Solid-Fuel Rockets & Rocket Engines

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Chapter 1

Solid-Fuel Rocket

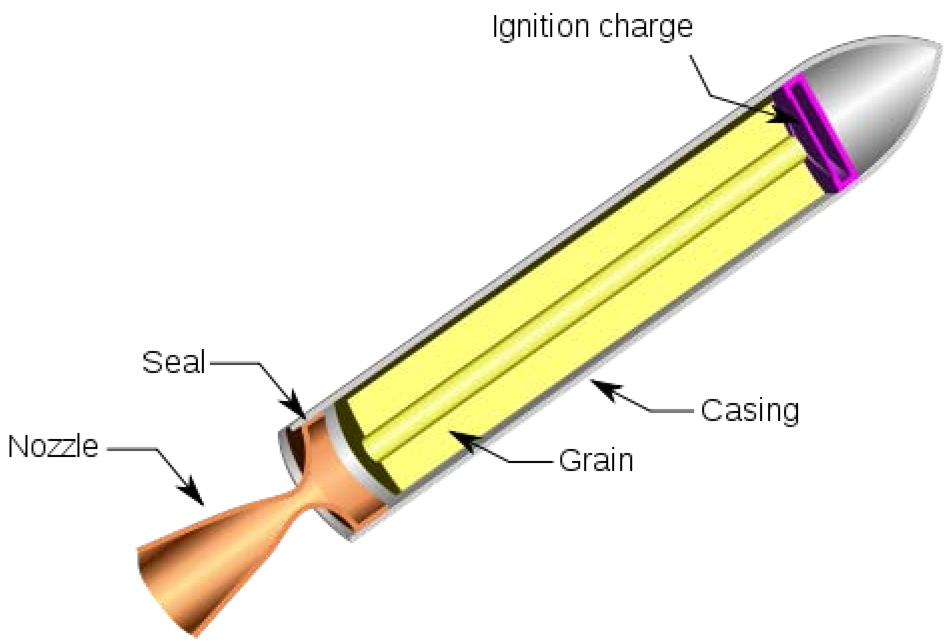
The Space Shuttle is launched with the help of two solid-fuel boosters known as SRBs

A **solid rocket** or a **solid-fuel rocket** is a rocket with a motor that uses solid propellants (fuel/oxidizer). The earliest rockets were solid-fuel rockets powered by gunpowder; they were used by the Indians, Chinese, Mongols and Arabs, in warfare as early as the 13th century.

All rockets used some form of solid or powdered propellant up until the 20th century, when liquid rockets and hybrid rockets offered more efficient and controllable alternatives. Solid rockets are still used today in model rockets and on larger applications for their simplicity and reliability.

Since solid-fuel rockets can remain in storage for long periods, and then reliably launch on short notice, they have been frequently used in military applications such as missiles. The lower performance of solid propellants (as compared to liquids) does not favor their use as primary propulsion in modern medium-to-large launch vehicles customarily used to orbit commercial satellites and launch major space probes. Solids are, however, frequently used as strap-on boosters to increase payload capacity or as spin-stabilized add-on upper stages when higher-than-normal velocities are required. Solid rockets *are* used as light launch vehicles for low Earth orbit (LEO) payloads under 2 tons or escape payloads up to 1000 pounds.

Basic concepts



Solid Rocket Motor.

A simple solid rocket motor consists of a casing, nozzle, grain (propellant charge), and igniter.

The grain behaves like a solid mass, burning in a predictable fashion and producing exhaust gases. The nozzle dimensions are calculated to maintain a design chamber pressure, while producing thrust from the exhaust gases.

Once ignited, a simple solid rocket motor cannot be shut off, because it contains all the ingredients necessary for combustion within the chamber in which they are burned. More advanced solid rocket motors can not only be throttled but also be extinguished and then re-ignited by controlling the nozzle geometry or through the use of vent ports. Also, pulsed rocket motors that burn in segments and that can be ignited upon command are available.

Modern designs may also include a steerable nozzle for guidance, avionics, recovery hardware (parachutes), self-destruct mechanisms, APUs, controllable tactical motors, controllable divert and attitude control motors, and thermal management materials.

Design

Design begins with the total impulse required, which determines the fuel/oxidizer mass. Grain geometry and chemistry are then chosen to satisfy the required motor characteristics.

The following are chosen or solved simultaneously. The results are exact dimensions for grain, nozzle, and case geometries:

- The grain burns at a predictable rate, given its surface area and chamber pressure.
- The chamber pressure is determined by the nozzle orifice diameter and grain burn rate.
- Allowable chamber pressure is a function of casing design.
- The length of burn time is determined by the grain 'web thickness'.

The grain may or may not be bonded to the casing. Case-bonded motors are much more difficult to design, since the deformation, under operating conditions, of the case and the grain must be compatible.

Common modes of failure in solid rocket motors include fracture of the grain, failure of case bonding, and air pockets in the grain. All of these produce an instantaneous increase in burn surface area and a corresponding increase in exhaust gas and pressure, which may potentially induce rupture of the casing.

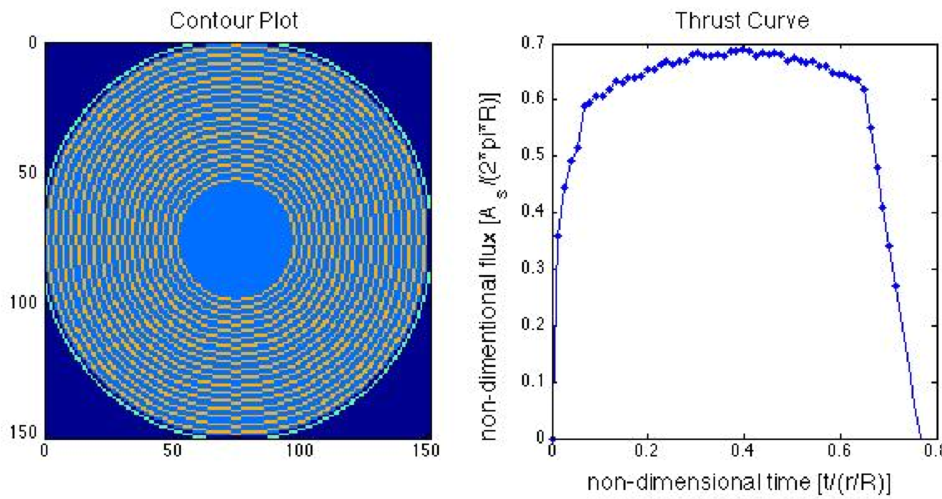
Another failure mode is casing seal design. Seals are required in casings that have to be opened to load the grain. Once a seal fails, hot gas will erode the escape path and result in failure. This was the cause of the Space Shuttle *Challenger* disaster.

Grain geometry

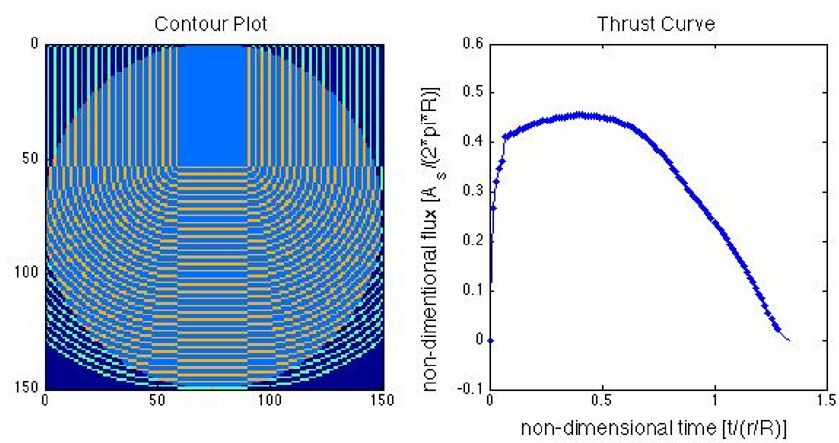
Solid rocket fuel deflagrates from the surface of exposed propellant in the combustion chamber. In this fashion, the geometry of the propellant inside the rocket motor plays an important role in the overall motor performance. As the surface of the propellant burns, the shape evolves (a subject of study in internal ballistics), most often changing the propellant surface area exposed to the combustion gases. The mass flux (kg/sec) [and, therefore, pressure] of combustion gases generated is a function of the instantaneous surface area A_s, (m^2), and linear burn rate b_r(m/sec):

$$\dot{m} = \rho \cdot A_s \cdot b_r$$

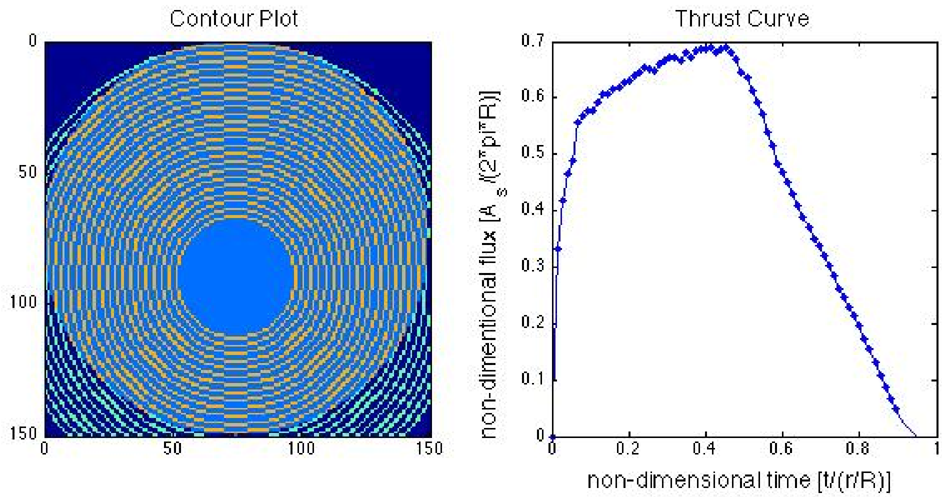
Several geometric configurations are often used depending on the application and desired thrust curve:



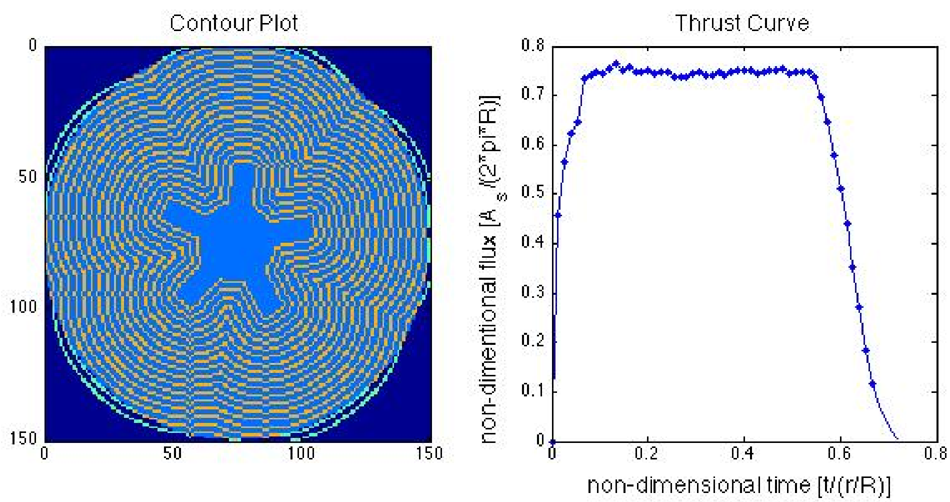
Circular Bore Simulation



C-Slot Simulation



Moon Burner Simulation



5-point Finocyl Simulation

- Circular Bore: if in BATES configuration, produces progressive-regressive thrust curve.
- End Burner: propellant burns from one axial end to other producing steady long burn, though has thermal difficulties, CG shift.
- C-Slot: propellant with large wedge cut out of side (along axial direction), producing fairly long regressive thrust, though has thermal difficulties and asymmetric CG characteristics.
- Moon Burner: off-center circular bore produces progressive-regressive long burn, though has slight asymmetric CG characteristics
- Finocyl: usually a 5- or 6-legged star-like shape that can produce very level thrust, with a bit quicker burn than circular bore due to increased surface area.

Casing

The casing may be constructed from a range of materials. Cardboard is used for small black powder model motors, whereas aluminum is used for larger composite-fuel hobby motors. Steel is used for the space shuttle boosters. Filament wound graphite epoxy casings are used for high-performance motors.

The casing must be designed to withstand the pressure and resulting stresses of the rocket motor, possibly at elevated temperature. For design, the casing is considered a pressure vessel.

To protect the casing from corrosive hot gases, a sacrificial thermal liner on the inside of the casing is often implemented, which ablates to prolong the life of the motor casing.

Nozzle

A convergent-divergent design accelerates the exhaust gas out of the nozzle to produce thrust. The nozzle must be constructed from a material that can withstand the heat of the combustion gas flow. Often, heat-resistant carbon-based materials are used, such as amorphous graphite or carbon-carbon.

Some designs include directional control of the exhaust. This can be accomplished by gimballing the nozzle, as in the Space Shuttle SRBs, by the use of jet vanes in the exhaust similar to those used in the V-2 rocket, or by liquid injection thrust vectoring (LITV).

An early Minuteman first stage used a single motor with four gimballed nozzles to provide pitch, yaw, and roll control.

LITV consists of injecting a liquid into the exhaust stream after the nozzle throat. The liquid then vaporizes, and in most cases chemically reacts, adding mass flow to one side of the exhaust stream and thus providing a control moment. For example, the Titan IIIC solid boosters injected nitrogen tetroxide for LITV; the tanks can be seen on the sides of the rocket between the main center stage and the boosters .

Performance

A typical well designed ammonium perchlorate composite propellant (APCP) first stage motor may have a vacuum specific impulse (Isp) as high as 285.6 sec (Titan IVB SRMU). This compares to 339.3 sec for kerosene/liquid oxygen (RD-180) and 452.3 sec for hydrogen/oxygen (Block II SSME) bipropellant engines. Upper stage specific impulses are somewhat greater: as much as 303.8 sec for APCP (Orbus 6E), 359 sec for kerosene/oxygen (RD-0124) and 465.5 sec for hydrogen/oxygen (RL10B-2). Propellant fractions are usually somewhat higher for (non-segmented) solid propellant first stages than for upper stages. The 117,000 pound Castor 120 first stage has a propellant mass fraction of 92.23% while the 31,000 pound Castor 30 upper stage recently developed for Orbital Science's Taurus II COTS (International Space Station resupply) launch vehicle has a 91.3% propellant fraction with 2.9% graphite epoxy motor casing, 2.4% nozzle, igniter and thrust vector actuator, and 3.4% non-motor hardware including such things as payload mount, interstage adapter, cable raceway, instrumentation, etc. Castor 120 and Castor 30 are 93 and 92 inches in diameter, respectively, and serve as stages on the Athena IC and IIC commercial launch vehicles. A four stage Athena II using Castor 120s as both first and second stages became the first *commercially developed* launch vehicle to launch a lunar probe (*Lunar Prospector*) in 1998.

Solid rockets can provide high thrust for relatively low cost. For this reason, solids have been used as initial stages in rockets (the classic example being the Space Shuttle), while reserving high specific impulse engines, especially less massive hydrogen fueled engines for higher stages. In addition, solid rockets have a long history as the final boost stage for satellites due to their simplicity, reliability, compactness and reasonably high mass fraction . A spin-stabilized solid rocket motor is sometimes added when extra velocity is required, such as for a mission to a comet or the outer solar system, because a spinner does not require a guidance system (on the newly added stage). Thiokol's extensive family of mostly titanium-cased *Star* space motors has been widely used, especially on Delta launch vehicles and as spin-stabilized upper stages to launch satellites from the cargo bay of the Space Shuttle. *Star* motors have propellant fractions as high as 94.6% but add-on structures and equipment reduce the operating mass fraction by 2% or more.

Higher performing solid rocket propellants are used in large strategic missiles (as opposed to commercial launch vehicles). HMX, $C_4H_8N_4(NO_2)_4$, a nitramine with greater energy than ammonium perchlorate, is the main ingredient in NEPE-75 propellant used in the Trident II D-5 Fleet Ballistic Missile. It is because of explosive hazard that the higher energy military solid propellants are not used in commercial launch vehicles except when the LV is an adapted ballistic missile already containing HMX propellant (example: Minotaur IV and V based on retired Peacekeeper ICBMs). The Naval Air Weapons Station at China Lake, CA developed a new compound, $C_6H_6N_6(NO_2)_6$, called simply CL-20 (China Lake compound #20). Compared to HMX, CL-20 has 14% more energy per mass, 20% more energy per volume, and a higher oxygen-to-fuel ratio. One of the motivations for development of these very high energy density military solid propellants is to achieve mid-course exo-atmospheric ABM capability from missiles small enough to fit in existing ship-based below-deck vertical launch tubes and air-mobile truck-mounted launch tubes. CL-20 propellant compliant with Congress' 2004 insensitive munitions (IM) law has been demonstrated and may, as its cost comes down, be suitable for use in commercial launch vehicles, with a very significant increase in performance compared with the currently favored APCP solid propellants.

An attractive attribute for military use is the ability for solid rocket propellant to remain loaded in the rocket for long durations and then reliably launched at a moment's notice.

Propellant families

Black Powder (BP) Propellants

Composed of charcoal (fuel), potassium nitrate (oxidizer), and sulfur (additive), BP is one of the oldest pyrotechnic compositions with application to rocketry. In modern times, black powder finds use in low-power model rockets (such as Estes and Quest rockets), as it is cheap and fairly easy to produce. The fuel grain is typically a mixture of pressed fine powder (into a solid, hard slug), with a burn rate that is highly dependent upon exact composition and operating conditions. Due to its sensitivity to fracture (and, therefore, catastrophic failure upon ignition) and poor performance (specific impulse around 80 sec), BP does not typically find use in motors above 40 Ns.

Zinc-Sulfur (ZS) Propellants

Composed of powdered zinc metal and powdered sulfur (oxidizer), ZS is another pressed propellant that does not find any practical application outside of specialized amateur rocketry circles due to its poor performance (as most ZS burns outside the combustion chamber) and incredibly fast linear burn rates on the order of 2 m/s. ZS is most often made novelty propellant as the rocket accelerates extremely quickly leaving a spectacular large orange fireball behind it.

"Candy" propellants

In general, candy propellants are an oxidizer (typically potassium nitrate) and a sugar fuel (typically dextrose, sorbitol, or sucrose) that are cast into shape by gently melting the propellant constituents together and pouring or packing the amorphous colloid into a mold. Candy propellants generate a low-medium specific impulse of roughly 130 sec and, thus, are implemented primarily only with amateur and experimental rocketeers.

Double-Base (DB) Propellants

DB propellants are composed of two monopropellant fuel components where one typically acts as a high-energy (yet unstable) monopropellant and the other acts as a lower-energy stabilizing (and gelling) monopropellant. In typical circumstances, nitroglycerin is dissolved in a nitrocellulose gel and solidified with additives. DB propellants are implemented in applications where minimal smoke is required yet medium-high performance (I_{sp} of roughly 235 sec) is required. The addition of metal fuels (such as aluminum) can increase the performance (around 250 sec), though metal oxide nucleation in the exhaust can turn the smoke opaque.

Composite propellants

A powdered oxidizer and powdered metal fuel are intimately mixed and immobilized with a rubbery binder (that also acts as a fuel). Composite propellants are often either ammonium nitrate-based (ANCP) or ammonium perchlorate-based (APCP). Ammonium nitrate composite propellant often uses magnesium and/or aluminum as fuel and delivers medium performance (I_{sp} of about 210 sec) whereas Ammonium Perchlorate Composite Propellant often uses aluminum fuel and delivers high performance (vacuum I_{sp} up to 296 sec with a single piece nozzle or 304 sec with a high area ratio telescoping nozzle). Composite propellants are cast, and retain their shape after the rubber binder, such as Hydroxyl-terminated polybutadiene (HTPB), cross-links (solidifies) with the aid of a curative additive. Because of its high performance, moderate ease of manufacturing, and moderate cost, APCP finds widespread use in space rockets, military rockets, hobby and amateur rockets, whereas cheaper and less efficient ANCP finds use in amateur rocketry and gas generators. Ammonium dinitramide, $NH_4N(NO_2)_2$, is being considered as a 1-to-1 chlorine-free substitute for ammonium perchlorate in composite propellants. Unlike ammonium nitrate, ADN can be substituted for AP without a loss in motor performance.

In 2009, a group succeeded in creating a propellant of water and nanoaluminum (ALICE).

The Constellation program uses a mix of aluminum, ammonium perchlorate, a polymer of polybutadiene and acrylonitrile, epoxy and iron oxide.

High-Energy Composite (HEC) propellants

Typical HEC propellants start with a standard composite propellant mixture (such as APCP) and add a high-energy explosive to the mix. This extra component usually is in the form of small crystals of RDX or HMX, both of which have higher energy than ammonium perchlorate. Despite a modest increase in specific impulse, implementation is limited due to the increased hazards of the high-explosive additives.

Composite Modified Double Base propellants

Composite modified double base propellants start with a nitrocellulose/nitroglycerin double base propellant as a binder and add solids (typically ammonium perchlorate and powdered aluminum) normally used in composite propellants. The ammonium perchlorate makes up the oxygen deficit introduced by using nitrocellulose, improving the overall specific impulse. The aluminum also improves specific impulse as well as combustion stability. High performing propellants such as NEPE-75 used in Trident II D-5, replace most of the AP with HMX, further increasing specific impulse. The mixing of composite and double base propellant ingredients has become so common as to blur the functional definition of double base propellants.

Minimum-signature (*smokeless*) propellants

One of the most active areas of solid propellant research is the development of high-energy, minimum-signature propellant using CL-20 (China Lake compound #20), $C_6H_6N_6(NO_2)_6$, which has 14% higher energy per mass and 20% higher energy density than HMX. The new propellant has been successfully developed and tested in tactical rocket motors. The propellant is non-polluting: acid free, solid particulates free, and lead free. It is also smoke free and has only a faint shock diamond pattern that is visible in the otherwise transparent exhaust. Without the bright flame and dense smoke trail produced by the burning of aluminized propellants, these smokeless propellants all but eliminate the risk of giving away the positions from which the missiles are fired. The new CL-20 propellant is shock-insensitive (hazard class 1.3) as opposed to current HMX smokeless propellants which are highly detonable (hazard class 1.1). CL-20 is considered a major breakthrough in solid rocket propellant technology but has yet to see widespread use because costs remain high.

Hobby and amateur rocketry

Solid propellant rocket motors can be bought for use in model rocketry; they are normally small cylinders of black powder fuel with an integral nozzle and sometimes a small charge that is set off when the propellant is exhausted after a time delay. This charge can be used to trigger a camera, or deploy a parachute. Without this charge and delay, the motor may ignite a second stage (black powder only).

In mid- and high-power rocketry, commercially made APCP motors are widely used. They can be designed as either single-use or reloadables. These motors are available in impulse ranges from "D" to "O", from several manufacturers. They are manufactured in standardized diameters, and varying lengths depending on required impulse. Standard motor diameters are 18, 24, 29, 38, 54, 75, 98, and 150 millimeters. Different propellant formulations are available to produce different thrust profiles, as well as "special effects" such as colored flames, smoke trails, or large quantities of sparks (produced by adding titanium sponge to the mix).

Designing solid rocket motors is particularly interesting to amateur rocketry enthusiasts. The design of a successful solid-fuel motor requires application of continuum mechanics, combustion chemistry, materials science, fluid dynamics (including compressible flow), heat transfer, geometry (particle spectrum packing), and machining. The vast majority of amateur-built rocket motors utilize a composite propellant, most commonly APCP.

Advanced research

- Environmentally sensitive fuel formulations such as ALICE propellant
- Ramjets with solid fuel
- Variable thrust designs based on variable nozzle geometry
- Hybrid rockets that use solid fuel and throttleable liquid or gaseous oxidizer

Chapter 2

Shavit

Shavit

Function	Expendable launch vehicle
Manufacturer	Israel Aerospace Industries
Country of origin	Israel
Size	
Height	26.4 m (86.6 ft)
Diameter	2.3 m (7.5 ft)
Mass	30,500 - 70,000 kg (67,200 - 154,000 lb)
Stages	4
Capacity	
Payload to LEO	350-800kg (770-1760 lb)

Launch history	
Status	Active
Launch sites	Palmachim Airbase
Total launches	9
Successes	6
Failures	3
Maiden flight	19 September 1988
First Stage (Shavit LeoLink LK-1) - LK-1	
Engines	LK-1
Thrust	774.0 kN (174,002 lb_f)
Specific impulse	268 sec
Burn time	55 seconds
Fuel	HTPB
First Stage (Shavit LeoLink LK-2) - Castor 120	
Engines	
Thrust	1650.2kN (370,990 lbf)
Specific impulse	280 sec
Burn time	82 sec
Fuel	HTPB polymer, Class1.3 C
Second Stage - LK-1	
Engines	1 LK-1
Thrust	774.0 kN
Specific impulse	268 sec
Burn time	55 sec
Fuel	HTPB
Third Stage - RSA-3-3	
Engines	1 RSA-3-3
Thrust	58.8 kN
Specific impulse	298 sec
Burn time	94 seconds
Fuel	Solid
Fourth Stage - LK-4	
Engines	1 LK-4
Thrust	402 kN
Specific impulse	200 sec
Burn time	800 seconds
Fuel	N_2O_4/UDMH

Shavit (Hebrew: "comet" - שביט) is a space launch vehicle produced by Israel to launch small satellites into low earth orbit. It was first launched on September 19, 1988 (carrying an Ofeq satellite payload), making Israel the eighth country to have a space launch capability after the USSR, United States, France, Japan, People's Republic of China, United Kingdom and India.

Shavit rockets are launched from Palmachim Airbase by the Israeli Space Agency into a retrograde orbit over the Mediterranean Sea to prevent debris coming down in populated areas and also to avoid flying over nations hostile to Israel to the east; this results in a lower payload-to-orbit than east-directed launches would allow. The launcher consists of three stages powered by solid fuel rocket motors, with an optional liquid fuel fourth stage, and is manufactured by IAI.

Development

The development of Shavit began in 1982. Shavit was a three-stage, solid-propellant launcher designed to carry payloads up to 250 kg into low earth orbit. It was speculated for some time and later confirmed that the first two stages of the Shavit were that of the Jericho II missile.

Shavit was first launched in 1988 and because of its geographic location and hostile relations with surrounding countries, Israel had to launch it to the west, over the Mediterranean Sea, in order to avoid flying over those hostile territories to its east. The practice has continued ever since.

Vehicle Description

The first of the Shavit vehicles were a small, 3-stage, solid propellant booster based on the 2-stage Jericho-II ballistic missile and developed under the general management of Israeli Aircraft Industries and in particular its MBT System and Space Technology subsidiary. Israel Military Industries produces the first and second stage motors, while RAFAEL is responsible for the third stage motor.

A planned commercial Shavit upgrade was called Next. This name is no longer used, and this proposed upgrade configuration is now called Shavit-2. Both first and second stages of the Shavit-2 use the stretched motor design of the Shavit-1 first stage.

Proposed LK civilian launch variants

In 1998 Israel Space Agency partnered with U.S. Coleman Research Corporation (now a division of L-3 Communications) to develop the LK family of small launch vehicles. In 2001 a new French joint-venture, LeoLink, between Astrium and Israel Aircraft Industries, was created to market the LK variant. It is believed that in 2002 development of the LK variant was discontinued.

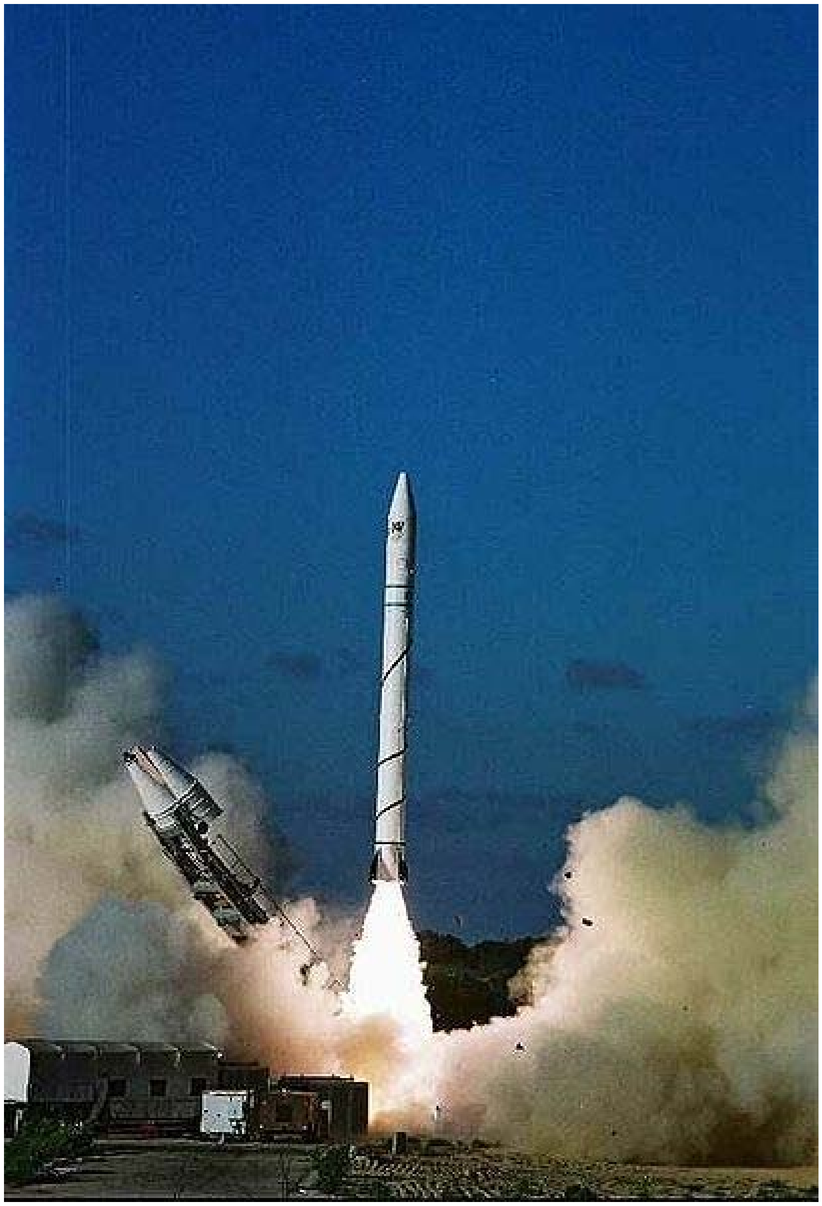
The LK-1 was closely based on the Shavit-2, but with motors and other components built in the United States to satisfy U.S. government requirements. The LK-2 was a larger vehicle using a Thiokol Castor 120 motor as its first stage. The third stage was either a standard AUS-51 motor built under license by Atlantic Research Corp., or a Thiokol Star 48 motor. All launch vehicles would have had a small monopropellant hydrazine fourth stage.

- LK-A - for 350 kg-class satellites in 240 x 600 km elliptical polar orbits.
- LK-1 - for 350 kg-class satellites in 700 km circular polar orbits.
- LK-2 - for 800 kg-class satellites in 700 km circular polar orbits.

A Shavit LK air launched satellite launcher was proposed by ISA and Israel Aircraft Industries (IAI). The booster would have been a standard Shavit-1 or Shavit-2 without a first stage that would be dropped from a Hercules C-130 aircraft.

Launch history

The Shavit has been launched nine times, placing the payload into orbit six times. On the third, fifth and seventh flights, the vehicle failed before reaching space. Most non-Israeli satellites are launched eastward to gain a boost from the Earth's rotational speed. However, the Shavit is launched westward (retrograde orbit) over the Mediterranean to avoid flying and dropping spent rocket stages over populated areas in Israel and neighboring Arab countries. The Shavit is also said to be made available for commercial launches in the near future. Of the seven launches two are the basic Shavit, four are the Shavit-1 and the last one being Shavit-2. The September 2004 failure of the Shavit resulted in the destruction of the $100 million Ofeq 6 spy satellite. Israel has announced that it may use the Polar Satellite Launch Vehicle, developed by India's ISRO for future Ofeq launches.



Variant	Date of Launch	Launch Location	Payload	Mission Status
Shavit	19 September 1988	Palmachim Airbase	Ofeq 1	Success, experimental payload
Shavit	3 April 1990	Palmachim Airbase	Ofeq 2	Success, experimental payload

Shavit	15 September 1994	Palmachim Airbase	Ofeq ?	Failure, unknown payload
Shavit-1	5 April 1995	Palmachim Airbase	Ofeq 3	Success, first operational satellite in orbit
Shavit-1	22 January 1998	Palmachim Airbase	Ofeq 4	Failure
Shavit-1	28 May 2002	Palmachim Airbase	Ofeq 5	Success, second operational satellite in orbit
Shavit-1	6 September 2004	Palmachim Airbase	Ofeq 6	Failure
Shavit-2	11 June 2007	Palmachim Airbase	Ofeq 7	Success, third operational satellite in orbit
Shavit-2	22 June 2010	Palmachim Airbase	Ofeq 9	Success

Criticism

The September 2004 failure of the Shavit resulted in the destruction of the $100 million Ofeq 6 spy satellite. Israel has announced that it will use the Polar Satellite Launch Vehicle, developed by India's ISRO for future Ofeq launches. There was widespread criticism about that decision of using PSLV for Ofeq launches as some quarters wanted to use only Shavit for launch for reasons of national pride. The followup Ofek-7 was successfully launched on a Shavit rocket in 2007. Israel also launched TecSAR SAR satellite on India's PSLV on January 21, 2008.

Comparable solid fuel rockets

- ASLV
- Minotaur
- Mu
- Pegasus
- Start-1
- VLS-1

Chapter 3

Scout (Rocket Family)

Scout launch (NASA)

The **Scout** family of rockets were launch vehicles designed to place small satellites into orbit around the Earth. The Scout multistage rocket was the first (and for a long time, the only) orbital launch vehicle to be entirely composed of solid fuel stages.

The original Scout (an acronym for Solid Controlled Orbital Utility Test system) was designed in 1957 at the NACA Langley center. Scouts were used from 1961 until 1994. To enhance reliability the development team opted to use "off the shelf" hardware, originally produced for military programs. According to the NASA fact sheet:

... the first stage motor was a combination of the Jupiter Senior and the Navy Polaris; the second stage came from the Army MGM-29 Sergeant; and the third and fourth stage motors were designed by Langley engineers who adapted a version of the Navy Vanguard.

The first successful orbital launch of a Scout, on February 16, 1961, delivered Explorer 9, a 7-kg satellite used for atmospheric density studies, into orbit. The final launch of a Scout, using a Scout G-1, was on May 9, 1994. The payload was the Miniature Sensor Technology Integration 2 (*MSTI-2*) military spacecraft with a mass of 163 kg, which remained in orbit until 1998.

The standard Scout launch vehicle was a solid propellant, four-stage booster system, approximately 75 feet (23 m) in length with a launch weight of 47,398 pounds (21,500 kilograms.)

Technical data

Scout A overview

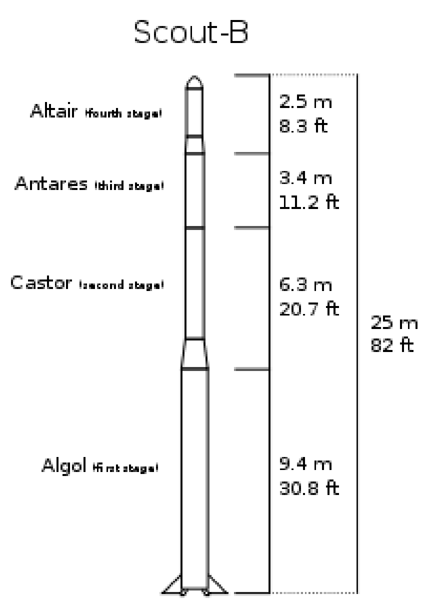


Diagram showing the Scout B rocket.

- Thrust at liftoff: 513.40 kN (52,352 kgf)
- Mass at launch: 17,850 kg
- Diameter: 1.01 m

- Length: 25.00 m

Scout A stages

Stage 1: *Algol*

- Gross Mass: 11,600 kg
- Empty Mass: 1,650 kg
- Vacuum thrust: 564.25 kN (57,537 kgf)
- Burn time: 47 s
- Diameter: 1.01 m
- Span: 1.01 m
- Length: 9.09 m

Stage 2: *Castor*

- Derived from Sergeant missile
- Gross Mass: 4,424 kg
- Empty Mass: 695 kg
- Vacuum thrust: 258.92 kN (26,402 kgf)
- Burn time: 37 s
- Diameter: 0.79 m
- Span: 0.79 m
- Length: 6.04 m

Stage 3: *Antares*

- Gross Mass: 1,400 kg
- Empty Mass: 300 kg
- Vacuum thrust: 93.09 kN (9,493 kgf)
- Burn time: 36 s
- Diameter: 0.78 m
- Span: 0.78 m
- Length: 2.90 m

Stage 4: *Altair*

- Gross Mass: 275 kg
- Empty Mass: 37 kg
- Vacuum thrust: 22.24 kN (2,268 kgf)
- Burn time: 28 s
- Diameter: 0.64 m
- Span: 0.64 m
- Length: 2.53 m

NASA use

In the late 1950s, NASA established the Scout program to develop a multistage solid-propellant space booster and research rocket. The U.S. Air Force also participated in the program, but different requirements led to some divergence in the development of NASA and USAF Scouts.

The basic NASA Scout configuration, from which all variants were derived, was known as Scout-X1. It was a four-stage rocket, which used the following motors:

- 1st stage: Aerojet General Algol
- 2nd stage: Thiokol XM33 Castor
- 3rd stage: Allegany Ballistics Laboratory X-254 Antares
- 4th stage: Allegany Ballistics Laboratory X-248 Altair

Scout's first-stage motor was based on an earlier version of the Navy's Polaris missile motor; the second-stage motor was developed from the Army's Sergeant surface-to-surface missile; and the third- and fourth-stage motors were adapted by NASA's Langley Research Center; Hampton, VA, from the Navy's Vanguard missile.

Satellites that were put into orbit

- San Marco 1, the first Italian satellite (in 1964), launched by an Italian crew.
- San Marco 2, the second Italian satellite (in 1967) and first in the world launched from a sea platform. Three more San Marco satellites would use Scout rockets.
- AEREOS and AEROS B atmospheric research
- Ariel 3 - the first satellite designed and constructed in the United Kingdom, and four other Ariel satellites.
- Magsat - The first globally complete 3D map of Earth's magnetic fields.
- Transit satellites - A prototype satellite, Transit 5A, was launched 1962-12-19 by a Scout X-3. On four different flights, Scout rockets placed two Transit satellites in orbit with a single launch. The last of these, on 1988-08-25, launched Transit-O 31 and Transit O-25 on a Scout G rocket.

Scout designations

The Scout-X1 first flew successfully on 1960-10-10, after an earlier failure in July 1960. The rocket's first stage had four stabilizing fins, and the vehicle incorporated a gyro-based guidance system for attitude stabilization to keep the rocket on course.

Some other Scout designations were:

- The Scout X-2 which in 1962 introduced the Antares IIB stage upgrade. On 1962-08-23 a Scout X-2 was used for the first successful launch of a DMSP satellite, lifting off from Point Arguello near Vandenberg Air Force Base.
- The Scout X-3 which in 1963 introduced the Algol IIA upgrade.

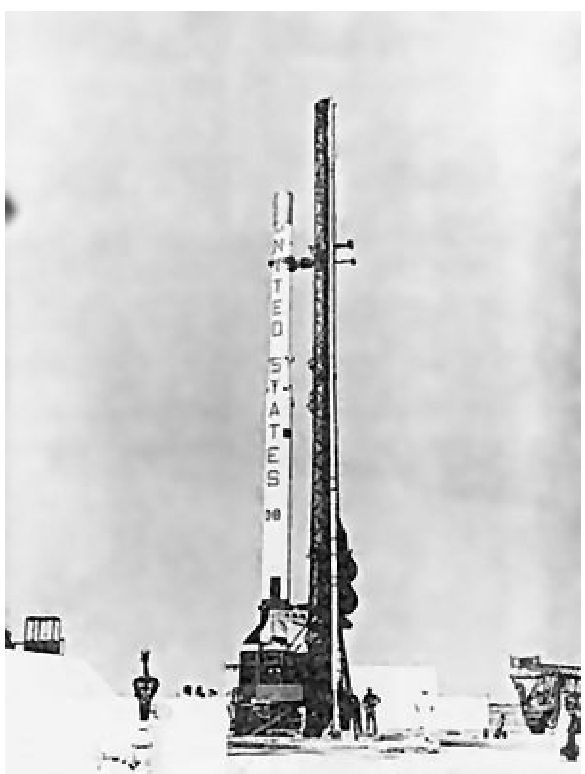
- The Scout A-1 and B-1 which in 1965 introduced the Castor IIA and Altair III upgrades, respectively.
- The Scout D-1 which in 1972 introduced the Algol III upgrade.
- The Scout G flew from 1974 until the Scout's retirement in 1994. It was rated to orbit a 210 kg payload.

Military use — Blue Scout I

The USAF Scout program was known as HETS (Hyper Environmental Test System) or System 609A, and the rockets were generally referred to as Blue Scout. The prime contractor for the NASA Scout was LTV, but the Blue Scout prime contractor was Ford Aeronutronics.

By using different combinations of rocket stages, the USAF created several different Blue Scout configurations. One of these was the XRM-89 Blue Scout I, which was a three-stage vehicle, using Castor 2 and an Antares 1A stages, but omitting the basic Scout's Altair 4th stage. The first launch of an XRM-89 occurred on 1961-01-07, and was mostly successful. On that flight, the XRM-89 carried a variety of experiments to measure rocket performance and high-altitude fields and particle radiation. The payload was located in a recoverable reentry capsule, but the capsule sank before it could be recovered from the water. The only other XRM-89 launches (in May 1961 and April 1962) were unsuccessful, and the Blue Scout I program was terminated in 1962.

Blue Scout II



Mercury-Scout 1, an Air Force Blue Scout II launched for NASA

The XRM-90 Blue Scout II was a rocket of the U.S. Air Force's System 609A Blue Scout family. The XRM-90 was a four-stage rocket, which used the same stages as the basic NASA Scout. It was nevertheless not identical to the latter, because the 4th stage was hidden in a payload fairing with the same diameter as the 3rd stage, and the first stage nozzle used a flared tail skirt between the fins. Externally, the XRM-90 was indistinguishable from the XRM-89 Blue Scout I.

The first XRM-90 launch occurred on 1961-03-03, followed by a second one on 1961-04-12. Both sub-orbital flights were successful, and measured radiation levels in the Van Allen belts. The second Blue Scout II also carried a micrometeorite sampling experiment, but the recovery of the reentry capsule failed. The third XRM-90 was used by NASA in November 1961 for Mercury-Scout 1. This was an attempt to orbit a communications payload for Project Mercury, but the rocket failed after 28 seconds of flight. The USAF subsequently abandoned the XRM-89 Blue Scout I and XRM-90 Blue Scout II vehicles, and shifted to the RM-91/SLV-1B Blue Scout Junior instead.

Blue Scout Junior

Blue Scout Junior

The XRM-91 Blue Scout Junior (sometimes called Journeyman B) was a rocket of the U.S. Air Force's System 609A Blue Scout family. Although the Blue Scout Junior had sufficient impulse to have put a small satellite in low-earth orbit, it was not used as an orbital launch vehicle. The XRM-91 did not resemble the other Scout variants externally, because the usual first Scout stage (an Aerojet General Algol) was not used. Instead, the four-stage Blue Scout Junior used Scout's 2nd and 3rd stages (Castor and Antares) as the first two stages, and added an Aerojet General Alcor and a spherical NOTS Cetus in a common nose fairing. The XRM-91 also lacked the gyro-stabilization and guidance system of the RM-89 Blue Scout I and RM-90 Blue Scout II, making it a completely unguided rocket. It relied on second-stage fins and two spin motors to achieve a stable flight trajectory.

The first launch of an XRM-91 occurred on September 21, 1960, making it actually the first Blue Scout configuration to fly. The flight was planned to make radiation and magnetic field measurements at distances of up to 26 700 km (16 600 miles) from earth, and while the rocket did indeed achieve this altitude, the telemetry system failed so that no data was received. The second launch in November ended with a failure during second stage burn. The third flight was to measure particle densities in the Van Allen belts and reached a distance of 225 000 km (140 000 miles), but again a telemetry failure prevented the reception of scientific data. The fourth and final XRM-91 mission in December 1961 also carried particle detectors, and was the only completely successful flight of the initial Blue Scout Junior program.

The Blue Scout Junior was regarded by the USAF as the most useful of the various Blue Scout configurations. It was used (in slightly modified form) between 1962 and 1965 by the Air Force as the SLV-1B/C launch vehicle for suborbital scientific payloads. The SLV-1C was also chosen as the rocket for the MER-6A interim ERCS (Emergency Rocket Communications System) vehicle. NASA used a three-stage Blue Scout Junior configuration (omitting the Cetus 4th stage) as the RAM B.

Chapter 4

Graphite-Epoxy Motor & M-V

Graphite-Epoxy Motor

A "GEM 40" Solid rocket booster is hoisted for attachment to a Delta II.

A **Graphite-Epoxy Motor** (GEM) is a high-performance, solid rocket motor, used for supplemental thrust on several launch vehicles, including the Boeing Delta II and Delta IV. They are designed to allow launch vehicles to deliver larger payloads to orbit. The name "Graphite-Epoxy Motor" refers specifically to solid motors produced by Alliant Techsystems, although boosters of similar construction are used on other launch vehicles.

Background

A solid rocket motor consists primarily of a casing that is packed with propellant grain (a mixture of a solid fuel, such as a rubber or aluminum, and an oxidizer, such as ammonium perchlorate), and a nozzle at the aft end of the motor. The casing is crucial for the solid motor because it contains the pressure of the burning solid fuel; if the casing was not strong enough, the motor would rupture and explode.

Before the development of Graphite-Epoxy Motors (GEMs), the company's (first Thiokol, now Alliant Techsystems) Castor boosters used steel solid motors used to produce extra thrust and boost payload capacity. By using a lighter material, the motor could be made larger to contain more propellant, increasing thrust and payload capacity, without increasing weight excessively. However, simply using thinner steel would not work, as the steel would be insufficiently strong to contain the burning fuel.

Stronger composites eventually enabled the construction of motors that were lighter than the older steel-case motors while still retaining the strength necessary to contain the pressure. The graphite-epoxy composite is lighter than steel, allowing the composite motor to be larger, improving thrust and performance. For example, the GEM-40 motors used on the Delta II are 6 feet longer than the Castor IVA motors they replace, allowing them to produce over 6,000 pounds more thrust and burn seven seconds longer, while still weighing over 200 pounds less.

The first flight of a GEM occurred on 26 November 1990. Nine GEMs were used as boosters for a Delta II launch vehicle (Delta 201), launching a NAVSTAR GPS satellite.

Variants

A Boeing Delta IV launching with two GEM-60 solid motors.

Alliant Techsystems manufactures GEMs for Boeing's Delta II, III, and IV launch vehicles in the following sizes:

- **GEM-40**, 40-inch-diameter (1,000 mm) solid motor used on Delta II beginning in 1990. Delta II vehicles can use three, four, or nine GEM-40s. A special version of the GEM-40, the GEM-40VN (Vectorable Nozzle), is used as the first stage in the United States' Missile Defense System interceptor vehicle, for destroying nuclear warheads.

- **GEM-46 (GEM-LDXL)**, lengthened 46-inch-diameter (1,200 mm) solid motor developed for Delta III. This solid motor variant also includes Thrust Vector Control (TVC), which helps to steer the vehicle by changing (or vectoring) the direction of the thrust. With the discontinuation of the Delta III, the GEM-46 motors (without TVC) are also used on Delta II vehicles to boost payload capacity further. A Delta II with GEM-46 motors is considered a "Heavy" variant. Both Delta III and Delta II-Heavy use nine GEM-46s.
- **GEM-60**, 60-inch-diameter (1,500 mm) solid motor used on the Delta IV family of launch vehicles. These motors are available with and without TVC. A Delta IV can have two or four GEM-60s, and a Delta IV with these motors is classified as a Delta IV Medium+ launch vehicle.

Aerojet also manufactures strap-on solid rocket boosters for the Atlas V, although they do not use the "GEM" name. They refer to them simply as the Atlas V's Solid Rocket Boosters.

Reliability

Graphite-Epoxy Motors have proven themselves to be reliable; however they are not infallible. On 17 January 1997, a Delta II (Delta 241) exploded 13 seconds after launch due to a rupture in a graphite-epoxy casing. The failure was a result of the casing having been damaged at some point, either during manufacturing or installation — the investigation could not determine the exact cause of the damage. When the motor ignited, the pressure inside the casing built up until the damaged casing could not hold in the pressure of the burning fuel and exploded, destroying the launch vehicle.

M-V

M-V

The third M-V launches with the ASTRO-E spacecraft.

Function	All-solid small orbital launch vehicle
Manufacturer	Nissan Motors (-2000) IHI AEROSPACE (-2006)
Country of origin	Japan
Size	
Height	30.8 m (101 ft)
Diameter	2.5 m (8.2 ft)
Mass	137,500 - 139,000 kg (303,100 - 306,000 lb)
Stages	3 or 4
Capacity	
Payload to LEO	1,800 kg (3,900 lb)
Payload to Polar LEO	1,300 kg (2,800 lb)
Launch history	
Status	Retired
Launch sites	Kagoshima
Total launches	7 (**M-V:** 4, **M-V KM:** 3)
Successes	6 (**M-V:** 3, **M-V KM:** 3)
Failures	1 (**M-V**)
Maiden flight	**M-V:** 10 February 2000 **M-V KM:** 12 February 1997
Last flight	**M-V:** 22 September 2006 **M-V KM:** 9 May 2003
Notable payloads	HALCA, Nozomi, ASTRO-E, Hayabusa Suzaku, AKARI Hinode
First stage - M-14	
Engines	1 Solid
Thrust	3,780.345 kN (849,855 lb_f)
Specific impulse	246 sec
Burn time	46 seconds
Fuel	Solid
Second stage - M-24	
Engines	1 Solid
Thrust	1,245.287 kN (279,952 lb_f)
Specific impulse	203 sec
Burn time	71 seconds
Fuel	Solid
Third stage - M-34	
Engines	1 Solid
Thrust	294 kN (66,093 lb_f)

Specific impulse	301 sec
Burn time	102 seconds
Fuel	Solid
	Fourth stage - KM-V1
Engines	1 Solid
Thrust	51.9 kN (11,668 lb_f)
Specific impulse	298 sec
Burn time	73 seconds
Fuel	Solid

M-V rocket with the ASTRO-E satellite (Febr. 2000)

The **M-V** rocket, also called **M-5** or **Mu-5**, was a Japanese solid-fuel rocket designed to launch scientific satellites. It was a member of the Mu family of rockets. The Institute of Space and Astronautical Science (ISAS) began developing the M-V in 1990 at a cost of 15 billion yen. It has three stages and is 30.7 meters high, 2.5 meters in diameter, and weighs about 140 tonnes (310,000 pounds). It was capable of launching a satellite weighing 1.8 tonnes (2 short tons) into an orbit as high as 250 km (155 miles).

The first M-V rocket launched the HALCA radio astronomy satellite in 1997, and the second the Nozomi Mars explorer in July 1998. The third rocket attempted to launch the Astro-E X-ray satellite on February 10, 2000 but failed.

ISAS recovered from this setback and launched Hayabusa to 25143 Itokawa in 2003.

The following M-V launch was the scientific Astro-E2 satellite, a replacement for Astro-E, which took place on July 10, 2005.

The final launch was that of the Hinode (SOLAR-B) spacecraft, along with the SSSat microsat and a nanosatellite, HIT-SAT, on 22 September 2006.

National security considerations

Solid fuel rockets are the design of choice for military applications as they can remain in storage for long periods, and then reliably launch at short notice.

Lawmakers made national security arguments for keeping Japan's solid-fuel rocket technology alive after ISAS was merged into the Japan Aerospace Exploration Agency, which also has the H-IIA liquid-fuelled rocket, in 2003. The ISAS director of external affairs, Yasunori Matogawa, said, "It seems the hard-line national security proponents in parliament are increasing their influence, and they aren't getting much criticism…I think we're moving into a very dangerous period. When you consider the current environment and the threat from North Korea, it's scary."

The M-V design could be weaponised quickly although this would be politically difficult.

M-V flights

Date (UTC)	Flight	Payload	Result
February 12, 1997 04:50:00	M-V-1	Muses B (HALCA)	Success

July 3, 1998 18:12:00	M-V-3	Planet B (Nozomi)	Success
February 10, 2000 01:30:00	M-V-4	ASTRO-E	Failure
May 9, 2003 04:29:25	M-V-5	Muses C (Hayabusa)	Success
July 10, 2005 03:30:00	M-V-6	ASTRO-E2 (Suzaku)	Success
February 21, 2006 21:28:00	M-V-8	ASTRO-F (Akari) CUTE-1.7-APD SSP (solar sail sub payload)	Success SSP failed to open completely
September 22, 2006 21:36	M-V-7	Solar-B (Hinode) HIT-SAT SSSAT (solar sail)	Success SSSat failed after launch

Following program

A follow on to the M-V is being developed, called the *Epsilon Rocket* (formerly *Advanced Solid Rocket*), with a lower 1.2 tonne LEO payload capability. The development aim is to reduce costs, primarily by using the H-IIA solid rocket booster as the first stage and through shorter launch preparation time.

Comparable solid fuel rockets

- Athena II
- Epsilon
- Minotaur IV
- Taurus
- Vega

Chapter 5

Castor (Rocket Stage) & Crow (Missile)

Castor (Rocket Stage)

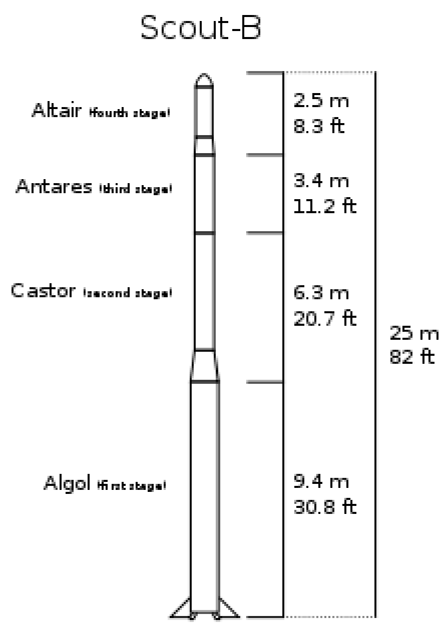


Diagram showing the use of a Castor as the second stage of a Scout-B vehicle

The **Castor** family of solid-fuel rocket stages and boosters were built by Thiokol (now ATK) and used on a variety of launch vehicles. They were initially developed as the second stage motor of the Scout rocket. The design was based the MGM-29 Sergeant, a surface-to-surface missile developed for the United States Army at the Jet Propulsion Laboratory.

Versions

Castor 1

The Castor 1 was first used for a successful suborbital launch of a Scout X-1 rocket on September 2, 1960.

It was 19.42 feet (5.92 m) long, 2.6 feet (0.79 m) in diameter, and had a burn time of 27 seconds. Castor 1 stages were also used as strap-on boosters for launch vehicles using Thor first stages, including the Delta-D. (A Delta-D was used in 1964 to launch Syncom-3, the first satellite placed in a geostationary orbit.) Castor 1 stages were used in 141 launch attempts of Scout and Delta rockets, only 2 of which were failures. They were also used on some thrust-assisted Thor-Agena launchers. The last launch using a Castor 1 was in 1971.

Castor 2

The Castor 2 was an upgraded version of the Castor 1. It was first used on a Scout in 1965, and continued to be used on Scouts until the last Scout launch, in 1994. Castor 2 stages were also used as the strap-on boosters for the Delta-E, and for the Japanese-built N-I, N-II and H-I rockets. It retained the same diameter as the Castor 1, and was from 5.96 m to 6.27 m in length.

Castor 4

The Castor 4, along with its A and B variants, were expanded to 1.02 m in diameter. They were used as strap-ons on some Delta, Delta II, Atlas IIAS, and Athena launch vehicles. They were also planned to serve as the first stage of the Spanish Capricornio booster, however, no such flights occurred before the project was cancelled.

Castor 4B is used in the European Maxus Programme, with launches from Esrange in Sweden.

The H-IIA rockets flown by JAXA use either two or four strap-on boosters developed and produced by Alliant Techsystems. These boosters use motors which are modified versions of the Castor IVA-XL motor design. These motors are 38 feet long and approximately 48 inches in diameter.

A Castor 120 that will be used as Stage 0 of a Taurus XL rocket

Castor 120

An unrelated development, the Castor 120 is a derivative of the first-stage motor of the MX ("Peacekeeper") missile. "120" refers to the planned weight, in thousands of pounds, of the booster at project inception. The actual product turned out lighter than this, however. It was first used as the first-stage motor of Lockheed Martin's Athena I, and later the first and second stages of Athena II. After a test launch in August 1995, the first launch of a customer payload took place on August 22, 1997, when an Athena was used to launch the NASA Lewis satellite.

Castor 30

An upper stage based on the Castor 120 is under development for use on the Taurus II launch vehicle.

Crow (Missile)

Crow	
 Guided version of Crow on F-4B Phantom	
Type	Experimental missile
Place of origin	United States
Service history	
In service	1961-1965
Used by	United States Navy
Production history	
Manufacturer	Naval Air Missile Test Center
Number built	7
Specifications (Unguided version)	
Weight	300 pounds (140 kg)
Length	9 feet 7 inches (2.91 m)
Diameter	8 inches (200 mm)
Engine	Rocket-ramjet; rocket, 10,000 lb_f (44.5 kN) ramjet, 220 lb_f (0.98 kN)
Wingspan	28 inches (710 mm)
Propellant	Solid fuel
Operational range	97 nautical miles (180 km; 112 mi)
Speed	Mach 3
Guidance system	None
Launch platform	F4D Skyray F-4 Phantom II

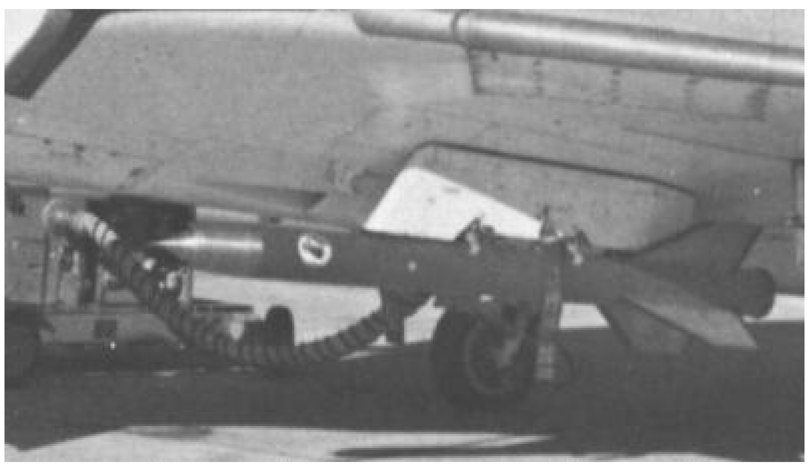
The **Creative Research On Weapons** or **Crow** program was an experimental missile project developed by the United States Navy's Naval Air Missile Test Center during the late 1950s. Intended to evalulate the solid-fueled integral rocket/ramjet (SFIRR) method of propulsion as well as solid-fueled ramjet engines, flight tests were conducted during the early 1960s with mixed success.

Development and RARE

Studies of the rocket-ramjet and solid-fueled ramjet concepts begain at the U.S. Navy's Naval Air Missile Test Center - later the Naval Missile Center - at Point Mugu, California in 1956, with the intent of increasing the range of small air-to-air missiles through using the combined ramjet and rocket propulsion system with solid fuels only. Following extensive ground testing, the concept was considered promising enough for a flight-test vehicle to be constructed to fully evalulate the new engine.

The first flight test vehicle, known as **Ram Air Rocket Engine** or **RARE**, was developed by the Naval Ordnance Test Station at China Lake, California. RARE was constructed using a conventional five-inch (127mm) rocket tube, 10 feet (3.0 m) in length and weighing 153 pounds (69 kg). Rocket-sled tests conducted during 1956 indicated that the rocket-ramjet configuration would be stable; three flight tests were conducted between 1959 and 1960, with the RARE rocket reaching speeds of Mach 2.3.

Crow I



Crow I on F4D Skyray

Even as testing of RARE was undertaken, the Naval Air Missile Test Center was developing their own test vehicle. Known as CROW, or Creative Research on Weapons, the NAMTC vehicle was intended to demonstrate that a solid-fueled rocket-ramjet was capable of delivering a reasonable payload. A simple unguided rocket, the first Crow vehicle, known as **Crow I**, was intended for aerial launch at low supersonic speed and an altitude of 50,000 feet (15,000 m). After launch, the booster acted as a ordinary solid-fueled rocket; however upon burnout of the booster stage, the rocket's casing acted as the duct for a ramjet engine, with remaining solid fuel being mixed with the incoming air to provide thrust.

The first flight test, from a Douglas F4D Skyray launch aircraft, was undertaken on January 19, 1961; due to a flaw in the launch mechanism, the rocket failed to ignite, and the test was a failure. Modifications were made, and that November two successful flights of the Crow I vehicle were conducted.

Controlled Crow

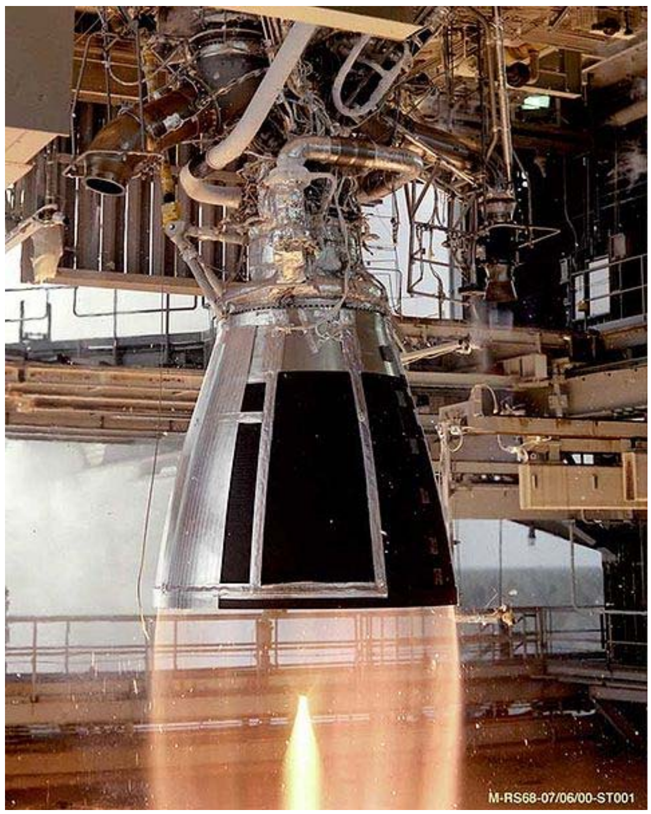
With the ballistic Crow I having proved the propulsion concept sound, follow-up work on a modification of the vehicle to provide guidance was undertaken. The missile was fitted with a simple autopilot, utilising infrared horizon-scanning to maintain the missile's attitude in flight.

Captive flight tests of Crow began in February 1963 aboard a F-4B Phantom II carrier aircraft; on May 29, the first test launch was attempted, with three further launches taking place through May 1965. None of first three attempted flights were successful, however; malfunctions in the rocket motor, autopilot, and controls plagued the program. The fourth flight test proved more successful, and Crow was considered to have met the project goals.

The Crow project successfully established the solid-fueled rocket-ramjet as a viable method of propulsion; consideration of Crow for use as an air-to-air missile or target drone was undertaken, but this was not pursued.

Chapter 6

Rocket Engine



RS-68 being tested at NASA's Stennis Space Center. The nearly transparent exhaust is due to this engine's exhaust being mostly superheated steam (water vapor from its propellants, hydrogen and oxygen)

Viking 5C rocket engine

A **rocket engine**, or simply "rocket," is a jet engine that uses only propellant mass for forming its high speed propulsive jet. Rocket engines are reaction engines and obtain thrust in accordance with Newton's third law. Since they need no external material to form their jet, rocket engines can be used for spacecraft propulsion as well as terrestrial uses, such as missiles. Most rocket engines are internal combustion engines, although non combusting forms also exist.

Rocket engines as a group have the highest exhaust velocities, are by far the lightest, and are the most energy efficient (at least at very high speed) of all types of jet engines.

However, for the thrust they give, due to the high exhaust velocity and relatively low specific energy of rocket propellant, they consume propellant very rapidly.

Terminology

Chemical rockets are rockets powered by exothermic chemical reactions of the propellant.

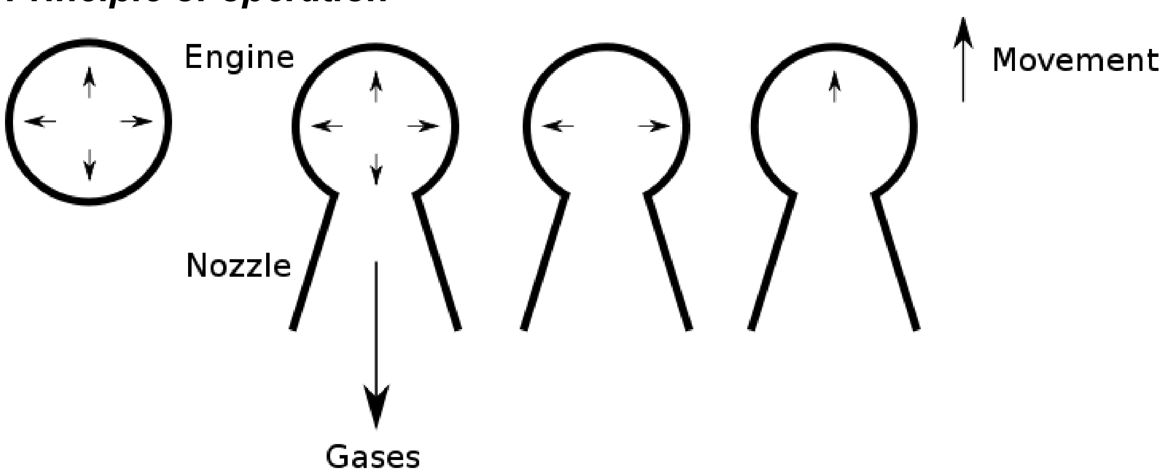
Rocket motor (or **solid-propellant rocket motor**) is a synonymous term with rocket engine that usually refers to solid rocket engines.

Liquid rockets (or **liquid-propellant rocket engine**) use one or more liquid propellants that are held in tanks prior to burning.

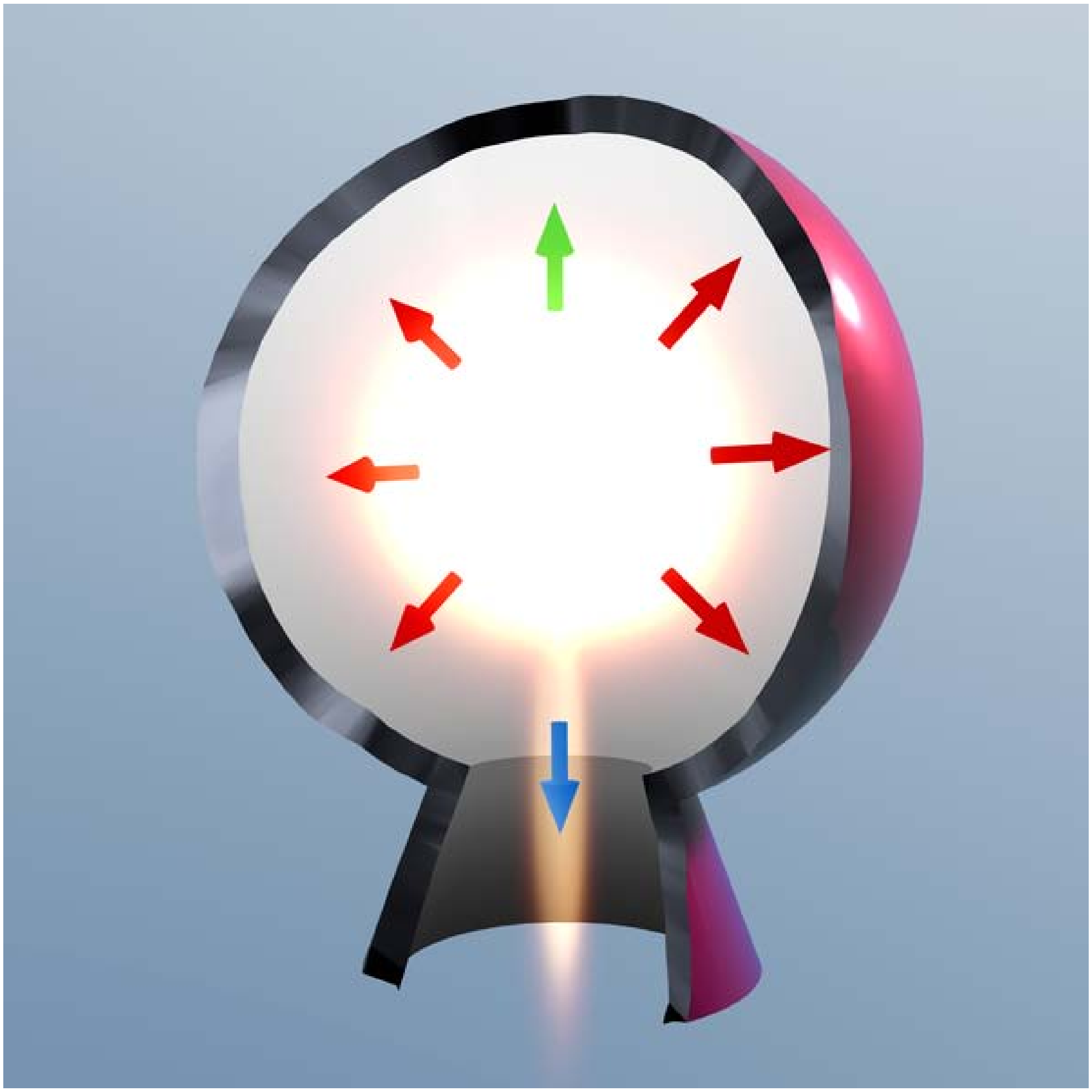
Hybrid rockets have a solid propellant in the combustion chamber and a second liquid or gas propellant is added to permit it to burn.

Thermal rockets are rockets where the propellant is inert, but is heated by a power source such as solar or nuclear power or beamed energy.

Principle of operation



How rocket engines work



Rocket engines give part of their thrust due to unopposed pressure on the combustion chamber

Rocket engines produce thrust by the expulsion of a high-speed fluid exhaust. This fluid is nearly always a gas which is created by high pressure (10-200 bar) combustion of solid or liquid propellants, consisting of fuel and oxidiser components, within a combustion chamber.

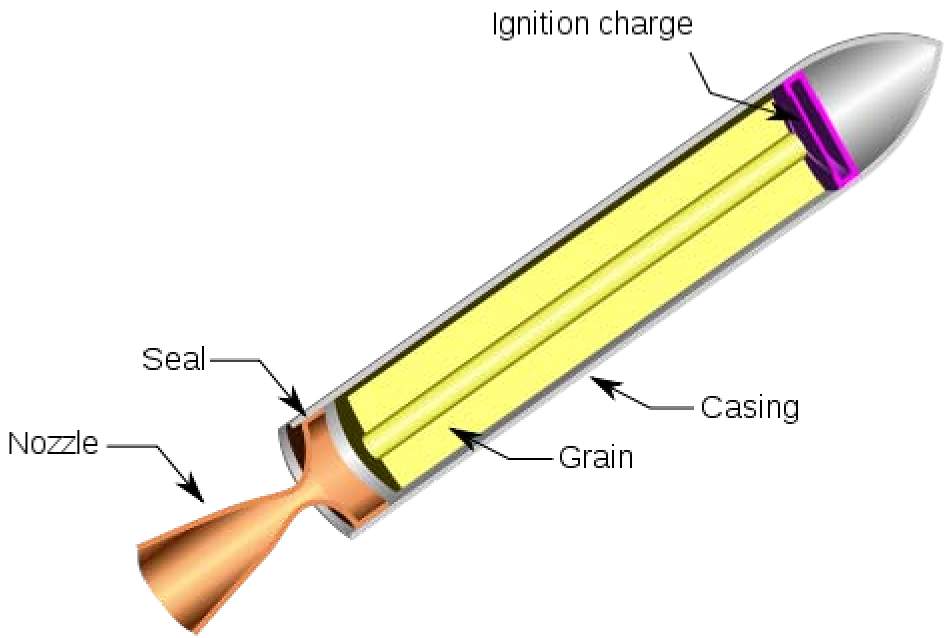
The fluid exhaust is then passed through a supersonic propelling nozzle which uses heat energy of the gas to accelerate the exhaust to very high speed, and the reaction to this pushes the engine in the opposite direction.

In rocket engines, high temperatures and pressures are highly desirable for good performance as this permits a longer nozzle to be fitted to the engine, which gives higher exhaust speeds, as well as giving better thermodynamic efficiency.

Introducing propellant into a combustion chamber

Rocket propellant is mass that is stored, usually in some form of propellant tank, prior to being ejected from a rocket engine in the form of a fluid jet to produce thrust.

Chemical rocket propellants are most commonly used, which undergo exothermic chemical reactions which produce hot gas which is used by a rocket for propulsive purposes. Alternatively, a chemically inert reaction mass can be heated using a high-energy power source via a heat exchanger, and then no combustion chamber is used.



A solid rocket motor.

Solid rocket propellants are prepared as a mixture of fuel and oxidizing components called 'grain' and the propellant storage casing effectively becomes the combustion chamber. Liquid-fueled rockets typically pump separate fuel and oxidiser components into the combustion chamber, where they mix and burn. Hybrid rocket engines use a combination of solid and liquid or gaseous propellants. Both liquid and hybrid rockets use *injectors* to introduce the propellant into the chamber. These are often an array of simple jets- holes through which the propellant escapes under pressure; but sometimes may be more complex spray nozzles. When two or more propellants are injected the jets usually deliberately collide the propellants as this breaks up the flow into smaller droplets that burn more easily.

Combustion chamber

For chemical rockets the combustion chamber is typically just a cylinder, and flame holders are rarely used. The dimensions of the cylinder are such that the propellant is able to combust thoroughly; different propellants require different combustion chamber sizes for this to occur. This leads to a number called L^* :

$$L^* = \frac{V_c}{A_t}$$

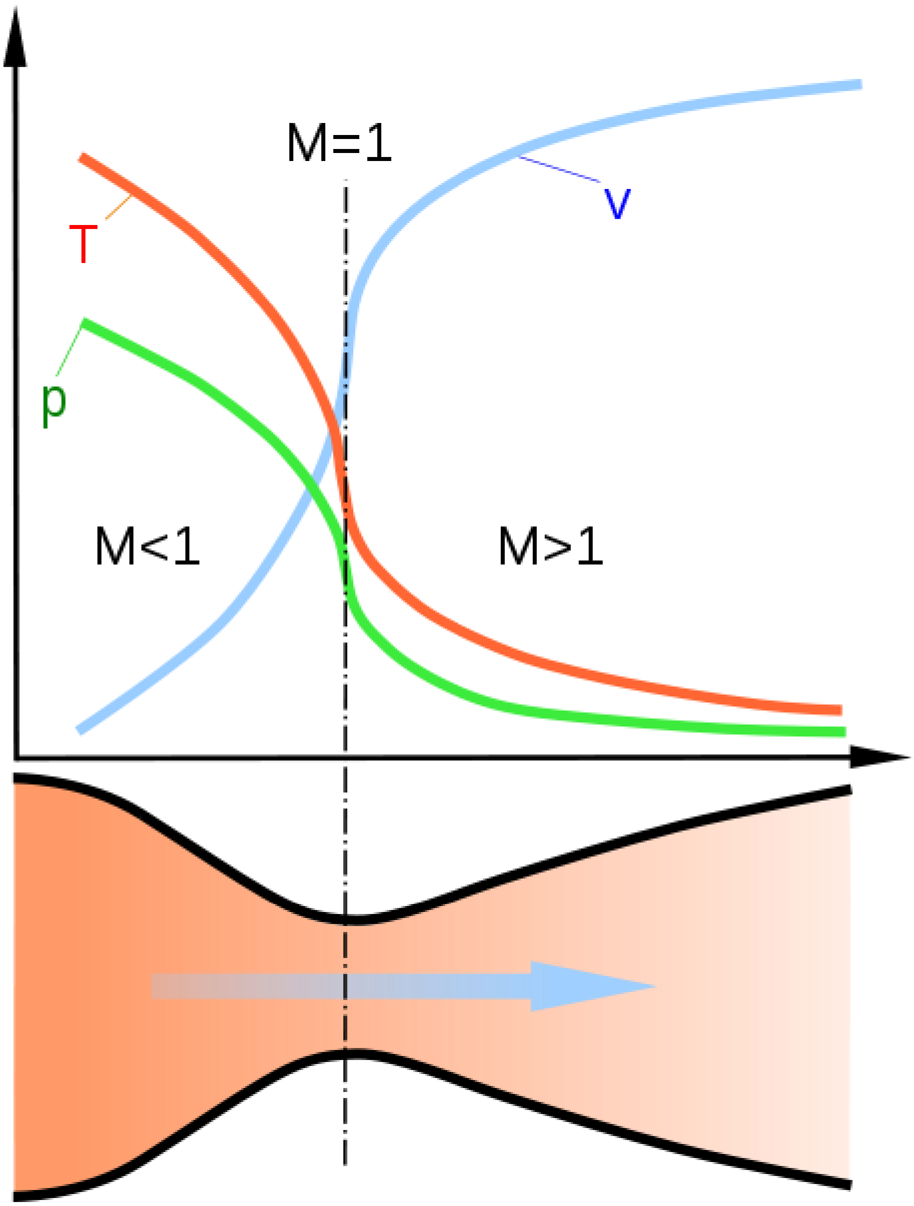
where:

- V_c is the volume of the chamber
- A_t is the area of the throat

L* is typically in the range of 25–60 inches (0.63–1.5 m).

The combination of temperatures and pressures typically reached in a combustion chamber is usually extreme by any standards. Unlike in air-breathing jet engines, no atmospheric nitrogen is present to dilute and cool the combustion, and the temperature can reach true stoichiometric. This, in combination with the high pressures, means that the rate of heat conduction through the walls is very high.

Rocket nozzles



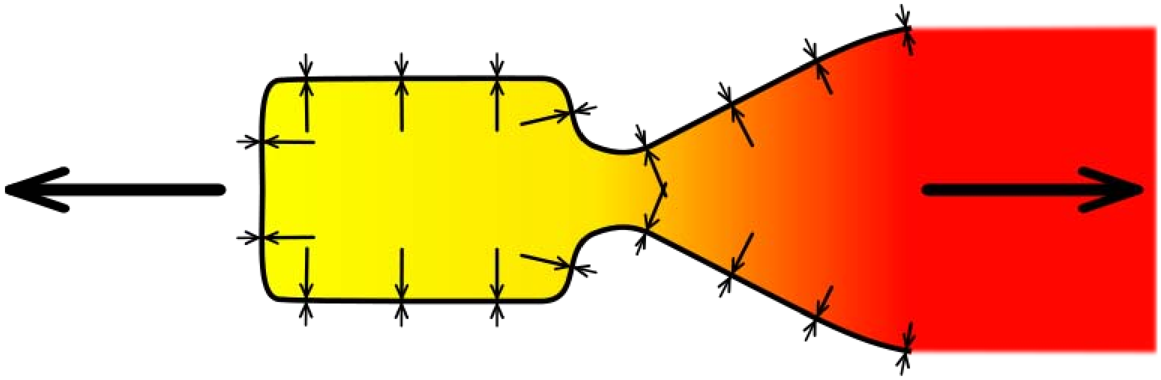
Typical temperatures (T) and pressures (p) and speeds (v) in a De Laval Nozzle

The large bell or cone shaped expansion nozzle gives a rocket engine its characteristic shape.

In rockets the hot gas produced in the combustion chamber is permitted to escape from the combustion chamber through an opening (the "throat"), within a high expansion-ratio 'de Laval' nozzle.

Provided sufficient pressure is provided to the nozzle (about 2.5-3x above ambient pressure) the nozzle *chokes* and a supersonic jet is formed, dramatically accelerating the gas, converting most of the thermal energy into kinetic energy.

The exhaust speeds vary, depending on the expansion ratio the nozzle is designed to give, but exhaust speeds as high as ten times the speed of sound of sea level air are not uncommon.



Rocket thrust is caused by pressures acting in the combustion chamber and nozzle. From Newton's third law, equal and opposite pressures act on the exhaust, and this accelerates it to high speeds.

About half of the rocket engine's thrust comes from the unbalanced pressures inside the combustion chamber and the rest comes from the pressures acting against the inside of the nozzle. As the gas expands (adiabatically) the pressure against the nozzle's walls forces the rocket engine in one direction while accelerating the gas in the other.

Propellant efficiency

For a rocket engine to be propellant efficient, it is important that the maximum pressures possible be created on the walls of the chamber and nozzle by a specific amount of propellant; as this is the source of the thrust. This can be achieved by all of:

- heating the propellant to as high a temperature as possible (using a high energy fuel, containing hydrogen and carbon and sometimes metals such as aluminium, or even using nuclear energy)
- using a low specific density gas (as hydrogen rich as possible)
- using propellants which are, or decompose to, simple molecules with few degrees of freedom to maximise translational velocity

Since all of these things minimise the mass of the propellant used, and since pressure is proportional to the mass of propellant present to be accelerated as it pushes on the engine, and since from Newton's third law the pressure that acts on the engine also reciprocally acts on the propellant, it turns out that for any given engine the speed that the propellant leaves the chamber is unaffected by the chamber pressure (although the thrust is proportional). However, speed is significantly affected by all three of the above factors and the exhaust speed is an excellent measure of the engine propellant efficiency. This is termed *exhaust velocity*, and after allowance is made for factors that can reduce it, the

effective exhaust velocity is one of the most important parameters of a rocket engine (although weight, cost, ease of manufacture etc. are usually also very important).

For aerodynamic reasons the flow goes sonic ("chokes") at the narrowest part of the nozzle, the 'throat'. Since the speed of sound in gases increases with the square root of temperature, the use of hot exhaust gas greatly improves performance. By comparison, at room temperature the speed of sound in air is about 340 m/s while the speed of sound in the hot gas of a rocket engine can be over 1700 m/s; much of this performance is due to the higher temperature, but additionally rocket propellants are chosen to be of low molecular mass, and this also gives a higher velocity compared to air.

Expansion in the rocket nozzle then further multiplies the speed, typically between 1.5 and 2 times, giving a highly collimated hypersonic exhaust jet. The speed increase of a rocket nozzle is mostly determined by its area expansion ratio—the ratio of the area of the throat to the area at the exit, but detailed properties of the gas are also important. Larger ratio nozzles are more massive but are able to extract more heat from the combustion gases, increasing the exhaust velocity.

Nozzle efficiency is affected by operation in the atmosphere because atmospheric pressure changes with altitude; but due to the supersonic speeds of the gas exiting from a rocket engine, the pressure of the jet may be either below or above ambient, and equilibrium between the two is not reached at all altitudes.

Back pressure and optimal expansion

For optimal performance the pressure of the gas at the end of the nozzle should just equal the ambient pressure: if the exhaust's pressure is lower than the ambient pressure, then the vehicle will be slowed by the difference in pressure between the top of the engine and the exit; on the other hand, if the exhaust's pressure is higher, then exhaust pressure that could have been converted into thrust is not converted, and energy is wasted.

To maintain this ideal of equality between the exhaust's exit pressure and the ambient pressure, the diameter of the nozzle would need to increase with altitude, giving the pressure a longer nozzle to act on (and reducing the exit pressure and temperature). This increase is difficult to arrange in a lightweight fashion, although is routinely done with other forms of jet engines. In rocketry a lightweight compromise nozzle is generally used and some reduction in atmospheric performance occurs when used at other than the 'design altitude' or when throttled. To improve on this, various exotic nozzle designs such as the plug nozzle, stepped nozzles, the expanding nozzle and the aerospike have been proposed, each providing some way to adapt to changing ambient air pressure and each allowing the gas to expand further against the nozzle, giving extra thrust at higher altitudes.

When exhausting into a sufficiently low ambient pressure (vacuum) several issues arise. One is the sheer weight of the nozzle- beyond a certain point, for a particular vehicle, the extra weight of the nozzle outweighs any performance gained. Secondly, as the exhaust gases adiabatically expand within the nozzle they cool, and eventually some of the chemicals can freeze, producing 'snow' within the jet. This causes instabilities in the jet and must be avoided.

On a De Laval nozzle, exhaust gas flow detachment will occur in a grossly over-expanded nozzle. As the detachment point will not be uniform around the axis of the engine, a side force may be imparted to the engine. This side force may change over time and result in control problems with the launch vehicle.

Thrust vectoring

Many engines require the overall thrust to change direction over the length of the burn. A number of different ways to achieve this have been flown:

- The entire engine is mounted on a hinge or gimbal and any propellant feeds reach the engine via low pressure flexible pipes or rotary couplings.
- Just the combustion chamber and nozzle is gimbled, the pumps are fixed, and high pressure feeds attach to the engine
- multiple engines (often canted at slight angles) are deployed but throttled to give the overall vector that is required, giving only a very small penalty
- fixed engines with vernier thrusters
- high temperature vanes held in the exhaust that can be tilted to deflect the jet

Overall rocket engine performance

Rocket technology can combine very high thrust (meganewtons), very high exhaust speeds (around 10 times the speed of sound in air at sea level) and very high thrust/weight ratios (>100) *simultaneously* as well as being able to operate outside the atmosphere, and while permitting the use of low pressure and hence lightweight tanks and structure.

Rockets can be further optimised to even more extreme performance along one or more of these axes at the expense of the others.

Specific impulse

The most important metric for the efficiency of a rocket engine is impulse per unit of propellant, this is called specific impulse (usually written I_{sp}). This is either measured as a speed (the *effective exhaust velocity* V_e in metres/second or ft/s) or as a time (seconds). An engine that gives a large specific impulse is normally highly desirable.

The specific impulse that can be achieved is primarily a function of the propellant mix (and ultimately would limit the specific impulse), but practical limits on chamber pressures and the nozzle expansion ratios reduce the performance that can be achieved.

Typical performances of common propellants

Propellant mix	Vacuum Isp (seconds)	Effective exhaust velocity (m/s)
liquid oxygen/ liquid hydrogen	455	4462
liquid oxygen/ kerosene (RP-1)	358	3510
nitrogen tetroxide/ hydrazine	305	2993

n.b. All performances at a nozzle expansion ratio of 40

Net thrust

Below is an approximate equation for calculating the net thrust of a rocket engine:

$$F_n = \dot{m}\, V_e = \dot{m}\, V_{e-act} + A_e(P_e - P_{amb})$$

where:

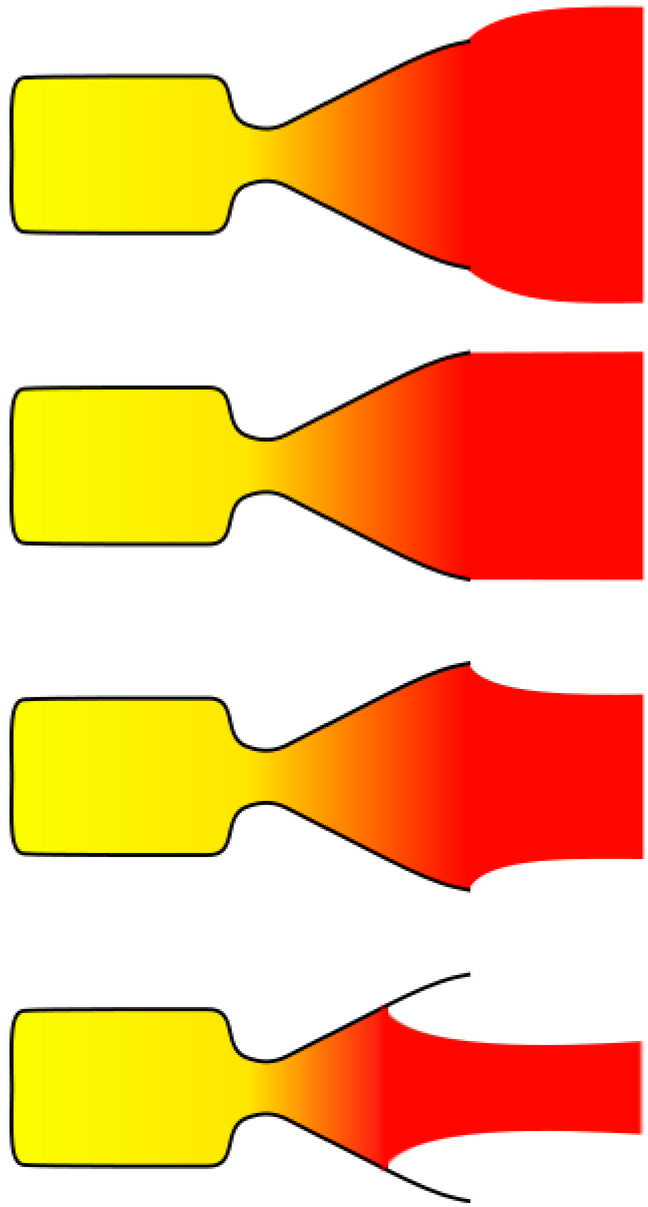
$\dot{m}$ =exhaust gas mass flow
V_e =effective exhaust velocity
V_{e-act} =actual jet velocity at nozzle exit plane
A_e =flow area at nozzle exit plane (or the plane where the jet leaves the nozzle if separated flow)
P_e =static pressure at nozzle exit plane
P_{amb} =ambient (or atmospheric) pressure

Since, unlike a jet engine, a conventional rocket motor lacks an air intake, there is no 'ram drag' to deduct from the gross thrust. Consequently the net thrust of a rocket motor is equal to the gross thrust (apart from static back pressure).

The $\dot{m}\, V_{e-act}$term represents the momentum thrust, which remains constant at a given throttle setting, whereas the $A_e(P_e - P_{amb})$term represents the pressure thrust term. At full throttle, the net thrust of a rocket motor improves slightly with increasing altitude, because as atmospheric pressure decreases with altitude, the pressure thrust term increases. At the surface of the Earth the pressure thrust may be reduced by up to 30%,depending on the engine design. This reduction drops roughly exponentially to zero with increasing altitude.

Maximum thrust for a rocket engine is achieved by maximizing the momentum contribution of the equation without incurring penalties from over expanding the exhaust.

This occurs when $P_e = P_{amb}$. Since ambient pressure changes with altitude, most rocket engines spend very little time operating at peak efficiency.



If the pressure of the exhaust jet varies from atmospheric pressure, nozzles can be said to be (top to bottom):

Underexpanded
Ambient
Overexpanded
Grossly overexpanded

If under or overexpanded then loss of efficiency occurs, grossly overexpanded nozzles lose less efficiency, but can cause mechanical issues with the nozzle. Rockets become progressively more underexpanded as they gain altitude. Note that almost all rocket engines will be momentarily grossly overexpanded during startup in an atmosphere.

Vacuum Isp

Due to the specific impulse varying with pressure, a quantity that is easy to compare and calculate with is useful. Because rockets choke at the throat, and because the supersonic exhaust prevents external pressure influences travelling upstream, it turns out that the pressure at the exit is ideally exactly proportional to the propellant flow $\dot{m}$, provided the mixture ratios and combustion efficiencies are maintained. It is thus quite usual to rearrange the above equation slightly:

$$F_{vac} = C_f \dot{m} c^*$$

and so define the *vacuum Isp* to be:

$$V_{evac} = C_f c^*$$

Where:

$c^* =$ the speed of sound constant at the throat
$C_f =$ the thrust coefficient constant of the nozzle (typically about 2)

And hence:

$$F_n = \dot{m} V_{evac} - A_e P_{amb}$$

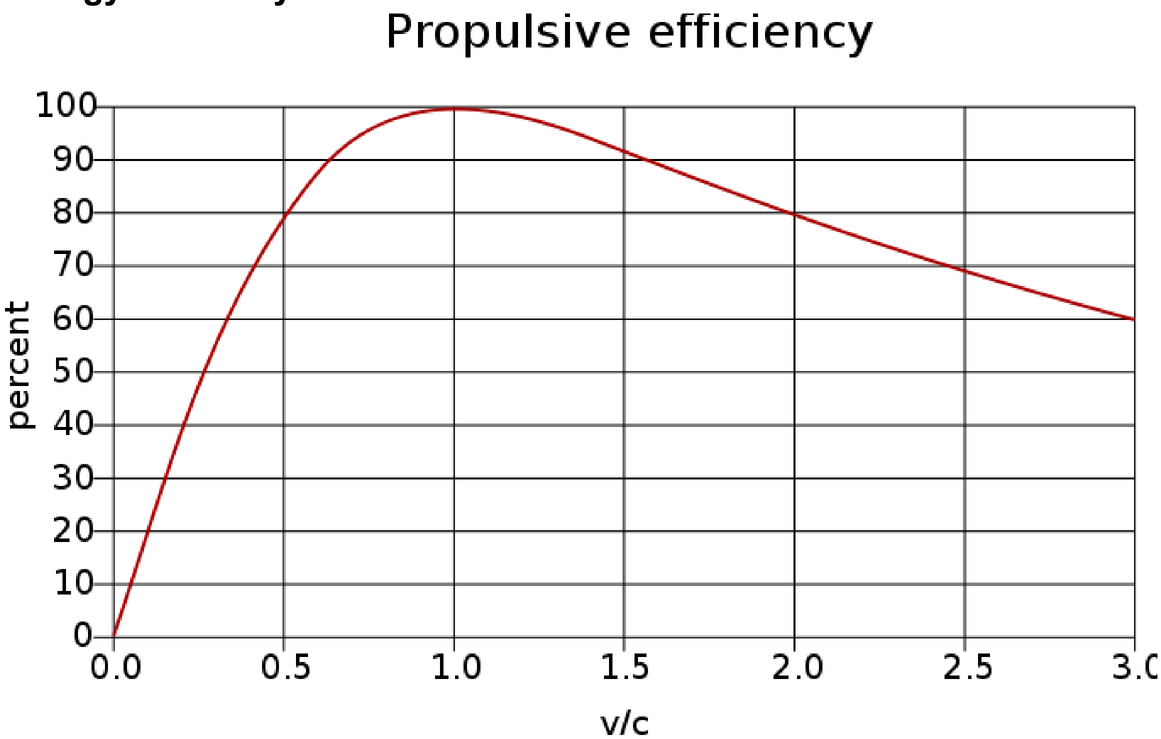
Throttling

Rockets can be throttled by controlling the propellant combustion rate $\dot{m}$(usually measured in kg/s or lb/s). In liquid and hybrid rockets, the propellant flow entering the chamber is controlled using valves, in solid rockets it is controlled by changing the area of propellant that is burning and this can be designed into the propellant grain (and hence cannot be controlled in real-time).

Rockets can usually be throttled down to an exit pressure of about one-third of ambient pressure (often limited flow separation in nozzles) and up to a maximum limit determined only by the mechanical strength of the engine.

In practice, the degree to which rockets can be throttled varies greatly, but most rockets can be throttled by a factor of 2 without great difficulty; the typical limitation is combustion stability, as for example, injectors need a minimum pressure to avoid triggering damaging oscillations (chugging or combustion instabilities); but injectors can often be optimised and tested for wider ranges. Solid rockets can be throttled by using shaped grains that will vary their surface area over the course of the burn.

Energy efficiency



Rocket energy efficiency as a function of vehicle speed divided by effective exhaust speed

Rocket engine nozzles are surprisingly efficient heat engines for generating a high speed jet, as a consequence of the high combustion temperature and high compression ratio. Rocket nozzles give an excellent approximation to adiabatic expansion which is a reversible process, and hence they give efficiencies which are very close to that of the Carnot cycle. Given the temperatures reached, over 60% efficiency can be achieved with chemical rockets.

For a *vehicle* employing a rocket engine the energetic efficiency is very good if the vehicle speed approaches or somewhat exceeds the exhaust velocity (relative to launch); but at low speeds the energy efficiency goes to 0% at zero speed (as with all jet propulsion.)

Thrust to weight ratio

Rockets, of all the jet engines, indeed of essentially all engines, have the highest thrust to weight ratio. This is especially true for liquid rocket engines.

This high performance is due to the small volume of pressure vessels that make up the engine- the pumps, pipes and combustion chambers involved. The lack of inlet duct and the use of dense liquid propellant allows the pressurisation system to be small and lightweight, whereas duct engines have to deal with air which has a density about one thousand times lower.

Jet or Rocket engine	Mass, kg	Jet or rocket thrust, kN	Thrust-to-weight ratio
RD-0410 nuclear rocket engine	2000	35.2	1.8
J-58 (SR-71 Blackbird jet engine)	2722	150	5.2
Concorde's Rolls-Royce/Snecma Olympus 593 turbojet with reheat	3175	169.2	5.4
RD-0750 rocket engine, three-propellant mode	4621	1413	31.2
RD-0146 rocket engine	260	98	38.5
Space Shuttle's SSME **rocket engine**	3177	2278	73.2
RD-180 rocket engine	5393	4152	78.6
F-1 (Saturn V first stage)	8391	7740.5	94.1
NK-33 rocket engine	1222	1638	136.8

Rocket thrusts are vacuum thrusts unless otherwise noted

Of the liquid propellants used, density is worst for liquid hydrogen. Although this propellant is marvellous in many ways, it has a very low density, about one fourteenth that of water. This makes the turbopumps and pipework larger and heavier, and this is reflected in the thrust-to-weight ratio of engines that use it (for example the SSME) compared to those that do not (NK-33).

Cooling

For efficiency reasons, and because they physically can, rockets run with combustion temperatures that can reach ~3500 K (~5800 °F)(~3227 °C).

Most other jet engines have gas turbines in the hot exhaust. Due to their larger surface area, they are harder to cool and hence there is a need to run the combustion processes at much lower temperatures, losing efficiency. In addition duct engines use air as an oxidant, which contains 80% largely unreactive nitrogen, which dilutes the reaction and lowers the temperatures. Rockets have none of these inherent disadvantages.

Therefore in rockets temperatures employed are very often far higher than the melting point of the nozzle and combustion chamber materials, two exceptions are graphite and tungsten (~1200 K for copper), however both are subject to oxidation if not protected. Indeed many construction materials can make perfectly acceptable propellants in their own right. It is important that these materials be prevented from combusting, melting or vaporising to the point of failure. This is sometimes somewhat facetiously termed an 'engine rich exhaust'. Materials technology could potentially place an upper limit on the exhaust temperature of chemical rockets.

Alternatively, rockets may use more common construction materials such as aluminium, steel, nickel or copper alloys and employ cooling systems that prevent the construction material itself becoming too hot. Regenerative cooling, where the propellant is passed through tubes around the combustion chamber or nozzle, and other techniques, such as curtain cooling or film cooling, are employed to give longer nozzle and chamber life. These techniques ensure that a gaseous thermal boundary layer touching the material is kept below the temperature which would cause the material to catastrophically fail.

In rockets, the heat fluxes that can pass through the wall are among the highest in engineering, fluxes are generally in the range of 1-200 MW/m^2. The strongest heat fluxes are found at the throat, which often sees twice that found in the associated chamber and nozzle. This is due to the combination of high speeds (which gives a very thin boundary layer), and although lower than the chamber, the high temperatures seen there.

In rockets the coolant methods include:

1. uncooled (used for short runs mainly during testing)
2. ablative walls (walls are lined with a material that is continuously vaporised and carried away).
3. radiative cooling (the chamber becomes almost white hot and radiates the heat away)
4. dump cooling (a propellant, usually hydrogen, is passed around the chamber and dumped)
5. regenerative cooling (liquid rockets use the fuel, or occasionally the oxidiser, to cool the chamber via a cooling jacket before being injected)
6. curtain cooling (propellant injection is arranged so the temperature of the gases is cooler at the walls)
7. film cooling (surfaces are wetted with liquid propellant, which cools as it evaporates)

In all cases the cooling effect that prevents the wall from being destroyed is caused by a thin layer of insulating fluid (a boundary layer) that is in contact with the walls that is far cooler than the combustion temperature. Provided this boundary layer is intact the wall will not be damaged.

Disruption of the boundary layer may occur during cooling failures or combustion instabilities, and wall failure typically occurs soon after.

With regenerative cooling a second boundary layer is found in the coolant channels around the chamber. This boundary layer thickness needs to be as small as possible, since the boundary layer acts as an insulator between the wall and the coolant. This may be achieved by making the coolant velocity in the channels as high as possible.

In practice, regenerative cooling is nearly always used in conjunction with curtain cooling and/or film cooling.

Liquid fueled engines are often run fuel rich, which results in lower temperature combustion. Cooler exhaust reduces heat loads on the engine allowing lower cost materials, a simplified cooling system, and a lower performance engine.

Mechanical issues

Rocket combustion chambers are normally operated at fairly high pressure, typically 10-200 bar (1 to 20 MPa, 150-3000 psi). When operated within significant atmospheric pressure, higher combustion chamber pressures give better performance by permitting a larger and more efficient nozzle to be fitted without it being grossly overexpanded.

However, these high pressures cause the outermost part of the chamber to be under very large hoop stresses – rocket engines are pressure vessels.

Worse, due to the high temperatures created in rocket engines the materials used tend to have a significantly lowered working tensile strength.

In addition, significant temperature gradients are set up in the walls of the chamber and nozzle, these cause differential expansion of the inner liner that create internal stresses.

Acoustic issues

In addition, the extreme vibration and acoustic environment inside a rocket motor commonly result in peak stresses well above mean values, especially in the presence of organ pipe-like resonances and gas turbulence.

Combustion instabilities

The combustion may display undesired instabilities, of sudden or periodic nature. The pressure in the injection chamber may increase until the propellant flow through the injector plate decreases; a moment later the pressure drops and the flow increases, injecting more propellant in the combustion chamber which burns a moment later, and again increases the chamber pressure, repeating the cycle. This may lead to high-amplitude pressure oscillations, often in ultrasonic range, which may damage the motor. Oscillations of ±200 psi at 25 kHz were the cause of failures of early versions of the Titan II missile second stage engines. The other failure mode is a deflagration to detonation transition; the supersonic pressure wave formed in the combustion chamber may destroy the engine.

The combustion instabilities can be provoked by remains of cleaning solvents in the engine, reflected shock wave, initial instability after ignition, explosion near the nozzle that reflects into the combustion chamber, and many more factors. In stable engine designs the oscillations are quickly suppressed; in unstable designs they persist for prolonged periods. Oscillation suppressors are commonly used.

Periodic variations of thrust, caused by combustion instability or longitudinal vibrations of structures between the tanks and the engines which modulate the propellant flow, are known as "pogo oscillations" or "pogo", named after the pogo stick.

Three different types of combustion instabilities occur:

Chugging

This is a low frequency oscillation at a few Hertz in chamber pressure usually caused by pressure variations in feed lines due to variations in acceleration of the vehicle. This can cause cyclic variation in thrust, and the effects can vary from merely annoying to actually damaging the payload or vehicle. Chugging can be minimised by using gas-filled damping tubes on feed lines of high density propellants.

Buzzing

This can be caused due to insufficient pressure drop across the injectors. It generally is mostly annoying, rather than being damaging. However, in extreme cases combustion can end up being forced backwards through the injectors – this can cause explosions with monopropellants.

Screeching

This is the most immediately damaging, and the hardest to control. It is due to acoustics within the combustion chamber that often couples to the chemical combustion processes that are the primary drivers of the energy release, and can lead to unstable resonant "screeching" that commonly leads to catastrophic failure due to thinning of the insulating thermal boundary layer. Such effects are very difficult to predict analytically during the design process, and have usually been addressed by expensive, time consuming and extensive testing, combined with trial and error remedial correction measures.

Screeching is often dealt with by detailed changes to injectors, or changes in the propellant chemistry, or vaporizing the propellant before injection, or use of Helmholtz dampers within the combustion chambers to change the resonant modes of the chamber.

Testing for the possibility of screeching is sometimes done by exploding small explosive charges outside the combustion chamber with a tube set tangentially to the combustion chamber near the injectors to determine the engine's impulse response and then evaluating the time response of the chamber pressure- a fast recovery indicates a stable system.

Exhaust noise

For all but the very smallest sizes, rocket exhaust compared to other engines is generally very noisy. As the hypersonic exhaust mixes with the ambient air, shock waves are formed. The Space Shuttle generates over 200 dB(A) of noise around its base.

The Saturn V launch was detectable on seismometers a considerable distance from the launch site. The sound intensity from the shock waves generated depends on the size of the rocket and on the exhaust velocity. Such shock waves seem to account for the characteristic crackling and popping sounds produced by large rocket engines when heard live. These noise peaks typically overload microphones and audio electronics, and so are generally weakened or entirely absent in recorded or broadcast audio reproductions. For large rockets at close range, the acoustic effects could actually kill.

More worryingly for space agencies, such sound levels can also damage the launch structure, or worse, be reflected back at the comparatively delicate rocket above. This is why so much water is typically used at launches. The water spray changes the acoustic qualities of the air and reduces or deflects the sound energy away from the rocket.

Generally speaking noise is most intense when a rocket is close to the ground, since the noise from the engines radiates up away from the plume, as well as reflecting off the ground. Also, when the vehicle is moving slowly, little of the chemical energy input to the engine can go into increasing the kinetic energy of the rocket (since useful power P transmitted to the vehicle is $P = F * V$ for thrust F and speed V). Then the largest portion of the energy is dissipated in the exhaust's interaction with the ambient air, producing noise. This noise can be reduced somewhat by flame trenches with roofs, by water injection around the plume and by deflecting the plume at an angle.

Testing

Rocket engines are usually statically tested at a test facility before being put into production. For high altitude engines, either a shorter nozzle must be used, or the rocket must be tested in a large vacuum chamber.

Safety

Rockets have a reputation for unreliability and danger; especially catastrophic failures. Contrary to this reputation, carefully designed rockets can be made arbitrarily reliable. In military use, rockets are not unreliable. However, one of the main non-military uses of rockets is for orbital launch. In this application, the premium is on minimum weight, and it is difficult to achieve high reliability and low weight simultaneously. In addition, if the number of flights launched is low, there is a very high chance of a design, operations or manufacturing error causing destruction of the vehicle. Essentially all launch vehicles are test vehicles by normal aerospace standards (as of 2006).

The X-15 rocket plane achieved a 0.5% failure rate, with a single catastrophic failure during ground test, and the SSME has managed to avoid catastrophic failures in over 350 engine-flights.

Chemistry

Rocket propellants require a high specific energy (energy per unit mass), because ideally all the reaction energy appears as kinetic energy of the exhaust gases, and exhaust velocity is the single most important performance parameter of an engine, on which vehicle performance depends.

Aside from inevitable losses and imperfections in the engine, incomplete combustion, etc., after specific reaction energy, the main theoretical limit reducing the exhaust velocity obtained is that, according to the laws of thermodynamics, a fraction of the chemical energy may go into rotation of the exhaust molecules, where it is unavailable for producing thrust. Monatomic gases like helium have only three degrees of freedom, corresponding to the three dimensions of space, $\{x,y,z\}$, and only such spherically symmetric molecules escape this kind of loss. A diatomic molecule like H_2 can rotate about either of the two axes perpendicular to the one joining the two atoms, and as the equipartition law of statistical mechanics demands that the available thermal energy be divided equally among the degrees of freedom, for such a gas in thermal equilibrium 3/5 of the energy can go into unidirectional motion, and 2/5 into rotation. A triatomic molecule like water has six degrees of freedom, so the energy is divided equally among rotational and translational degrees of freedom. For most chemical reactions the latter situation is the case. This issue is traditionally described in terms of the ratio, gamma, of the specific heat of the gas at constant volume to that at constant pressure. The rotational energy loss is largely recovered in practice if the expansion nozzle is large enough to allow the gases to expand and cool sufficiently, the function of the nozzle being to convert the random thermal motions of the molecules in the combustion chamber into the unidirectional translation that produces thrust. As long as the exhaust gas remains in equilibrium as it expands, the initial rotational energy will be largely returned to translation in the nozzle.

Although the specific reaction energy per unit mass of reactants is key, low mean molecular weight in the reaction products is also important in practice in determining exhaust velocity. This is because the high gas temperatures in rocket engines pose serious problems for the engineering of survivable motors. Because temperature is proportional to the mean *energy per molecule*, a given amount of energy distributed among more molecules of lower mass permits a higher exhaust velocity at a given temperature. This means low atomic mass elements are favoured. Liquid hydrogen (LH2) and oxygen (LOX, or LO2), are the most effective propellants in terms of exhaust velocity that have been widely used to date, though a few exotic combinations involving boron or liquid ozone are potentially somewhat better in theory if various practical problems could be solved.

It is important to note in computing the specific reaction energy, that the *entire mass of the propellants, including both fuel and oxidizer*, must be included. The fact that air-breathing engines are typically able to obtain oxygen "for free" without having to carry it along, accounts for one factor of why air-breathing engines are very much more propellant-mass efficient, and one reason that rocket engines are far less suitable for most ordinary terrestrial applications. Fuels for automobile or turbojet engines, utilize atmospheric oxygen and so have a much better effective energy output per unit mass of propellant that must be carried, but are similar per unit mass of fuel.

Computer programs that predict the performance of propellants in rocket engines are available.

Ignition

With liquid and hybrid rockets, immediate ignition of the propellant(s) as they first enter the combustion chamber is essential.

With liquid propellants (but not gaseous), failure to ignite within milliseconds usually causes too much liquid propellant to be within the chamber, and if/when ignition occurs the amount of hot gas created will often exceed the maximum design pressure of the chamber. The pressure vessel will often fail catastrophically. This is sometimes called a *hard start*.

Ignition can be achieved by a number of different methods; a pyrotechnic charge can be used, a plasma torch can be used, or electric spark plugs may be employed. Some fuel/oxidizer combinations ignite on contact (hypergolic), and non-hypergolic fuels can be "chemically ignited" by priming the fuel lines with hypergolic propellants (popular in Russian engines).

Gaseous propellants generally will not cause hard starts, with rockets the total injector area is less than the throat thus the chamber pressure tends to ambient prior to ignition and high pressures cannot form even if the entire chamber is full of flammable gas at ignition.

Solid propellants are usually ignited with one-shot pyrotechnic devices.

Once ignited, rocket chambers are self sustaining and igniters are not needed. Indeed chambers often spontaneously reignite if they are restarted after being shut down for a few seconds. However, when cooled, many rockets cannot be restarted without at least minor maintenance, such as replacement of the pyrotechnic igniter.

Plume physics

Armadillo aerospace's quad vehicle showing visible banding (shock diamonds) in the exhaust plume

Rocket plume varies depending on the rocket engine, design altitude, altitude, thrust and other factors.

Carbon rich exhausts from kerosene fuels are often orange in colour due to the black body radiation of the unburned particles, in addition to the blue Swan bands. Peroxide oxidiser based rockets and hydrogen rocket plumes contain largely steam and are nearly invisible to the naked eye but shine brightly in the ultraviolet and infrared. Plumes from solid rockets can be highly visible as the propellant frequently contains metals such as elemental aluminium which burns with an orange-white flame and adds energy to the combustion process.

Some exhausts, notably alcohol fuelled rockets, can show visible shock diamonds. These are due to cyclic variations in the plume pressure relative to ambient creating shock waves that form 'mach disks'.

The shape of the plume varies from the design altitude, at high altitude all rockets are grossly under-expanded, and a quite small percentage of exhaust gases actually end up expanding forwards.

Types of rocket engines

Physically powered

Type	Description	Advantages	Disadvantages
water rocket	Partially filled pressurised carbonated drinks container with tail and nose weighting	Very simple to build	Altitude typically limited to a few hundred feet or so (world record is 623 meters/2044 feet)
cold gas thruster	A non combusting form, used for vernier thrusters	Non contaminating exhaust	Extremely low performance
hot water rocket	Hot water is stored in a tank at high temperature/pressure and turns to steam in nozzle	Simple, fairly safe, under 200 seconds Isp	Low overall performance due to heavy tank

Chemically powered

Type	Description	Advantages	Disadvantages
Solid rocket	Ignitable, self sustaining solid fuel/oxidiser mixture ("grain") with central hole and nozzle	Simple, often no moving parts, reasonably good mass fraction, reasonable I_{sp}. A thrust schedule can be designed into the grain.	Once lit, extinguishing it is difficult although often possible, cannot be throttled in real time; handling issues from ignitable mixture, lower performance than liquid rockets, if grain cracks it can block nozzle with disastrous results, cracks burn and widen during burn. Refuelling grain harder than simply filling tanks, Lower specific Impulse than Liquid Rockets.
Hybrid rocket	Separate oxidiser/fuel, typically oxidiser is liquid and kept in a tank, the other solid with central hole	Quite simple, solid fuel is essentially inert without oxidiser, safer; cracks do not escalate, throttleable and easy to switch off.	Some oxidisers are monopropellants, can explode in own right; mechanical failure of solid propellant can block nozzle (very rare with rubberised propellant), central hole widens over burn

			and negatively affects mixture ratio.
Monopropellant rocket	Propellant such as Hydrazine, Hydrogen Peroxide or Nitrous Oxide, flows over catalyst and exothermically decomposes and hot gases are emitted through nozzle	Simple in concept, throttleable, low temperatures in combustion chamber	catalysts can be easily contaminated, monopropellants can detonate if contaminated or provoked, I_{sp} is perhaps 1/3 of best liquids
Liquid Bipropellant rocket	Two fluid (typically liquid) propellants are introduced through injectors into combustion chamber and burnt	Up to ~99% efficient combustion with excellent mixture control, throttleable, can be used with turbopumps which permits incredibly lightweight tanks, can be safe with extreme care	Pumps needed for high performance are expensive to design, huge thermal fluxes across combustion chamber wall can impact reuse, failure modes include major explosions, a lot of plumbing is needed.
Dual mode propulsion rocket	Rocket takes off as a bipropellant rocket, then turns to using just one propellant as a monopropellant	Simplicity and ease of control	Lower performance than bipropellants
Tripropellant rocket	Three different propellants (usually hydrogen, hydrocarbon and liquid oxygen) are introduced into a combustion chamber in variable mixture ratios, or multiple engines are used with fixed propellant mixture ratios and throttled or shut down	Reduces take-off weight, since hydrogen is lighter; combines good thrust to weight with high average I_{sp}, improves payload for launching from Earth by a sizeable percentage	Similar issues to bipropellant, but with more plumbing, more R&D

Air-augmented rocket	Essentially a ramjet where intake air is compressed and burnt with the exhaust from a rocket	Mach 0 to Mach 4.5+ (can also run exoatmospheric), good efficiency at Mach 2 to 4	Similar efficiency to rockets at low speed or exoatmospheric, inlet difficulties, a relatively undeveloped and unexplored type, cooling difficulties, very noisy, thrust/weight ratio is similar to ramjets.
Turborocket	A combined cycle turbojet/rocket where an additional oxidizer such as oxygen is added to the airstream to increase maximum altitude	Very close to existing designs, operates in very high altitude, wide range of altitude and airspeed	Atmospheric airspeed limited to same range as turbojet engine, carrying oxidizer like LOX can be dangerous. Much heavier than simple rockets.
Precooled jet engine / LACE (combined cycle with rocket)	Intake air is chilled to very low temperatures at inlet before passing through a ramjet or turbojet engine. Can be combined with a rocket engine for orbital insertion.	Easily tested on ground. High thrust/weight ratios are possible (~14) together with good fuel efficiency over a wide range of airspeeds, mach 0-5.5+; this combination of efficiencies may permit launching to orbit, single stage, or very rapid intercontinental travel.	Exists only at the lab prototyping stage. Examples include RB545, SABRE, ATREX

Electrically powered

Type	Description	Advantages	Disadvantages
Resistojet rocket (electric heating)	A monopropellant is electrically heated by a filament for extra performance	Higher I_{sp} than monopropellant alone, about 40% higher.	Uses a lot of power and hence gives typically low thrust
Arcjet rocket (chemical burning aided by electrical discharge)	Similar to resistojet in concept but with inert propellant, except an arc is used	1600 seconds I_{sp}	Very low thrust and high power, performance is similar to Ion drive.

	which allows higher temperatures		
Pulsed plasma thruster (electric arc heating; emits plasma)	Plasma is used to erode a solid propellant	High I_{sp} , can be pulsed on and off for attitude control	Low energetic efficiency
Variable specific impulse magnetoplasma rocket	Microwave heated plasma with magnetic throat/nozzle	Variable I_{sp} from 1000 seconds to 10,000 seconds	similar thrust/weight ratio with ion drives (worse), thermal issues, as with ion drives very high power requirements for significant thrust, really needs advanced nuclear reactors, never flown, requires low temperatures for superconductors to work

Solar powered

The Solar thermal rocket would make use of solar power to directly heat reaction mass, and therefore does not require an electrical generator as most other forms of solar-powered propulsion do. A solar thermal rocket only has to carry the means of capturing solar energy, such as concentrators and mirrors. The heated propellant is fed through a conventional rocket nozzle to produce thrust. The engine thrust is directly related to the surface area of the solar collector and to the local intensity of the solar radiation and inversely proportional to the I_{sp}.

Type	Description	Advantages	Disadvantages
Solar thermal rocket	Propellant is heated by solar collector	Simple design. Using hydrogen propellant, 900 seconds of I_{sp} is comparable to Nuclear Thermal rocket, without the problems and complexity of controlling a fission reaction. Ability to productively utilize waste gaseous hydrogen—an inevitable byproduct of long-term liquid hydrogen storage in the radiative heat environment of space—for both orbital stationkeeping and attitude control.	Only useful once in space, as thrust is fairly low, but hydrogen has not been traditionally thought to be easily stored in space, otherwise moderate/low I_{sp} if higher–molecular-mass propellants are used. Using higher–molecular-weight propellants, for example water, lowers performance.

Beam powered

Type	Description	Advantages	Disadvantages
light beam powered rocket	Propellant is heated by light beam (often laser) aimed at vehicle from a distance, either directly or indirectly via heat exchanger	simple in principle, in principle very high exhaust speeds can be achieved	~1 MW of power per kg of payload is needed to achieve orbit, relatively high accelerations, lasers are blocked by clouds, fog, reflected laser light may be dangerous, pretty much needs hydrogen monopropellant for good performance which needs heavy tankage, some designs are limited to ~600 seconds due to reemission of light since propellant/heat exchanger gets white hot
microwave beam powered rocket	Propellant is heated by microwave beam aimed at vehicle from a distance	microwaves avoid reemission of energy, so ~900 seconds exhaust speeds might be achieveable	~1 MW of power per kg of payload is needed to achieve orbit, relatively high accelerations, microwaves are absorbed to a degree by rain, reflected microwaves may be dangerous, pretty much needs hydrogen monopropellant for good performance which needs heavy tankage, transmitter diameter is measured in kilometres to achieve a fine enough beam to hit a vehicle at up to 100 km.

Nuclear powered

Nuclear propulsion includes a wide variety of propulsion methods that use some form of nuclear reaction as their primary power source. Various types of nuclear propulsion have been proposed, and some of them tested, for spacecraft applications:

Type	Description	Advantages	Disadvantages
Radioisotope rocket/"Poodle thruster" (radioactive decay energy)	Heat from radioactive decay is used to heat hydrogen	about 700–800 seconds, almost no moving parts	low thrust/weight ratio.
Nuclear thermal rocket (nuclear fission energy)	propellant (typ. hydrogen) is passed through a nuclear reactor to heat to high temperature	I_{sp} can be high, perhaps 900 seconds or more, above unity thrust/weight ratio with some designs	Maximum temperature is limited by materials technology, some radioactive particles can be present in exhaust in some designs, nuclear reactor shielding is

			heavy, unlikely to be permitted from surface of the Earth, thrust/weight ratio is not high.
Gas core reactor rocket (nuclear fission energy)	Nuclear reaction using a gaseous state fission reactor in intimate contact with propellant	Very hot propellant, not limited by keeping reactor solid, I_{sp} between 1500 and 3000 seconds but with very high thrust	Difficulties in heating propellant without losing fissionables in exhaust, massive thermal issues particularly for nozzle/throat region, exhaust almost inherently highly radioactive. Nuclear lightbulb variants can contain fissionables, but cut I_{sp} in half.
Fission-fragment rocket (nuclear fission energy)	Fission products are directly exhausted to give thrust		Theoretical only at this point.
Fission sail (nuclear fission energy)	A sail material is coated with fissionable material on one side	No moving parts, works in deep space	Theoretical only at this point.
Nuclear salt-water rocket (nuclear fission energy)	Nuclear salts are held in solution, caused to react at nozzle	Very high I_{sp}, very high thrust	Thermal issues in nozzle, propellant could be unstable, highly radioactive exhaust. Theoretical only at this point.
Nuclear pulse propulsion (exploding fission/fusion bombs)	Shaped nuclear bombs are detonated behind vehicle and blast is caught by a 'pusher plate'	Very high I_{sp}, very high thrust/weight ratio, no show stoppers are known for this technology	Never been tested, pusher plate may throw off fragments due to shock, minimum size for nuclear bombs is still pretty big, expensive at small scales, nuclear treaty issues, fallout when used below Earth's magnetosphere.
Antimatter catalyzed nuclear pulse propulsion (fission and/or fusion energy)	Nuclear pulse propulsion with antimatter assist for smaller bombs	Smaller sized vehicle might be possible	Containment of antimatter, production of antimatter in macroscopic quantities isn't currently feasible. Theoretical only at this point.
Fusion rocket (nuclear fusion energy)	Fusion is used to heat propellant	Very high exhaust velocity	Largely beyond current state of the art.
Antimatter rocket	Antimatter	Extremely	Problems with antimatter

(annihilation energy)	annihilation heats propellant	energetic, very high theoretical exhaust velocity	production and handling; energy losses in neutrinos, gamma rays, muons; thermal issues. Theoretical only at this point

History of rocket engines

According to the writings of the Roman Aulus Gellius, in c. 400 BC, a Greek Pythagorean named Archytas, propelled a wooden bird along wires using steam. However, it would not appear to have been powerful enough to take off under its own thrust.

The *aeolipile* described in the first century BC (often known as *Hero's engine*) essentially consists of a steam rocket on a bearing. It was created almost two millennia before the Industrial Revolution but the principles behind it were not well understood, and its full potential was not realized for a millennium.

The availability of black powder to propel projectiles was a precursor to the development of the first solid rocket. Ninth Century Chinese Taoist alchemists discovered black powder in a search for the Elixir of life; this accidental discovery led to fire arrows which were the first rocket engines to leave the ground.

Rocket engines were also brought in use by Tippu Sultan, The king of Mysore. These rockets could be of various sizes, but usually consisted of a tube of soft hammered iron about 8" long and 1½ - 3" diameter, closed at one end and strapped to a shaft of bamboo about 4 ft. long. The iron tube acted as a combustion chamber and contained well packed black powder propellant. A rocket carrying about one pound of powder could travel almost 1,000 yards. These 'rockets', fitted with swords used to travel long distance, several meters above in air before coming down with swords edges facing the enemy. These rockets were used against British empire very effectively.

Slow development of this technology continued up to the later 20th Century, when the writings of Konstantin Tsiolkovsky first talked about liquid fuelled rocket engines.

These independently became a reality thanks to Robert Goddard. Goddard also used a De Laval nozzle for the first time on a rocket, doubling the thrust and multiplying up the efficiency by several times.

During the late 1930s, German scientists, such as Wernher von Braun and Hellmuth Walter, investigated installing liquid-fuelled rockets in military aircraft (Heinkel He 112, He 111, He 176 and Messerschmitt Me 163). The turbopump was first employed by German scientists in WWII. Until then cooling the nozzle was problematic, and the A4 ballistic missile used dilute alcohol for the fuel, which reduced the combustion temperature sufficiently.

Staged combustion (*Замкнутая схема*) was first proposed by Alexey Isaev in 1949. The first staged combustion engine was the S1.5400 used in the Soviet planetary rocket, designed by Melnikov, a former assistant to Isaev. About the same time (1959), Nikolai Kuznetsov began work on the closed cycle engine NK-9 for Korolev's orbital ICBM, GR-1. Kuznetsov later evolved that design into the NK-15 and NK-33 engines for the unsuccessful Lunar N1 rocket.

In the West, the first laboratory staged-combustion test engine was built in Germany in 1963, by Ludwig Boelkow.

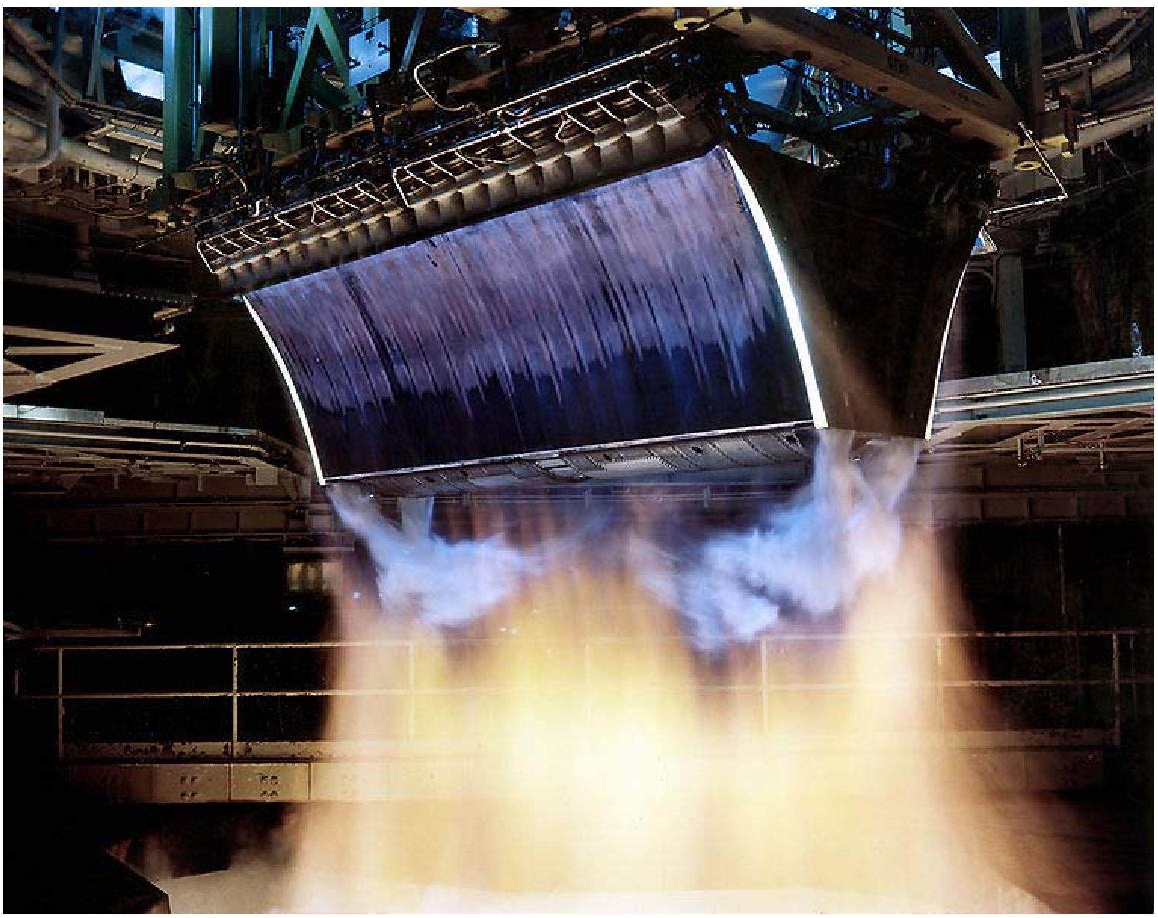
Hydrogen peroxide / kerosene fuelled engines such as the British Gamma of the 1950s used a closed-cycle process (arguably not *staged combustion*, but that's mostly a question of semantics) by catalytically decomposing the peroxide to drive turbines *before* combustion with the kerosene in the combustion chamber proper. This gave the efficiency advantages of staged combustion, whilst avoiding the major engineering problems.

Liquid hydrogen engines were first successfully developed in America, the RL-10 engine first flew in 1962. Hydrogen engines were used as part of the Project Apollo; the liquid hydrogen fuel giving a rather lower stage mass and thus reducing the overall size and cost of the vehicle.

The Space Shuttle's SSME is the highest ground-launched specific impulse rocket engine to fly.

Chapter 7

Aerospike Engine



XRS-2200 linear aerospike engine for the X-33 program being tested

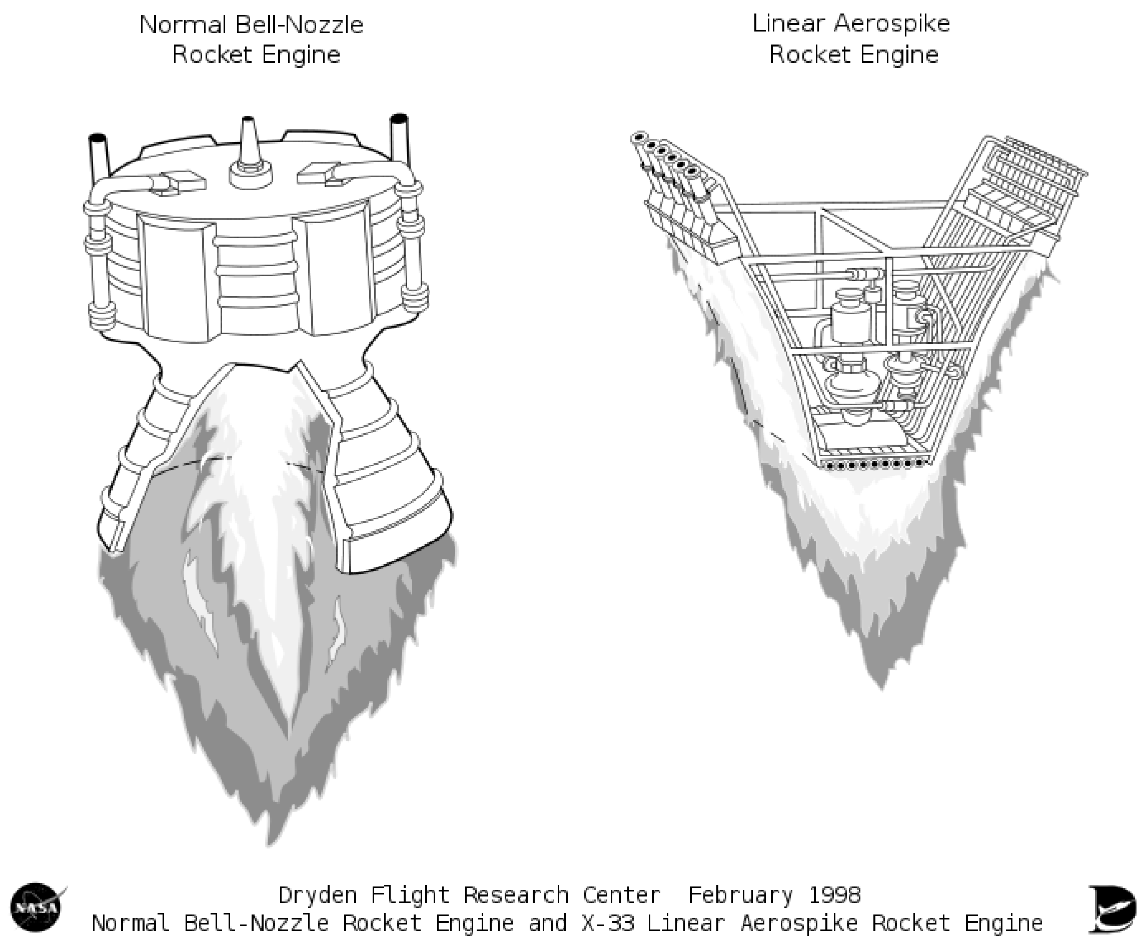
The **aerospike engine** is a type of rocket engine that maintains its aerodynamic efficiency across a wide range of altitudes through the use of an aerospike nozzle. It is a member of the class of altitude compensating nozzle engines. A vehicle with an aerospike engine uses 25–30% less fuel at low altitudes, where most missions have the greatest need for thrust. Aerospike engines have been studied for a number of years and are the baseline engines for many single-stage-to-orbit (SSTO) designs and were also a strong contender for the Space Shuttle main engine. However, no engine is in commercial production. The best large-scale aerospikes are still only in testing phases.

The terminology in the literature surrounding this subject is somewhat confused—the term *aerospike* was originally used for a truncated plug nozzle with a very rough conical taper and some gas injection, forming an "air spike" to help make up for the absence of the plug tail. However, frequently, a full-length plug nozzle is now called an aerospike.

Conventional designs

The basic concept of any engine bell is to efficiently direct the flow of exhaust gases from the rocket engine into one direction. The exhaust, a high-temperature mix of gases, has an effectively random momentum distribution, and if it is allowed to escape in that form, only a small part of the flow will be moving in the correct direction to contribute to forward thrust.

Principles



It is like and unlike a bell engine.

The aerospike attempts to avoid the problems above. Instead of firing the exhaust out of a small hole in the middle of a bell, it is fired along the outside edge of a wedge-shaped protrusion, the "spike". The spike forms one side of a virtual bell, with the other side being formed by the outside air—thus the "aerospike".

The idea behind the aerospike design is that at low altitude the ambient pressure compresses the wake against the nozzle. The recirculation in the base zone of the wedge can then raise the pressure there to near ambient. Since the pressure on top of the engine is ambient, this means that base gives no overall thrust (but it also means that this part of the nozzle doesn't *lose* thrust by forming a partial vacuum, thus the base part of the nozzle can be ignored at low altitude).

As the spacecraft climbs to higher altitudes, the air pressure holding the exhaust against the spike decreases, but the pressure on top of the engine decreases at the same time, so this is not detrimental. Further, although the base pressure drops, the recirculation zone keeps the pressure on the base up to a fraction of 1 bar, a pressure that is not balanced by the near vacuum on top of the engine; this difference in pressure gives extra thrust at altitude, contributing to the altitude compensating effect. This produces an effect like that of a bell that grows larger as air pressure falls, providing altitude compensation.

The disadvantages of aerospikes seem to be extra weight for the spike, and increased cooling requirements due to the extra heated area. Further, the larger cooled area can reduce performance below theoretical levels by reducing the pressure against the nozzle. Also, aerospikes work relatively poorly between Mach 1-3, where the airflow around the vehicle has reduced pressure, and this reduces the thrust.

Variations

Rocketdyne's J-2T-250k annular aerospike test firing.

Several versions of the design exist, differentiated by their shape. In the **toroidal aerospike** the spike is bowl-shaped with the exhaust exiting in a ring around the outer rim. In theory this requires an infinitely long spike for best efficiency, but by blowing a small amount of gas out the center of a shorter truncated spike, something similar can be achieved.

In the **linear aerospike** the spike consists of a tapered wedge-shaped plate, with exhaust exiting on either side at the "thick" end. This design has the advantage of being stackable, allowing several smaller engines to be placed in a row to make one larger engine while augmenting steering performance with the use of individual engine throttle control.

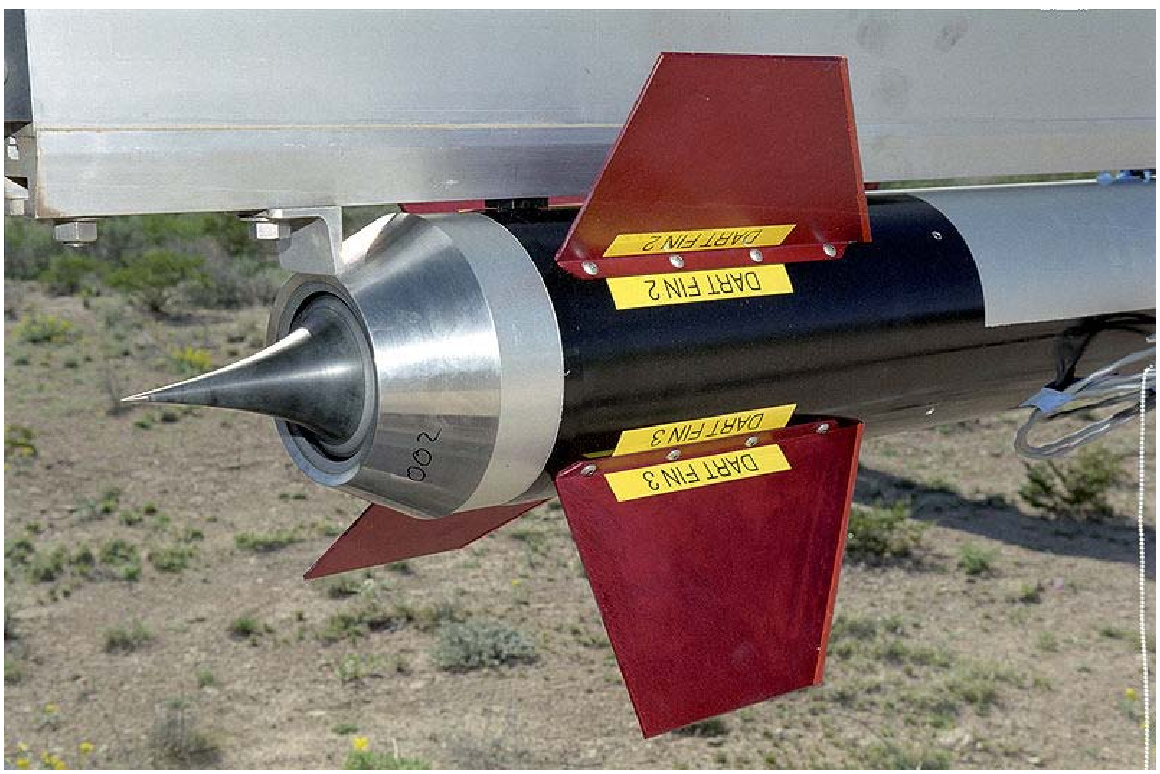
Performance

Rocketdyne conducted a lengthy series of tests in the 1960s on various designs. Later models of these engines were based on their highly reliable J-2 engine machinery and provided the same sort of thrust levels as the conventional engines they were based on; 200,000 lbf (890 kN) in the **J-2T-200k**, and 250,000 lbf (1.1 MN) in the **J-2T-250k** (the T refers to the toroidal combustion chamber). Thirty years later their work was dusted off again for use in NASA's X-33 project. In this case the slightly upgraded J-2S engine machinery was used with a linear spike, creating the **XRS-2200**. After more development and considerable testing, this project was cancelled when the X-33's composite fuel tanks repeatedly failed.

Three XRS-2200 engines were built during the X-33 program and underwent testing at NASA's Stennis Space Center. The single-engine tests were a success, but the program was halted before the testing for the 2-engine setup could be completed. The XRS-2200 produces 204,420 lbf (909,300 N) thrust with an I_{sp} of 339 seconds at sea level, and 266,230 lbf (1,184,300 N) thrust with an I_{sp} of 436.5 seconds in a vacuum.

The RS-2200 Linear Aerospike Engine was derived from the XRS-2200. The RS-2200 was to power the VentureStar single-stage-to-orbit vehicle. In the latest design, seven RS-2200s producing 542,000 pounds of thrust each would boost the VentureStar into low earth orbit. The development on the RS-2200 was formally halted in early 2001 when the X-33 program did not receive Space Launch Initiative funding. Lockheed Martin chose to not continue the VentureStar program without any funding support from NASA.

Although the cancelling of the X-33 program was a setback for aerospike engineering, it is not the end of the story. A milestone was achieved when a joint academic/industry team from California State University, Long Beach (CSULB) and Garvey Spacecraft Corporation successfully conducted a flight test of a liquid-propellant powered aerospike engine in the Mojave Desert on September 20, 2003. CSULB students had developed their Prospector 2 (P-2) rocket using a 1,000 lb_f (4.4 kN) LOX/ethanol aerospike engine. This work on aerospike engines is ongoing; Prospector-10, a ten-chamber aerospike engine, was test-fired June 25, 2008.

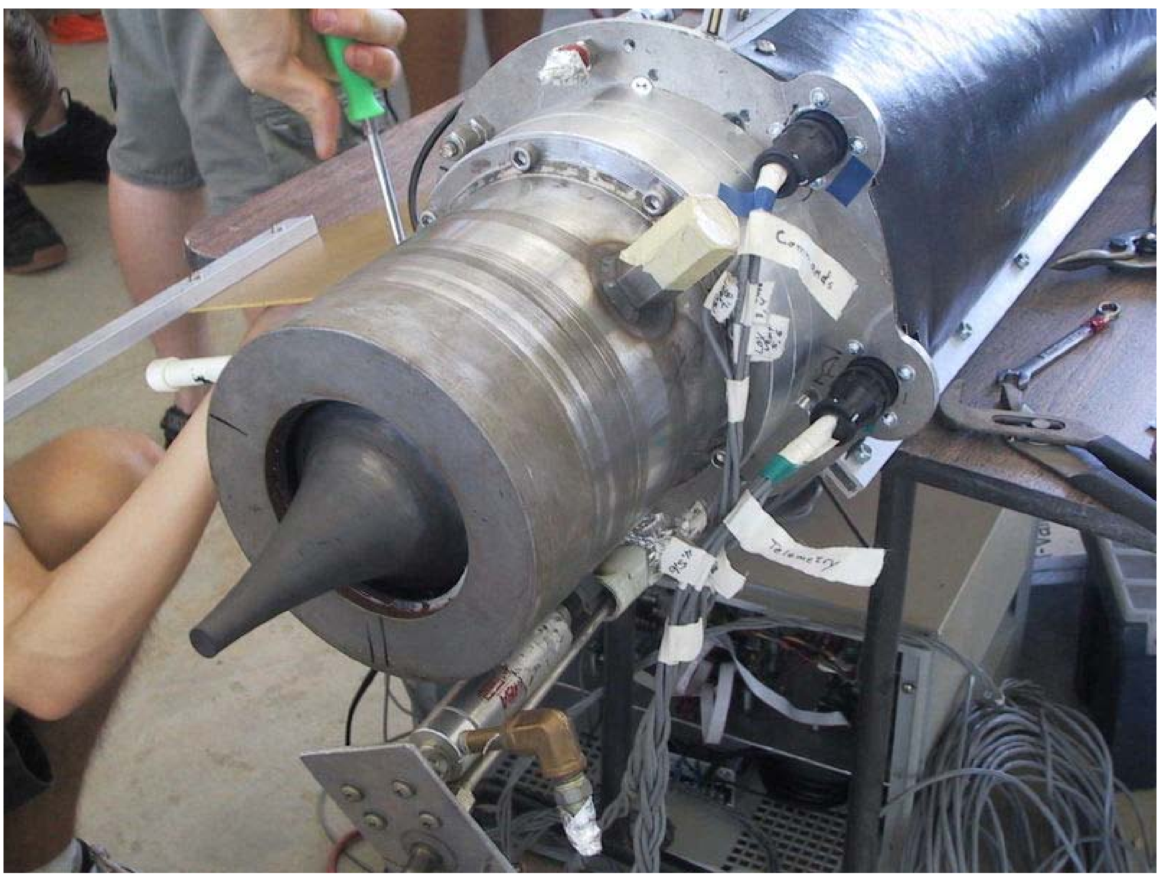


NASA's Toroidal aerospike nozzle

Further progress came in March 2004 when two successful tests were carried out at the NASA Dryden Flight Research Centre using small-scale rockets manufactured by Blacksky Corporation, based in Carlsbad, California. The aerospike nozzles and solid rocket motors were developed and built by Cesaroni Technology Incorporated. The two rockets were solid-fuel powered and fitted with non-truncated toroidal aerospike nozzles. They reached an apogee of 26,000 ft (7,900 m) and speeds of about Mach 1.5.

Small-scale aerospike engine development using a hybrid rocket propellant configuration has been ongoing by members of the Reaction Research Society.

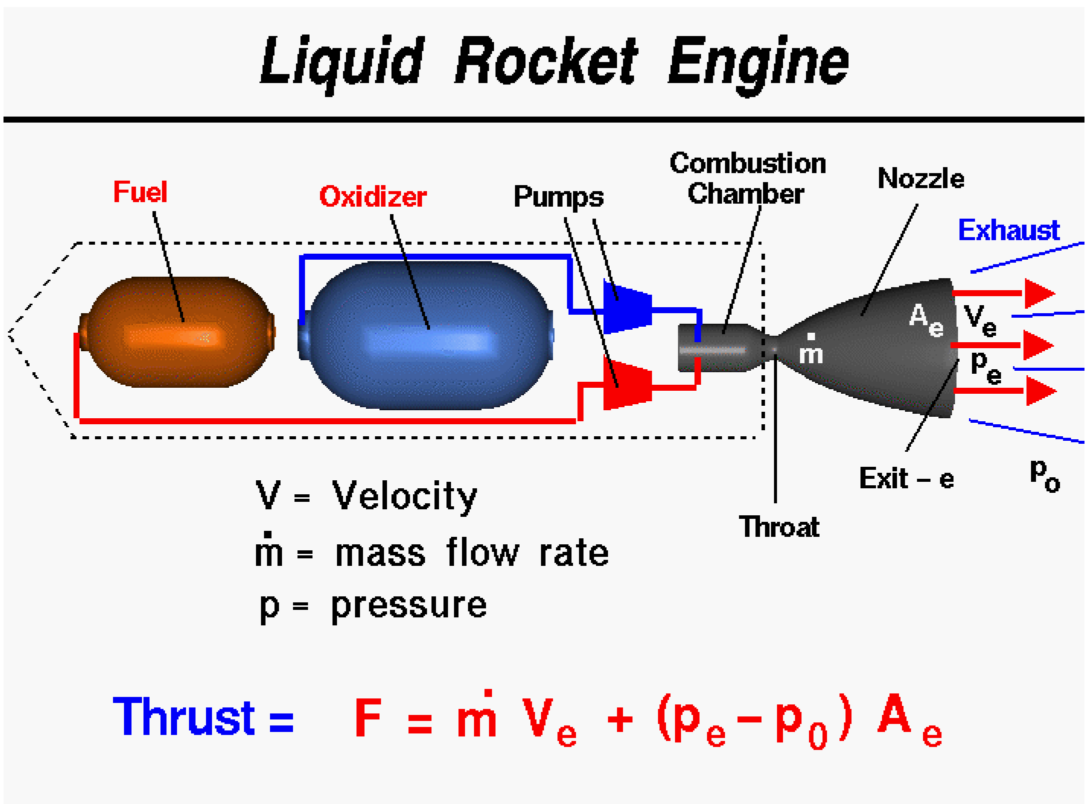
RS-2200 linear aerospike test firing



CSULB aerospike engine

Chapter 8

Liquid-Propellant Rocket



Schematic of a pumped bipropellant rocket

A **liquid-propellant rocket** or a **liquid rocket** is a rocket with an engine that uses propellants in liquid form. Liquids are desirable because their reasonably high density allows the volume of the propellant tanks to be relatively low, and it is possible to use lightweight pumps to pump the propellant from the tanks into the engines, which means that the propellants can be kept under low pressure. This permits the use of low mass propellant tanks, permitting a high mass ratio for the rocket.

Liquid rockets have been built as monopropellant rockets using a single type of propellant, bipropellant rockets using two types of propellant, or more exotic tripropellant rockets using three types of propellant. **Bipropellant liquid rockets** generally use one liquid fuel and one liquid oxidizer, such as liquid hydrogen or a hydrocarbon fuel such as RP-1, and liquid oxygen. This example also shows that liquid-propellant rockets sometimes use cryogenic rocket engines, where fuel or oxidizer are gases liquefied at very low temperatures.

Liquid propellants are also sometimes used in hybrid rockets, in which they are combined with a solid or gaseous propellant.

History

Robert H. Goddard, bundled against the cold New England weather of March 16, 1926, holds the launching frame of his most notable invention — the first liquid rocket.

The idea of liquid rocket as understood in the modern context first appears in the book *The Exploration of Cosmic Space by Means of Reaction Devices* , by Konstantin Tsiolkovsky. This seminal treatise on astronautics was published in 1903.

The only known claim to liquid propellant rocket engine experiments in the nineteenth century was made by a Peruvian scientist named Pedro Paulet. However, he did not immediately publish his work. In 1927 he wrote a letter to a newspaper in Lima, claiming he had experimented with a liquid rocket engine while he was a student in Paris three decades earlier. Historians of early rocketry experiments, among them Max Valier and Willy Ley, have given differing amounts of credence to Paulet's report. Paulet described laboratory tests of liquid rocket engines, but did not claim to have flown a liquid rocket.

The first flight of a vehicle powered by a liquid-rocket took place on March 16, 1926 at Auburn, Massachusetts, when American professor Robert H. Goddard launched a rocket which used liquid oxygen and gasoline as propellants. The rocket, which was dubbed "Nell", rose just 41 feet during a 2.5-second flight that ended in a cabbage field, but it was an important demonstration that liquid rockets were possible.

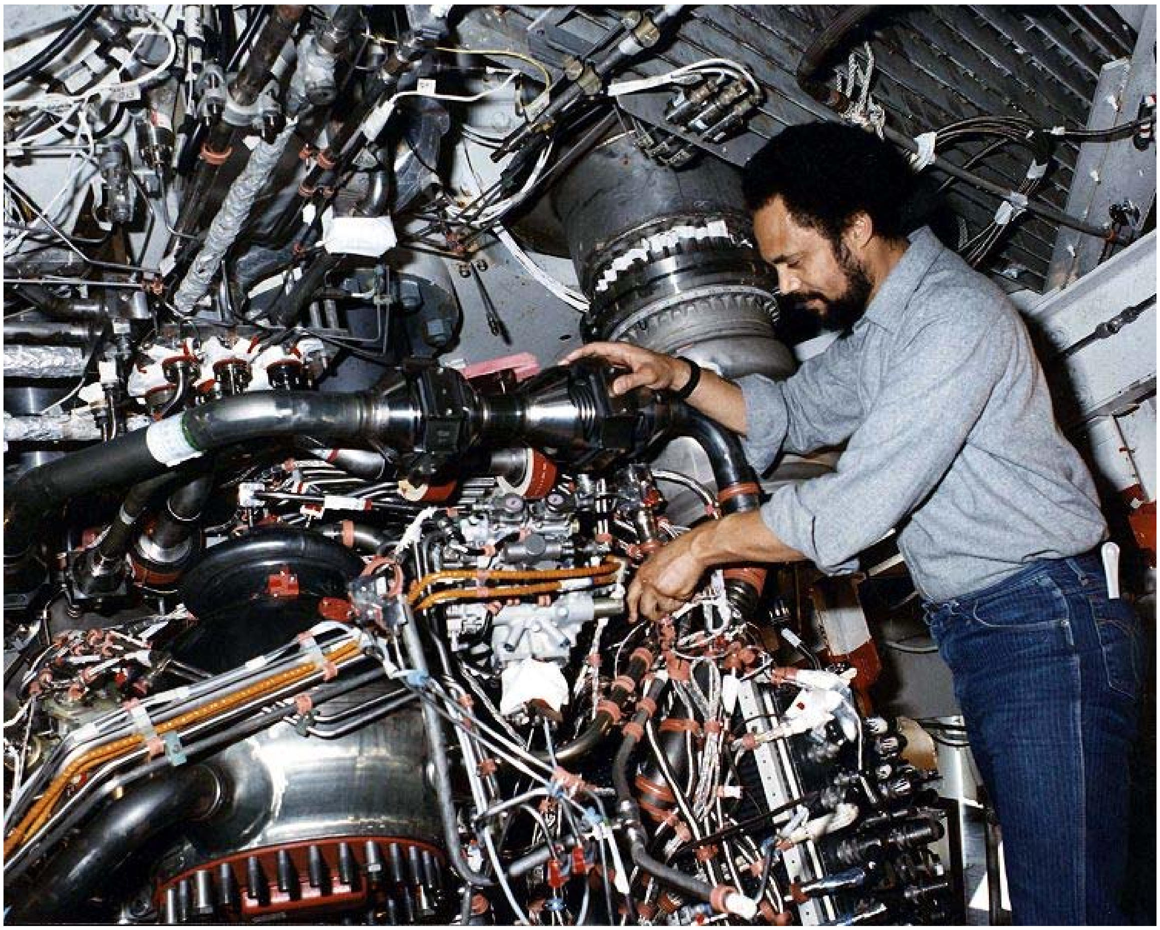
After Goddard's success, German engineers and scientists became enthralled with liquid fuel rockets and design better liquid fuel rockets testing them in the early 1930s in a field near Berlin.

Advantages of liquid rockets

- Liquid systems usually have the advantage of higher specific impulse (energy content).
- Tankage efficiency: Unlike gases, a typical liquid propellant has a density similar to water, approximately 0.7-1.4g/cm³ (except liquid hydrogen which has a much lower density), while requiring only relatively modest pressure to prevent vapourisation. This combination of density and low pressure permits very lightweight tankage; approximately 1% of the contents for dense propellants and around 10% for liquid hydrogen (due to its low density and the mass of the required insulation).
  For injection into the combustion chamber the propellant pressure needs to be greater than the chamber pressure at the injectors; this can be achieved with a pump. Suitable pumps usually use turbopumps due to their high power and lightweight, although reciprocating pumps have been employed in the past. Turbopumps are usually extremely lightweight and can give excellent performance; with an on-Earth weight well under 1% of the thrust. Indeed, overall rocket engine thrust to weight ratios including a turbopump have been as high as 133:1 with the Soviet NK-33 rocket engine.
  Alternatively, a heavy tank can be used, and the pump foregone; but the delta-v that the stage can achieve is often much lower due to the extra mass of the tankage reducing performance; but for high altitude or vacuum use the tankage mass can be acceptable.

- Liquid propellant rockets can be throttled in realtime, and have good control of mixture ratio; they can also be shut down, and, with a suitable ignition system or self-igniting propellant, restarted.
- A liquid rocket engine (LRE) can be tested prior to use, whereas for a solid rocket motor a rigorous quality management must be applied during manufacturing to insure high reliability.
- A LRE can be reused for several flights, like in the Space Shuttle.

Disadvantages of liquid rockets



Bipropellant liquid rockets are simple in concept but due to high temperatures and high speed moving parts, very complex in practice.

Use of liquid propellants can be associated with a number of issues:

- Because the propellant is a very large proportion of the mass of the vehicle, the center of mass shifts significantly rearward as the propellant is used; one will typically lose control of the vehicle if its center mass gets too close to the center of drag.

- When operated within an atmosphere, pressurization of the typically very thin-walled propellant tanks must guarantee positive gauge pressure at all times to avoid catastrophic collapse of the tank.
- Liquid propellants are subject to *slosh*, which has frequently led to loss of control of the vehicle. This can be controlled with slosh baffles in the tanks as well as judicious control laws in the guidance system.
- Liquid propellants often need ullage motors in zero-gravity or during staging to avoid sucking gas into engines at start up. They are also subject to vortexing within the tank, particularly towards the end of the burn, which can also result in gas being sucked into the engine or pump.
- Liquid propellants can leak, especially hydrogen, possibly leading to the formation of an explosive mixture.
- Turbopumps to pump liquid propellants are complex to design, and can suffer serious failure modes, such as overspeeding if they run dry or shedding fragments at high speed if metal particles from the manufacturing process enter the pump.
- Cryogenic propellants, such as liquid oxygen, freezes atmospheric water vapour into very hard crystals. This can damage or block seals and valves and can cause leaks and other failures. Avoiding this problem often requires lengthy *chilldown* procedures which attempt to remove as much of the vapour from the system as possible. Ice can also form on the outside of the tank, and later fall and damage the vehicle. External foam insulation can cause issues as shown by the Space Shuttle Columbia disaster. Non-cryogenic propellants do not cause such problems.
- Non-storable liquid rockets require considerable preparation immediately before launch. This makes them less practical than solid rockets for most weapon systems.

Propellants

Thousands of combinations of fuels and oxidizers have been tried over the years. Some of the more common and practical ones are:

- liquid oxygen (LOX, O_2) and liquid hydrogen (LH2, H_2) - Space Shuttle main engines, Ariane 5 main stage and the Ariane 5 ECA second stage, the first stage of the Delta IV, the upper stages of the Ares I, Saturn V, Saturn IB, and Saturn I as well as Centaur rocket stage, the first stage and second stage of the H-II, H-IIA, H-IIB
- liquid oxygen (LOX) and kerosene or RP-1 - Saturn V, Zenit rocket, R-7 Semyorka family of Soviet boosters which includes Soyuz, Delta, Saturn I, and Saturn IB first stages, Titan I and Atlas rockets
- liquid oxygen (LOX) and alcohol (ethanol, C_2H_5OH) - early liquid fueled rockets, like German (World War II) A-4, aka V-2, and Redstone
- liquid oxygen (LOX) and gasoline - Robert Goddard's first liquid-fuel rocket
- T-Stoff (80% hydrogen peroxide, H_2O_2 as the oxidizer) and C-Stoff (methanol, CH_3OH, and hydrazine hydrate, $N_2H_4 \cdot n(H_2O$ as the fuel) - Walter Werke HWK

109-509 engine used on Messerschmitt Me 163B Komet a rocket fighterplane of (World War II)

- nitric acid (HNO_3) and kerosene - Soviet Scud-A, aka SS-1
- inhibited red fuming nitric acid (IRFNA, $HNO_3 + N_2O_4$) and unsymmetric dimethyl hydrazine (UDMH, $(CH_3)_2N_2H_2$) Soviet Scud-B,-C,-D, aka SS-1-c,-d,-e
- nitric acid 73% with dinitrogen tetroxide 27% (=AK27) and kerosene/gasoline mixture - various Russian (USSR) cold-war ballistic missiles, Iran: Shahab-5, North Korea: Taepodong-2
- hydrogen peroxide and kerosene - UK (1970s) Black Arrow, USA Development (or study): BA-3200
- hydrazine (N_2H_4) and red fuming nitric acid - Nike Ajax Antiaircraft Rocket
- Aerozine 50 and dinitrogen tetroxide - Titans 2–4, Apollo lunar module, Apollo service module, interplanatary probes (Such as Voyager 1 and Voyager 2)
- Unsymmetric dimethylhydrazine (UDMH) and dinitrogen tetroxide - Proton rocket and various Soviet rockets
- monomethylhydrazine (MMH, $(CH_3)HN_2H_2$) and dinitrogen tetroxide - Space Shuttle orbiter's Orbital maneuvering system (OMS) engines and Reaction control system (RCS) thrusters.

One of the most efficient mixtures, oxygen and hydrogen, suffers from the extremely low temperatures required for storing hydrogen and oxygen as liquids (around 20 K or −253 °C)) and low fuel density (70 kg/m^3), necessitating large and heavy tanks. The use of lightweight foam to insulate the cryogenic tanks caused problems for the Space Shuttle Columbia's STS-107 mission, as a piece broke loose, damaged its wing and caused it to break up and be destroyed on atmospheric reentry.

For storable ICBMs and interplanetary spacecraft, storing cryogenic propellants over extended periods is awkward and expensive. Because of this, mixtures of hydrazine and its derivatives in combination with nitrogen oxides are generally used for such rockets. Hydrazine has its own disadvantages, being a very caustic and volatile chemical as well as being toxic and carcinogenic. Consequently, hybrid rockets have recently been the vehicle of choice for low-budget private and academic developments in aerospace technology. Also the RP-1/LOX combination has become a popular choice for reliable and cost-sensitive commercial spaceflight applications.

Injectors

The injector implementation in liquid rockets determines the percentage of the theoretical performance of the nozzle that can be achieved. A poor injector performance causes unburnt propellant to leave the engine, giving extremely poor efficiency.

Additionally, injectors are also usually key in reducing thermal loads on the nozzle; by increasing the proportion of fuel around the edge of the chamber, this gives much lower temperatures on the walls of the nozzle.

Types of injectors

Injectors can be as simple as a number of small diameter holes arranged in carefully constructed patterns through which the fuel and oxidiser travel. The speed of the flow is determined by the square root of the pressure drop across the injectors, the shape of the hole and other details such as the density of the propellant.

The first injectors used on the V-2 created parallel jets of fuel and oxidizer which then combusted in the chamber. This gave quite poor efficiency.

Injectors today classically consist of a number of small holes which aim jets of fuel and oxidiser so that they collide at a point in space a short distance away from the injector plate. This helps to break the flow up into small droplets that burn more easily.

The main type of injectors are

- Shower Head type
- Self Impinging doublet type
- Cross impinging triplet type
- Centrifugal or Swirling type

Other injector types include the pintle injector, which potentially permits good mixture control over a wide range of flow rates. The pintle injector was used on the Apollo Lunar Module engines and the Merlin and Kestrel engines designed by SpaceX.

The Space Shuttle Main Engine uses a system of fluted posts, which use heated hydrogen from the preburner to vaporize the liquid oxygen flowing through the center of the posts and this improves the rate and stability of the combustion process; previous engines such as the F-1 used for the Apollo program had significant issues with oscillations that led to destruction of the engines, but this was not a problem in the SSME due to this design detail.

Valentin Glushko invented the centrifugal injector in the early 1930s, and it has been almost universally used in Russian engines. Rotational motion is applied to the liquid (and sometimes the two propellants are mixed), then it is expelled through a small hole, where it forms a cone-shaped sheet that rapidly atomizes. Goddard's first liquid fuel engine used a single impinging injector. German scientists in WWII experimented with impinging injectors on flat plates, used successfully in the Wasserfall missile.

Combustion stability

To avoid instabilities such as *chugging* which is a relatively low speed oscillation the engine must be designed with enough pressure drop across the injectors to render the flow largely independent of the chamber pressure. This is normally achieved by using at least 20% of the chamber pressure across the injectors.

Nevertheless, particularly in larger engines, a high speed combustion oscillation is easily triggered, and these are not well understood. These high speed oscillations tend to disrupt the gas side boundary layer of the engine, and this can cause the cooling system to rapidly fail, destroying the engine. These kinds of oscillations are much more common on large engines, and plagued the development of the Saturn V, but were finally overcome.

Some combustion chambers, such as the SSME uses Helmholtz resonators as damping mechanisms to stop particular resonant frequencies from growing.

To prevent these issues the SSME injector design instead went to a lot of effort to vapourise the propellant prior to injection into the combustion chamber. Although many other features were used to ensure that instabilities could not occur, later research showed that these other features were unnecessary, and the gas phase combustion worked reliably.

Testing for stability often involves the use of small explosives. These are detonated within the chamber during operation, and causes an impulsive excitation. By examining the pressure trace of the chamber to determine how quickly the effects of the disturbance die away, it is possible to estimate the stability and redesign features of the chamber if required.

Engine cycles

For liquid propellant rockets four different ways of powering the injection of the propellant into the chamber are in common use.

Generally speaking, pumping losses are small compared to the heat energy lost in the nozzle. For atmospheric use, high pressure engine cycles are desirable as it improves the efficiency of the nozzle. For vacuum use, pumps aren't usually required.

- pressure fed cycle- the propellants are forced in from pressurised (relatively heavy) tanks. The heavy tanks mean that a relatively low pressure is optimal. The pressurant used is frequently helium due to its lack of reactivity.
- expander cycle - cryogenic fuel is used to cool the walls of the combustion chamber and nozzle. Absorbed heat vaporizes and expands the fuel which is then used to drive the turbopumps before it enters the combustion chamber. No heat or propellant is lost, so efficiency is very high. Pump power and combustion pressure are constrained by available heat transfer.
- gas generator cycle - a small percentage of the propellants are burnt in a preburner to power a turbopump and then exhausted through a separate nozzle, or low down on the main one. This usually gives a small reduction in performance.
- staged combustion cycle - the high pressure outlet from the turbopump is fed back to power a burner which then powers the turbopump in a self starting cycle. The still high pressure exhaust from the turbine is then fed directly into the main chamber, thus essentially all the energy goes through the nozzle, giving no pumping losses at all, and permitting very high pressures.

Cooling

Injectors are commonly laid out so that a fuel-rich layer is created at the combustion chamber wall. This reduces the temperature there, and downstream to the throat and even into the nozzle and permits the combustion chamber to be run at higher pressure, which permits a higher expansion ratio nozzle to be used which gives a higher ISP and better system performance. A liquid rocket engine often employs regenerative cooling, which uses the fuel or the oxidiser to cool the chamber and nozzle.

Ignition

Ignition can be performed in many ways, but perhaps more so with liquid propellants than other rockets a consistent and significant ignitions source is required; a delay of ignition (in some cases as small as) a few tens of milliseconds can cause overpressure of the chamber due to excess propellant. A hard start can even cause an engine to explode.

Generally, ignition systems try to apply flames across the injector surface, with a mass flow of approximately 1% of the full mass flow of the chamber.

Safety interlocks are sometimes used to ensure the presence of an ignition source before the main valves open; however reliability of the interlocks can in some cases be lower than the ignition system. Thus it depends on whether the system must fail safe, or whether overall mission success is more important. Interlocks are rarely used for upper, unmanned stages where failure of the interlock would cause loss of mission, but are present on the SSME, to shut the engines down prior to liftoff of the Space Shuttle. In addition, detection of successful ignition of the igniter is surprisingly difficult, some systems use thin wires that are cut by the flames, pressure sensors have also seen some use.

Methods of ignition include pyrotechnic, electrical (spark or hot wire), and chemical. Hypergolic propellants have the advantage of self igniting, reliably and with less chance of hard starts. In the 1940s, the Russians began to start engines with hypergolic fuel, then switch over to the primary propellants after ignition. This was also used on the American F-1 rocket engine on the Apollo program.

Chapter 9

LE Engines: LE-5 & LE-7

LE-5

LE-5

The LE-5 liquid rocket engine and its derivative models were developed in Japan to meet the need for an upper stage propulsion system for the H-I and H-II series of launch vehicles. It is a bipropellant design, using LH2 and LOX. Primary design and production work was carried out by Mitsubishi Heavy Industries. In terms of liquid rockets, it is a fairly small engine, both in size and thrust output, being in the 20,000 lbf (89,000 N) and the more recent models the 30,000 lbf (130 kN) thrust class. The motor is capable of multiple restarts, due to a spark ignition system as opposed to the single use pyrotechnic or hypergolic igniters commonly used on some contemporary engines. Though rated for up to 16 starts and 40+ minutes of firing time, on the H-II the engine is considered expendable, being used for one flight and jettisoned. It is sometimes started only once for a nine-minute burn, but in missions to GTO the engine is often fired a second time to inject the payload into the higher orbit after a temporary low Earth orbit has been established.

LE-5

The original LE-5 was built as a third stage engine for the H-I launch vehicle. It used a fairly conventional gas generator cycle.

LE-5A

The LE-5A was a heavily redesigned version of the LE-5 intended for use on the new H-II launch vehicle's second stage. The major difference is that the operation of the engine was switched from the gas generator to expander bleed cycle. The LE-5A was the first expander bleed cycle engine to be put into operational service. Cryogenic liquid hydrogen fuel for the cycle is drawn through tubes and passages in both the engine's nozzle and combustion chamber where the hydrogen heats up incredibly while simultaneously cooling those components. The heating of the initially cold fuel causes it to become significantly pressurized and it is utilized to drive the turbine for the propellant pumps.

LE-5B

The LE-5B was a farther modified version of the LE-5A. The changes focused on lowering the per-unit cost of the engine while continuing to increase reliability. The modifications veered towards simplification and cheaper production where possible at the cost of actually lowering the specific impulse to 447 seconds, the lowest of all three models. However, it produced the highest thrust of the three and was significantly cheaper. The primary change from the 5A model was that the 5B's expander bleed system circulated fuel around only the combustion chamber as opposed to the both the chamber and the nozzle in the 5A. Alterations to the combustion chamber cooling passages and constituent materials were made with special emphasis on effective heat transfer to allow this method to be successful.

Specifications

LE-5 Model Specifications

LE-5 Model	*(units)*	**LE-5**	**LE-5A**	**LE-5B**
Operational Cycle	-	Gas Generator	Expander Bleed (Nozzle/Chamber)	Expander Bleed (Chamber)

Rated Thrust	kN (lbs)	102.9 (23,100)	121.5 (27,300)	137.2 (30,800)
Mixture Ratio	Oxidizer to Fuel	5.5	5	5
Expansion Ratio	-	140	130	110
Specific Impulse (Isp)	Seconds	450	452	447
Chamber Pressure	MPa (PSI)	3.65 (529)	3.98 (577)	3.58 (519)
LH2 Rotational Speed	rpm	50,000	51,000	52,000
LOX Rotational Speed	rpm	16,000	17,000	18,000
Length	m (ft)	2.68 (8.84)	2.69 (8.88)	2.79 (9.21)
Weight	kg (lbs)	255 (562)	248 (547)	285 (628)

LE-7

LE-7

LE-7 (Nagoya City Science Museum, 2006

Country of origin Japan

Designed by	JAXA
Manufacturer	Mitsubishi Heavy Industries
Application	Booster
Status	Succeeded by LE-7A upgrade
Liquid-fuelled engine	
Propellant	LOX / LH2
Cycle	Staged combustion
Configuration	
Chamber	1
Nozzle Area ratio	52:1
Performance	
Thrust(Vac)	1,078 kN (242,000 lbf)
Thrust(SL)	843.5 kN (189,600 lbf)
Thrust-to-weight ratio	64.13
Chamber pressure	12.7 MPa (1,840 psi)
I_{sp}(Vac)	446 s
I_{sp}(SL)	349 s
Dimensions	
Length	3.4 m
Dry weight	1,714 kg (3,780 lb)

LE-7A (Marunouchi JAXAi, 2009)

The **LE-7** and its succeeding upgrade model the LE-7A are staged combustion cycle LH2/LOX liquid rocket engines produced in Japan for the H-II series of launch vehicles. Design and production work was all done domestically in Japan, the first major (main/first-stage) liquid rocket engine with that claim, in a collaborative effort from the National Space Development Agency (NASDA), Aerospace Engineering Laboratory (NAL), Mitsubishi Heavy Industries, and Ishikawajima-Harima. NASDA and NAL have since been integrated into JAXA. However, a large part of the work was contracted to Mitsubishi, with Ishikawajima-Harima providing turbomachinery, and the engine is often referred to as the **Mitsubishi LE-7(A)**.

LE-7

The original LE-7 was designed to be a high efficiency, medium-sized motor with sufficient thrust for use on the H-II, and classified as expendable since the engine was non-recoverable after launch.

H-II Flight 8, only operational LE-7 failure

The fuel turbopump had an issue using the originally designed inducer (a propeller-like axial pump used to raise the inlet pressure of the propellant ahead of the main turbopumps to prevent cavitation) where the inducer would itself begin to cavitate and cause an imbalance resulting in excessive vibration. A comprehensive post-flight analysis of the unsuccessful 8th H-II launch, including a deep ocean retrieval of the wreckage, determined that fatigue due to this vibration was the cause of premature engine failure.

LE-7A

The LE-7A is an upgraded model from the LE-7 rocket engine. Basic design is unchanged from the original model. The 7A had additional engineering effort placed on cost cutting, reliability, and performance developments. The renovation was undertaken to mate it with the likewise improved H-IIA launch vehicle, with the common goal being a more reliable, more powerful and flexible, and more cost effective launch system.

Changes / improvements

Specific emphasis was placed on reducing or the amount of required welding by allowing for more machined or cast components, and to simplify as many of the remaining welds as possible. This resulted in a substantial rework of the pipe routing (which makes the outward appearance of the two models considerably different). To combat the fuel inducer complications described above, the fuel inducer was redesigned for the 7A. The oxidizer inducer was also redesigned, but this was primarily to poor performance at low inlet pressures as opposed to reliability concerns. The fuel turbopump itself was also the subject of various durability enhancements. Additionally the combustion chamber/injector assembly underwent a number of small changes, like decreasing the number of injector elements, to reduce machining complexity (and thus cost) and improve reliability. While these changes overall resulted in a drop in maximum specific impulse to 440 seconds (basically making the engine less fuel efficient), the trade off for lower cost and enhanced reliability was considered acceptable.

New nozzle design (side-loading problem)

For the new engine model, a nozzle extension was designed that could be added to the base of the new standard “short” nozzle when extra performance was required. But when the engine was fitted with the nozzle extension, the 7A encountered a new problem with unprecedented side-loads and irregular heating on the nozzle strong enough to damage the gimbal actuators and regenerative cooling tubes during startup. Meticulous computational fluid dynamics (CFD) work was able to sufficiently replicate and trace the dangerous transient loading and a new one-piece “long” nozzle with full regenerative cooling (as opposed to the original short nozzle with a separate film-cooled extension) was designed to mitigate the problem. Before this new nozzle was ready, some H-IIA’s were launched using only the short nozzle. The 7A no longer uses a separate nozzle extension in any configuration.

Use on H-IIB

The new H-IIB launch vehicles uses two LE-7A engines in its first stage.

LE-7A specifications

- Operational Cycle: staged combustion
- Fuel: hydrogen
- Oxidizer: liquid oxygen
- Mixture ratio (oxidizer to fuel): 5.90
- Short nozzle:
    - Rated thrust (sea level): 843 kN (190,000 lbf)
    - Rated thrust (vacuum): 1,074 kN (241,000 lbf)
    - Specific impulse (sea level):
    - Specific impulse (vacuum): 429 seconds
- Long nozzle:
    - Rated thrust (sea level): 870 kN (200,000 lbf)
    - Rated thrust (vacuum): 1,098 kN (247,000 lbf)
    - Specific impulse (sea level): 338 seconds
    - Specific impulse (vacuum): 440 seconds
- Dry mass: 1,800 kg (4,000 lb)
- Length:
    - short nozzle = 3.2 m
    - long nozzle = 3.7 m
- Throttle capability: 72-100%
- Thrust-to-weight: 65.9
- Nozzle area ratio: 51.9:1
- Combustion chamber pressure: 12.0 MPa (1,740 psi)
- Liquid hydrogen turbopump: 41,900 rpm
- Liquid oxygen turbopump: 18,300 rpm

Chapter 10

RD-170 & RD-180 (Rocket Engine)

RD-170

RD-170

RD-170 rocket engine model on exhibition in Saint Petersburg's Museum of Space and Missile Technology.

Country of origin	Soviet Union/Russian Federation
Manufacturer	NPO Energomash
Application	Main engine
Liquid-fuelled engine	
Propellant	LOX / RP-1 (Soviet/Russian equivalent)
Cycle	Staged combustion

Nozzle Area ratio	36.87
Performance	
Thrust(Vac)	1,773,000 lbf (7.887 MN)
Thrust(SL)	1,697,300 lbf (7.550 MN)
Thrust-to-weight ratio	82
Chamber pressure	245 bar
I_{sp}(Vac)	338 s (3,315 N·s/kg)
I_{sp}(SL)	309 s (3,030 N·s/kg)
Burn time	150 s

The **RD-170** (**РД-170, Ракетный Двигатель-170, Rocket Engine-170**) is the world's most powerful liquid-fuel rocket engine, designed and produced in the USSR by NPO Energomash for use with Energia launch vehicle. This bipropellant engine burns the Russian equivalent of RP-1 fuel and LOX oxidizer in four combustion chambers supplied by a single turbo pump according to a staged combustion cycle and is rated at ~20,000,000 HP.

Shared turbopump

Several Soviet and Russian rocket engines use the approach of clustering small combustion chambers around a single turbine and pump. During the early 1950s, many Soviet engine designers, including Glushko, faced problems of combustion instability, while designing bigger thrust chambers. At that time they solved the problem by using a cluster of smaller thrust chambers.

Variants

The RD-170 is now out of production, but it forms the basis for a family of modern rocket engines.

RD-171

RD-171 model

One RD-170 variant, the RD-171, is currently used in the Zenit rocket. While the RD-170 had nozzles which swiveled on only one axis, the RD-171 swivels on two axes. Models called the RD-172 and RD-173 were proposed upgrades providing additional thrust, but they were never built.

RD-180

This variant uses only 2 combustion chambers instead of the 4 of the RD-170. The RD-180 used on the Atlas V, replaced the three engines used on early Atlas rockets with a single engine and achieved significant payload and performance gains. This engine has also been chosen to be the main propulsion system for the first stage of the new Russian Rus-M rocket.

RD-191

Yet another variant, the RD-191, will be used in the Russian Angara rocket, which is currently under development.

RD-151

The RD-151 is the RD-191 with thrust reduced to 170 tonnes. This engine was fire-tested on July 30, 2009. The first flight test of this engine was conducted on August 25, 2009 as part of the first launch of South Korean Naro-1 rocket. The first stage of the Naro-1 rocket is made of the universal rocket module (URM) from the Angara rocket.

Specifications

- 4 combustion chambers, 4 nozzles
- 1 set of turbines and pumps - Turbine produces approximately 257,000 hp (192 MW); equivalent to the power output of 3 nuclear powered icebreakers
- Ignition: Hypergolic
- Vacuum thrust of 1,773,000 lbf (7,887 kN)
- Vacuum Isp of 338 s (3,315 N·s/kg)
- Sea Level Isp of 309 s (3,030 N·s/kg)
- Weight: 9,750 kg (21,500 lb)
- Thrust to weight ratio: 82

RD-180

RD-180 (РД-180)	
 RD-180 test firing, November 4, 1998 at the Marshall Space Flight Center Advanced Engine Test Facility.	
Country of origin	Russia
Date	1999
Designed by	NPO Energomash
Manufacturer	NPO Energomash
Application	Booster
Predecessor	RD-170
Status	In use
Liquid-fuelled engine	
Propellant	LOX / RP-1
Cycle	Staged combustion
Configuration	
Chamber	2
Nozzle Area ratio	36.87
Performance	
Thrust(Vac)	933,400 lbf (4.15 MN)
Thrust(SL)	860,568 lbf (3,83 MN)
Thrust-to-weight ratio	78.44
Chamber pressure	3,868 psia (26.7 MPa, 266.8 bar)
I_{sp}(Vac)	338 sec (3,313 N·s/kg)
I_{sp}(SL)	311 sec (3,053 N·s/kg)

Dimensions	
Length	140 in (3.56 m)
Diameter	124 in (3.15 m)
Dry weight	12,081 lb (5,480 kg)

The **RD-180** (**РД-180, Ракетный Двигатель-180, Rocket Engine-180**) is a Russian dual-combustion chamber, dual nozzle, rocket engine, derived from the RD-170 used in Soviet Zenit rockets.

Specifications

The combustion chambers of the RD-180 share a single turbopump unit, much like in its predecessor, the four-chambered RD-170. The RD-180 is fueled by a kerosene / LOX mixture and uses an extremely efficient, high-pressure staged combustion cycle. The engine runs with an oxidizer to fuel ratio of 2.72 and (like its progenitor the RD-170) is unique in that it employs a LOX-rich preburner, unlike typical fuel rich US designs. The thermodynamics of the cycle allow a LOX-rich preburner to be more powerful per unit weight, but with the drawback that high pressure, high temperature gaseous oxygen must be transported throughout the engine. The movements of the engine nozzles are controlled by four hydraulic actuators. Another unique feature of RD-180 is its capability of being throttled from 40% to 100% of rated thrust.

Prospective use

RD-180 is planned to be used with a new family of Rus-M Russian space launch vehicles, currently being proposed by Roskosmos contractors.

RD-180 licensed for use in US

During the early 90s General Dynamics Space Systems Division (later purchased by Lockheed Martin) acquired the rights to use the RD-180 in the Evolved Expendable Launch Vehicle (EELV) and the Atlas program. As these programs were conceived to support United States government launches as well as commercial launches, it was also arranged for the RD-180 to be co-produced by Pratt & Whitney. However all production to date has taken place in Russia. The engine is currently sold by a joint venture between the Russian developer and producer of the engine NPO Energomash and Pratt & Whitney, called RD AMROSS.

The RD-180 was first deployed on the Atlas IIA-R vehicle, which was the Atlas IIA vehicle with the Russian (hence the R) engine replacing the previous main engine. This vehicle was later renamed the Atlas III. An additional development program was undertaken to certify the engine for use on the modular Common Core Booster primary stage of the Atlas V rocket.

Jerry Grey, a consultant to the American Institute of Aeronautics and Astronautics and Universities Space Research Association and a former professor of aerospace engineering at Princeton University, suggests using the RD-180 for a prospective NASA heavy-lift launch vehicle. For those who might be concerned about too much reliance on Russia, he points out that RD Amross "is very close to producing a U.S.-built version of the RD-180, and with some infusion of NASA funding could be manufacturing that engine (and perhaps even a 1,700,000 pounds-force (7.6 MN) thrust equivalent of the RD-170) in a few years."

Despite the availability of necessary documentation and legal rights for producing RD-180 in the United States, NASA is considering to develop an indigenous core stage engine that would be "capable of generating high levels of thrust approximately equal to or exceeding the performance of the Russian-built engine." NASA intends to produce a fully operational engine by 2020, or sooner if it can establish a partnership with the U.S. Defense Department.

Chapter 11

Rocket Engine Test Facility & Vulcain

Rocket Engine Test Facility

Rocket firing at the WSTF

A **rocket engine test facility** is a location where rocket engines may be tested on the ground, under controlled conditions. A ground test program is generally required before the engine is certified for flight. Ground testing is very inexpensive in comparison to the cost of risking an entire mission or the lives of a flight crew.

The test conditions available are usually described as *sea level ambient* or *altitude*. Sea level testing is useful for evaluations of start characteristics for rockets launched from the ground. However, sea level testing does not provide a true simulation of the majority of the operating environment of the rocket. Better simulations are provided by altitude test facilities.

Sea level tests

Sea Level engine test stand at the SSC

The advantage of sea level static testing is simplicity. The facility must restrain the rocket and direct the rocket exhaust safely toward the open atmosphere. Structural integrity, system operations, and sea level thrust can be measured and verified. However, rockets are primarily intended for operations in very thin or no atmosphere. Systems that work well on the ground may behave very differently in space.

A typical sea level test stand may be designed to restrain the rocket engine in either a horizontal or vertical position. Liquid rocket engines are usually fired in a vertical position because the propellant pump intakes are designed to draw fuel from the bottoms of the fuel tanks. The effect of the propellant weight on the thrust measurement system (TMS) must be accounted for as the engine is firing. The rocket exhaust is directed into a flame bucket or trench. The flame trench is designed to redirect the hot exhaust to a safe direction and is protected by a water deluge system that both cools the exhaust and also reduces the sound pressure level (loudness). The sound pressure level of large rocket engines has been measured at greater than 200 decibels — one of the loudest man-made sounds on earth.

Solid rocket engines may be fired in either a vertical or horizontal orientation. The thrust measurement system does not need to account for the changing weight of the rocket in a horizontal position. The associated flame trench need not be so sturdy as with a vertical test stand, however a water system may be less effective at reducing the sound pressure level.

All test stands require safety provisions to protect against the destructive potential of an unplanned engine detonation. The safety provisions generally include siting the stand some minimum distance from inhabited areas or other critical facilities, placing the stand behind a thick concrete blast wall or earthen berm, and using some form of inerting system (either gaseous nitrogen or helium) to eliminate the buildup of explosive mixtures.

Altitude tests

The advantage of altitude testing is to obtain a better simulation of the rocket's operating environment. Air pressure decreases with increasing altitude. Effects of the lower air pressure include higher rocket thrust and lower heat transfer.

An altitude facility is much more complex than a sea level facility. The rocket is installed inside an enclosed chamber which is evacuated to a minimum pressure level before rocket firing. A typical chamber operating pressure of 0.16 psia (equivalent to an altitude of 100,000 feet) is established inside the chamber by some form of mechanical pumping. Mechanical pumping is typically provided by steam ejector/diffusers. If the products of combustion from the rocket firing include flammable or explosive materials, the chamber must be inerted, typically with gaseous nitrogen (GN2). The inerting process prevents build-up of potentially explosive materials inside the chamber or exhaust ducting.

Rocket ground test facilities

Test facilities in the United States

- ATK Promontory Point, Utah
- Rocketdyne Santa Susana Field Laboratory (Closed)
- NASA
    - Rocket Engine Test Facility
    - Marshall Space Flight Center
    - Plum Brook Station
    - White Sands Test Facility (WSTF)
    - John C. Stennis Space Center
- USAF Arnold Engineering Development Center

- New Mexico Tech Energetic Materials and Testing Research Center EMRTC Rocket Engine Test Site
- Air Force Research Laboratory at Edwards AFB, CA
- Pratt & Whitney Space Propulsion test facility
- SpaceX Test Facility, McGregor, TX

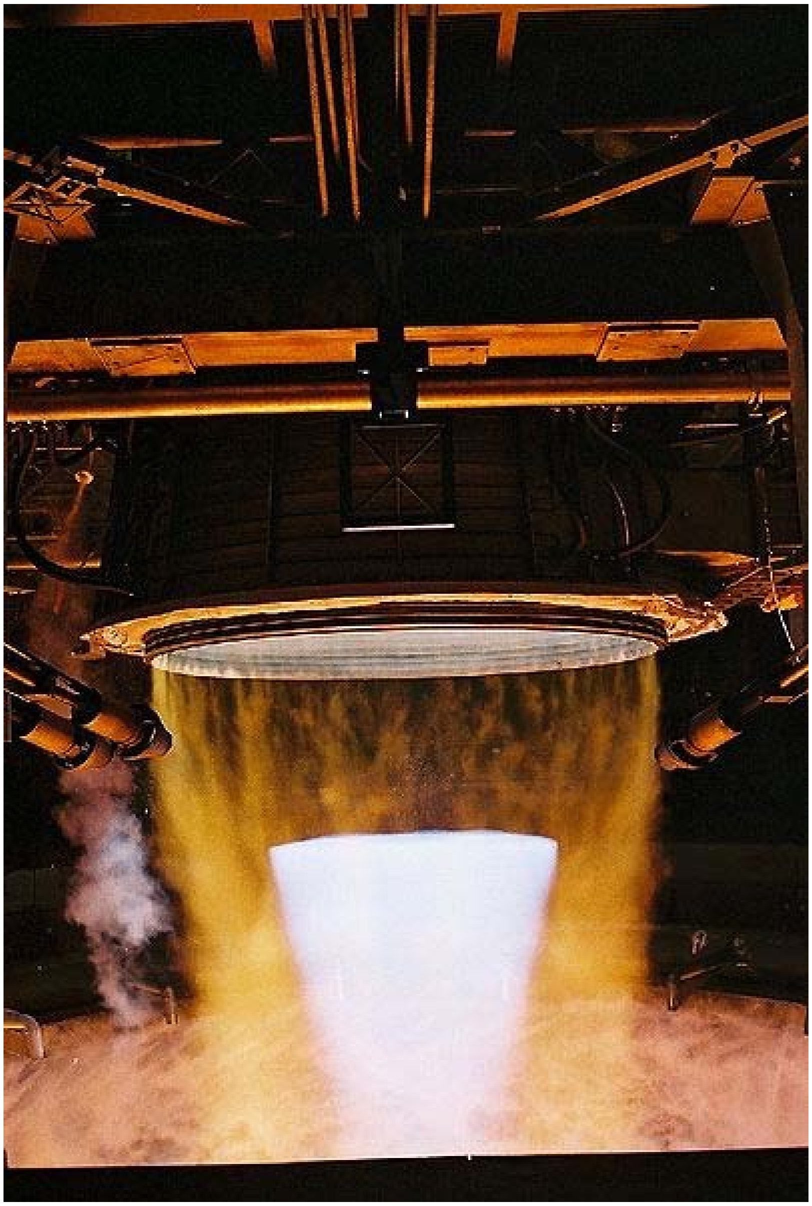
Rocket ground test facilities outside the United States

- Lampoldshausen - Baden-Württemberg, Germany, European Union
- Liquid Propulsion Systems Centre - Mahendragiri,]],Tamil Nadu, India
- NII-229 (NIIKhIMMash) - Zagorsk, Moscow Oblast, Russia
- RAF Spadeadam - (No longer in use) United Kingdom.

Vulcain

The Vulcain engine in a museum.

The Vulcain 2 engine.



Test firing of the Vulcain 2 engine in May 2004.

Vulcain is a family of European cryogenic first stage rocket engines for the Ariane 5.

History

The development of Vulcain, assured by a European collaboration, began in 1988 with the Ariane 5 rocket program. It first flew in 1996 powering the ill-fated flight 501 without being the cause of the disaster, and had its first successful flight in 1997 (flight 502). In 2002 the upgraded Vulcain 2 with 20% more thrust first flew on flight 517, although a problem with the engine turned the flight into a failure. The cause was due to flight loads being much higher than expected, as the inquiry board concluded. Subsequently, the nozzle has been redesigned, reinforcing the structure and improving the thermal situation of the tube wall, enhancing hydrogen coolant flow as well as applying thermal barrier coating to the flame-facing side of the coolant tubes, reducing heat load. The first successful flight of the (partially redesigned) Vulcain 2 occurred in 2005 on flight 521.

Future development

Although different upgrades to the engine have been proposed, there is no current program to develop an uprated version of the engine. If there will ever be one, it is likely that the new engine would be introduced after the "PA batch" of 30 Ariane 5 ECAs ordered on 10 May 2004 will be expended.

On 2007-06-17 Volvo Aero announced that in spring of 2008 it expected to hot-fire test a Vulcain 2 nozzle manufactured with a new "sandwich" technology.

Overview

Engine performance

	Vulcain	Vulcain 2
Vacuum thrust	1120 kN	1340 kN
Sea level thrust	800 kN	900 kN
Chamber pressure	114 bar	117 bar
Expansion ratio	45:1	60:1
Nominal specific impulse	433 s	431 s

The Vulcain engines are gas-generator cycle cryogenic rocket engines fed with liquid oxygen and liquid hydrogen. They feature regenerative cooling through a tube wall design, and the Vulcain 2 introduced a particular film cooling for the lower part of the nozzle, where exhaust gas from the turbine is re-injected in the engine They power the first stage of the Ariane 5 launcher, the EPC (Étage Principal Cryotechnique, main cryogenic stage) and provide 8% of the total lift-off thrust (the rest being provided by the two solid rocket boosters). The engine operating time is 600 s in both configurations. 3 m tall and 1.76 m in diameter, the engine weighs 1686 kg and provides 137 t of thrust in its latest version . The oxygen turbopump rotates at 13600 rpm with a power of 3 MW while the hydrogen turbopump rotates at 34000 rpm with 12 MW of power. The total mass flow rate is 235 kg/s, of which 41.2 kg/s are of hydrogen.

Contractors

The main contractor for the Vulcain engines is Snecma Moteurs (France), which also provides the liquid hydrogen turbopump. The liquid oxygen turbopump is responsibility of Avio (Italy), the gas turbines that power the turbopumps and the nozzle are developed by Volvo (Sweden).