pressure of the gas
ρ (rho) is the density of the gas
v is the velocity of the gas

To keep the combustion rate of the fuel constant, the pressure and temperature in the engine must also be constant. This is problematic because the airflow control systems that would facilitate this are not physically possible in a scramjet launch vehicle due to the large speed and altitude range involved, meaning that it must travel at an altitude specific to its speed. Because air density reduces at higher altitudes, a scramjet must climb at a specific rate as it accelerates to maintain a constant air pressure at the intake. This optimal climb/descent profile is called a "constant dynamic pressure path". It is thought that scramjets might be operable up to an altitude of 75 km.

Fuel injection and management is also potentially complex. One possibility would be that the fuel be pressurized to 100 bar by a turbo pump, heated by the fuselage, sent through the turbine and accelerated to higher speeds than the air by a nozzle. The air and fuel stream are crossed in a comb like structure, which generates a large interface. Turbulence due to the higher speed of the fuel leads to additional mixing. Complex fuels like kerosene need a long engine to complete combustion.

The minimum Mach number at which a scramjet can operate is limited by the fact that the compressed flow must be hot enough to burn the fuel, and have pressure high enough that the reaction be finished before the air moves out the back of the engine. Additionally, in order to be called a scramjet, the compressed flow must still be supersonic after combustion. Here two limits must be observed: Firstly, since when a supersonic flow is compressed it slows down, the level of compression must be low enough (or the initial speed high enough) not to slow the gas below Mach 1. If the gas within a scramjet goes below Mach 1 the engine will "choke", transitioning to subsonic flow in the combustion chamber. This effect is well known amongst experimenters on scramjets since the waves caused by choking are easily observable. Additionally, the sudden increase in pressure and temperature in the engine can lead to an acceleration of the combustion, leading to the combustion chamber exploding.

Secondly, the heating of the gas by combustion causes the speed of sound in the gas to increase (and the Mach number to decrease) even though the gas is still travelling at the same speed. Forcing the speed of air flow in the combustion chamber under Mach 1 in this way is called "thermal choking". It is clear that a pure scramjet can operate at Mach numbers of 6-8, but in the lower limit, it depends on the definition of a scramjet. Certainly there are designs where a ramjet transforms into a scramjet over the Mach 3-6 range (Dual-mode scramjets). In this range however, the engine is still receiving significant thrust from subsonic combustion of "ramjet" type.

The high cost of flight testing and the unavailability of ground facilities have hindered scramjet development. A large amount of the experimental work on scramjets has been undertaken in cryogenic facilities, direct-connect tests, or burners, each of which simulates one aspect of the engine operation. Further, vitiated facilities, storage heated facilities, arc facilities and the various types of shock tunnels each have limitations which have prevented perfect simulation of scramjet operation. The HyShot flight test showed the relevance of the 1:1 simulation of conditions in the T4 and HEG shock tunnels, despite having cold models and a short test time. The NASA-CIAM tests provided similar verification for CIAM's C-16 V/K facility and the Hyper-X project is expected to provide similar verification for the Langley AHSTF, CHSTF and 8 ft (2.4 m) HTT.

Computational fluid dynamics has only recently reached a position to make reasonable computations in solving scramjet operation problems. Boundary layer modeling, turbulent mixing, two-phase flow, flow separation, and real-gas aerothermodynamics continue to be problems on the cutting edge of CFD. Additionally, the modeling of kinetic-limited combustion with very fast-reacting species such as hydrogen makes severe demands on computing resources. Reaction schemes are numerically stiff requiring reduced reaction schemes.

Much of scramjet experimentation remains classified. Several groups including the US Navy with the SCRAM engine between 1968–1974 and the Hyper-X program with the X-43A have claimed successful demonstrations of scramjet technology. Since these results have not been published openly, they remain unverified and a final design method of scramjet engines still does not exist.

The final application of a scramjet engine is likely to be in conjunction with engines which can operate outside the scramjet's operating range. Dual-mode scramjets combine subsonic combustion with supersonic combustion for operation at lower speeds, and rocket-based combined cycle (RBCC) engines supplement a traditional rocket's propulsion with a scramjet, allowing for additional oxidizer to be added to the scramjet flow. RBCCs offer a possibility to extend a scramjet's operating range to higher speeds or lower intake dynamic pressures than would otherwise be possible.

Advantages and disadvantages of scramjets

Advantages

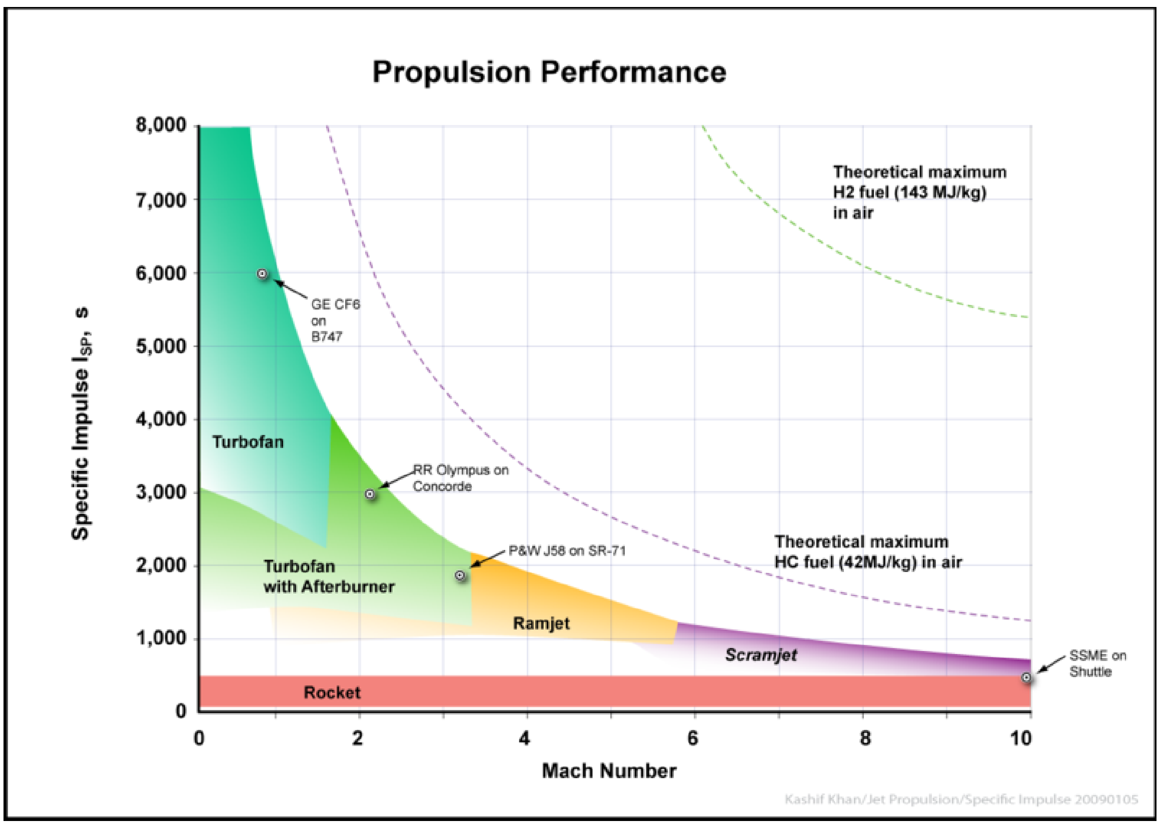
1. Does not have to carry oxygen
2. No rotating parts makes it easier to manufacture
3. Has a higher specific impulse (change in momentum per unit of propellant) than a conventional engine; could provide between 1000 and 4000 seconds, while a rocket only provides 600 seconds or less
4. Higher speed could mean cheaper access to outer space in the future

Special cooling and materials

Unlike a rocket that quickly passes mostly vertically through the atmosphere or a turbojet or ramjet that flies at much lower speeds, a hypersonic airbreathing vehicle optimally flies a "depressed trajectory", staying within the atmosphere at hypersonic speeds. Because scramjets have only mediocre thrust-to-weight ratios, acceleration would be limited. Therefore time in the atmosphere at hypersonic speed would be considerable, possibly 15–30 minutes. Similar to a reentering space vehicle, heat insulation would be a formidable task. The time in the atmosphere would be greater than that for a typical space capsule, but less than that of the space shuttle.

New materials offer good insulation at high temperature, but they often sacrifice themselves in the process. Therefore studies often plan on "active cooling", where coolant circulating throughout the vehicle skin prevents it from disintegrating. Often the coolant is the fuel itself, in much the same way that modern rockets use their own fuel and oxidizer as coolant for their engines. All cooling systems add weight and complexity to a launch system. The cooling of scramjets in this way may result in greater efficiency, as heat is added to the fuel prior to entry into the engine, but result in increased complexity and weight which ultimately could outweigh any performance gains.

Engine weight and efficiency



The specific impulse of various engines

The performance of a launch system is complex and depends greatly on its weight. Normally craft are designed to maximise range (R), orbital radius (R) or payload mass fraction (Γ) for a given engine and fuel. This results in tradeoffs between the efficiency of the engine (takeoff fuel weight) and the complexity of the engine (takeoff dry weight), which can be expressed by the following:

$$\Pi_e + \Pi_f + \frac{1}{\Gamma} = 1$$

Where :

- $\Pi_e = \frac{m_{\text{empty}}}{m_{\text{initial}}}$ is the empty mass fraction, and represents the weight of the superstructure, tankage and engine.
- $\Pi_f = \frac{m_{\text{fuel}}}{m_{\text{initial}}}$ is the fuel mass fraction, and represents the weight of fuel, oxidiser and any other materials which are consumed during the launch.
- $\Gamma = \frac{m_{\text{initial}}}{m_{\text{payload}}}$ is initial mass ratio, and is the inverse of the payload mass fraction. This represents how much payload the vehicle can deliver to a destination.

A scramjet increases the mass of the engine Π_e over a rocket, and decreases the mass of the fuel Π_f. It can be difficult to decide whether this will result in an increased Γ (which would be an increased payload delivered to a destination for a constant vehicle takeoff weight). The logic behind efforts driving a scramjet is (for example) that the reduction in fuel decreases the total mass by 30%, while the increased engine weight adds 10% to the vehicle total mass. Unfortunately the uncertainty in the calculation of any mass or efficiency changes in a vehicle is so great that slightly different assumptions for engine efficiency or mass can provide equally good arguments for or against scramjet powered vehicles.

Additionally, the drag of the new configuration must be considered. The drag of the total configuration can be considered as the sum of the vehicle drag (D) and the engine installation drag (D_e). The installation drag traditionally results from the pylons and the coupled flow due to the engine jet, and is a function of the throttle setting. Thus it is often written as:

$D_e = \varphi_e F$ Where:

- φ_e is the loss coefficient
- F is the thrust of the engine

For an engine strongly integrated into the aerodynamic body, it may be more convenient to think of (D_e) as the difference in drag from a known base configuration.

The overall engine efficiency can be represented as a value between 0 and 1 (η_0), in terms of the specific impulse of the engine:

$$\eta_0 = \frac{g_0 V_0}{h_{PR}} \cdot I_{sp} = \frac{\text{Thrust Power}}{\text{Chemical energy rate}}$$

Where:

- g_0 is the acceleration due to gravity at ground level
- V_0 is the vehicle speed
- I_{sp} is the specific impulse
- h_{PR} is fuel heat of reaction

Specific impulse is often used as the unit of efficiency for rockets, since in the case of the rocket, there is a direct relation between specific impulse, specific fuel consumption and exhaust velocity. This direct relation is not generally present for airbreathing engines, and so specific impulse is less used in the literature. Note that for an airbreathing engine, both η_0 and I_{sp} are a function of velocity.

The specific impulse of a rocket engine is independent of velocity, and common values are between 200 and 600 seconds (450s for the space shuttle main engines). The specific impulse of a scramjet varies with velocity, reducing at higher speeds, starting at about 1200s, although values in the literature vary.

For the simple case of a single stage vehicle, the fuel mass fraction can be expressed as:

$$\Pi_f = 1 - \exp\left[-\frac{\left(\frac{V_{initial}^2}{2} - \frac{V_i^2}{2}\right) + \int g\,dr}{\eta_0 h_{PR}\left(1 - \frac{D+D_e}{F}\right)}\right]$$

Where this can be expressed for single stage transfer to orbit as:

$$\Pi_f = 1 - \exp\left[-\frac{g_0 r_0 \left(1 - \frac{1}{2}\frac{r_0}{r}\right)}{\eta_0 h_{PR}\left(1 - \frac{D+D_e}{F}\right)}\right]$$

or for level atmospheric flight from air launch (missile flight):

$$\Pi_f = 1 - \exp\left[-\frac{g_0 R}{\eta_0 h_{PR}\left(1 - \phi_e\right)\frac{C_L}{C_D}}\right]$$

Where R is the range, and the calculation can be expressed in the form of the Breguet range formula:

$$\Pi_f = 1 - e^{-BR}$$

$$B = \frac{g_0}{\eta_0 h_{PR} \left(1 - \phi_e\right) \frac{C_L}{C_D}}$$

Where:

- C_L is the lift coefficient
- C_D is the drag coefficient

This extremely simple formulation, used for the purposes of discussion assumes:

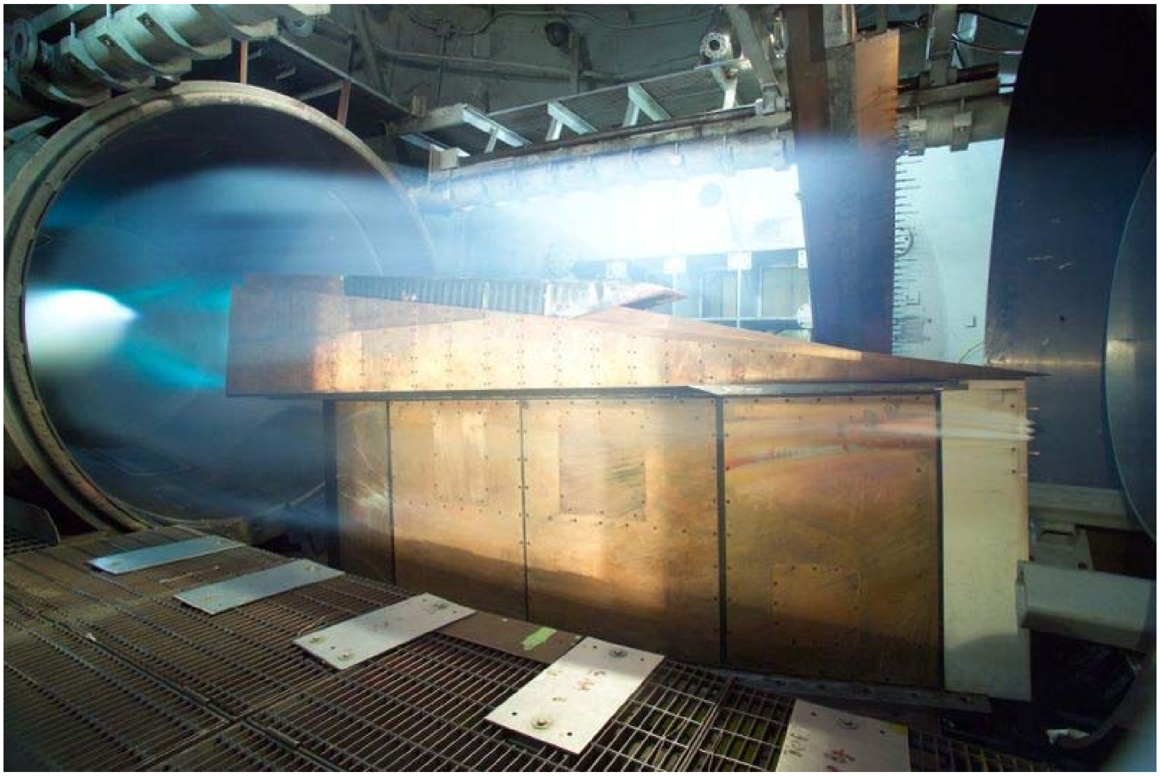
- Single stage vehicle
- No aerodynamic lift for the transatmospheric lifter

However they are true generally for all engines.

Additional propulsion requirements

A scramjet cannot produce efficient thrust unless boosted to high speed, around Mach 5, although depending on the design it could act as a ramjet at low speeds. A horizontal take-off aircraft would need conventional turbofan or rocket engines to take off, sufficiently large to move a heavy craft. Also needed would be fuel for those engines, plus all engine-associated mounting structure and control systems. Turbofan engines are heavy and cannot easily exceed about Mach 2-3, so another propulsion method would be needed to reach scramjet operating speed. That could be ramjets or rockets. Those would also need their own separate fuel supply, structure, and systems. Many proposals instead call for a first stage of droppable solid rocket boosters, which greatly simplifies the design.

Testing difficulties



Test of Pratt & Whitney Rocketdyne SJY61 scramjet engine for the Boeing X-51

Unlike jet or rocket propulsion systems facilities which can be tested on the ground, testing scramjet designs use extremely expensive hypersonic test chambers or expensive launch vehicles, both of which lead to high instrumentation costs. Tests using launched test vehicles very typically end with destruction of the test item and instrumentation.

Advantages and disadvantages for orbital vehicles

Propellant

An advantage of a hypersonic airbreathing (typically scramjet) vehicle like the X-30 is avoiding or at least reducing the need for carrying oxidizer. For example the space shuttle external tank holds 616,432 kg of liquid oxygen (LOX) and 103,000 kg of liquid hydrogen (LH2) while having an empty weight of 30,000 kg. The orbiter gross weight is 109,000 kg with a maximum payload of about 25,000 kg and to get the assembly off the launch pad the shuttle uses two very powerful solid rocket boosters with a weight of 590,000 kg each. If the oxygen could be eliminated, the vehicle could be lighter at liftoff and maybe carry more payload. That would be an advantage, but the central motivation in pursuing hypersonic airbreathing vehicles would be to reduce cost.

On the other hand, scramjets spend more time in the atmosphere and require more hydrogen fuel to deal with aerodynamic drag. Whereas liquid oxygen is quite a dense fluid, liquid hydrogen has much lower density and takes up much more volume. This means that the vehicle using this fuel becomes much bigger and gives even more drag.

Thrust-to-weight ratio

A rocket has the advantage that its engines have *very* high thrust-weight ratios (~100:1), while the tank to hold the liquid oxygen approaches a tankage ratio of ~100:1 also. Thus a rocket can achieve a very high mass fraction (Takeoff rocket mass:unfuelled rocket mass+fuel+oxidiser+structure+engines+payload), which improves performance. By way of contrast the projected thrust/weight ratio of scramjet engines of about 2 mean a very much larger percentage of the takeoff mass is engine (ignoring that this fraction increases anyway by a factor of about four due to the lack of onboard oxidiser). In addition the vehicle's lower thrust does not necessarily avoid the need for the expensive, bulky, and failure prone high performance turbopumps found in conventional liquid-fuelled rocket engines, since most scramjet designs seem to be incapable of orbital speeds in airbreathing mode, and hence extra rocket engines are needed.

Need for additional engines to reach orbit

Scramjets might be able to accelerate from approximately Mach 5-7 to around somewhere between half of orbital velocity and orbital velocity (X-30 research suggested that Mach 17 might be the limit compared to an orbital speed of Mach 25, and other studies put the upper speed limit for a pure scramjet engine between Mach 10 and 25, depending on the assumptions made). Generally, another propulsion system (very typically rocket is proposed) is expected to be needed for the final acceleration into orbit. Since the delta-V is moderate and the payload fraction of scramjets high, lower performance rockets such as solids, hypergolics, or simple liquid fueled boosters might be acceptable.

Reentry

The scramjet's heat-resistant underside potentially doubles as its reentry system if a single-stage-to-orbit vehicle using non-ablative, non-active cooling is visualised. If an ablative shielding is used on the engine it will probably not be usable after ascent to orbit. If active cooling is used the loss of all fuel during the burn to orbit will also mean the loss of all cooling for the thermal protection system.

Costs

Reducing the amount of fuel and oxidizer does not necessarily improve costs as rocket propellants are comparatively very cheap. Indeed, the unit cost of the vehicle can be expected to end up far higher, since aerospace hardware cost is about two orders of magnitude higher than liquid oxygen, fuel and tankage, and scramjet hardware seems to be much heavier than rockets for any given payload. Still, if scramjets enable reusable vehicles, this could theoretically be a cost benefit. Whether equipment subject to the extreme conditions of a scramjet can be reused sufficiently many times is unclear; all flown scramjet tests only survive for short periods and have never been designed to survive a flight to date.

The eventual cost of such a vehicle is the subject of intense debate since even the best estimates disagree whether a scramjet vehicle would be advantageous. It is likely that a scramjet vehicle would need to lift more load than a rocket of equal takeoff weight in order to be equally as cost efficient (if the scramjet is a non-reusable vehicle).

Applications

Seeing its potential, organizations around the world are researching scramjet technology. Scramjets will likely propel missiles first, since that application requires only cruise operation instead of net thrust production. Much of the money for the current research comes from governmental defense research contracts.

Space launch vehicles may or may not benefit from having a scramjet stage. A scramjet stage of a launch vehicle theoretically provides a specific impulse of 1000 to 4000 s whereas a rocket provides less than 450 s while in the atmosphere, potentially permitting much cheaper access to space. A scramjet's specific impulse decreases rapidly with speed, however, and the vehicle would suffer from a relatively low lift to drag ratio.

One issue is that scramjet engines are predicted to have exceptionally poor thrust to weight ratio- around 2. This compares very unfavorably with the 50-100 of a typical rocket engine. This is compensated for in scramjets partly because the weight of the vehicle would be carried by aerodynamic lift rather than pure rocket power (giving reduced 'gravity losses'), but scramjets would take much longer to get to orbit due to lower thrust which greatly offsets the advantage. The takeoff weight of a scramjet vehicle is significantly reduced over that of a rocket, due to the lack of onboard oxidiser, but increased by the structural requirements of the larger and heavier engines.

Whether this vehicle would be reusable or not is still a subject of debate and research.

An aircraft using this type of jet engine could dramatically reduce the time it takes to travel from one place to another, potentially putting any place on Earth within a 90 minute flight. However, there are questions about whether such a vehicle could carry enough fuel to make useful length trips, and there are issues with sonic booms.

There are also questions as to how realistic such a proposal is that revolve around costs (capital and maintenance) of technology that is yet to be developed.

Progress in the 2000s

Artist's conception of the NASA X-43 with scramjet attached to the underside

In the 2000s, significant progress was made in the development of hypersonic technology, particularly in the field of scramjet engines.

US efforts are probably the best funded, and the Hyper-X team claimed the first flight of a thrust-producing scramjet powered vehicle with full aerodynamic maneuvering surfaces in 2004 with the X-43A. However, the first group to demonstrate a scramjet working in an atmospheric test was a project by a joint British and Australian team from UK defense company QinetiQ and the University of Queensland. The HyShot project demonstrated scramjet combustion on July 30, 2002. The scramjet engine worked effectively and demonstrated supersonic combustion in action. However, the engine was not designed to provide thrust to propel a craft. It was designed more or less as a technology demonstrator.

On Friday, June 15, 2007, the US Defense Advanced Research Project Agency (DARPA), in cooperation with the Australian Defence Science and Technology Organization (DSTO), announced a successful scramjet flight at Mach 10 using rocket engines to boost the test vehicle to hypersonic speeds.

On May 22, 2009, Woomera hosted the first successful test flight of a hypersonic aircraft in HIFire. The launch was one of 10 planned test flights. The series of up to 10 planned hypersonic flight experiments are part of a joint research program between the Defence Science and Technology Organisation and the US Air Force, designated as the Hypersonic International Flight Research Experimentation (HIFiRE). HIFiRE is investigating hypersonics technology (the study of flight exceeding five times the speed of sound) and its application to advanced scramjet-powered space launch vehicles — the objective is to support the new Boeing X-51 scramjet demonstrator while also building a strong base of flight test data for quick-reaction space launch development and hypersonic "quick-strike" weapons.

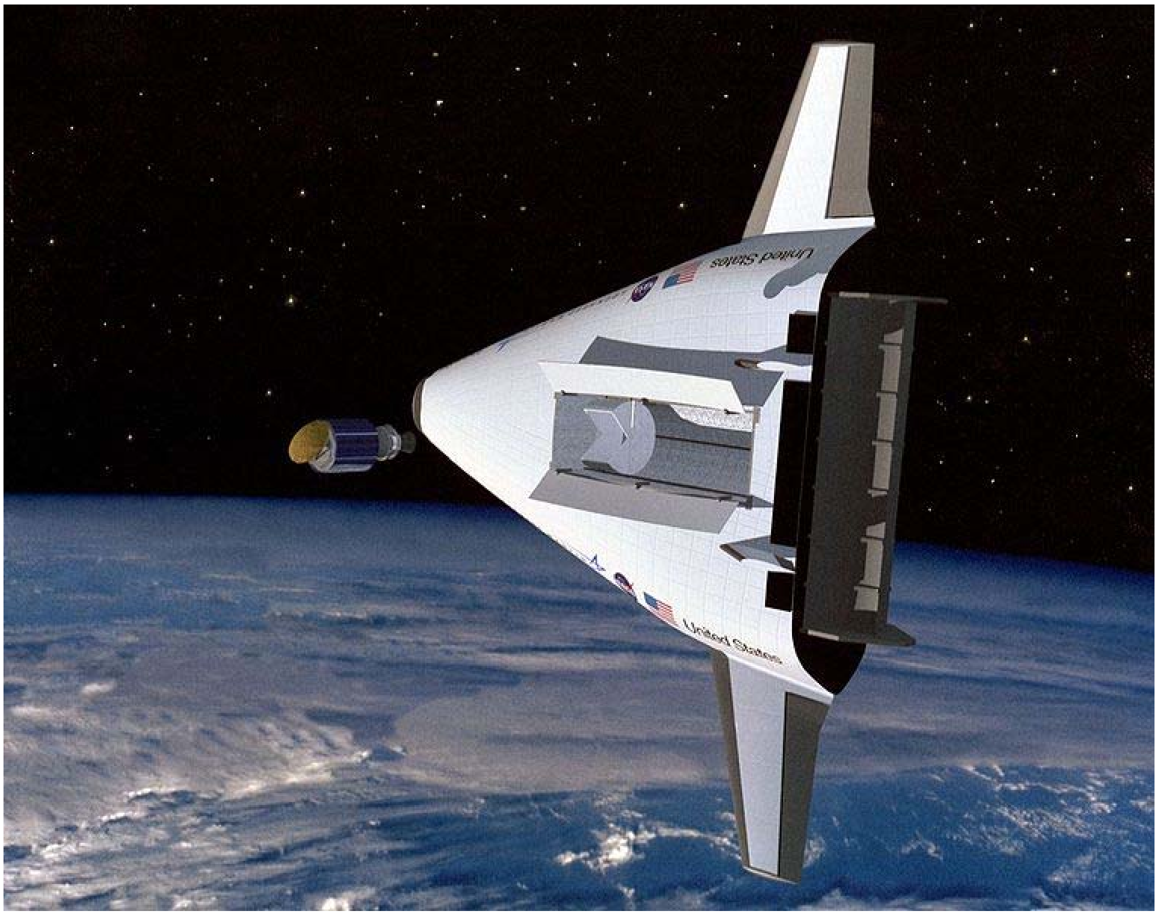
On 22 and 23 March 2010, Australian and American defence scientists successfully tested a (HIFiRE) hypersonic rocket. It reached an atmospheric velocity of "more than 5,000 kilometres per hour" after taking off from the Woomera Test Range in outback South Australia.

On May 27, 2010, NASA and the United States Air Force successfully flew the X-51A Waverider for approximately 200 seconds at Mach 5, setting a new world record hypersonic airspeed. The Waverider flew autonomously before losing acceleration for an unknown reason and destroying itself as planned. The test was declared a success. The X-51A was carried aboard a B-52, accelerated to Mach 4.5 via a solid rocket booster, and then ignited the Pratt & Whitney Rocketdyne scramjet engine to reach Mach 5 at 70,000 feet.

On 16 November 2010, Australian scientists have successfully demonstrated that the high-speed flow in a naturally non-burning scramjet engine can be ignited using a pulsed laser source.

Chapter 5

Single-stage-to-Orbit



The VentureStar was a proposed SSTO spaceplane

A **single-stage-to-orbit** (or **SSTO**) vehicle reaches orbit from the surface of a body without jettisoning hardware, expending only propellants and fluids. The term usually, but not exclusively, refers to reusable vehicles. No Earth-launched SSTO launch vehicles have ever been constructed. Current orbital launches are either performed by multi-stage fully or partially expendable rockets, or by the Space Shuttle which is multi-stage and partially reusable. Several research spacecraft have been designed and partially or completely constructed, including Skylon, the DC-X, the X-33, and the Roton SSTO.

However, despite showing some promise, none of them has come close to achieving orbit yet due to problems with finding the most efficient propulsion system.

Single-stage-to-orbit has been achieved from the Moon by both the Apollo program's Lunar Module and several robotic spacecraft of the Soviet Luna programme; the lower lunar gravity and absence of any significant atmosphere makes this much easier than from Earth.

Approaches to SSTO

There have been various approaches to SSTO, including pure rockets that are launched and land vertically, air-breathing scramjet-powered vehicles that are launched and land horizontally, nuclear-powered vehicles, and even jet-engine-powered vehicles that can fly into orbit and return landing like an airliner, completely intact.

For rocket-powered SSTO, the main challenge is achieving a high enough mass-ratio to carry sufficient propellant to achieve orbit, plus a meaningful payload weight. One possibility is to give the rocket an initial speed with a space gun, as planned in the Quicklaunch project.

For air-breathing SSTO, the main challenge is system complexity and associated research and development costs, material science, and construction techniques necessary for surviving sustained high-speed flight within the atmosphere, *and* achieving a high enough mass-ratio to carry sufficient propellant to achieve orbit, plus a meaningful payload weight. Air-breathing designs typically fly at supersonic or hypersonic speeds, and usually include a rocket engine for the final burn for orbit.

Whether rocket-powered or air-breathing, a reusable vehicle must be rugged enough to survive multiple round trips into space without adding excessive weight or maintenance. In addition a reusable vehicle must be able to reenter without damage, and land safely.

Features of SSTO

The goals of *fully reusable* SSTO vehicles include lower operating costs, improved safety, and better reliability than current launch vehicles. The ultimate goal for an SSTO vehicle would be airliner-like operations.

However, even a non-reusable single-stage vehicle might be worth building, since it would have a much lower part count, and may therefore be cheaper to design and build.

For pure rocket approaches Tsiolkovsky's rocket equation shows that dead weight will prevent reaching orbit unless the ratio of propellant to structural mass (called mass ratio) is very high — between about 10 and 25 (i.e. 24 parts propellant weight to 1 part structural weight; depending on propellant choice).

It is extremely difficult to design a structure which is strong, safe, very light, and economical to build. Designers often liken the task to designing and building an egg shell. The problem originally seemed insuperable, and drove all early designers to multistage rockets.

Multistage rockets are able to reach orbital velocity because they discard structural weight during boost. Thus a single-stage rocket is at a disadvantage because it must carry its entire vehicle mass to orbit, which in turn reduces payload capacity. On the other hand, a single-stage vehicle does not have to carry a second stage, so the vehicle is easier to make lightweight.

Alternatively, since expendable multistage rockets entail discarding costly structure and engines, if the stages could be reused, this could permit much cheaper operation since the parts costs would be amortized over many flights.

One problem with multistage reusable rockets is the difficulty of reusing even the first stage, and the development cost of such a large device. Analysis shows the optimum staging velocity (the speed at which the first stage is dropped) is very high — possibly 3.65 km/s (12,000 feet per second). This means after separation, the large first stage is at high altitude and headed downrange very fast, which makes it difficult to turn around and get back to the launch point. The stage also must reenter without damage from a speed as high as Mach 10.

The reusable first stage would be very large, nearly the size of a Saturn V to lift an orbiter the size of the current shuttle. Because development cost of aerospace vehicles has historically been related to weight, it is assumed that such a vehicle would be extremely expensive to develop.

Some approaches envisioned parachutes to gently lower a reusable first stage. However, for most US launches the trajectory is over the Atlantic ocean, and complex liquid-fueled stages are easily damaged by a salt water landing.

These problems with the multistage approach drive the design path toward SSTO.

All these complications drove designers to consider a single reusable stage as this:

- Avoids discarding expensive engines and structure (vs. expendable).
- Avoids difficulty of retrieving the large first stage (vs. reusable multistage).
- Avoids increased development cost of two separate vehicles (vs. reusable multistage).

If an SSTO vehicle were combined with reliable systems and lower maintenance design of a more automated nature, it could greatly reduce operational costs. Fully recoverable SSTO craft can be flown on short test missions, and developed incrementally, since no hardware is expended in test flights.

On the other hand, an SSTO vehicle needs to lift its entire structure into orbit. To reach orbit with a useful payload, the rocket requires careful and extensive engineering to save weight. Although an SSTO rocket might theoretically be built, margins would be likely to be very thin: even comparatively minor problems could mean that the craft may fail to achieve the necessary mass-fraction to reach orbit with useful payload.

While single-stage rockets were once thought to be beyond reach, but advances in materials technology and construction techniques have shown them to be possible. For example, calculations show that the Titan II first stage, launched on its own, would have a 25-to-1 ratio of fuel to vehicle hardware. It has a sufficiently efficient engine to achieve orbit, but without carrying much payload.

Dense versus hydrogen fuels

Hydrogen might seem the obvious fuel for SSTO vehicles. When burned with oxygen, hydrogen gives the highest specific impulse of any commonly used fuel: around 450 seconds, compared with up to 350 seconds for kerosene.

Hydrogen has the following advantages:

- Hydrogen has nearly 30% higher specific impulse (about 450 seconds vs. 350 seconds) than most dense fuels.
- Hydrogen is an excellent coolant.
- The gross mass of hydrogen stages is lower than dense-fuelled stages for the same payload.

However, hydrogen also has these disadvantages:

- Very low density (about 1/7 of the density of kerosene) — requiring a very large tank,
- Deeply cryogenic — must be stored at very low temperatures and thus needs heavy insulation,
- Escapes very easily from the smallest gap,
- Wide combustible range — easily ignited and burns with a dangerously invisible flame,
- Tends to condense oxygen which can cause flammability problems,
- Has a large coefficient of expansion for even small heat leaks.

These issues can be dealt with, but at extra cost.

While kerosene tanks can be 1% of the weight of their contents, hydrogen tanks often must weigh 10% of their contents. This is because of both the low density and the additional insulation required to minimize boiloff (a problem which does not occur with kerosene and many other fuels). The low density of hydrogen further affects the design of the rest of the vehicle — pumps and pipework need to be much larger in order to pump the fuel to the engine. The end result is the thrust/weight ratio of hydrogen-fueled engines is 30–50% lower than comparable engines using denser fuels.

This inefficiency indirectly affects gravity losses as well; the vehicle has to hold itself up on rocket power until it reaches orbit. The lower excess thrust of the hydrogen engines due to the lower thrust/weight ratio means that the vehicle must ascend more steeply, and so less thrust acts horizontally. Less horizontal thrust results in taking longer to reach orbit, and gravity losses are increased by at least 300 meters per second. While not appearing large, the mass ratio to delta-v curve is very steep to reach orbit in a single stage, and this makes a 10% difference to the mass ratio on top of the tankage and pump savings.

The overall effect is that there is surprisingly little difference in overall performance between SSTOs that use hydrogen and those that use denser fuels, except that hydrogen vehicles may be rather more expensive to develop and buy. Careful studies have shown that some dense fuels (for example liquid propane) exceed the performance of hydrogen fuel when used in an SSTO launch vehicle by 10% for the same dry weight.

Operational experience with the DC/X experimental rocket has caused a number of SSTO advocates to reconsider hydrogen as a satisfactory fuel. The late Max Hunter, while employing hydrogen fuel in the DC/X, often said that he thought the first successful orbital SSTO would more likely be fueled by propane.

One engine for all altitudes

Some SSTO vehicles use the same engine for all altitudes, which is a problem for traditional engines with a bell-shaped nozzle. Depending on the atmospheric pressure, different bell shapes are optimal. Engines operating in the lower atmosphere have shorter bells than those designed to work in vacuum. Having a bell not optimized for the height makes the engine less efficient.

One possible solution would be to use an aerospike engine, which can be effective in a wide range of ambient pressures. In fact, a linear aerospike engine was used in the X-33 design.

Other solutions involve using multiple engines and other altitude adapting designs such as double-mu bells or extensible bell sections.

Still, at very high altitudes, the extremely large engine bells tend to expand the exhaust gases down to near vacuum pressures. As a result, these engine bells are counterproductive due to their excess weight. Some SSTO vehicles simply use very high pressure engines which permit high ratios to be used from ground level. This gives good performance, negating the need for more complex solutions.

Airbreathing SSTO

Some designs for SSTO attempt to use airbreathing jet engines that collect oxidiser and reaction mass from the atmosphere to reduce the take-off weight of the vehicle.

Some of the issues with this approach are:

- No known air breathing engine is capable of operating at orbital speed within the atmosphere (for example hydrogen fueled scramjets seem to have a top speed of about Mach 17). This means that rockets must be used for the final orbital insertion.
- Rocket thrust needs the orbital mass to be as small as possible to minimize propellant weight.
- Oxidiser tanks are very lightweight when empty, approximately 1% of their contents, so the reduction in orbital weight by airbreathing is small, whereas air-breathing engines have a poor thrust/weight ratio which tends to increase the orbital mass.
- Very high speeds in the atmosphere necessitate very heavy thermal protection systems, which makes reaching orbit even harder.
- While at lower speeds, air-breathing engines are very efficient, the efficiency (Isp) and thrust levels of air-breathing jet engines drop considerably at high speed (above Mach 5–10 depending on the engine) and begin to approach that of rocket engines or worse.
- Lift to drag ratios of vehicles at hypersonic speeds are poor whereas since acceleration is a vector, the effective lift to drag ratios of rocket vehicles at high g is not dissimilar.

Thus with for example scramjet designs (e.g. X-43) the mass budgets do not seem to close for orbital launch.

Similar issues occur with single stage vehicles attempting to carry conventional jet engines to orbit- the weight of the jet engines is not compensated by the reduction in propellant sufficiently.

On the other hand LACE-like precooled airbreathing designs such as the Skylon spaceplane (and ATREX) which transition to rocket thrust at rather lower speeds (Mach 5.5) do seem to give, on paper at least, an improved orbital mass fraction over pure rockets (even multistage rockets) sufficiently to hold out the possibility of full reusability with better payload fraction.

It is important to note that mass fraction is an important concept in the engineering of a rocket. However, mass fraction may have little to do with the costs of a rocket, as the costs of fuel are very small when compared to the costs of the engineering program as a whole. As a result, a cheap rocket with a poor mass fraction may be able to deliver more payload to orbit with a given amount of money than a more complicated, more efficient rocket.

Launch assists

Many vehicles are only narrowly suborbital, so practically anything that gives a relatively small delta-v increase can be helpful, and outside assistance for a vehicle is therefore desirable.

Proposed launch assists include:

- sled launch (maglev, rail etc.)
- aircraft tow
- in-flight fueling
- Lofstrom launch loop/space fountains

And on-orbit resources such as:

- hypersonic tethers
- tugs

Nuclear propulsion

Due to weight issues such as shielding, many nuclear propulsion systems are unable to lift their own weight, and hence are unsuitable for launching to orbit. However some designs such as the Orion project and some nuclear thermal designs do have a thrust to weight ratio in excess of 1, enabling them to lift off. Clearly one of the main issues with nuclear propulsion would be safety, both during a launch for the passengers, but also in case of a failure during launch. No current program is attempting nuclear propulsion from Earth's surface.

Beam-powered propulsion

Because they can be more energetic than the potential energy that chemical fuel allows for, some laser or microwave powered rocket concepts have the potential to launch vehicles into orbit, single stage. In practice, this area is relatively undeveloped, and current technology falls far short of this.

Comparison with the Shuttle

The high cost per launch of the Space Shuttle sparked interest throughout the 1980s in designing a cheaper successor vehicle. Several official design studies were done, but most were basically smaller versions of the existing Shuttle concept.

Most cost analysis studies of the Space Shuttle have shown that workforce is by far the single greatest expense. Early shuttle discussions speculated airliner-type operation, with a two-week turnaround. However, senior NASA planners envisioned no more than 10 to 12 flights per year for the entire shuttle fleet. The absolute maximum flights per year for the entire fleet was limited by external tank manufacturing capacity to 24 per year.

Very efficient (hence complex and sophisticated) main engines were required to fit within the available vehicle space. Likewise the only known suitable lightweight thermal protection was delicate, maintenance-intensive silica tiles. These and other design decisions resulted in a vehicle that requires great maintenance after every mission. The engines are removed and inspected, and prior to the new "block II" main engines, the turbopumps were removed, disassembled and rebuilt. While Space Shuttle Atlantis was refurbished and relaunched in 53 days between missions STS-51-J and STS-61-B, generally months are required to repair an orbiter for a new mission. Given that there are 25,000 people working on Shuttle operations, the payroll alone is the Shuttle's single biggest operating cost.

Many in the aerospace community concluded that an entirely self-contained, reusable single-stage vehicle could solve these problems. The idea behind such a vehicle is to reduce the processing requirements from those of the Shuttle.

Examples

The early Atlas rocket is an expendable SSTO by some definitions. It is a **"stage-and-a-half"** rocket, jettisoning two of its three engines during ascent but retaining its fuel tanks and other structural elements. However, by modern standards the engines ran at low pressure and thus not particularly high specific impulse and were not especially lightweight; using engines operating with a higher specific impulse would have eliminated the need to drop engines in the first place.

The first stage of the Titan II had the mass ratio required for single-stage-to-orbit capability with a small payload. A rocket stage is not a complete launch vehicle, but this demonstrates that an expendable SSTO was probably achievable with 1962 technology.

The Apollo Lunar Module was a true SSTO vehicle, albeit on the moon. It achieved lunar orbit using a single stage.

A detailed study into SSTO vehicles was prepared by Chrysler Corporation's Space Division in 1970–1971 under NASA contract NAS8-26341. Their proposal (Shuttle SERV) was an enormous vehicle with more than 50,000 kg of payload, utilizing jet engines for (vertical) landing. While the technical problems seemed to be solvable, the USAF required a winged design (for cross range) that led to the Shuttle as we know it today.

The unmanned DC-X technology demonstrator, originally developed by McDonnell Douglas for the Strategic Defense Initiative (SDI) program office, was an attempt to build a vehicle that could lead to an SSTO vehicle. The one-third-size test craft was operated and maintained by a tiny crew of three people based out of a trailer, and the craft was once relaunched less than 24 hours after landing. Although the test program was not without mishap (including a minor explosion), the DC-X demonstrated that the maintenance aspects of the concept were sound. That project was cancelled when it crashed on the fourth flight after transferring management from the Strategic Defense Initiative Organization to NASA.

The Aquarius Launch Vehicle was designed to bring bulk materials to orbit as cheaply as possible.

Current Development

Current private SSTO projects include the Japanese Kankoh-maru project and the Skylon.

Skylon

The British Government partnered with the ESA in 2010 to promote a single-stage to orbit spaceplane concept called Skylon. This design was pioneered by Reaction Engines Limited, a company founded by Alan Bond after HOTOL was canceled. The Skylon spaceplane has been positively received by the British government, and the British Interplanetary Society. Pending a successful engine test in June 2011, the company will begin Phase 3 of development with the first orders expected around 2011-2013.

Alternative approaches to cheap spaceflight

Many studies have shown that regardless of selected technology, the most effective cost reduction technique is economies of scale. Merely launching a large total quantity reduces the manufacturing costs per vehicle, similar to how the mass production of automobiles brought about great increases in affordability.

Using this concept, some aerospace analysts believe the way to lower launch costs is the exact opposite of SSTO. Whereas reusable SSTOs would reduce per launch costs by making a reusable high-tech vehicle that launches frequently with low maintenance, the "mass production" approach views the technical advances as a source of the cost problem in the first place. By simply building and launching large quantities of rockets, and hence launching a large volume of payload, costs can be brought down. This approach was attempted in the late ’70s, early ’80s in West Germany with the Democratic Republic of the Congo-based OTRAG rocket and could have been successful if the project was not killed following political pressure from France and the Soviet Union.

A related idea is to obtain economies of scale from building simple, massive, multi-stage rockets using cheap, off-the-shelf parts. The vehicles would be dumped into the ocean after use. This strategy is known as the "big dumb booster" approach.

This is somewhat similar to the approach some previous systems have taken, using simple engine systems with "low-tech" fuels, as the Russian and Chinese space programs still do. These nations' launches are significantly cheaper than their Western counterparts.

Chapter 6

Space Gun

A **space gun** is a method of launching an object into outer space using a large gun, or cannon. It provides a method of non-rocket spacelaunch.

In the HARP Project a U.S. Navy 16 inch (406 mm) 100 caliber gun (40 m) was used to fire a 180 kilogram slug at 3,600 meters per second, reaching an apogee of 180 kilometers, hence performing a suborbital spaceflight.

However, a space gun has never been successfully used to launch an object into orbit.

Technical issues

The large g-force experienced by a ballistic projectile would likely mean that a space gun would be incapable of safely launching humans or delicate instruments, rather being restricted to freight or ruggedized satellites.

Atmospheric drag also makes it more difficult to control the trajectory of any projectile launched, subjects the projectile to extremely high forces, and causes severe energy losses that may not be easily overcome. The lower troposphere is the densest layer of the atmosphere, and some of these issues may be mitigated by using a space gun with a "gun barrel" reaching above it (e.g. a gun emplacement on a mountaintop).

A space gun, by itself, is generally not capable of placing objects into stable orbit around the planet, unless the objects are able to perform course corrections after launch.

If acceptable solutions to these fundamental issues could be achieved, a space gun could offer access to space at an unprecedented low cost.

Getting to orbit

A space gun, by itself, is not capable of placing objects into stable orbit. The laws of gravitation make it impossible to reach a stable orbit without an active payload which performs orbital correction burns to change the shape of its orbit after launch. The orbit is a parabolic orbit, a hyperbolic orbit, or part of an elliptic orbit which ends at the planet's surface at the point of launch or another point. This means that an uncorrected ballistic payload will always strike the planet within its first orbit unless the velocity was so high as to reach or exceed escape velocity.

Isaac Newton avoided this objection in his thought experiment by positing an impossibly tall mountain from which his cannon was fired. The projectile, however, would still tend to circle the planet and strike the point of launch.

As a result, all payloads intended to reach a closed orbit would have at least to perform some sort of course correction to create another orbit that does not intersect the planet's surface. In addition a rocket can be used for additional boost, as planned in the Quicklaunch project.

It is conceivable that in a multi-body gravitational system, like the Earth-Moon system, that a trajectory could be found that does not re-intersect the Earth's surface, although these paths would likely not be very simple nor desirable, and would require much more energy.

Acceleration

A space gun with a "gun barrel" of length (l), and the needed velocity (v_e), the acceleration (a) is provided by the following formula:

$$a = \frac{v_e^2}{2l}$$

For instance, with a space gun with a vertical "gun barrel" through both the Earth's crust and the troposphere, totalling ~60 km of length (l), and a velocity (v_e) enough to escape the Earth's gravity (escape velocity, which is 11.2 km/s on Earth), the acceleration (a) would theoretically be more than 1000 m/s^2, which is more than 100 g-forces, which is about 3 times the human tolerance to g-forces of maximum 20 to 35 *g* during the ~10 seconds such a firing would take.

Any doubling of the barrel length would theoretically cut the generated g-force in half.

Practical attempts

The German V-3 cannon program (less well known than the V-2 rocket or V-1 flying bomb), during the Second World War was an attempt to build something approaching a space gun. Based in the Pas-de-Calais area of France it was planned to be more devastating than the other Nazi 'Vengeance weapons'. It was destroyed by RAF bombing using 'Tallboy' blockbuster bombs in July 1944.

Project HARP, a prototype of a space gun

Two sections of the Project Babylon gun

On the practical side, the most prominent recent attempt to make a space gun was artillery engineer Gerald Bull's Project Babylon, which was also known as the 'Iraqi supergun' by the media. During Project Babylon, Bull used his experience from Project HARP to build a massive cannon for Saddam Hussein leader of Ba'athist Iraq. This gun, had it been completed, would have been the first true space gun capable of launching objects into space. However, Bull was assassinated before the project was completed.

Since Bull's death, few have seriously attempted to build a space gun. Perhaps most promisingly, the US Ballistic Missile Defense program sponsored the Super High Altitude Research Project in the 1980s. Developed at Lawrence Livermore Laboratory, it is a light gas gun and has been used to test fire objects at Mach 9. One of the lead developers John Hunter has since founded the Jules Verne Launcher Company in 1996, though has as yet been unable to find funding for the multi-billion dollar project. He has now founded the Quicklaunch company.

Ram accelerators have also been proposed as an alternative to light gas guns. Other proposals use electromagnetic techniques for accelerating the payload, such as coilguns and railguns.

Chapter 7

Lynx Rocketplane

The Lynx rocketplane in flight (artists' conception) - XCOR Aerospace

The **Lynx rocketplane** is a suborbital horizontal-takeoff, horizontal-landing (HTHL), rocket-powered spaceplane being developed by the California-based company XCOR to compete in the emerging suborbital space flight market. The Lynx is projected to carry one pilot, a ticketed passenger, and/or a payload or small satellites above 100 km altitude. As of December 2008, the passenger ticket was to cost $95,000. The Lynx was initially announced on March 26, 2008, with plans for an operational vehicle within two years. That date has since fallen to late 2011.

Description

The Lynx will have four liquid rocket engines at the rear of the fuselage burning a mixture of LOX-Kerosene and each of them will produce 2,900 pounds-force (13,000 N) of thrust.

Mark I Prototype

- Maximum Altitude: 62 km (203,000 ft)
- Primary Internal Payload: 120 kg (260 lb)
- External Dorsal Mounted Pod: 280 kg (620 lb)
- Secondary payload spaces include a small area inside the cockpit behind the pilot or outside the vehicle in two areas in the aft fuselage fairing.

Mark II Production Model

- Maximum Altitude: +100 km (330,000 ft)
- Primary Internal Payload: 120 kg (260 lb)
- External Dorsal Mounted Pod: 650 kg (1,400 lb) and is large enough to hold a two stage carrier to launch a microsatellite or multiple nanosatellites into low Earth orbit.
- Secondary payload spaces include the same as the Mark I.
- Non-toxic (non-hydrazine) reaction control system (RCS) thrusters, type 3N22

Test program

Tests of the XR-5K18 main engine began in 2008 and, as of February 2011, are largely complete.

As of February 2011, the vehicle aerodynamic design has completed two rounds of wind tunnel testing. A third and final round of tests is planned for later in 2011.

Flight tests of the Mark I prototype are expected to start in 2011.

Operations

Artists' depiction of Lynx on ground with people - XCOR Aerospace

The first engine hot fire tests were conducted on December 15, 2008. Wind tunnel tests were conducted in July 2009.

NASA sRLV program

As of March 2011, XCOR has submitted the Lynx as a reusable launch vehicle for carrying research payloads in response to NASA's suborbital reusable launch vehicle (sRLV) solicitation, which is a part of NASA's Flight Operations Program. XCOR projects 110 km (68 mi) altitude in flights of 30 to 45 minutes duration, while carrying up to 140 kg (310 lb) internal—or 650 kg (1,400 lb) external—of research payload. Flights will provide up to three minutes of microgravity below 0.01 g

Commercial operations

According to XCOR, the Lynx will fly four or more times a day, and will also have the capacity to deliver payloads into space. A Lynx prototype called Mark I is expected to perform its first test flight in early 2011, followed with a flight of the Mark II production model nine to eighteen months after. XCOR currently plans to have the Lynx's initial flights from the Mojave Air and Spaceport in Mojave, California or any licensed spaceport with a 2,400 meter (7900 ft) runway. Beginning in January, 2014 the Lynx is expected to be flying suborbital space tourism flights and scientific research missions from a new spaceport on the Caribbean island of Curaçao.

Because it lacks any propulsion system other than its rocket engines, the Lynx will have to be towed to the end of the runway. Once positioned on the runway, the pilot will ignite the four rocket engines and begin a steep climb. The engines will be shut off at approximately 138,000 feet (42 km) and Mach 2. The spaceplane will then continue to climb, unpowered until it reaches an apogee of approximately 200,000 feet (61 km). The spacecraft will experience a little over four minutes of weightlessness before re-entering the Earth's atmosphere. The occupants of the Lynx may experience up to four times normal gravity during re-entry. Once it has completed re-entry, the Lynx will then glide down and perform an unpowered landing. The total flight time is projected to last about 30 minutes. The Lynx is expected to be able to perform 40 flights before maintenance is required.

The occupants would wear pressure suits made by Orbital Outfitters.

The successor to the Mark II is planned to be a two stage fully-reusable orbital vehicle that takes off and lands horizontally.

Chapter 8

Expendable Launch System

A Delta II ELV launching the *Dawn* spacecraft from CCAFS SLC-17

An **expendable launch system** is a launch system that uses an **expendable launch vehicle** (ELV) to carry a payload into space. The vehicles used in expendable launch systems are designed to be used only once (i.e. they are "expended" during a single flight), and their components are not recovered for re-use after launch. The vehicle typically consists of several rocket stages, discarded one by one as the vehicle gains altitude and speed.

Design rationale

The ELV design differs from that of reusable launch systems, where the vehicle is launched and recovered more than once. Reuse might seem to make systems like the Space Shuttle more cost effective than ELVs, but in practice launches using ELVs have been less expensive than Shuttle launches. Most satellites are currently launched using expendable launchers; they are perceived as having a low risk of mission failure, a short time to launch and a relatively low cost.

History

Many orbital expendable launchers are derivatives of 1950s-era ballistic missiles. Many see this as unfortunate because cost was not a major consideration in their design. A prime example of this is the Titan IV, probably the costliest per-unit launch vehicle in history (perhaps following the Space Shuttle).

On the other hand, a reusable launcher such as the Shuttle requires a heavier structure and a recovery system (wings, thermal protection system, wheels, etc) that reduce payload capacity. The Shuttle additionally carries a crew (though not inherent to a reusable system) whose weight, supplies and life support systems further decrease payload capacity.

A Shuttle orbiter is a major national asset, and its high cost (far more than a single expendable launch vehicle) and presence of a crew require stringent "man rated" flight safety precautions that increase launch and payload costs. Only five orbiters were built, and the unexpected loss of two (Challenger and Columbia) significantly impacted the capacity and viability of the Shuttle program. Each loss also resulted in an extended hiatus in Shuttle flights compared to that following most expendable launch failures, each of which impacted only that model of launcher.

For these reasons it is generally agreed that the Space Shuttle has not delivered on its original promise to reduce the costs of constructing and launching payloads into orbit. The Shuttle was originally intended to replace expendable launchers in the launching of satellites, but after the loss of **Challenger** the Shuttle was reserved for previously planned missions and those requiring a crew.

Spacecraft launched by the Shuttle included several TDRSS communications relays heavily used by the Shuttle program itself, a series of commercial communication satellites, and the interplanetary probes Magellan, Galileo and Ulysses. Several classified military payloads were also carried.

Development

European sponsorship

On March 26, 1980, the European Space Agency and the Centre National d'Etudes Spatiales (CNES) created Arianespace, the world's first commercial space transportation company. Arianespace produces, operates and markets the Ariane launcher family. By 1995 Arianespace lofted its 100th satellite and by 1997 the Ariane rocket had its 100th launch. Arianespace's 23 shareholders represent scientific, technical, financial and political entities from 10 different European countries. The major shareolder is the CNES, with 34.68% of capital.

American deregulation

From the beginning of the Shuttle program until the *Challenger* disaster in 1986, it was the policy of the United States that NASA be the public-sector provider of U.S. launch capacity to the world market. Initially NASA subsidized satellite launches with the intention of eventually pricing Shuttle service for the commercial market at long-run marginal cost.

On October 30, 1984, United States President Ronald Reagan signed into law the Commercial Space Launch Act. This enabled an American industry of private operators of expendable launch systems. Prior to the signing of this law, all commercial satellite launches in the United States were limited to NASA's Space Shuttle.

On November 5, 1990, United States President George H. W. Bush signed into law the Launch Services Purchase Act. The Act, in a complete reversal of the earlier Space Shuttle monopoly, ordered NASA to purchase launch services for its primary payloads from commercial providers whenever such services are required in the course of its activities.

Russian privatization

The Russian government sold part of its stake in RSC Energia to private investors in 1994. Energia together with Khrunichev constituted most of the Russian manned space program. In 1997, the Russian government sold off enough of its share to lose the majority position.

American subsidization

In 1996 the United States government selected Lockheed Martin and Boeing to each develop Evolved Expendable Launch Vehicles (EELV) to compete for launch contracts and provide assured access to space. The government's acquisition strategy relied on the strong commercial viability of both vehicles to lower unit costs. This anticipated market demand did not materialize, but both the Delta IV and Atlas V EELVs remain in active service.

Launch alliances

Since 1995 Khrunichev's Proton rocket is marketed through International Launch Services while the Soyuz rocket is marketed via Starsem. Energia builds the Soyuz rocket and owns part of the Sea Launch project which flies the Ukrainian Zenit rocket.

In 2003 Arianespace joined with Boeing Launch Services and Mitsubishi Heavy Industries to create the Launch Services Alliance. In 2005, continued weak commercial demand for EELV launches drove Lockheed Martin and Boeing to propose a joint venture called the United Launch Alliance to monopolize the United States government launch market.

Today many commercial space transportation companies offer launch services to satellite companies and government space organizations around the world. In 2005 there were 18 total commercial launches and 37 non-commercial launches. Russia flew 44% of commercial orbital launches, while Europe had 28% and the United States had 6%.

Chapter 9

Ariane 5

Ariane 5

Ariane 5 mock-up (Photo taken at Cité de l'espace)

Function	Heavy launch vehicle
Manufacturer	Astrium for ESA and Arianespace
Country of origin	Europe

Size

Height	46-52 m (151-170 ft)
Diameter	5.4 m (17.7 ft)
Mass	777,000 kg (1,712,000 lb)
Stages	2
Capacity	
Payload to LEO	**G:** 16,000 kg **ES:** 21,000 kg
Payload to GTO	**G:** 6,200 kg **G+:** 6,950 kg **GS:** 6,100 kg **ECA:** 10,500 kg
Launch history	
Status	Active
Launch sites	ELA-3, Guiana Space Centre
Total launches	56 (**G:** 16, **G+:** 3, **GS:** 6) (**ECA:** 29, **ES:** 2)
Successes	52 (**G:** 13, **G+:** 3, **GS:** 6) (**ECA:** 28, **ES:** 2)
Failures	2 (**G:** 1, **ECA:** 1)
Partial failures	2 (**G**)
Maiden flight	**G:** 4 June 1996 **G+:** 2 March 2004 **GS:** 11 August 2005 **ECA:** 11 December 2002 **ES:** 9 March 2008
Last flight	**G:** 27 September 2003 **G+:** 18 December 2004 **GS:** 18 December 2009
Notable payloads	Rosetta Automated Transfer Vehicle Herschel Space Observatory Planck
Boosters (Stage 0) - P230	
№ boosters	2
Engines	1 Solid
Thrust	6,470 kN (1,450,000 lbf)
Specific impulse	275 s
Burn time	129 seconds
Fuel	Solid
First stage (Ariane 5G)	
Engines	1 Vulcain
Thrust	1,114 kN (250,000 lbf)
Specific impulse	430 s
Burn time	589 seconds

Fuel	LH2/LOX
First stage (Ariane 5 ECA)	
Engines	1 Vulcain 2
Thrust	1,340 kN (301,000 lbf)
Specific impulse	431 s
Burn time	650 seconds
Fuel	LH2/LOX
Second stage (Ariane 5G)	
Engines	1 Aestus
Thrust	27.4 kN (6,160 lbf)
Specific impulse	324 s
Burn time	1,100 seconds
Fuel	N2O4/MMH
Second stage (Ariane 5 ECA)	
Engines	1 HM7-B
Thrust	64.7 kN (14,500 lbf)
Specific impulse	446 s
Burn time	960 seconds
Fuel	LH2/LOX

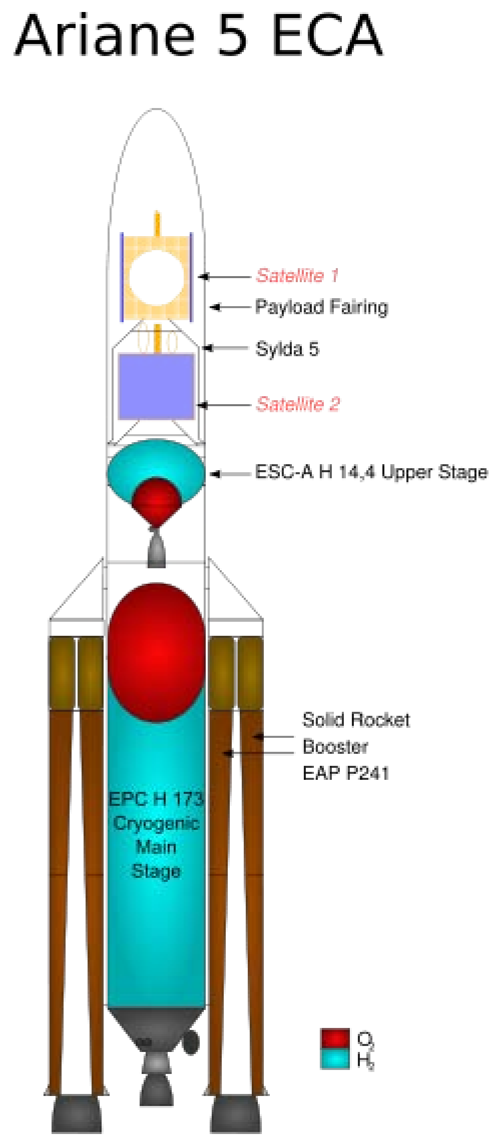
Ariane 5 is an expendable launch system used to deliver payloads into geostationary transfer orbit (GTO) or low Earth orbit (LEO). Ariane 5 rockets are manufactured under the authority of the European Space Agency (ESA) and the Centre National d'Etudes Spatiales (CNES). Astrium, an EADS company, is the prime contractor for the vehicles, leading a consortium of sub-contractors. Ariane 5 is operated and marketed by Arianespace as part of the *Ariane* programme. Astrium builds the rockets in Europe and Arianespace launches them from the Guiana Space Centre in French Guiana.

Ariane 5 succeeded Ariane 4, but was not derived from it directly. Ariane 5 has been refined since the first launch in successive versions, "G", "G+", "GS", "ECA", and most recently, "ES". ESA originally designed Ariane 5 to launch the manned mini shuttle Hermes, and thus intended it to be "human rated" from the beginning.

Two satellites can be mounted using a SYLDA carrier (*SYstème de Lancement Double Ariane*). Three main satellites are possible depending on size using SPELTRA (*Structure Porteuse Externe Lancement TRiple Ariane*). Up to eight secondary payloads, usually small experiment packages or minisatellites, can be carried with an ASAP (*Ariane Structure for Auxiliary Payloads*) platform.

By mid 2007, Arianespace has ordered a total of 99 Ariane 5 launchers from Astrium. The first batch ordered in 1995 consisted of 14 launchers, while the second—P2—batch ordered in 1999 consisted of 20 launchers. A third—PA—batch consisting of 25 ECA and 5 ES launchers was ordered in 2004. The latest batch ordered in mid 2007 consist of another 35 ECA launchers. Through these orders, the Ariane 5 will be the workhorse of Arianespace at least through 2015.

Vehicle Description



Cut drawing of an Ariane 5 ECA

Vulcain engine

Cryogenic main stage

Ariane 5's cryogenic H158 main stage (H173 for Ariane 5 ECA) is called the EPC (*Étage Principal Cryotechnique* - Cryotechnic Main Stage). It consists of a large tank 30.5 metres high with two compartments, one for 130 tonnes of liquid oxygen and one for 25 tonnes of liquid hydrogen, and a Vulcain engine at the base with thrust of 115 tonnes-force (1.13 meganewtons). This part of the first stage weighs about 15 tonnes when empty.

Solid boosters

Attached to the sides are two solid rocket boosters (SRBs or EAPs from the French *Étages d'Accélération à Poudre*), P238 (P241 for Ariane 5 ECA), each weighing about 277 tonnes full. Each delivers a thrust of about 630 tonnes-force (6.2 MN). These SRBs are usually allowed to sink to the bottom of the ocean, but like the Space Shuttle Solid Rocket Boosters they can be recovered with parachutes, and this is occasionally done for post-flight analysis. (Unlike Space Shuttle SRBs Ariane 5 boosters are not reused.) The most recent attempt was for the first Ariane 5 ECA mission. One of the two boosters was successfully recovered and returned to the Guiana Space Center for analysis. Prior to that mission, the last such recovery and testing was done in 2003.

The French M51 SLBM shares a substantial amount of technology with these boosters.

In March 2000 the nose cone of an Ariane 5 booster washed ashore on the South Texas coast, and was recovered by beachcombers.

Second stage

The second stage is on top of the main stage and below the payload. The Ariane 5G used the EPS (*Étage à Propergols Stockables* - Storable Propellant Stage), which is fueled by monomethylhydrazine (MMH) and nitrogen tetroxide, whereas the Ariane 5 ECA uses the ESC (*Étage Supérieur Cryotechnique* - Cryogenic Upper Stage), which is fueled by liquid hydrogen and liquid oxygen.

The EPS upper stage is capable of re-ignition, which has been demonstrated twice. The first demonstration occurred during flight V26, which was launched on 5 October 2007. This was purely to test the engine, and occurred after the payloads had been deployed. The first operational use of restart capability as part of a mission, came on 9 March 2008, when two burns were made to deploy the first Automated Transfer Vehicle into a circular parking orbit.

Fairing

The payload and all upper stages are covered at launch by a fairing, which splits off once sufficient altitude has been reached (typically above 150 km). Ariane 5G+ used and Ariane 5 GS and ES use an improved EPS upper stage.

Variants

- The original version is dubbed Ariane 5 **G** (Generic) with a launch mass of 737 tonnes. Its payload capability to geostationary transfer orbit (GTO) was initially specified as 5,970 kg (13,200 lb), but was increased after the qualification flights to 6,200 kg (14,000 lb).

- The Ariane 5 **G+** had an improved second stage, with a GTO capacity of 6,950 kg (15,300 lb) for a single payload. It flew three times in 2004.

- It was replaced in 2005 by the Ariane 5 **GS**, with the same solid EAP as the Ariane 5 ECA and a modified first Stage with a Vulcain 1B engine. It can carry a single payload of 6,100 kg (13,000 lb) to GTO.

- The Ariane 5 **ECA** (*Evolution Cryotechnique type A*) has a GTO launch capacity of 10,000 kg (22,000 lb) for dual payloads or 10,500 kg (23,000 lb) for a single payload. This variant uses a new Vulcain 2 first-stage engine, and an ESC-A (*Etage Supérieur Cryogénique*-A) second stage, powered by an HM-7B engine, weighing 2,100 kg (4,600 lb) and carrying 14,000 kg (31,000 lb) of cryogenic propellant. The second stage was previously used as the third stage of Ariane 4; in ECA use, the tanks are modified to shorten stage length. The revised Vulcain has a longer, more efficient nozzle with more efficient flow cycle and denser propellant ratio. The new ratio demanded length modifications to the first-stage tanks. Also, the solid EAP casings have been lightened with new welds, and packed with more propellant. The ESC-A cryogenic second stage does not improve the performance to Low Earth orbit compared to Ariane 5G, and for this reason the Ariane 5 ECA will not be used to launch the Automated Transfer Vehicle (ATV).

- The Ariane 5 **ES** (*Evolution Storable*) is used to launch the Automated Transfer Vehicle (ATV) into a 260 km circular low Earth orbit inclined at 51.6°. It includes all the performance improvements of Ariane 5 **ECA** on EPC (*Etage Principal Cryogénique*—main stage) and EAP (*Etage d'Accélération à Poudre*—solid rocket booster) stages while the second stage is the EPS (*Etage à Propergols Stockable*) used on Ariane 5 **GS** variants. It is estimated that the Ariane 5 **ES** can put up to 21,000 kg (46,000 lb) in LEO. The first such launch occurred at 04:03 GMT on 9 March 2008.

Comparable rockets: Delta IV · Atlas V · Chang Zheng 5 · Angara · Proton · Falcon 9 · H-IIB

Future developments

Ariane 5 ECB

Ariane 5 **ECB** was planned to have an ESC-B upper stage using a new Vinci expander cycle type engine. The GTO capacity was to increase to 12,000 kg, but ECB was put on hold due to budget cuts.

At an ESA conference (December 2005) in Berlin there was no decision to restart or cancel the program, meaning it is currently on hold. The Vinci engine, which is designed to power the Ariane 5 ECB upper stage, is still being developed, though at a slower pace. At the ESA's Council of Ministers 25–26 November 2008 there was an agreement for the funding of a modernized second stage. On December 21, 2009 EADS Astrium was awarded a €200 million contract to develop the ECB.

Unlike the HM7-B engine, which is not capable of engine restarting, the Vinci engine is capable to do as such for up to five times, allowing increased flexibility, from additional payload capacity to possible direct-to-geosynchronous orbit (GSO) missions. The first successful test firing of this campaign took place on 27 May 2010, while the first flight test of the Vinci engine is not expected until 2016 or 2017.

Solid propellant stage

Work on the Ariane 5 EAP motors have been continued in the Vega programme. The Vega 1st stage engine—the P80 engine—is a shorter derivation of the EAP. The P80 booster casing is made of filament wound graphite epoxy, much lighter than the current stainless steel casing. A new composite steerable nozzle has been developed while new thermal insulation material and a narrower throat improve the expansion ratio and subsequently the overall performance. Additionally, the nozzle now has electromechanical actuators which have replaced the heavier hydraulic ones used for thrust vector control.

These developments will probably later make their way back into the Ariane programme. The incorporation of the ESC-B with the improvements to the solid motor casing and an uprated Vulcain engine would deliver 27,000 kilograms (60,000 lb) to LEO. This would be developed for any lunar missions but the performance of such a design may not be possible if the higher Max-Q for the launch of this rocket poses a constraint on the mass delivered to orbit.

Notable launches

Launch of the 34th Ariane 5 at Kourou

Ariane 5's first test flight (Ariane 5 Flight 501) on 4 June 1996 failed, with the rocket self-destructing 37 seconds after launch because of a malfunction in the control software. A data conversion from 64-bit floating point value to 16-bit signed integer value to be stored in a variable representing horizontal bias caused a processor trap (operand error) because the floating point value was too large to be represented by a 16-bit signed integer. The software was originally written for the Ariane 4 where efficiency considerations (the computer running the software had an 80% maximum workload requirement) led to 4 variables being protected with a handler while 3 others, including the horizontal bias variable, were left unprotected because it was thought that they were "physically limited or that there was a large margin of error". The software, written in Ada, was included in the Ariane 5 through the reuse of an entire Ariane 4 subsystem despite the fact that the particular software containing the bug, which was just a part of the subsystem, was not required by the Ariane 5 because it has a different preparation sequence than the Ariane 4.

The second test flight (L502, on 30 October 1997) was a partial failure. The Vulcain nozzle caused a roll problem, leading to premature shutdown of the core stage. The upper stage operated successfully but could not reach the intended orbit.

A subsequent test flight (L503, on 21 October 1998) proved successful and the first commercial launch (L504) occurred on 10 December 1999 with the launch of the XMM-Newton X-ray observatory satellite.

Another partial failure occurred on 12 July 2001, with the delivery of two satellites into an incorrect orbit, at only half the height of the intended GTO. The ESA Artemis telecommunications satellite was able to reach its intended orbit on 31 January 2003, through the use of its experimental ion propulsion system.

The next launch did not occur until 1 March 2002, when the Envisat environmental satellite successfully reached an orbit 800 km above the Earth in the 11th launch. At 8111 kg, it was the heaviest single payload until the launch of the first ATV on March 9, 2008 (~9000 kg).

The first launch of the ECA variant on 11 December 2002 ended in failure when a main booster problem caused the rocket to veer off-course, forcing its self-destruction three minutes into the flight. Its payload of two communications satellites (Stentor and Hot Bird 7), valued at about EUR 630 million, was lost in the ocean. The fault was determined to have been caused by a leak in coolant pipes allowing the nozzle to overheat. After this failure, Arianespace SA delayed the expected January 2003 launch for the Rosetta mission to 26 February 2004, but this was again delayed to early March 2004 due to a minor fault in the foam that protects the cryogenic tanks on the Ariane 5. As of November 2010, the failure of the first ECA launch was the last failure of an Ariane 5; since then, all subsequent launches have been successful, with 40 consecutive successes that stretch back to 9 April 2003 with the launch of INSAT-3A and Galaxy 12 satellites.

On 27 September 2003 the last Ariane 5 G boosted three satellites (including the first European lunar probe, SMART-1), in Flight 162. On 18 July 2004 an Ariane 5 G+ boosted what was at the time the heaviest telecommunication satellite ever, Anik F2, weighing almost 6,000 kg.

The first successful launch of the Ariane 5 ECA took place on 12 February 2005. The payload consisted of the XTAR-EUR military communications satellite, a 'SLOSHSAT' small scientific satellite and a MaqSat B2 payload simulator. The launch had been originally scheduled for October 2004, but additional testing and the military requiring a launch at that time (of an Helios 2A observation satellite) delayed the attempt.

On 11 August 2005, the first Ariane 5GS (featuring the Ariane 5 ECA's improved solid motors) boosted Thaïcom-4/iPStar-1, the heaviest telecommunications satellite to date at 6,505 kg, into orbit.

On 16 November 2005, the third Ariane 5 ECA launch (the second successful ECA launch) took place. It carried a dual payload consisting of Spaceway-F2 for DirecTV and Telkom-2 for PT Telekomunikasi of Indonesia. This was the rocket's heaviest dual payload to date, at more than 8,000 kg.

On 27 May 2006, an Ariane 5 ECA rocket set a new commercial payload lifting record of 8.2 tonnes. The dual-payload consisted of the Thaicom 5 and Satmex 6 satellites.

On 4 May 2007 the Ariane 5 ECA set another new commercial record, lifting into transfer orbit the Astra 1L and Galaxy 17 communication satellites with a combined weight of 8.6 tonnes, and a total payload weight of 9.4 tonnes. This record was again broken by another Ariane 5 ECA, launching the Skynet 5B and Star One C1 satellites, on 11 November 2007. The total payload weight for this launch was 9,535 kg.

On 9 March 2008, the first Ariane 5 ES-ATV was launched to deliver the first ATV called *Jules Verne* to the International Space Station. The ATV was the heaviest payload ever launched by an European rocket, providing supplies to the space station with necessary propellant, water, air and dry cargo. This was the first operational Ariane mission which involved an engine restart in the upper stage. (The ES-ATV Aestus EPS upper stage was restartable while the ECA HM7-B engine was not.)

On 1 July 2009, an Ariane 5 ECA launched TerreStar-1, the largest commercial telecommunication satellite ever built.

On 28 October 2010, an Ariane 5 ECA launched Eutelsat's W3B (part of its W Series of satellites) and Broadcasting Satellite System Corporation (B-SAT)'s BSAT-3b satellites into orbit. However, the W3B satellite failed to operate shortly after the successful launch and was written off as a total loss due to an oxidizer leak in the satellite's main propulsion system. The BSAT-3b satellite, however, is operating normally.

Ariane 5 flights

Date & Time (UTC)	Flight (Vol)	Configuration	Serial number	Payload	Result	#
1996-06-04 12:34:06	V-89	5G	501	Cluster	Failure	1
1997-10-30 13:43:00	V-101	5G	502	MaqSat-H, TEAMSAT, MaqSat-B, YES	Partial failure	2
1998-10-21 16:37:21	V-112	5G	503	MaqSat 3, ARD	Success	3
1999-12-10 14:32:07	V-119	5G	504	XMM-Newton	Success	4
2000-03-21 23:28:19	V-128	5G	505	INSAT-3B, AsiaStar	Success	5
2000-09-14 22:54:07	V-130	5G	506	Astra 2B, GE-7	Success	6
2000-11-16 01:07:07	V-135	5G	507	PAS-1R, Amsat P3D, STRV 1C, STRV 1D	Success	7
2000-12-20 00:26:00	V-138	5G	508	Astra 2D, GE-8, LDREX	Success	8
2001-03-08 22:51:00	V-140	5G	509	Eurobird 1, BSAT-2a	Success	9
2001-07-12 22:58:00	V-142	5G	510	Artemis, BSAT-2b	Partial failure	10
2002-03-01 01:07:59	V-145	5G	511	Envisat	Success	11
2002-07-05 23:22:00	V-153	5G	512	Stellat 5, N-Star c	Success	12
2002-08-28 22:45:00	V-155	5G	513	Atlantic Bird 1, Meteosat 8	Success	13
2002-12-11 22:22:00	V-157	5ECA	517	Hot Bird 7, Stentor	Failure	14
2003-04-09 22:52:19	V-160	5G	514	INSAT-3A, Galaxy 12	Success	15
2003-06-11 22:38:15	V-161	5G	515	Optus and Defence C1, BSAT-2c	Success	16
2003-09-27 23:14:46	V-162	5G	516	INSAT-3E, eBird 1, SMART-1	Success	17
2004-03-02 07:17:44	V-158	5G+	518	Rosetta	Success	18
2004-07-18 00:44:00	V-163	5G+	519	Anik F2	Success	19

2004-12-18 16:26:00	V-165	5G+	520	Helios 2A, Essaim 1, 2, 3, 4, PARASOL, Nanosat 01	Success	20
2005-02-12 21:03:00	V-164	5ECA	521	XTAR-EUR, Maqsat-B2, Sloshsat	Success	21
2005-08-11 08:20:00	V-166	5GS	523	Thaicom 4	Success	22
2005-10-13 22:32:00	V-168	5GS	524	Syracuse 3A, Galaxy 15	Success	23
2005-11-16 23:46:00	V-167	5ECA	522	Spaceway F2, TELKOM-2	Success	24
2005-12-21 22:33:00	V-169	5GS	525	INSAT-4A, Meteosat 9	Success	25
2006-03-11 22:32:50	V-170	5ECA	527	Spainsat, Hot Bird 7A	Success	26
2006-05-26 21:09	V-171	5ECA	529	Satmex 6, Thaicom 5	Success	27
2006-08-11 22:15	V-172	5ECA	531	JCSAT-10, Syracuse 3B	Success	28
2006-10-13 20:56:00	V-173	5ECA	533	DirecTV-9S, Optus D1, LDREX-2	Success	29
2006-12-08 22:08:00	V-174	5ECA	534	WildBlue 1, AMC-18	Success	30
2007-03-11 22:03	V-175	5ECA	535	Skynet 5A, INSAT-4B	Success	31
2007-05-04 22:29	V-176	5ECA	536	Astra 1L, Galaxy 17	Success	32
2007-08-14 23:44	V-177	5ECA	537	Spaceway-3, BSAT-3A	Success	33
2007-10-05 22:02	V-178	5GS	526	Intelsat 11, Optus D2	Success	34
2007-11-14 22:06	V-179	5ECA	538	Skynet 5B, Star One C1	Success	35
2007-12-21 21:41	V-180	5GS	530	RASCOM-QAF 1, Horizons-2	Success	36
2008-03-09 04:03	V-181	5ES	528	ATV-1 "Jules Verne"	Success	37
2008-04-18 22:17	V-182	5ECA	539	Star One C2, Vinasat-1	Success	38
2008-06-12 22:05	V-183	5ECA	540	Turksat 3A, Skynet 5C	Success	39
2008-07-07	V-184	5ECA	541	Badr-6, ProtoStar I	Success	40

21:47						
2008-08-14 20:44	V-185	5ECA	542	AMC-21, Superbird 7	Success	41
2008-12-20 22:35	V-186	5ECA	543	Eutelsat W2M, Hot Bird 9	Success	42
2009-02-12 22:09	V-187	5ECA	545	Hot Bird 10 , NSS-9 , Spirale A , Spirale B	Success	43
2009-05-14 13:12	V-188	5ECA	546	Herschel, Planck	Success	44
2009-07-01 19:52	V-189	5ECA	547	TerreStar-1	Success	45
2009-08-21 22:09	V-190	5ECA	548	JCSAT-12, Optus D3	Success	46
2009-10-01 21:59	V-191	5ECA	549	Amazonas 2, COMSATBw-1	Success	47
2009-10-29 20:00	V-192	5ECA	550	NSS-12, Thor 6	Success	48
2009-12-18 16:26	V-193	5GS	532	Helios 2B	Success	49
2010-05-21 22:01	V-194	5ECA	551	Astra 3B, COMSATBw-2	Success	50
2010-06-26 21:41	V-195	5ECA	552	Arabsat-5A, COMS-1	Success	51
2010-08-04 20:59	V-196	5ECA	554	Nilesat 201, RASCOM-QAF 1R	Success	52
2010-10-28 21:51	V-197	5ECA	555	Eutelsat W3B, BSAT-3b	Success	53
2010-11-26 18:39	V-198	5ECA	556	Intelsat 17, HYLAS 1	Success	54
2010-12-29 21:27	V-199	5ECA	557	Koreasat 6 , HispaSat-1E	Success	55
2011-02-16 21:50	V-200	5ES	544	ATV-2 "Johannes Kepler"	Success	56

Chapter 10

Atlas V

Atlas V

Launch of the Mars Reconnaissance Orbiter, 11:43:00 am GMT August 12, 2005 on the first Atlas V rocket used by NASA. The rocket is in the 401 configuration.

Function	EELV/Medium-heavy launch vehicle
Manufacturer	United Launch Alliance
Country of origin	United States
Cost per launch (2011)	$187M, 401 configuration
Size	
Height	58.3 m (191.2 ft)
Diameter	3.81 m (12.49 ft)
Mass	334,500 kg (737,400 lb)
Stages	2
Capacity	

Payload to LEO	9,750–29,420 kg (21,490–64,860 lb)
Payload to GTO	4,750–13,000 kg (10,470–28,660 lb)
Launch history	
Status	Active
Launch sites	SLC-41, Cape Canaveral SLC-3E, Vandenberg AFB
Total launches	23 (**401:** 10, **411:** 2, **421:** 3, **431:** 2) (**501:** 2, **521:** 2, **531:** 1, **551:** 1)
Successes	22 (**401:** 9, **411:** 2, **421:** 3, **431:** 2) (**501:** 2, **521:** 2, **531:** 1, **551:** 1)
Partial failures	1 (**401**)
Maiden flight	**401:** 21 August 2002 **411:** 20 April 2006 **421:** 10 October 2007 **431:** 11 March 2005 **501:** 22 April 2010 **521:** 17 July 2003 **531:** 14 August 2010 **551:** 19 January 2006
Notable payloads	Mars Reconnaissance Orbiter New Horizons Lunar Reconnaissance Orbiter Solar Dynamics Observatory Boeing X-37B
Boosters (Not Heavy) - Aerojet	
№ boosters	1 to 5
Engines	1 Solid
Thrust	1,270 kN (285,500 lb_f)
Specific impulse	275 seconds
Burn time	94 seconds
Fuel	Solid
Boosters (Atlas V Heavy (5HX)) - Atlas CCB	
N^o boosters	2
Engines	1 RD-180
Thrust	4,152 kN (933,406 lb_f)
Specific impulse	311 seconds
Burn time	253 seconds
Fuel	RP-1/LOX
First stage - Atlas CCB	
Engines	1 RD-180
Thrust	4,152 kN (933,400 lb_f)
Specific impulse	311 seconds

Burn time	253 seconds
Fuel	RP-1/LOX
Second stage (Atlas V XX1) - Centaur	
Engines	1 RL10A
Thrust	99.2 kN (22,300 lb_f)
Specific impulse	451 seconds
Burn time	842 seconds
Fuel	LH2/LOX
Second stage (Atlas V XX2) - Centaur	
Engines	2 RL10A
Thrust	147 kN (41,600 lb_f)
Specific impulse	449 seconds
Burn time	421 seconds
Fuel	LH2/LOX

Atlas V is an active expendable launch system in the Atlas rocket family. Atlas V was formerly operated by Lockheed Martin, and is now operated by the Lockheed Martin-Boeing joint venture United Launch Alliance. Each Atlas V rocket uses a Russian-built RD-180 engine burning kerosene and liquid oxygen to power its first stage and an American-built RL10 engine burning liquid hydrogen and liquid oxygen to power its Centaur upper stage. The RD-180 engines are provided by RD AMROSS and the RL10 engines by Pratt & Whitney Rocketdyne. Some configurations also use strap-on booster rockets made by Aerojet. The payload fairings, which can be 4 or 5 meters in diameter and three lengths, are made by RUAG Space. The rocket is assembled in Decatur, Alabama; Harlingen, Texas; San Diego, California; and at United Launch Alliance's headquarters near Denver, Colorado.

In its 22 launches, from its maiden launch in August 2002 to August 2010, Atlas V has had a near-perfect success rate. On one flight, NRO L-30 on June 15, 2007, an upper-stage anomaly occurred when the engine in the vehicle's Centaur upper stage shut down early, leaving the payload—a pair of ocean surveillance satellites—in a lower than intended orbit. However, the customer, the National Reconnaissance Office, categorized the mission as a success. Atlas V has made 13 successful flights since the anomaly.

History

The Atlas V is the newest member of the Atlas family. Compared to the Atlas III vehicle, there are numerous changes. Compared to the Atlas II, it is a near-redesign. There was no Atlas IV.

1. The *"1.5 staging"* technique was dropped on the Atlas III, in favor of a more-advanced RD-180 engine.
2. The main-stage diameter increased from 10 feet to 12.5 feet. As with the Atlas III, the different mixture ratio of the engine called for a larger oxygen tank (relative to the fuel tank) compared to western engines and stages.

3. First-stage tanks no longer use stainless steel monocoque "balloon" construction. The tanks are isogrid aluminum and are stable when unpressurized.
4. Use of aluminum, with a higher thermal conductivity than stainless steel, requires insulation for the liquid oxygen. The tanks are covered in a polyurethane-based layer.
5. Accommodation points for parallel stages, both smaller solids and identical liquids, are built into first-stage structures.

The Atlas V was developed by Lockheed Martin Commercial Launch Services as part of the US Air Force Evolved Expendable Launch Vehicle (EELV) program. The term *expendable launch vehicle* means each vehicle is only used once. Launches are from Space Launch Complex 41 at Cape Canaveral Air Force Station and Space Launch Complex 3-E at Vandenberg Air Force Base. Lockheed Martin Commercial Launch Services continues to market the Atlas V to commercial customers worldwide.

Core stage of an Atlas V being raised to a vertical position

An Atlas V 551 with the New Horizons Deep Space Probe launches from Launch Pad 41 in Cape Canaveral

The MRO spacecraft launches on August 12, 2005 at 11:43 UTC to the Planet Mars on board an Atlas V 401 Launch Vehicle

X-37B OTV-1 (Orbital Test Vehicle) was the military spaceplane within the April 22, 2010 launch.

The first Atlas V was launched on August 21, 2002, and all subsequent launches have been successful except for the 2007 anomaly. The Atlas V family uses a single-stage Atlas main engine, the Russian RD-180 and the newly developed Common Core Booster (CCB) with up to five Aerojet made strap-on solid rocket boosters. The CCB is 12.5 ft (3.8 m) in diameter by 106.6 ft (32.5 m) long and uses 627,105 lb (284,450 kg) of liquid oxygen and RP-1 rocket fuel propellants. The booster operates for about four minutes, providing about 4 meganewtons (860,000 lbf) of thrust at start, the major part of this thrust, 4.152 meganewtons being provided by Russian RD-180 engine.

The Centaur upper stage uses a pressure stabilized propellant tank design and cryogenic propellants. The Centaur stage for Atlas V is stretched 5.5 ft (1.68 m) and is powered by either one or two Pratt & Whitney RL10A-4-2 engines, each engine developing a thrust of 99.2 kN (22,300 lbf). Operational and reliability upgrades are enabled with the RL10A-4-2 engine configuration. The inertial navigation unit (INU) located on the Centaur provides guidance and navigation for both Atlas and Centaur, and controls both

Atlas and Centaur tank pressures and propellant use. The Centaur engines are capable of multiple in-space starts, making possible insertion into low-earth parking orbit, followed by a coast period and then insertion into GTO. A subsequent third burn following a multi-hour coast can permit direct injection of payloads into geostationary orbit. As of 2006, the Centaur vehicle had the highest proportion of burnable propellant relative to total mass of any modern hydrogen upper stage and hence can deliver substantial payloads to a high energy state.

Many systems on the Atlas V have been the subject of upgrade and enhancement both prior to the first Atlas V flight and since that time. Work on a new Fault Tolerant INU (FTINU) started in 2001 to enhance mission reliability for Atlas vehicles by replacing the existing non-redundant navigation and computing equipment with a fault tolerant unit. The upgraded FTINU first flew in 2005, and in 2010 a follow-on order for more FTINU units was awarded.

On April 14, 2008, Atlas V lifted its heaviest payload to date into orbit—a 14,625-pound (6,634 kg) telecommunications satellite built by Space Systems/Loral.

Core stage of an Atlas V being raised to a vertical position (May 6, 2005)

Single engine Centaur upper stage of an Atlas V rocket (May 10, 2005)

Launch of Atlas V taking up the Mars Reconnaissance Orbiter, August 12, 2005

The MRO Spaceprobe lauches on August the 12. 2005 at 11:43 UTC to the Planet Mars onbord a Atlas V (Modell 401) Launch Vehicle.

Atltas V Launch Vehicle lifted of with the MRO Mars Orbiter

2007 valve anomaly

The only anomalous event in the use of the Atlas V launch system occurred June 15, 2007, when the engine in the Centaur upper stage of an Atlas V shut down early, leaving its payload–a pair of NRO L-30 ocean surveillance satellites–in a lower-than-intended orbit. The cause of the anomaly was traced to a leaky valve. Replacing the valve led to a delay in the next Atlas V launch.

Salvaged Piece

In May, 2010 a piece of the external fairing cover of an Atlas V washed up on Hilton Head Island, South Carolina. It can be viewed at the island's Coastal Discovery Museum, Honey Horn Plantation site.

Future developments

Human rating Atlas V

ULA is doing design and simulation work to human-rate the Atlas V for carrying passengers. ULA won a 2010 small contract of US$6.7 million in the first phase of the NASA Commercial Crew Development Program (CCDev) to develop an Emergency Detection System (EDS) for human-rating the Atlas V launch vehicle. As of February 2011, ULA "is still finishing up work on its $6.7-million award... In December ULA carried out a demonstration of its Emergency Detection System ... The company said it received an extension from NASA until April 2011 'to enable us to finish critical timing analyses tasks' for [the] fault coverage analysis work."

NASA solicited proposals for CCDev phase 2 in October 2010, under which ULA made a proposal for funding to "finish designing a key safety system for potential commercial crew launches on its Atlas and Delta rocket fleet." While NASA's goal is get astronauts to orbit by 2015, ULA President and CEO Michael Gass has stated "I think we need to stretch our goals to have commercial crew service operating by 2014" and has committed ULA to meet that schedule. Other than the addition of the Emergency Detection System, no major changes are expected to the Atlas V rocket, but ground infrastructure modifications are planned. The most likely candidate for the human-rating is the 402 configuration, with built-in redundancy in the dual-RL-10 engines on the Centaur upper stage and no solid rocket boosters.

Atlas V 401 rocket launches the MRO Spaceprobe to the Planet Mars. Cape Cenaveral. August 12. 2005 11:43 UTC

Atlas V 401 rocket launches the MRO Spaceprobe to the Planet Mars. Cape Cenaveral.
August 12. 2005 11:43 UTC

One of five Solid Rocket Booster on a Truck bevor it is attached to an Atlas V 551

The fith SRB before it is attached to the Atlas V 551 like the other four Booster

Atlas V SRB lifted up in the assembly hall for menting to the Atlas V 551

Atlas V HLV

The Atlas V HLV (Heavy Lift Vehicle) would use three Common Core Booster (CCB) stages strapped together to provide the capability necessary to lift 25 tonne payload to low Earth orbit. Approximately 95% of the hardware required for the Atlas HLV has already been flown on the Atlas V single core vehicles.

A report, prepared by RAND Corporation for the Office of the Secretary of Defense in 2006, stated that Lockheed Martin had decided not to develop an Atlas V heavy-lift vehicle (HLV). The report recommended for the Air Force and the National Reconnaissance Office to "determine the necessity of an EELV heavy-lift variant, including development of an Atlas V Heavy", and to "resolve the RD-180 issue, including coproduction, Stockpile, or U.S. development of an RD-180 replacement."

The lifting capability of the Atlas V HLV is roughly equivalent to the Delta IV Heavy. The latter utilizes RS-68 engines developed and produced domestically by Pratt & Whitney Rocketdyne.

As of February 2008, the Atlas V HLV configuration was available to customers 30 months from date of order.

GX rocket

The Atlas V Common Core Booster was to have been used as the first stage of the joint US-Japanese GX rocket, which was scheduled to make its maiden flight in 2012. GX launches would have been from the Atlas V launch complex at Vandenberg AFB, SLC-3E.

In December 2009, Japanese government decided to cancel the GX project. Development of the LNG propulsion system will continue for other projects.

Atlas Phase 2

With the merger of Boeing and Lockheed-Martin space operations into United Launch Alliance, the Atlas V program gained access to the tooling and processes for 5-meter-diameter stages, used on Delta IV. At 5 meters, a stage can accept dual RD-180 engines. The proposed heavy-lift vehicle is "Atlas Phase 2" or "PH2." An Atlas V PH2-Heavy (three 5 m stages in parallel; six RD-180s) would be in the class of a Shuttle-derived or Ares V heavy lifter, as considered in the Augustine Report. The Atlas PH2 HLV would launch a payload mass of approximately 70 metric tons into an orbit of 28.5 degree-inclination.

The fifth and last SRB is attaching to an Atlas V 551 to launch New Horizons

Atlas V 551 rocket prepared for launch

New Horizons encapsulation in the Atlas V 551 Payload fairing

Technicians install strips of the New Horizons mission decal on the Atlas V 551 fairing

Atlas V 551 with the New Horizons Pluto probe on Launch Pad 41 in Cape Canaveral

Versions

Each Atlas booster has a three-digit version designation that is determined by the features of the rocket. The first digit is the diameter (in meters) of the nosecone fairing, and is always either '4' or '5'. The second digit is the number of solid rocket boosters attached to the base of the rocket, and can number anywhere from '0' through '3' with the 4 m fairing and '0' through '5' with the 5 m fairing. The third digit is the number of engines on the Centaur stage, either '1' or '2'. As of 2009, only the single-engine Centaur (SEC) has been used, and no launches using a dual-engine Centaur (DEC) are currently planned.

For example, if the Atlas V version is *552*, this means that the fairing is five meters, has five solid rocket boosters, and has two Centaur engines. If the Atlas V version is *431*, this means that the fairing is four meters, has three solid rocket boosters, and has a single Centaur engine.

The Atlas V has two general payload fairing sizes. The classic 4-meter fairing, used since the Atlas II, comes in regular and slightly stretched versions, and Lockheed Martin introduced a 5.4-meter (4.57 meters usable) payload fairing developed and built by RUAG Space (former Oerlikon Space) in Switzerland. The RUAG fairing is a composite design and is based on flight proven hardware. Three configurations will be manufactured to support Atlas V. The short and medium length configurations will be used on the Atlas V 500 series. The long configuration will be used on the Atlas V-Heavy. The classic fairing covers only the payload, leaving the Centaur stage exposed to open air. The RUAG fairing encloses the Centaur stage as well as the payload.

An agreement between Lockheed and Bigelow Aerospace in September 2006 could lead to a human-rated version of the Atlas V to tap into the potential space tourism market.

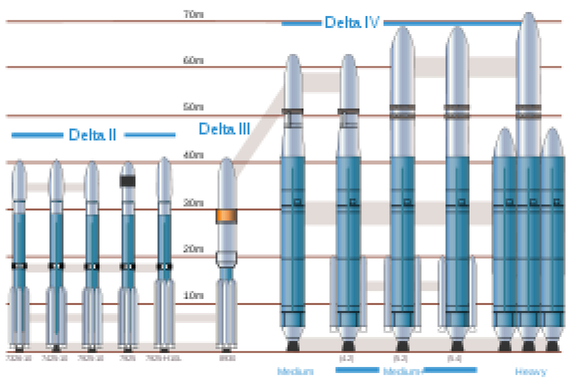
Versions: List Date: September 21, 2010

Version	Fairing	CCBs	SRBs	Upper stage	Payload to LEO	Payload to GTO	Launches to date
401	4 m	1	-	SEC	9,797 kg	4,750 kg	10
402	4 m	1	-	DEC	12,500 kg	-	0
411	4 m	1	1	SEC	12,150 kg	5,950 kg	2
421	4 m	1	2	SEC	14,067 kg	6,890 kg	3
431	4 m	1	3	SEC	15,718 kg	7,700 kg	2
501	5.4 m	1	-	SEC	8,123 kg	3,775 kg	2
502	5.4 m	1	-	DEC	10,300 kg	-	0
511	5.4 m	1	1	SEC	10,986 kg	5,250 kg	0
512	5.4 m	1	1	DEC	12,050 kg	-	0
521	5.4 m	1	2	SEC	13,490 kg	6,475 kg	2
522	5.4 m	1	2	DEC	13,950 kg	-	0
531	5.4 m	1	3	SEC	15,575 kg	7,475 kg	1
532	5.4 m	1	3	DEC	17,250 kg	-	0
541	5.4 m	1	4	SEC	17,443 kg	8,290 kg	0
542	5.4 m	1	4	DEC	18,750 kg	-	0
551	5.4 m	1	5	SEC	18,814 kg	8,900 kg	1
552	5.4 m	1	5	DEC	20,050 kg	-	0
Heavy (HLV/5H1))	5.4 m	3	-	SEC	-	13,605 kg	0
Heavy (HLV DEC/5H2))	5.4 m	3	-	DEC	29,400 kg	-	0

Chapter 11

Delta (Rocket Family)

Delta Family



The Delta rocket family.

Role	Expendable launch system with various applications
Manufacturer	United Launch Alliance
Introduced	1960
Status	active

Delta is a versatile family of expendable launch systems that has provided space launch capability in the United States since 1960. There have been more than 300 Delta rockets launched, with a 95 percent success rate. Two Delta launch systems – Delta II and Delta IV – are in active use. Delta rockets are currently manufactured and launched by the United Launch Alliance.

Delta origins

Delta rocket on display at the Goddard Space Flight Center in Maryland

The original Delta rockets used a modified version of the PGM-17 Thor, the first ballistic missile deployed by the United States Air Force, as their first stage. The Thor had been designed in the mid-1950s to reach Moscow from bases in Britain or similar allied nations, and the first wholly-successful Thor launch had occurred in September 1957. Subsequent satellite and space probe flights soon followed, using a Thor first stage with several different upper stages. The fourth upper stage used on the Thor was the Thor "Delta," delta being the fourth letter of the Greek alphabet. Eventually the entire Thor-Delta launch vehicle came to be called simply, "Delta."

NASA intended Delta as "an interim general purpose vehicle" to be "used for communication, meteorological, and scientific satellites and lunar probes during '60 and '61". The plan was to replace Delta with other rocket designs when they came on-line. The Delta design emphasized reliability rather than performance by replacing components which had caused problems on earlier Thor flights. NASA let the original Delta contract to the Douglas Aircraft Company in April 1959 for 12 vehicles of this design:

- Stage 1: Modified Thor IRBM with a Block I MB-3 engine producing 152,000 lbf (680 kN) thrust. (LOX/RP1 turbopump, gimbal mounted engine, two verniers for roll control)
- Stage 2: Modified Able. Pressure fed UDMH/nitric acid powered Aerojet AJ-10-118 engine producing 7,700 lbf (34 kN). This reliable engine cost $4 million to build and is still flying in modified form today. Gas jet attitude control system.
- Stage 3: Altair. A spin stabilized (via a turntable on top of the Able) at 100 rpm by two solid rocket motors before separation. One ABL X-248 solid rocket motor provided 2,800 lbf (12 kN) of thrust for 28 seconds. The stage weighed 500 pounds (230 kg) and was largely constructed of wound fiberglass.

These vehicles would be able to place 650 pounds (290 kg) into a 150 to 230 miles (240 to 370 km) LEO or 100 pounds (45 kg) into GTO. Eleven of the twelve initial Delta flights were successful. The total project development and launch cost came to $43 million, $3 million over budget. An order for 14 more vehicles was let before 1962.

Early Delta flights

No.	Date	Payload	Site	Outcome	Remarks
1	May 13, 1960	Echo 1	CCAFS LC 17A	failure	Launch at 9:16 p.m. GMT. Good first stage. Second stage attitude control system failure. Vehicle destroyed.
2	August 12, 1960	Echo 1A		success	Payload placed into 1,035 miles (1,666 km), 47 degree inclination orbit.
3	November 23, 1960	TIROS-2		success	

4	March 25, 1961	Explorer-10		success	78 pounds (35 kg) payload placed into elliptical 138,000 miles (222,000 km) orbit.
5	July 12, 1961	TIROS-3		success	
6	August 16, 1961	Explorer-12		success	Energetic Particle Explorers. EPE-A. Highly elliptical orbit.
7	February 8, 1962	TIROS-4		success	
8	March 7, 1962	OSO-1		success	Orbiting Solar Observatory. 345 miles (555 km), 33 degree orbit.
9	April 26, 1962	Ariel 1		success	Ariel 1 was later seriously damaged by the Starfish Prime nuclear test.
10	June 19, 1962	TIROS-5		success	
11	July 10, 1962	Telstar 1		success	Also later damaged by the Starfish Prime high altitude nuclear event.
12	September 18, 1962	TIROS-6		success	

Delta evolution

Launch of the first Skynet satellite by Delta rocket in 1969 from Cape Canaveral

Delta A

Block II MB-3 engine, 170,000 lbf (760 kN) vs. 152,000 lbf (680 kN)

13. EPE2
14. EPE3

Delta B

- Upgraded AJ10-118D upper stage—3-foot tank stretch, higher energy oxidizer, solid-state guidance system
- Delta program goes from 'interim' to 'operational' status.
- 200 pounds (91 kg) to GTO

15. 13 December 1962. *Relay 1*, second NASA communications satellite, NASA's first active one
16. 13 February 1963. pad 17b. *Syncom 1*; Thiokol Star 13B solid rocket as apogee kick motor
20. July 26, 1963. Syncom 2; heosynchronous orbit, but inclined 33° due to the limited performance of the Delta

Delta C

- Third stage Altair replaced with Altair 2—its engine having been developed as the ABL X-258 for the Scout vehicle; 3 in (76 mm) longer, 10% heavier, but 65% more total thrust

  Sample mission: OSO-4

Delta D

- Also known as Thrust Augmented Delta
- A Delta C with the Thrust Augmented Thor core plus three Castor 1 boosters

25. 19 August 1964. Syncom 3, the first geostationary communications satellite
30. 6 April 1965. Intelsat I

Delta E

- Also known as Thrust Augmented Improved Delta
- 1965
- 100 pounds (45 kg) more to GTO than Delta D
- Castor 2 vs. Castor 1 boosters; Same thrust, longer duration
- MB-3 Block III core engine, 2,000 lbf (8.9 kN) more thrust
- AJ10-118E second stage widened from 2.75 feet (0.84 m) to 4.58 feet (1.40 m) diameter; Double burn time
- Additional helium tanks allow for almost unlimited restarts.
- Two available third stages: Altair 2 or FW-4D; the latter caused the Delta to be known as a Delta E1
- New payload fairing from Agena

First Delta E. 6 November 1965; Launched GEOS 1

Delta G

- Two stage Delta Es.
- used for Biosatellite 1 and 2 flights

1. 14 December 1966. Biosatellite 1
2. 7 September 1967. Biosatellite 2

Delta J

- Used larger Thiokol Star 37D motor as third stage

4 July 1968; Explorer 38

Delta L

- Introduced Extended Long Tank first stage- 8 feet (2.4 m) diameter throughout
- FW-4d motor for third stage

Delta M

- Star 37D for stage 3

Delta N

- Two stage version of Delta M.
- There were nine Delta N launch attempts from 1968 until 1972; eight were successful.

'Super Six'

- Delta M or Delta N with three extra strap-ons
- 1,000 pounds (450 kg) to GTO

Launch reliability

From 1969 through 1978 (inclusive), Thor-Delta was NASA's most popular launcher, with 84 launch attempts. (Scout was the second-most used vehicle with 32 launches.) NASA used it to launch its own satellites, and also to launch satellites for other government agencies and foreign governments on a cost-reimbursable basis. Sixty-three of the satellites NASA attempted to launch were provided by other parties. Out of the 84 attempts there were seven failures or partial failures (91.6% successful).

Delta numbering system

In 1972, McDonnell Douglas introduced a four-digit numbering system to replace the letter-naming system. The new system could better accommodate the various changes and improvements to Delta rockets (and avoided the problem of a rapidly depleting alphabet). It specified (1) the tank and main engine type, (2) number of solid boosters, (3) second stage, and (4) third stage.

<table>
<tr><th>Number</th><th>First Digit (First stage/boosters)</th><th>Second Digit (Number of boosters)</th><th>Third Digit (Second Stage)</th><th>Fourth Digit (Third stage)</th><th>Letter (Heavy configuration)</th></tr>
<tr><td>0</td><td>Long Tank Thor
MB-3 engine
Castor 2 SRBs</td><td>No SRBs</td><td>Delta, with AJ-10 engines</td><td>No third stage</td><td rowspan="6">N/A</td></tr>
<tr><td>1</td><td>Extended Long Tank Thor
MB-3 engine
Castor 2 SRBs</td><td>N/A</td><td>Delta, with TR-201 engines</td><td>N/A</td></tr>
<tr><td>2</td><td>Extended Long Tank Thor
RS-27 engine
Castor 2 SRBs</td><td>2 SRBs (or LRBs in the case of the Delta IVH)</td><td>Delta K, with AJ-10 engines</td><td>FW-4D (unflown)</td></tr>
<tr><td>3</td><td>Extended Long Tank Thor
RS-27 engine
Castor 4 SRBs</td><td>3 SRBs</td><td>Delta III cryogenic upper stage, RL-10B-2 engine</td><td>Star 37D</td></tr>
<tr><td>4</td><td>Extended Long Tank Thor
MB-3 engine
Castor 4A SRBs</td><td>4 SRBs</td><td>Delta IV 4m diameter cryogenic upper stage, RL-10B-2 engine</td><td>Star 37E</td></tr>
<tr><td>5</td><td>Extended Long Tank Thor</td><td>N/A</td><td>Delta IV 5m diameter</td><td>Star 48B/PAM-</td></tr>
</table>

	RS-27 engine Castor 4A SRBs		cryogenic upper stage, RL-10B-2 engine	D	
6	Extra-Extended Long Tank Thor RS-27 engine Castor 4A SRBs	6 SRBs	N/A	Star 37FM	
7	Extra-Extended Long Tank Thor RS-27A engine GEM 40 SRBs	N/A		N/A	GEM 46 SRBs
8	Strengthened Extra-Extended Long Tank Thor RS-27A engine GEM 46 SRBs				N/A
9	Delta IV CBC RS-68 engine	9 SRBs			2 additional CBC Parallel first stages

This numbering system was to have been phased out in favor of a new system that was introduced in 2005. In practice, this system was never been used.

Number	First Digit (First stage/boosters)	Second Digit (Number of boosters)	Third Digit (Second Stage)	Fourth Digit (Third stage)	Letter (Heavy configuration)
0	N/A	No SRBs	N/A	No third stage	N/A
1		N/A		N/A	
2	Extra-Extended Long Tank Thor RS-27A engine	2 SRBs (or LRBs in the case of the	Delta K, with AJ-10 engines		GEM 46 SRBs

<table>
<tr><td></td><td>GEM 40 SRBs</td><td>Delta IVH)</td><td></td><td rowspan="3"></td><td></td></tr>
<tr><td>3</td><td>Strengthened Extra-Extended Long Tank Thor RS-27A engine GEM 46 SRBs</td><td>3 SRBs</td><td>N/A</td><td></td></tr>
<tr><td>4</td><td>Delta IV CBC RS-68 engine</td><td>4 SRBs</td><td>Delta IV 4m diameter cryogenic upper stage, RL-10B-2 engine</td><td>2 additional CBC Parallel first stages</td></tr>
<tr><td>5</td><td rowspan="5">N/A</td><td rowspan="4">N/A</td><td>Delta IV 5m diameter cryogenic upper stage, RL-10B-2 engine</td><td>Star 48B/PAM-D</td><td rowspan="5">N/A</td></tr>
<tr><td>6</td><td rowspan="4">N/A</td><td>Star 37FM</td></tr>
<tr><td>7</td><td rowspan="3">N/A</td></tr>
<tr><td>8</td></tr>
<tr><td>9</td><td>9 SRBs</td></tr>
</table>

Delta 904

On July 23, 1972, the launch of Landsat 1 marked the first use of nine strap-on boosters, and the new uprated second-stage engine (AJ 10-118F). This Thor-Delta model was designated the 904.

Delta 1000-Series

- Extended Long Tank with 8-foot-diameter (2.4 m) payload fairing; nicknamed "Straight-Eight".

- Nine Castor II strap-on solid boosters.
- The first successful 1000 series Thor-Delta launched Explorer 47 on September 22, 1972.

Delta 2000-Series

- Features new Rocketdyne RS-27 main engine on Extended Long Tank. Same constant eight-foot diameter.
- Delta 2910 boosters were used to launch both Landsat 2 in 1975 and Landsat 3 in 1978.
- A Delta 2914 was used 1978-04-07 to launch the Japanese BSE Broadcasting Satellite, also known as "Yuri 1".

Delta 3000-Series

- Introduced upgraded Castor IV solid motors. Same first stage as 1000- and 2000-series.
- Also introduced PAM (Payload Assist Module)/Star 48B solid-fueled kick motor. Later used as Delta II third stage.
- The Delta 3914 model was approved for launching U.S. government payloads in May 1976.

Delta 4000-Series

- Used old MB-3 main engine on Extended Long Tank with Castor IV motors.
- Only launched two missions.
- First use of a Delta-K second stage.

Delta 5000-Series

- Featured upgraded Castor IVA motors on Extended Long Tank first stage with RS-27 main engine.
- Only launched one mission.

Delta II series

The Delta II series consists of the retired Delta 6000, the active Delta 7000, and two variants (Lite and Heavy) of the latter.

Delta 6000-Series

When in 1986 the *Challenger* accident demonstrated that Delta launches would continue, the Delta II was developed.

- Introduced Extra Extended Long Tank first stage. 12 additional feet provide more propellant.

- Introduced Castor IVA boosters. Six ignite at takeoff, three ignite in flight.

Delta 7000-Series

- Introduces RS-27A main engine, modified for efficiency at high altitude, at some cost to low-altitude performance.
- Introduces GEM-40 (Graphite-Epoxy Motor) solid boosters from Hercules (now Alliant). Besides being longer, their lighter casings allow higher payload capability.

Delta II Med-Lite

A 7000-series with no third stage and fewer strap-ons (often three, sometimes four). Usually used for small NASA missions.

Delta II Heavy

A Delta II 792X with the enlarged GEM-46 boosters from Delta III.

Delta III (8000-Series)

A McDonnell Douglas/Boeing-developed program to keep pace with growing satellite masses:

- The two upper stages, with low-performance fuels, were replaced with a single cryogenic stage, improving performance and reducing recurring costs and pad labor. Engine was a single Pratt & Whitney RL10, from the Centaur upper stage. The hydrogen fuel tank, 4 meters in diameter in orange insulation, is exposed; the narrower oxygen tank and engine are covered until stage ignition. Fuel tank contracted to Mitsubishi, and produced using technologies from Japanese H-II launcher.
- To keep the stack short and resistant to crosswinds, the first-stage kerosene tank was widened and shortened, matching the upper-stage and fairing diameters.
- Nine enlarged GEM-46 solid boosters attached. Three have thrust-vectoring nozzles.

Of the three Delta III flights, the first two were failures and the third carried only a dummy (inert) payload.

Delta IV (9000-series)

As part of the Air Force's EELV (Evolved Expendable Launch Vehicle) program, McDonnell Douglas/Boeing proposed Delta IV. As the program implies, many components and technologies were borrowed from existing launchers. Both Boeing and

Lockheed Martin were contracted to produce their EELV designs. Delta IVs are produced in a new facility in Decatur, Alabama.

- First stage changed to liquid hydrogen fuel. Tank technologies derived from Delta III upper stage, but widened to 5 meters.
- Kerosene engine replaced with Rocketdyne RS-68, the first new, large liquid-fueled rocket engine designed in the US since the Space Shuttle Main Engine (SSME) in the '70s. Designed for low cost; has lower chamber pressure and efficiency than the SSME, and a much simpler nozzle. Thrust chamber and upper nozzle is a channel-wall design, pioneered by Soviet engines. Lower nozzle is ablatively cooled.
- Second stage and fairing taken from the Delta III in smaller (Delta IV Medium) models; widened to 5 meters in Medium+ and Heavy models.
- Medium+ models have two or four GEM-60 60-inch diameter solid boosters.
- Revised plumbing and electric circuits eliminate need for a launch tower.

The first stage is referred to as a common booster core (CBC); a Delta IV Heavy attaches two extra CBCs as boosters.

Future development

Currently development is focused on the Delta IV Heavy, which uses three Common Booster Cores to lift higher masses to orbit and escape velocity.

Chapter 12

Delta II

Delta II

A Delta II rocket launches from Cape Canaveral carrying the Dawn spacecraft.

Function	Launch vehicle
Manufacturer	United Launch Alliance (Boeing IDS)
Country of origin	United States
Cost per launch (1987)	US$36.7 million
Size	
Height	38.2 - 39 m (125.3 - 127 ft)
Diameter	2.44 m (8 ft)
Mass	151,700 - 231,870 kg (334,300 - 511,180 lb)

Stages	2 or 3
Capacity	
Payload to LEO	2,700 - 6,100 kg (5,960 - 13,440 lb)
Payload to GTO	900 - 2,170 kg (1,980 - 4,790 lb)
Payload to HCO	1,000 kg (2,200 lb)
Launch history	
Status	Active
Launch sites	Cape Canaveral SLC-17 Vandenberg AFB SLC-2W
Total launches	148 **Delta 6000:** 17 **Delta 7000:** 126 **Delta 7000H:** 5
Successes	146 **Delta 6000:** 17 **Delta 7000:** 124 **Delta 7000H:** 5
Failures	1 (Delta 7000)
Partial failures	1 (Delta 7000)
Maiden flight	**Delta 6000:** 14 February 1989 **Delta 7000:** 26 November 1990 **Delta 7000H:** 8 July 2003
Last flight	**Delta 6000:** 24 July 1992
Boosters (6000 Series) - Castor 4A	
№ boosters	3, 4 or 9
Engines	1 Solid
Thrust	478.3 kN (107,530 lb_f)
Specific impulse	266 sec
Burn time	56 seconds
Fuel	Solid
Boosters (7000 Series) - GEM 40	
N^o boosters	3, 4 or 9
Engines	1 Solid
Thrust	492.9 kN (110,800 lb_f)
Specific impulse	274 sec
Burn time	64 seconds
Fuel	solid
Boosters (7000 Heavy) - GEM 46	
N^o boosters	9
Engines	1 solid
Thrust	628.3 kN (141,250 lb_f)
Specific impulse	278 sec

Burn time	75 seconds
Fuel	solid
First stage - Thor/Delta XLT-C	
Engines	1 RS-27A
Thrust	1,054.2 kN (237,000 lb_f)
Specific impulse	302 sec
Burn time	265 seconds
Fuel	RP-1/LOX
Second stage - Delta K	
Engines	1 AJ-10
Thrust	43.6 kN (9,800 lb_f)
Specific impulse	319 sec
Burn time	431 seconds
Fuel	Dinitrogen tetroxide/Aerozine
Third stage - PAM-D (optional)	
Engines	1 Star 48B
Thrust	66.0 kN (14,837 lb_f)
Specific impulse	286 sec
Burn time	87 seconds
Fuel	Solid

Delta II is an American space launch system originally designed and built by McDonnell Douglas. Delta II is part of the Delta rocket family and has been in service since 1989. Delta II vehicles include the retired Delta 6000, the active Delta 7000, and two 7000 variants (light and heavy).

Delta II rockets were later built by Boeing Integrated Defense Systems until Delta rocket production became the responsibility of United Launch Alliance (ULA) on December 1, 2006. ULA now markets Delta II to U.S. government customers, and Boeing Launch Services (BLS) markets Delta II to commercial companies.

History

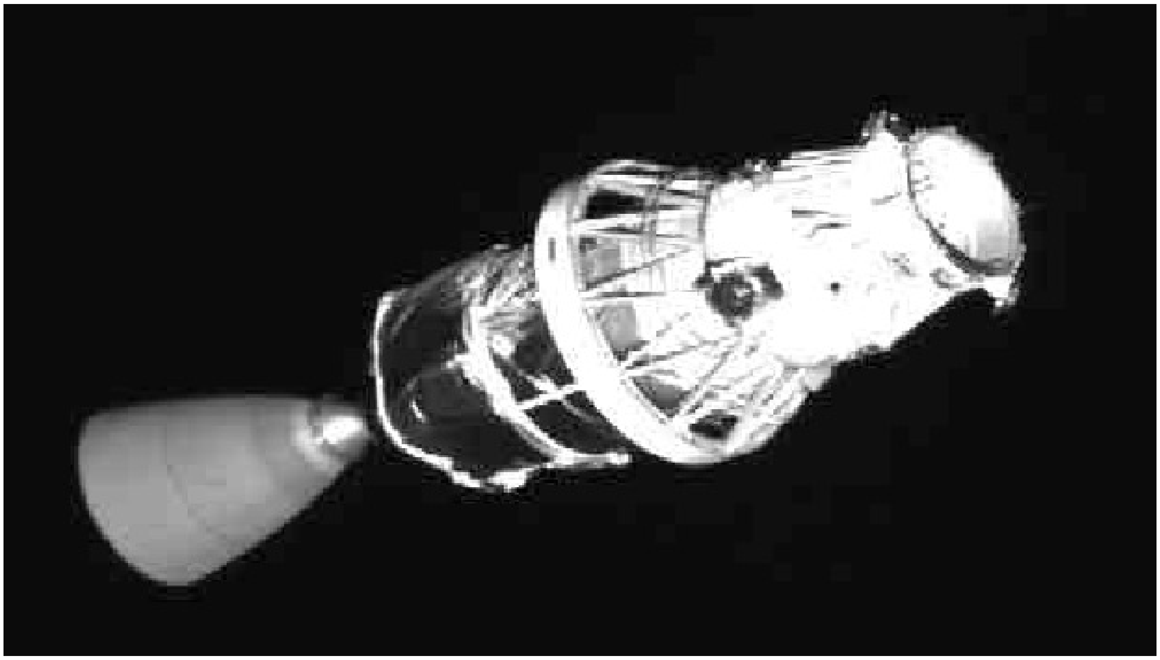
All United States expendable launch vehicles were to be phased out for the Space Shuttle, but in 1986 the *Challenger* accident restarted Delta development. The Delta II was specifically designed to accommodate the GPS Block II series of satellites. Delta IIs have successfully launched 125 projects (through August 2007), including several NASA missions to Mars:

- Mars Global Surveyor in 1996
- Mars Pathfinder in 1996
- Mars Climate Orbiter in 1998
- Mars Polar Lander in 1999
- Mars Odyssey in 2001

- Mars Exploration Rovers (MER-A, *Spirit* and MER-B, *Opportunity*) in 2003
- Mars Phoenix lander in 2007

Delta II manufacturing, assembly and integration currently take place in Decatur, Alabama; Harlingen, Texas; San Diego, California; and Denver, Colorado.

Vehicle description



A spent Delta-K second stage of the Delta II, photographed in orbit

Deltas are expendable launch vehicles (ELVs), which means they are used only once. Each Delta II launch vehicle consists of:

- Stage I: RP-1 and liquid oxygen tanks that feed the Rocketdyne RS-27 main engine for the ascent.
- Solid rocket booster motors: Used to increase thrust during the initial two minutes of flight. The medium-capacity Delta II has nine motors total (six fire on the ground, three in flight); the other models use only three or four.
- Interstage: A spacer between stage I and stage II. The first friction stir welded Interstage module has been launched in 1999.
- Stage II: Fuel and oxidizer tanks feeding a restartable, hypergolic Aerojet AJ10-118K engine that fires one or more times to insert the vehicle-spacecraft stack into low Earth orbit. This propellant mixture is highly corrosive and once loaded the launch must occur within approximately 37 days or the stage will have to be refurbished or replaced. This stage also contains the vehicle's "brains", a combined inertial platform and guidance system that controls all flight events.
- Stage III: Optional ATK-Thiokol solid rocket motor (some Delta II vehicles are two-stage only, and generally used for Earth-orbit missions) provides the majority

of the velocity change needed to leave Earth orbit and inject the spacecraft on a trajectory to Mars or other target beyond Earth orbit. It is connected to the spacecraft until it is done firing, and then separates. This stage is spin-stabilized and has no active guidance control; it depends on the second stage for proper orientation prior to Stage II/III separation. It also includes a yo-yo de-spin mechanism to slow the spin before spacecraft release, as many spacecraft cannot handle the high spin rates needed for stability of this stage.

- Payload fairing: Thin metal or composite payload fairing (aka "nose cone") to protect the spacecraft during the ascent through Earth's atmosphere.

Naming system

The Delta II family is more technically named by a four-digit system:

- The **first digit** is either 6 or 7, denoting the 6000- or 7000-series Deltas. The 6000-series, last flown in 1992, had an Extra Extended Long Tank first stage with RS-27 main engine, plus Castor IVA solid rocket boosters. The current model 7000-series have an RS-27A engine, with a longer nozzle for higher expansion ratio and better high-altitude performance, and GEM (Graphite-Epoxy Motor) boosters. GEMs are larger, and have a composite casing to reduce mass versus the steel-case Castors. In addition, two LR101-NA-11 vernier engines provide guidance for the first stage.

- The **second digit** indicates the number of boosters, usually 9. In such cases, six are lit at liftoff and three are lit one minute into flight. On vehicles with 3 or 4 boosters, all are ignited at liftoff.

- The **third digit** is 2, denoting a second stage with an Aerojet AJ10 engine. This engine is restartable, for complex missions. Only Deltas prior to the 6000-series used a different engine, the TR-201.

- The **last digit** denotes the third stage. 0 denotes no third stage, 5 indicates a Payload Assist Module (PAM) stage with Star 48B solid motor, 6 indicates a Star 37FM motor.

For example, a Delta 7925 has the later first stage, nine GEM boosters, and a PAM third stage. A Delta 7320 is a two-stage vehicle with three boosters.

- A Delta II-Heavy has the larger GEM-46 boosters, originally designed for the Delta III. These are designated 79*xx*H.

Three payload fairings are available. The original aluminum fairing, seen above, is 9.5 feet in diameter. A 10-foot fairing is made of composite, and can be distinguished by its tapering front and rear. A lengthened 10-foot fairing is used for the largest payloads.

Launch description

Launch vehicle build-up

A Delta II launch vehicle is assembled vertically on the launch pad. Assembly starts by hoisting the first stage into position. The solid rocket boosters are then hoisted into position and mated with the first stage. Launch vehicle build-up then continues with the second stage being hoisted atop the first stage.

Fueling

It takes approximately 20 minutes to load the first stage with 10,000 US gallons (37,900 L) of fuel.

Delta II launches

The Delta II system has been used for 146 launches. On September 18, 2007, Delta II completed its 75th consecutive successful launch. This is a record for modern launch vehicles. It is the most reliable launch vehicle currently in service, behind the (now retired) Tsyklon 2. Eight launches took place in 2007.

However, the Delta II system does not have a perfect success record. One mission, the launch of Koreasat-1, was a partial failure in which the satellite payload was able to compensate when the launch system placed the vehicle in an incorrect orbit.

Another failure, this time complete, occurred on January 17, 1997, when a Delta II 7925 carrying the first GPS Block IIR satellite, GPS IIR-1, exploded only 13 seconds after liftoff, raining flaming debris all over Launch Complex 17 at Cape Canaveral Air Force Station. No one was injured, and the launchpad itself was not seriously damaged, though several cars were destroyed and a few buildings were damaged. It was later determined that a "17-foot crack" in the rocket booster had caused the failure.

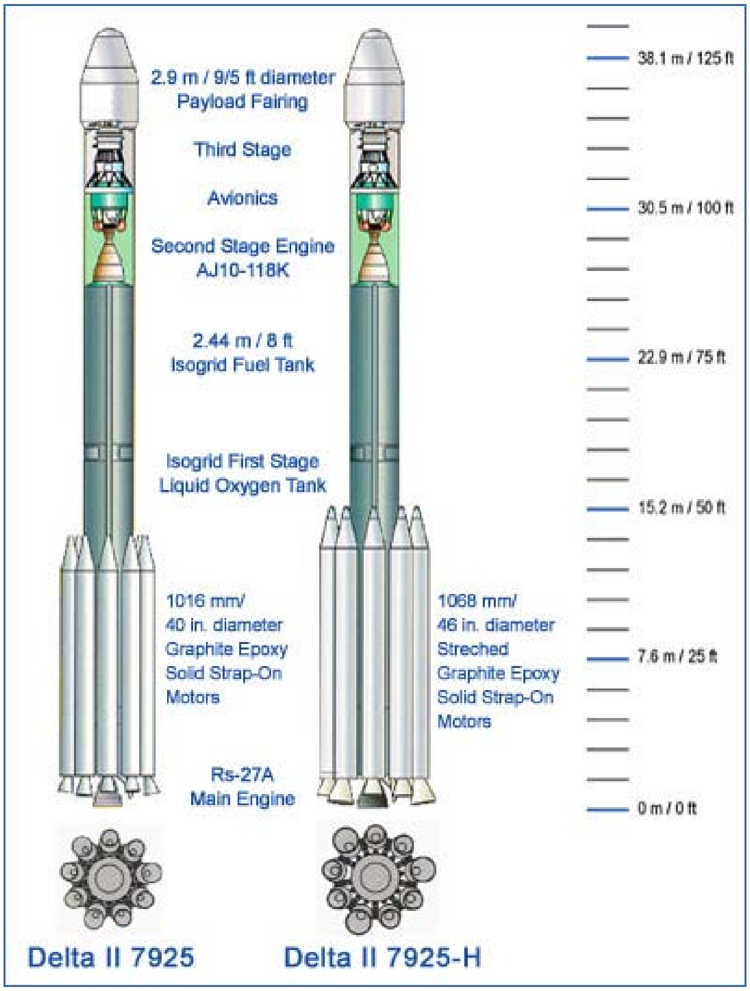
Notable payloads

- 2001 Mars Odyssey
- CONTOUR
- Dawn
- *Deep Impact*
- Deep Space 1
- GLAST
- Iridium
- Mars Climate Orbiter
- Mars Exploration Rovers
- Mars Global Surveyor
- Mars Pathfinder
- Mars Phoenix
- Mars Polar Lander
- MESSENGER
- NEAR
- Jason-2

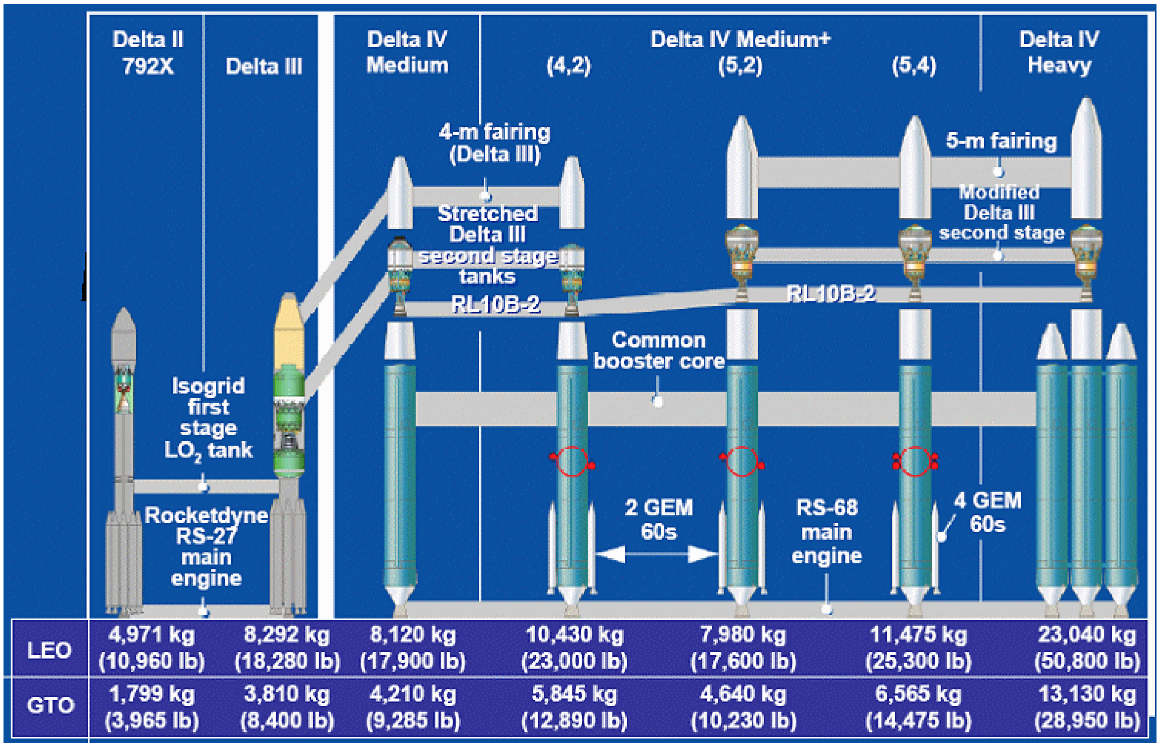
- Polar (satellite)
- ROSAT
- Spitzer Space Telescope (SIRTF)
- STEREO
- Swift
- THEMIS
- USA 193 (NROL-21)
- WIND
- WISE
- Kepler
- STSS-ATRR
- Genesis (spacecraft)

Between May 1997 and November 1998 Delta II vehicles placed 55 Iridium satellites into orbit.

Retirement of system



Comparison of standard vs. heavy Delta II



Delta rocket evolution

An article published by the *Wall Street Journal* speculates about the fate of the Delta II launch system after U.S. Air Force discontinues its use of the Delta II. Thomas Young, who was director of Goddard Space Flight Center from 1980 to 1982, is quoted as saying, "It's definitely an item people are quite worried about."

The final payload currently scheduled for Delta II is a NASA moon mission in 2011. ULA has indicated it has "around half a dozen" unsold Delta II rockets on hand. A spokesperson indicated ULA will change some aspects of the Delta II system once the current Medium Launch Vehicle 3 contract with the Air Force ends and requirements imposed by the contract are lifted. The Air Force contract required that Delta II be kept ready to launch within 40 days of call up, which led ULA to maintain two launch pads at Cape Canaveral. ULA indicated it would not continue to operate two launch pads.

In August 2009 the NASA assistant associate administrator for launch services indicated NASA might purchase additional Delta II launches beyond those it currently has planned. Seven Delta II flights are planned through 2011. Five additional Delta II vehicles have been built but remain unassigned to planned flights.

Comparable rockets

- Ariane 4 (retired)
- Falcon 9
- GSLV

- Soyuz-U
- Taurus II (whose development was started to fill the gap left by the retirement of the Delta II)

Chapter 13

Delta IV

Delta IV (Delta 9000)	
	
Delta IV Medium launch carrying DSCS III-B6	
Function	Orbital launch vehicle
Manufacturer	Boeing IDS United Launch Alliance
Country of origin	United States
Size	
Height	63 - 72 m (206 - 235 ft)
Diameter	5 m (16.4 ft)
Mass	249,500 - 733,400 kg (550,000 - 1,616,800 lb)
Stages	2

Capacity	
Payload to LEO	8,600 - 22,560 kg (18,900 - 49,740 lb)
Payload to GTO	3,900 - 12,980 kg (8,500 - 28,620 lb)
Launch history	
Status	Active
Launch sites	SLC-37B, Cape Canaveral SLC-6, Vandenberg AFB
Total launches	15 **Medium:** 3 **Medium+ (4,2):** 6 **Medium+ (5,4):** 1 **Heavy:** 5
Successes	14 **Medium:** 3 **Medium+ (4,2):** 6 **Medium+ (5,4):** 1 **Heavy:** 4
Partial failures	1 (**Heavy**)
Maiden flight	**Medium:** 11 March 2003 **Medium+ (4,2):** 20 November 2002 **Medium+ (5,4):** 6 December 2009 **Heavy:** 21 December 2004
Boosters (Medium+ Variants) - GEM 60	
№ boosters	Medium: 0; M+4,2: 2; M+5: 2 or 4
Engines	1 Solid
Thrust	826.6 kN (185,817 lb_f)
Specific impulse	275 sec
Burn time	90 seconds
Fuel	Solid
Boosters (Heavy) - CBC	
N^o boosters	2
Engines	1 RS-68
Thrust	3,312.8 kN (744,737 lb_f)
Specific impulse	410 sec
Burn time	249 seconds
Fuel	LH2/LOX
First stage - CBC	
Engines	1 RS-68
Thrust	3,312.8 kN (744,737 lb_f)
Specific impulse	410 sec
Burn time	259 seconds
Fuel	LH2/LOX
Second stage	
Engines	1 RL10-B-2

Thrust	110 kN (24,740 lb_f)
Specific impulse	462 sec
Burn time	850 - 1,125 seconds
Fuel	LH2/LOX

Delta IV is an active expendable launch system in the Delta rocket family. Delta IV uses rockets designed by Boeing's Integrated Defense Systems division and built in the United Launch Alliance (ULA) facility in Decatur, Alabama. Final assembly is completed at the launch site by ULA. The rockets were designed to launch payloads into orbit for the United States Air Force Evolved Expendable Launch Vehicle (EELV) program and commercial satellite business. Delta IV rockets are available in five versions: Medium, Medium+ (4,2), Medium+ (5,2), Medium+ (5,4), and Heavy, which are tailored to suit specific payload size and weight ranges. Delta IV was primarily designed to satisfy the needs of the U.S. military.

The rockets are assembled at the Horizontal Integration Facility for launches from SLC-37B at Cape Canaveral, and in a similar facility for launches from SLC-6 at Vandenberg AFB.

History

The Delta IV entered the space launch market at a period when global capacity was already much higher than demand. Furthermore, as an unproven design it has had difficulty finding a market in commercial launches, and the cost to launch a Delta IV is somewhat higher than that for competing vehicles. In 2003, Boeing pulled the Delta IV from the commercial market, citing low demand and high costs. In 2005, Boeing stated that it may return the Delta IV to commercial service; however as of 2006 no further announcements have been made regarding this. All but one of the first launches have been paid for by the U.S. Government, with a cost of between $140 million and $170 million.

Comparable rockets: Atlas V - Ariane 5 - Chang Zheng 5 - Angara - H-IIB - Proton - Falcon 9

Vehicle description

The first stage of a Delta IV consists of one, or in the Heavy variety three, Common Booster Cores (CBC) powered by a Rocketdyne RS-68 engine. Unlike many first-stage rocket engines, which use solid fuel or kerosene, the RS-68 engines burn liquid hydrogen and liquid oxygen.

In 2002 the RS-68 became the first large, liquid-fueled rocket engine designed in the U.S. since the Space Shuttle Main Engine (SSME) in the 1970s. The primary goal for the RS-68 was to reduce cost versus the SSME. Some sacrifice in chamber pressure and specific impulse was made, hurting efficiency; however, development time, part count, total cost, and assembly labor were reduced to a fraction of the SSME, despite the RS-68's significantly larger size. Typically, the RS-68 runs at 102% rated thrust for the first few minutes of flight, and then throttles down to 58% rated thrust before main engine cutoff. On the Heavy variant, the core CBC's engine throttles down to 58% rated thrust around 50 seconds after liftoff, while the strap-on CBCs remain at 102%. This allows the core CBC to conserve propellant and burn longer. After the strap-on CBCs separate, the core CBC's engine throttles back up to 102% before throttling back down to 58% prior to main engine cutoff.

The RS-68 engine is mounted to the lower thrust structure of the vehicle by a four-legged (quadrapod) thrust frame, and enclosed in a protective composite conical thermal shield. Above the thrust structure is an aluminum isogrid (a grid pattern machined out of the inside of the tank to reduce weight) liquid hydrogen tank, followed by a composite cylinder called the centerbody, an aluminum isogrid liquid oxygen tank, and a forward skirt. Along the back of the CBC is a cable tunnel to hold electrical and signal lines, and a tube to carry the liquid oxygen to the RS-68 from the tank. The CBC is of a constant, 5-meter, diameter.

The L-3 Communications Redundant Inertial Flight Control Assembly (RIFCA) guidance system used on the Delta IV is common to that carried on the Delta II, although the software is different because of the differences between the Delta II and Delta IV. The RIFCA features six ring laser gyroscopes and accelerometers each, to provide a higher degree of reliability.

Delta Cryogenic Second Stage

The upper stage of the Delta IV, or DCSS, is nearly identical to that of the Delta III, however the tanks are friction stir welded and either stretched (in 4-meter variants), or have a larger diameter (5-meter variants). The second stage is powered by a Pratt & Whitney RL-10B2 engine, which features an extendable carbon-carbon nozzle to improve specific impulse. Depending on variant, two different interstages are used to mate the first and second stages. A tapering interstage which narrows down from 5-meters to 4-meters in diameter is used on 4-meter variants, where a cylindrical interstage is used on 5-meter variants. Both interstages are built from composites.

To encapsulate the satellite payload, a variety of different payload fairings are available. A stretched Delta III 4-meter composite payload fairing is used on 4-meter variants, where an enlarged, 5-meter composite fairing is used on 5-meter variants. A longer version of the latter is standard on the Heavy variant, and a Boeing-built Titan-IV derived 5-meter, aluminum isogrid payload fairing is also available for the Heavy.

At over 63 meters in length, the Delta IV has been the tallest rocket in active use since its introduction.

Variants

Delta IV Small

During the Delta IV's development, a Small variant was considered. This would have featured the Delta II second stage, an optional Thiokol Star 48B third stage, and the Delta II payload fairing, all atop a single CBC. The Small variant was dropped by 1999. This was probably because the Delta II has a similar payload capability.

Delta IV Medium

The Delta IV Medium (Delta 9040) is the most basic Delta IV. It features a single CBC and a modified Delta III second stage, with 4-meter liquid hydrogen and liquid oxygen tanks and a 4-meter payload fairing derived from the Delta III fairing. The Delta IV Medium is capable of launching 4,210 kg (9,285 lb) to geosynchronous transfer orbit (GTO).

The Delta IV Medium+ (4,2) (Delta 9240) is similar to the Medium, but uses two Alliant-built 1.5-m (60-in) diameter solid rocket strap-on Graphite-Epoxy Motors (GEM-60s) to increase payload capacity to 5,845 kg (12,890 lb) to GTO.

The Delta IV Medium+ (5,2) (Delta 9250) is similar to the Medium+ (4,2), but has a 5-m–diameter payload fairing for larger payloads and a modified second stage with a 5-meter liquid hydrogen tank and stretched liquid oxygen tank. Because of the extra weight of the larger payload fairing and second stage, the Medium+ (5,2) can launch 4,640 kg (10,230 lb) to GTO, less than the Medium+ (4,2).

The Delta IV Medium+ (5,4) (Delta 9450) is similar to the Medium+ (5,2), but uses four GEM-60s instead of two, enabling it to lift 6,565 kg (14,475 lb) to GTO.

Delta IV Heavy

The Delta IV Heavy (Delta 9250H) is similar to the Medium+ (5,2), except that it uses two additional CBCs instead of using GEMs. These are strap-on boosters which are separated earlier in the flight than the center CBC. The Delta IV Heavy also features a stretched 5-meter composite payload fairing. An aluminum trisector (3 part) fairing derived from the Titan IV fairing is also available. This was first used on the DSP-23 flight.

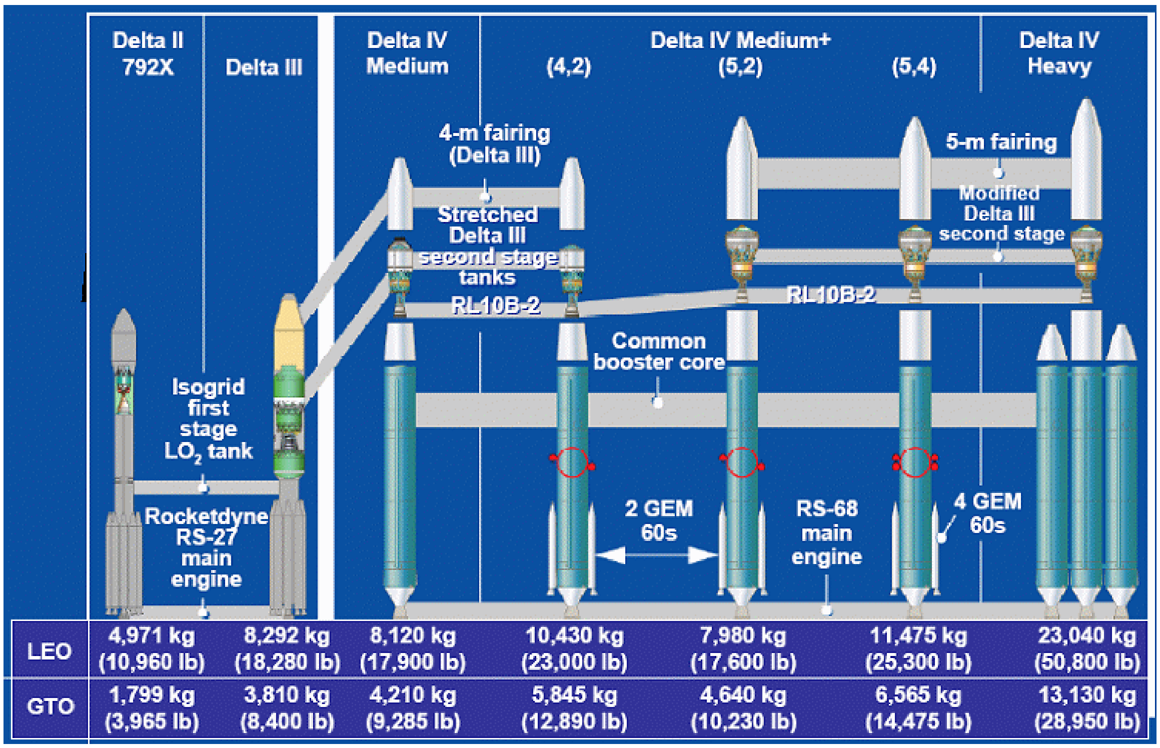
Capacity (separated spacecraft mass) of the Delta IV Heavy:

- geosynchronous transfer orbit (GTO) 13,130 kg (28,950 lb), more than any other currently available launch vehicle.
- geosynchronous orbit (GEO) 6,275 kg
- escape orbit 9,306 kg
- C_3 performance of 30 $km^2 s^{-2}$: 5,228 kg

- C_3 performance of 60 km²s^{-2}: 2,521 kg

The Heavy's total mass at launch is approximately 733,000 kg, much less than that of the Space Shuttle (2,040,000 kg).

Future variants



Delta IV evolution (US Govt)

Possible future upgrades for the Delta IV include adding extra strap-on solid motors to boost capacity, higher-thrust main engines, lighter materials, higher-thrust second stages, more (up to six) strap-on CBCs, and a cryogenic propellant cross feed from strap on boosters to the common core. These modifications could potentially increase the mass of the payload delivered to LEO to 100 tonnes.

At one point NASA had plans to use Delta IV to launch a Crew Exploration Vehicle. But with the change of the CEV from a winged or lifting body spacecraft to an Apollo-like capsule and a new launch vehicle based on Space Shuttle components, the only component from the Delta IV that NASA would adopt is the RS-68 engine that would be used to power the new cryogenically-fueled Ares V rocket.

In 2009 The Aerospace Corporation reported to NASA results of study intended to determine the feasibility of modifying Delta IV to be human-rated for use in NASA human spaceflight missions. According to Aviation Week the study, "found that a Delta IV heavy [...] could meet NASA's requirements for getting humans to low Earth orbit."

The possibility of an extra-heavy variant was indicated in a 2006 RAND Corporation study of national security launch requirements out to 2020, which noted, "...only the Delta IV Heavy has the performance to lift the ten NSS launch requirements that require a heavy-lift capability... the production capacity for Delta IV, with one possible exception, can satisfy the entire projected NSS launch demand. The exception involves the requirement to increase the Delta IV Heavy lift capability to accommodate a single NRO (National Reconnaissance Office) payload. The best solution to this requirement is currently under study."

An upgrade of the Delta IV Heavy, using the higher-performance RS-68A engine, is under development with initial availability in early 2011. This upgrade is planned to provide a roughly 13% improvement in payload capability to GTO. The new RS-68A is also planned to be used throughout the entire Delta IV family, where at 106% thrust it will provide a roughly 7–11% improvement in GTO payload (although this higher power level may require structural changes; running the engine at the current 102% produces a smaller improvement but requires less modification).

Another possible upgrade to the Delta IV family is the creation of new variants by the addition of extra solid motors. One such modification, the Medium+(4,4), would pair the four GEM-60s of the M+(5,4) with the upper stage and fairing of the (4,2). This would theoretically provide a GTO payload of 7,500 kg (16,600 lb) and an LEO payload of 14,800 kg (32,700 lb). This is the simplest variant to implement and is available within 36 months of the first order. Two other possible versions, the Medium+(5,6) and (5,8), would add two or four extra GEM-60s to the (5,4) variant, respectively. These would provide significantly higher performance (up to 9,200 kg/20,200 lb to GTO for the M+(5,8)) but would require more extensive modifications to the vehicle, such as adding the extra attach points and changes to cope with the different flight loads. They would also require pad and infrastructure changes. The Medium+(5,6) and (5,8) can be available within 48 months of the first order.

Launch sites

First Delta IV Heavy with three CBCs prior to launch

Delta IV launches occur from either of two rocket launch sites. On the East coast of the United States, Space Launch Complex 37 (SLC-37) at the Cape Canaveral Air Force Station. On the West coast, polar-orbit and high-inclination launches use Vandenberg Air Force Base's Space Launch Complex 6 (SLC-6) pad.

Launch facilities at both sites are similar. At the pad is a Mobile Service Tower (MST), which provides service access to the rocket and protection from the weather. There is a crane at the top of the MST, which allows the payload and GEM-60 solid motors to be attached to the vehicle. The MST is rolled away from the rocket several hours before launch. At Vandenberg, the launch pad also has a Mobile Assembly Shelter (MAS), which completely encloses the vehicle; at CCAFS, the vehicle is partly exposed near its bottom.

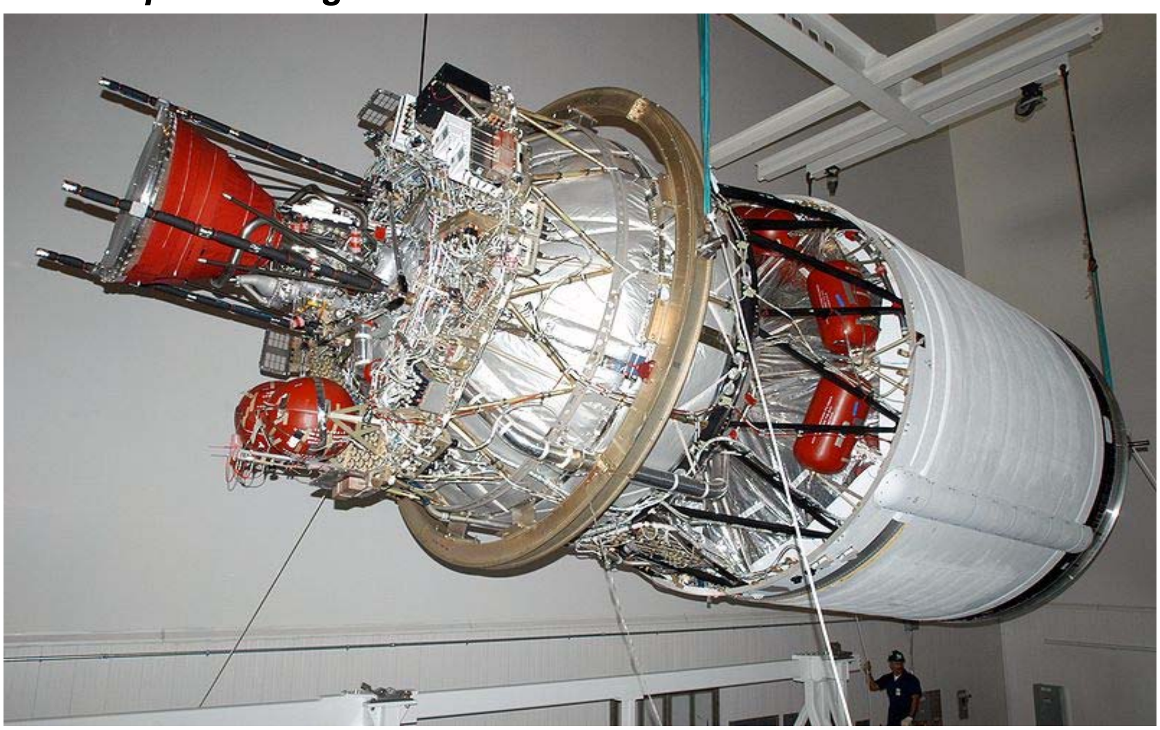
Beside the vehicle is a Fixed Umbilical Tower (FUT), which has two (VAFB) or three (CCAFS) swing arms. These arms carry electrical, hydraulic, environmental control, and other support functions to the vehicle through umbilical lines. The swing arms retract at T-0 seconds to prevent them from hitting the vehicle.

Under the vehicle is a Launch Table, with six Tail Service Masts (TSMs), two for each CBC. The Launch Table supports the vehicle on the pad, and the TSMs provide further support and fueling functions for the CBCs. The vehicle is mounted to the Launch Table by a Launch Mate Unit (LMU), which is attached to the vehicle by bolts that sever at launch. Behind the Launch Table is a Fixed Pad Erector (FPE), which uses two long-stroke hydraulic pistons to raise the vehicle to the vertical position after being rolled to the pad from the Horizontal Integration Facility (HIF). Beneath the Launch Table is a flame duct, which deflects the rocket's exhaust away from the rocket or facilities.

The Horizontal Integration Facility (HIF) is situated some distance from the pad. It is a large building that allows the Delta IV CBCs and second stages to be mated and tested before they are moved to the pad. The horizontal rocket assembly of the Delta IV are similar to the ones with the assembly of Soyuz launch vehicles; they are also assembled horizontally, unlike the current Space Shuttles, the past Saturn launch vehicles and the upcoming Ares I and Ares V, where they are assembled and rolled out to the launch pad entirely vertically.

Movement of the Delta IVs among the various facilities at the pad is facilitated by Elevating Platform Transporters (EPTs). These rubber-tired vehicles can be powered by either diesel engines or electric power. Diesel EPTs are used for moving the vehicles from the HIF to the pad, while electric EPTs are used in the HIF, where precision of movement is important.

Vehicle processing



Delta IV 4-Meter Second Stage

The Delta IVs are assembled using a process that Boeing claims reduces cost and expensive on-pad time. The CBCs are built in Boeing's factory in Decatur, Alabama. They are then loaded onto the M/V *Delta Mariner*, a roll-on/roll-off cargo vessel, and shipped to either launch pad. There, they are offloaded and rolled into a Horizontal Integration Facility (HIF), where they are mated with the second stages, which were shipped separately to the pad on the *Delta Mariner*. Also, in the HIF, the three CBCs of Heavy variant are mated to each other.

Various tests are performed, and then the vehicle is rolled horizontally to the pad, where the Fixed Pad Erector (FPE) is used to raise the vehicle to the vertical position, inside the MST. At this time, the GEM-60 solid motors, if any are required, are rolled to the pad and attached to the vehicle. After further testing, the payload (which has already been enclosed in its fairing) is transported to the pad, hoisted into the MST by a crane, and attached to the vehicle. Finally, on launch day, the MST is rolled away from the vehicle, and the vehicle is ready for launch.

History

Recent history

The United States Air Force (USAF) continues to fund Delta IV engineering, integration and infrastructure through contracts with Boeing Launch Services (BLS). On August 8, 2008 the USAF Space and Missile Systems Center increased the "cost plus award fee" contract with BLS for $1,656 million to extend the period of performance through the end of FY09. In addition a $557.1 million option was added to cover FY10.

Delta IV launches

List Date: January 21, 2011

No.	Date/Time (UTC)	Type	Serial-no.	Startplace	Payload	Type of payload	Orbit	Outcome	Remarks
1	2002-11-20 22:39	Medium+(4,2)	293	CCAFS SLC-37B	Eutelsat W5	Commercial communications satellite	GTO	Success	First Delta IV launch
2	2003-03-11 00:59	Medium	296	CCAFS SLC-37B	USA-167 (DSCS-3 A3)	Military communications satellite	GTO	Success	First Delta IV Medium launch First USAF EELV mission
3	2003-08-29 23:13	Medium	301	CCAFS SLC-37B	USA-170 (DSCS-3 B6)	Military communications satellite	GTO	Success	
4	2004-12-21 21:50	Heavy	310	CCAFS SLC-37B	DemoSat / 3CS-1 / 3CS-2	Demonstration payload	GSO (planned)	*Partial failure*	• First Delta IV Heavy launch • Premature cutoff of CBCs • DemoSat reached incorrect orbit, 3CS failed to reach orbit

5	2006-05-24 22:11	Medium+(4,2)	315	CCAFS SLC-37B	GOES 13 (GOES-N)	Weather satellite	GTO	Success	
6	2006-06-28 03:33	Medium+(4,2)	317	VAFB SLC-6	USA-184 (NROL-22)	Reconnaissance satellite	Molniya	Success	First Delta IV launch from Vandenberg
7	2006-11-04 13:53	Medium	320	VAFB SLC-6	USA-192 (DMSP F17)	Military weather satellite	SSO	Success	First Delta IV launch into a LEO/SSO
8	2007-11-11 01:50	Heavy	329	CCAFS SLC-37B	USA-197 (DSP-23)	Missile Warning satellite	GSO	Success	First Delta IV launch contracted by United Launch Alliance Launch delayed due to damage to launch pad caused by a liquid oxygen leak
9	2009-01-18 02:47	Heavy	337	CCAFS SLC-37B	USA-202 (NROL-26)	Reconnaissance satellite	GSO	Success	
10	2009-06-27 22:51	Medium+(4,2)	342	CCAFS SLC-37B	GOES 14 (GOES-O)	Weather satellite	GTO	Success	
11	2009-12-06 01:47	Medium+(5,4)	346	CCAFS SLC-37B	USA-211 (WGS-3)	Military communications satellite	GTO	Success	First Delta IV Medium+ (5,4) launch
12	2010-03-04 23:57	Medium+(4,2)	348	CCAFS SLC-37B	GOES 15 (GOES-P)	Weather satellite	GTO	Success	
13	2010-05-28 03:00	Medium+(4,2)	349	CCAFS SLC-37B	USA-213 (GPS IIF SV-1)	Navigation Satellite	MEO	Success	
14	2010-11-21 22:58	Heavy	351	CCAFS SLC-37B	USA-223 (NROL-32)	Reconnaissance satellite	*GSO*	Success	
15	2011-01-20 21:10	Heavy	352	VAFB SLC-6	USA-224 (NROL-49)	Reconnaissance satellite	*LEO*	Success	First Delta IV Heavy launch from Vandenberg

For 2010 and planned launches, see:
List of Thor and Delta launches (2010–2019)

Notable past launches

GOES-N launch on a Medium+ (4,2)

A unique aerial view of NROL-22 launch from SLC-6

The first payload launched with a Delta IV was the *Eutelsat W5* communications satellite. The launch vehicle was a Medium+ (4,2) variant, launched from Cape Canaveral. It carried the communications satellite into geostationary transfer orbit (GTO) on November 20, 2002.

Heavy Demo was the first launch of the Heavy variant in December 2004 after significant delays due to bad weather. Due to cavitation in the propellant lines, sensors registered depletion of propellant. The strap-on, and later core CBC engines shut down prematurely, even though sufficient propellant remained to continue the burn as scheduled. The second stage attempted to compensate for the under-burn, until it ran out of propellant. This flight was a test launch carrying a payload of:

- *DemoSat* – 6020 kg; an aluminum cylinder filled with 60 brass rods – planned to be carried to GEO; however due to the sensor faults, the satellite did not reach this orbit.
- *NanoSat-2*, carried to low Earth orbit (LEO) – a set of two very small satellites of 24 and 21 kg, nicknamed *Sparky* and *Ralphie* – planned to orbit for one day. Given the under-burn, the two most likely did not reach a stable orbit.

NROL-22 was the first Delta IV launched from SLC-6 at Vandenberg Air Force Base (VAFB). It was launched aboard a Medium+ (4,2) in June 2006 carrying a classified satellite for the U.S. National Reconnaissance Office (NRO).

DSP-23 was the first launch of a valuable payload aboard a Heavy vehicle. This was also the first Delta IV launch contracted by the United Launch Alliance, a joint venture between Boeing and Lockheed Martin. The main payload was the 23rd and final Defense Support Program missile-warning satellite, DSP-23. Launch from Cape Canaveral occurred at 01:50:00 GMT on November 11, 2007 (20:50 EST, November 10, 2007).

NROL-26 was the first "heavy" EELV launch for the NRO. It carried USA 202, a classified reconnaissance satellite, on a Delta IV Heavy that lifted off January 18, 2009 at 02:47 UTC.

NROL-32 was a "Heavy" launch, carrying a satellite for NRO. The payload is speculated to be the largest satellite sent into space. The rocket lifted off on November 21, 2010 at 22:58 UTC; the launch was delayed from October 19.

NROL-49 lifted off at 1:10 p.m. PST from Vandenberg AFB on January 20, 2011. NROL-49 was the first Delta IV Heavy mission to be launched out of Vandenberg. This mission is an NRO mission and its details are classified.

Planned launches

- Several GPS Block IIF satellites will be launched using Medium+ (4,2) rockets (as well as Atlas V 401 rockets). The first of these was launched in May 2010.