Rocketry Science

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First Edition, 2012

ISBN 978-81-323-4398-1

Published by:
White Word Publications
4735/22 Prakashdeep Bldg,
Ansari Road, Darya Ganj,
Delhi - 110002
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Chapter 1

Oberth Effect

In astronautics, the **Oberth effect** is where the use of a rocket engine when travelling at high speed generates much more useful energy than one at low speed. Oberth effect occurs because the propellant has more usable energy (due to its kinetic energy on top of its chemical potential energy) and it turns out that the vehicle is able to employ this kinetic energy to generate more mechanical power. It is named after Hermann Oberth, the Hungarian-born, German physicist and a founder of modern rocketry, who apparently first described the effect.

Oberth effect is used in a **powered flyby** or **Oberth maneuver** where the application of an impulse, typically from the use of a rocket engine, close to a gravitational body (where the gravity potential is low, and the speed is high) can give much more change in kinetic energy and final speed (i.e. higher specific energy) than the same impulse applied further from the body for the same initial orbit. For the Oberth effect to be most effective, the vehicle must be able to generate as much impulse as possible at the lowest possible altitude; thus the Oberth effect is often far less useful for low-thrust reaction engines such as ion drives, which have a low propellant flow rate.

Oberth effect also can be used to understand the behaviour of multi-stage rockets; the upper stage can generate much more usable kinetic energy than might be expected from simply considering the chemical energy of the propellants it carries.

Historically, a lack of understanding of this effect led early investigators to conclude that interplanetary travel would require completely impractical amounts of propellant, as without it, enormous amounts of energy are needed.

Description

Rocket engines produce the same force regardless of their velocity. A rocket acting on a fixed object, as in a static firing, does no useful work at all; the rocket's stored energy is entirely expended on accelerating its propellant to hypersonic speed. But when the rocket moves, the thrust of the rocket during any time interval acts through the distance the rocket and payload move during that time. Force acting through a distance is the definition of mechanical energy or work. So the farther the rocket and payload move during the burn, (i.e. the faster they move), the greater the kinetic energy imparted to the rocket and its payload.

This can be easily shown. The definition of mechanical work:

$$\Delta E = F \cdot s$$

Where:

> *E* is the energy (specifically the kinetic energy), F is the force- the thrust of the rocket, s is the distance

Differentiating with respect to time:

$$\frac{dE}{dt} = F \cdot \frac{ds}{dt}$$

Or:

$$\frac{dE}{dt} = F \cdot v$$

where v is the velocity.

dividing by the instantaneous mass m to express this in terms of specific energy (S)

$$\frac{dS}{dt} = \frac{F}{m} \cdot v = a \cdot v$$

where a is the acceleration vector.

Thus it can be readily seen that the rate of gain of specific energy of every part of the rocket is proportional to speed, and given this the equation can be integrated to calculate the overall increase in specific energy of the rocket.

However, integrating this is often unnecessary, if the burn is short. For example as a vehicle falls towards periapsis in any orbit (closed or escape orbits) the velocity relative to the central body increases. Briefly burning the engine (an 'impulsive burn') prograde at periapsis increases the velocity by the same increment as at any other time (Δv). However, since the vehicle's kinetic energy is related to the *square* of its velocity, this increase in velocity has a disproportionate effect on the vehicle's kinetic energy; leaving it with higher energy than if the burn were achieved at any other time.

It may seem that the rocket is getting energy for free, which would violate conservation of energy. However, any gain to the rocket's energy is balanced by an equal decrease in the energy the exhaust is left with. When expended lower in the gravitational field, even if the exhaust is left with more kinetic energy, it is left with less total energy.

At very high speed the mechanical power imparted to the rocket can even exceed the total power liberated in the combustion of the propellants, and this may also seem to violate conservation of energy. But the propellants in a fast moving rocket carry energy not only chemically but also in their own kinetic energy, which at speeds above a few km/s actually exceed the chemical component. When these propellants are burned, some of this kinetic energy is transferred to the rocket along with the chemical energy released by burning. This can make up for what seems like an extremely low efficiency early in the rocket's flight when it is moving only slowly. Most of the work done by a rocket early in flight is "invested" in the kinetic energy of the propellant not yet burned, part of which they will release later when they are burned.

Parabolic example

If the ship travels at velocity v at the start of a burn that changes the velocity by Δv, then the change in specific orbital energy (SOE) is

$$v\,\Delta v + \frac{(\Delta v)^2}{2}.$$

Once the space craft is far from the planet again, the SOE is entirely kinetic, since gravitational potential energy tends to zero. Therefore, the larger the v at the time of the burn, the greater the final kinetic energy, and the higher the final velocity.

The effect becomes more pronounced the closer to the central body, or more generally, the deeper in the gravitational field potential the burn occurs, since the velocity is higher there.

So if a spacecraft on a parabolic flyby of Jupiter with a periapsis velocity of 50 km/s, and it performs a 5 km/s burn, it turns out that the final velocity at great distance is 22.9 km/s; giving a multiplication of the burn by 4.6 times.

Detailed proof

If an impulsive burn of Δv is performed at periapsis in a parabolic orbit then the velocity at periapsis before the burn is equal to the escape velocity (V_{esc}), and the specific kinetic energy after the burn is:

$$\tfrac{1}{2}V^2$$

where $V = V_{esc} + \Delta v$

$$= \tfrac{1}{2}(V_{esc} + \Delta v)^2$$

$$= \tfrac{1}{2}V_{\text{esc}}^2 + \Delta v V_{\text{esc}} + \tfrac{1}{2}\Delta v^2$$

When the vehicle leaves the gravity field, the loss of specific kinetic energy is:

$$\tfrac{1}{2}V_{\text{esc}}^2$$

so it retains the energy:

$$\Delta v V_{\text{esc}} + \tfrac{1}{2}\Delta v^2$$

which is larger than the energy from a burn outside the gravitational field ($\tfrac{1}{2}\Delta v^2$) by:

$$\Delta v V_{\text{esc}}$$

It can then be easily shown that the impulse is multiplied by a factor of:

$$\sqrt{1 + \frac{2V_{\text{esc}}}{\Delta v}}$$

Plugging in 50 km/s escape velocity and 5 km/s burn we get a multiplier of 4.6.

Similar effects happen in closed and hyperbolic orbits.

Chapter 2

Rocket Artillery

M270 MLRS

Rocket artillery is a type of artillery equipped with rocket launchers instead of conventional guns or mortars.

Types of rocket artillery pieces include multiple rocket launchers.

History

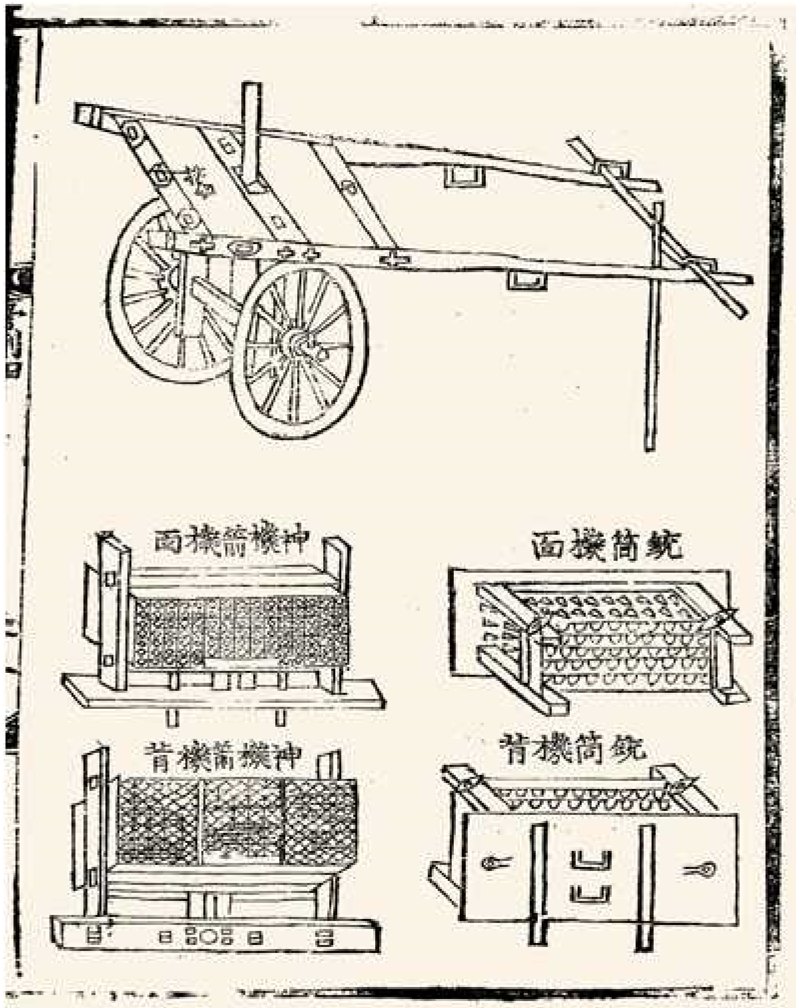


Illustration of a Korean rocket launcher of the 1500s

The use of rockets as some form of artillery dates back to medieval China where devices such as fire arrows were used (albeit mostly as a psychological weapon). Fire arrows were also used in multiple launch systems and transported via carts. Devices such as the Korean Hwacha were able to fire hundreds of fire arrows simultaneously. The use of medieval rocket artillery was picked up by the invading Mongols and spread to the Ottoman Turks who in turn used them on the European battlefield.

Metal-cylinder rocket artillery

The first iron-cased metal-cylinder rocket artillery were developed by Tipu Sultan, an Indian ruler of the Kingdom of Mysore, and his father Hyder Ali, in the 1780s. He successfully used these metal-cylinder rockets against the larger forces of the British East India Company during the Anglo-Mysore Wars. The Mysore rockets of this period were much more advanced than what the British had seen, chiefly because of the use of iron tubes for holding the propellant; this enabled higher thrust and longer range for the missile (up to 2 km range).

According to Stephen Oliver Fought and John F. Guilmartin, Jr. in *Encyclopedia Britannica* (2008): "Hyder Ali, prince of Mysore, developed war rockets with an important change: the use of metal cylinders to contain the combustion powder. Although the hammered soft iron he used was crude, the bursting strength of the container of black powder was much higher than the earlier paper construction. Thus a greater internal pressure was possible, with a resultant greater thrust of the propulsive jet. The rocket body was lashed with leather thongs to a long bamboo stick. Range was perhaps up to three-quarters of a mile (more than a kilometre). Although individually these rockets were not accurate, dispersion error became less important when large numbers were fired rapidly in mass attacks. They were particularly effective against cavalry and were hurled into the air, after lighting, or skimmed along the hard dry ground. Hyder Ali's son, Tippu Sultan, continued to develop and expand the use of rocket weapons, reportedly increasing the number of rocket troops from 1,200 to a corps of 5,000. In battles at Seringapatam in 1792 and 1799 these rockets were used with considerable effect against the British."

After Tipu's eventual defeat in the Fourth Anglo-Mysore War, the Mysore iron rockets were captured by the British. These rockets were influential in British rocket development, inspiring the Congreve rocket, which were soon put into use in the Napoleonic Wars, including at the Battle of Waterloo. Ironically, the technology of metal-cylinder missiles developed by Tipu Sultan contributed to the defeat of his ally Napoleon at Waterloo.

World War II

Modern rocket artillery was first employed during World War II, in the form of the German Nebelwerfer and Soviet Katyusha-series. The Soviet Katyushas, nicknamed by German troops Stalin Organs because of their visual resemblance to a church musical organ and alluding to the sound of the weapon's rockets, were mounted on trucks or light tanks, while the early German Nebelwerfer were mounted on a small wheeled carriage which was light enough to be moved by several men and could easily be deployed nearly anywhere, while also being towed by most vehicles. The Germans also had self-propelled rocket artillery in the form of the Panzerwerfer and Wurfrahmen 40 which equipped half-track armoured fighting vehicles. An oddity in the subject of rocket artillery during this time was the German "Sturmtiger", a vehicle based on the Tiger I heavy tank chassis that was armed with a 380 mm rocket mortar.

Rocket artillery used by US Navy in World War II

The Western Allies of World War II employed little rocket artillery. During later periods of the war, British and Canadian troops used the Land Mattress, a towed rocket launcher. The United States Army built and deployed a small number of T34 Calliope rocket artillery tanks (converted from M4 Sherman medium tanks) in France and Italy. In 1945, the British Army also fitted some M4 Shermans with two 60 lb RP3 rockets, the same as used on ground attack aircraft and known as *Tulip*.

In the Pacific, however, the US Navy made heavy use of rocket artillery, adding to the already intense bombardment by the guns of heavy warships to soften up Japanese-held islands before the US Marines would land. On Iwo Jima, the Marines made use of rocket artillery trucks in a similar fashion as the Soviet Katyusha, but on a smaller scale.

Post-World War II

Israel fitted some of their Sherman tanks with different rocket artillery. An unconventional Sherman conversion was the turretless **Kilshon** ("Trident") that launched a AGM-45 Shrike anti-radiation missile.

The Soviet Union continued its development of the Katyusha during the Cold War, and also exported them widely.

Modern rocket artillery such as the US M270 Multiple Launch Rocket System is highly mobile and are used in similar fashion to other self-propelled artillery. Global Positioning and Inertial Navigation terminal guidance systems have been introduced.

Rocket artillery vs tube artillery

- Rockets produce no recoil, while conventional artillery systems produce significant recoil. Unless firing within a very small arc with the possibility of wrecking a self propelled artillery system's vehicle suspension, gun artillery must usually be braced against recoil. In this state they are immobile, and can not change position easily. Rocket artillery is much more mobile and can change position easily. This "shoot-and-scoot" ability makes the platform difficult to target. A rocket artillery piece could, conceivably, fire on the move. Rocket systems produce a significant amount of backblast, however, which imposes its own restrictions. Launchers may be sighted by the firing arcs of the rockets, and their fire can damage themselves or neighbouring vehicles.

- Rocket artillery cannot usually match the accuracy and sustained rate of fire of conventional artillery. They may be capable of very destructive strikes by delivering a large mass of explosives simultaneously, thus increasing the shock effect and giving the target less time to take cover. Modern computer-controlled conventional artillery have recently begun to acquire the possibility to do something similar through MRSI but it is an open question if MRSI is really practical in a combat situation. On the other hand precision-guided rocket artillery demonstrates extreme accuracy, comparable with the best guided tube artillery systems.

- Rocket artillery typically has a very large fire signature, leaving a clear smoke-trail showing exactly where the barrage came from. Since the barrage does not take much time, however, the rocket artillery can move away quickly.

- Tube artillery can use a forward observer to correct fire, thus achieving further accuracy. This is usually not practical with rocket artillery.

- Tube artillery shells are typically cheaper and less bulky than rockets, so they can deliver a larger amount of explosive at the enemy per weight of ammunition or per money spent.

- While tube artillery shells are smaller than rockets, the gun itself must be very large to match the range of rockets. Therefore rockets typically have longer range while the rocket launchers remain small enough to mount on mobile vehicles. Extremely large guns like the Paris Gun have been rendered obsolete by long range missiles.

- If the artillery barrage was intended as a preparation for an attack, and it usually is, a short but intense barrage will give the enemy less time to prepare by, for instance, dispersing.

- The higher accuracy of gun artillery means that it can be used to attack an enemy close to a friendly force. This combined with the higher capacity for sustained fire makes cannon artillery more suitable than rocket artillery for defensive fire. It is also the only practicable system for counter-battery fire.

Chapter 3

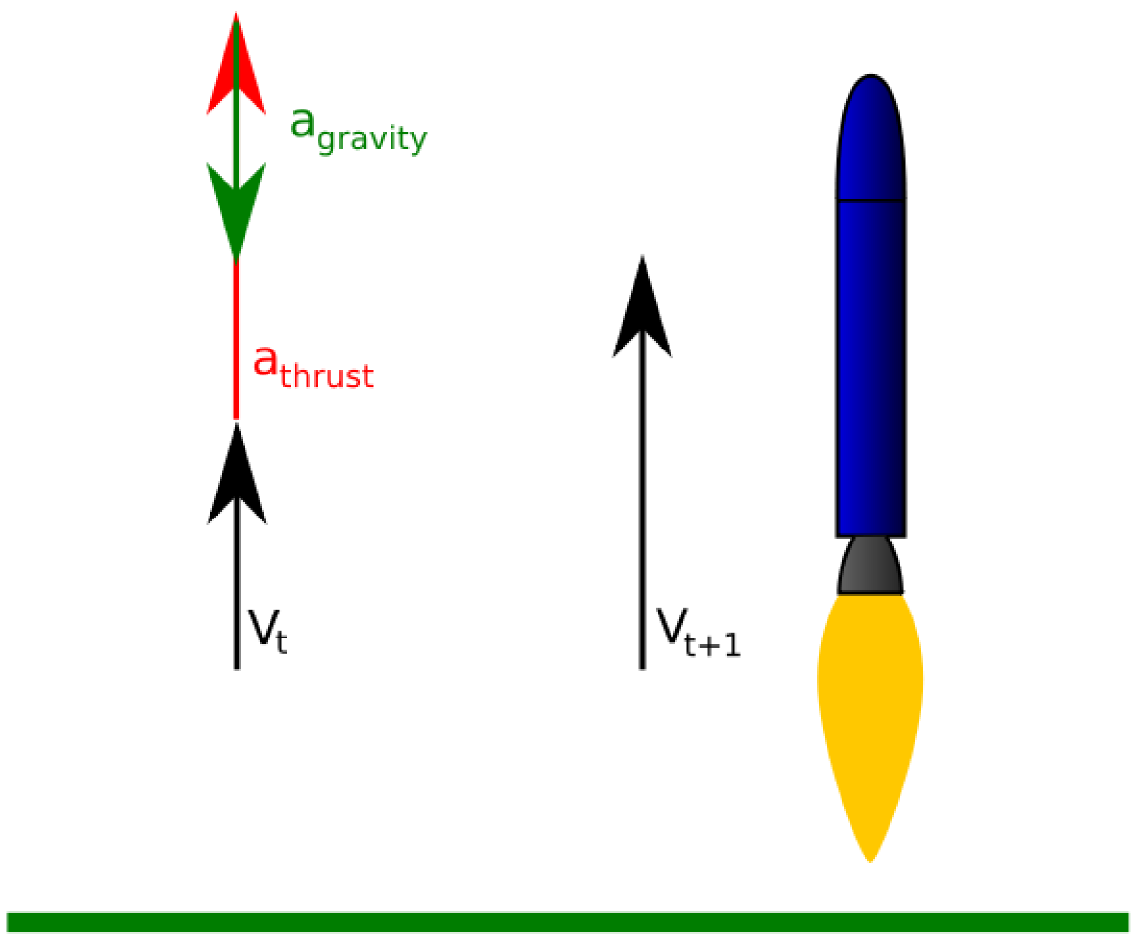
Gravity Turn

A **gravity turn** or **zero-lift turn** is a maneuver used in launching a spacecraft into, or descending from, an orbit around a celestial body such as a planet or a moon. This launch trajectory offers two main advantages over a thrust-controlled trajectory where the rocket's own thrust steers the vehicle. First, any thrust used to change the ship's direction does not accelerate the vehicle into orbit. This loss can be reduced by using gravity to steer the vehicle onto its desired trajectory. Second, and more importantly, because gravity does the steering during the initial ascent phase the vehicle can maintain low or even zero angle of attack. This minimizes transverse aerodynamic stress on the launch vehicle, allowing for a lighter launch vehicle.

The term gravity turn can also refer to the use of a planet's gravity to change a spacecraft's direction. When used in this context it is similar to a gravitational slingshot; the difference is that a gravitational slingshot often increases or decreases spacecraft velocity and changes direction while the gravity turn only changes direction.

Launch procedure

Vertical climb

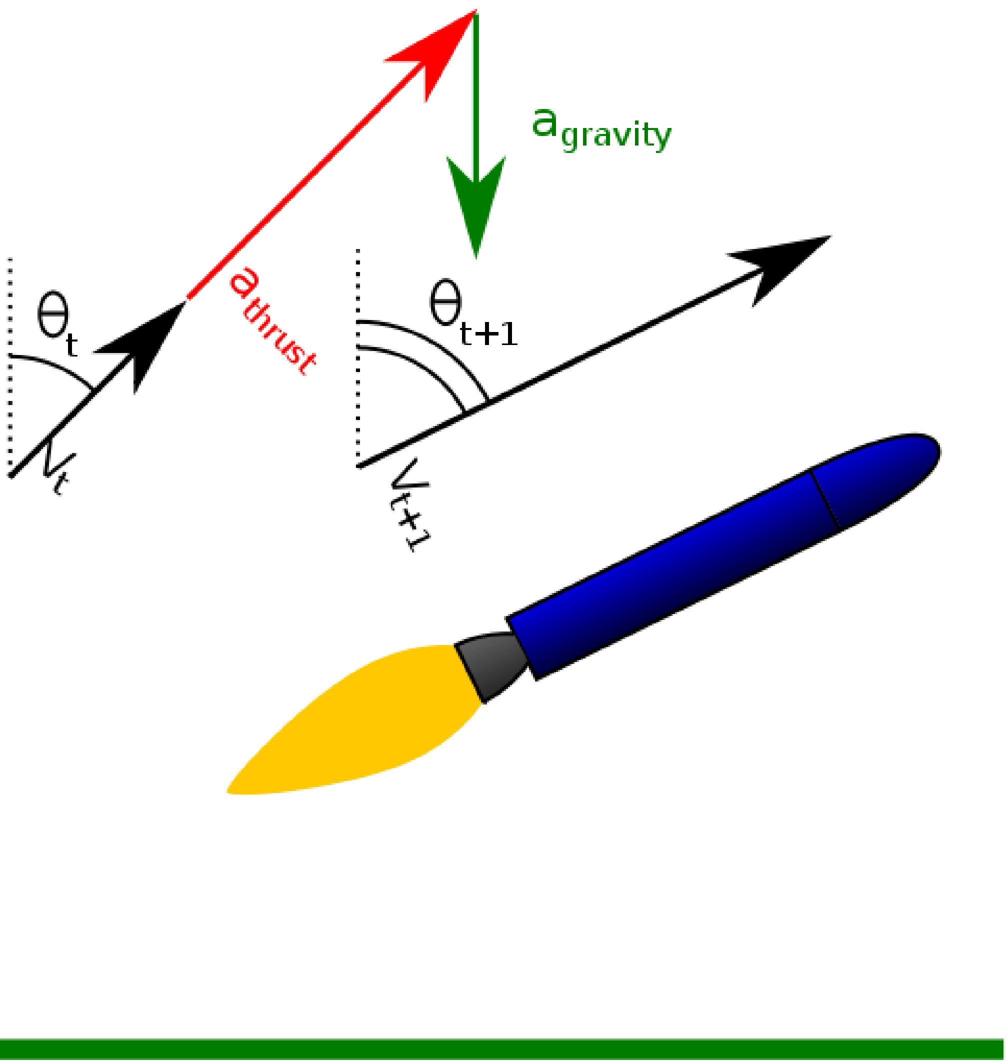


A diagram showing the velocity vectors for times t and $t + 1$ during the vertical climb phase. The launch vehicle's new velocity is the vector sum of its old velocity, the acceleration from thrust, and the acceleration of gravity. More formally $V_{t+1} = V_t + a_{thrust} * \Delta t + a_{gravity} * \Delta t$

The gravity turn is commonly utilized with launch vehicles such as a rocket or the Space Shuttle which launch vertically. The rocket begins by flying straight up, gaining both vertical speed and altitude. During this portion of the launch gravity acts directly against the thrust of the rocket, lowering its vertical acceleration. Losses associated with this slowing are known as gravity drag, and can be minimized by executing the next phase of the launch, the pitch over maneuver, as soon as possible. The pitch over should also be carried out while the vertical velocity is small to avoid large aerodynamic loads on the vehicle during the maneuver.

The pitch over maneuver consists of the rocket gimbaling its engine slightly to direct some of its thrust to one side. This force creates a net torque on the ship, turning it so that it no longer points vertically. The pitch over angle varies with the launch vehicle and is included in the rocket's initial guidance system, for some vehicles it is only a few degrees while other vehicles use relatively large angles (a few tens of degrees). After the pitch over is complete the engines are reset to point straight down the axis of the rocket again. This small steering maneuver is the only time during an ideal gravity turn ascent that thrust must be used for purposes of steering. This pitch over maneuver serves two purposes. First, it turns the rocket slightly so that its flight path is no longer vertical, and second, it places the rocket on the correct heading for its ascent to orbit. After the pitch over the rocket's angle of attack is adjusted to zero for the remainder of its climb to orbit. This zeroing of the angle of attack reduces lateral aerodynamic loads and produces negligible lift force during the ascent.

Downrange acceleration



A diagram showing the velocity vectors for times *t* and *t* + 1 during the downrange acceleration phase. As before the launch vehicle's new velocity is the vector sum of its old velocity, the acceleration from thrust, and the acceleration of gravity. Because gravity acts straight down, the new velocity vector is closer to being level with the horizon; gravity has "turned" the trajectory downward.

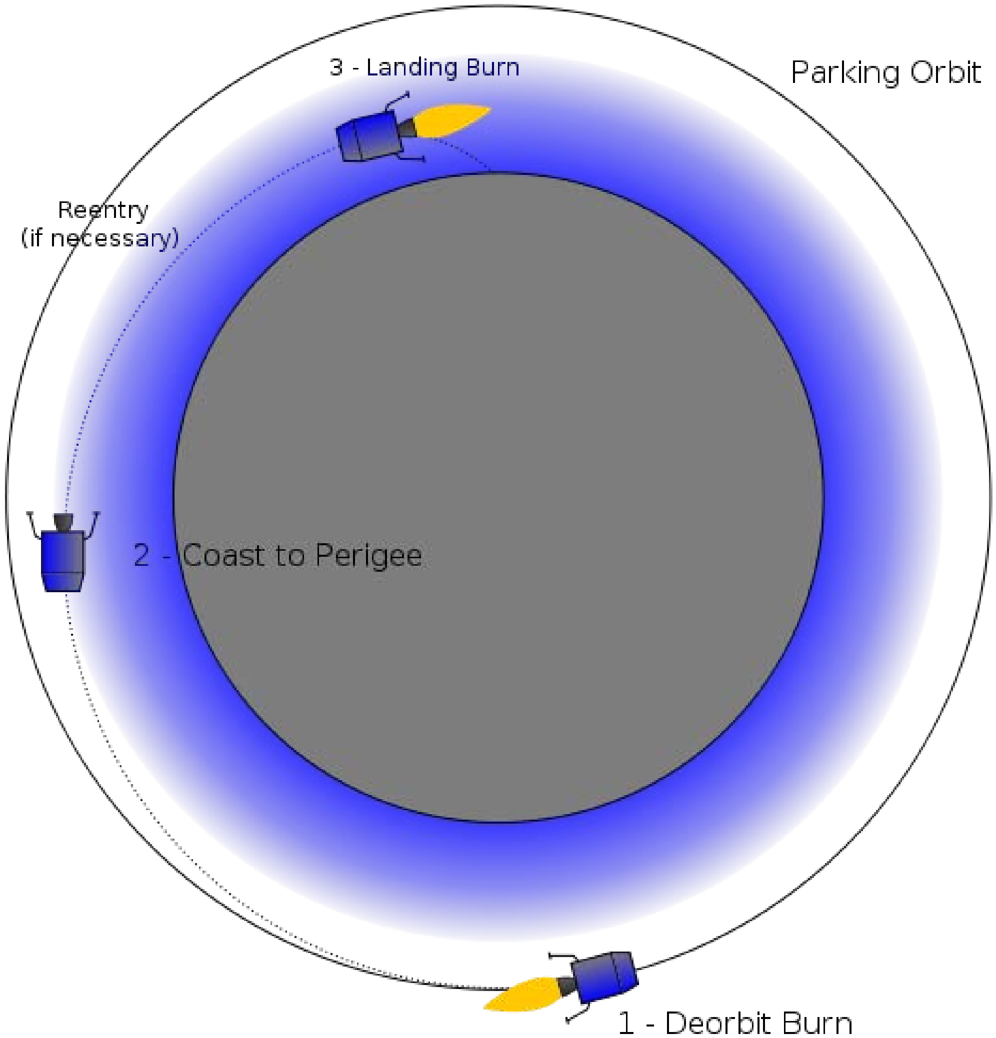
After the pitch over, the rocket's flight path is no longer completely vertical so gravity acts to turn the flight path back towards the ground. If the rocket were not producing thrust the flight path would be a simple parabola like a thrown ball, leveling off and then falling back to the ground. The rocket is producing thrust though, and rather than leveling off and then descending again, by the time the rocket levels off it has gained sufficient altitude and velocity to place it in a stable orbit.

If the rocket is a multi-stage system where stages fire sequentially, the rocket's ascent burn may not be continuous. Obviously some time must be allowed for stage separation and engine ignition between each successive stage, but some rocket designs call for extra free-flight time between stages. This is particularly useful in very high thrust rockets where if the engines were fired continuously the rocket would run out of fuel before leveling off and reaching a stable orbit above the atmosphere. The technique is also useful when launching from a planet with a thick atmosphere, such as the Earth. Since gravity turns the flight path during free flight the rocket can use a smaller initial pitch over angle, giving it higher vertical velocity, and taking it out of the atmosphere more quickly. This reduces both aerodynamic drag as well as aerodynamic stress during launch. Then later during the flight the rocket coasts between stage firings allowing it to level off above the atmosphere so when the engine fires again, at zero angle of attack, the thrust accelerates the ship horizontally, inserting it into orbit.

Descent and landing procedure

Because heat shields and parachutes cannot be used to land on an airless body such as the Moon, a powered descent with a gravity turn is a good alternative. The Apollo lunar module used a slightly modified gravity turn to land from lunar orbit. This was essentially a launch in reverse except that a landing spacecraft is lightest at the surface while a spacecraft being launched is heaviest at the surface. A computer program called Lander that simulated gravity turn landings applied this concept by simulating a gravity turn launch with a negative mass flow rate, i.e. the propellant tanks filled during the rocket burn. The idea of using a gravity turn maneuver to land a vehicle was originally developed for the Lunar Surveyor landings, although Surveyor made a direct approach to the surface without first going into lunar orbit.

Deorbit and reentry

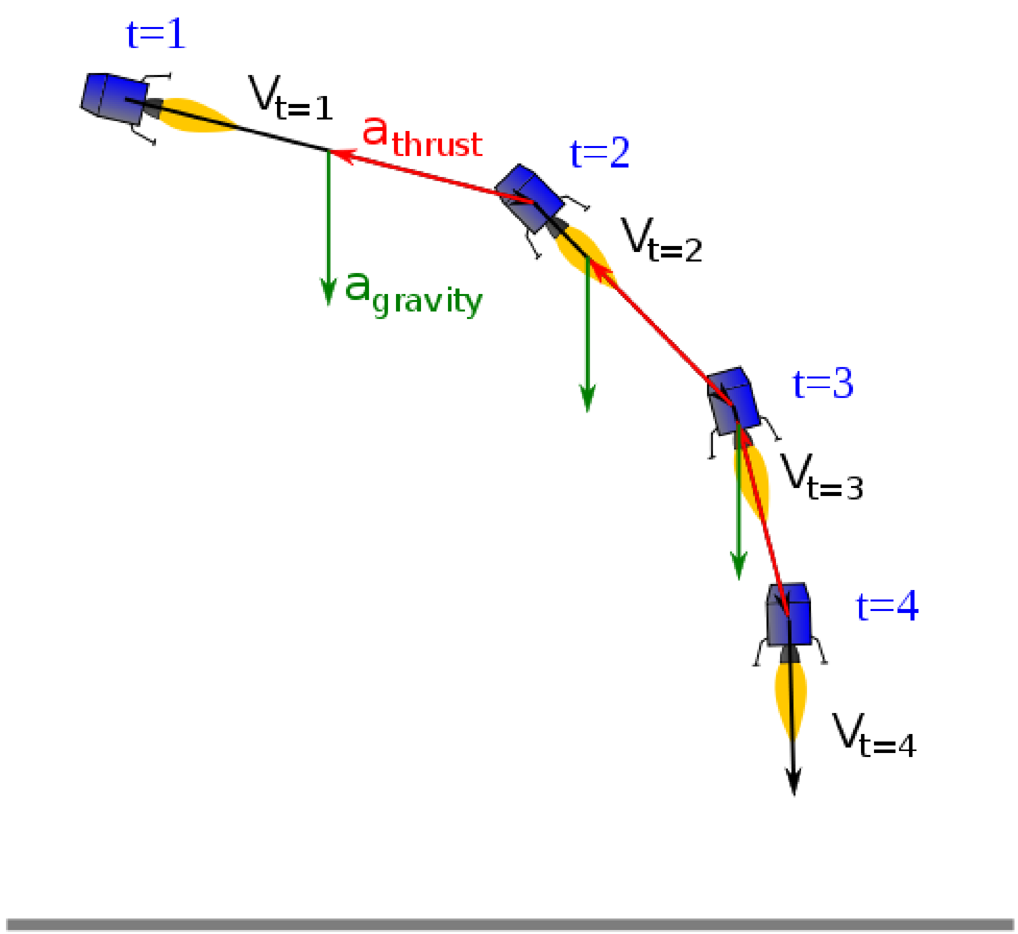


The deorbit, coast, and possible reentry phase leading up to the beginning of the final landing burn.

The vehicle begins by orienting for a retrograde burn to reduce its orbital velocity, lowering its point of periapsis to near the surface of the body to be landed on. If the craft is landing on a planet with an atmosphere such as Mars the deorbit burn will only lower periapsis into the upper layers of the atmosphere, rather than just above the surface as on an airless body. After the deorbit burn is complete the vehicle can either coast until it is nearer to its landing site or continue firing its engine while maintaining zero angle of attack. For a planet with an atmosphere the coast portion of the trip includes reentry through the atmosphere as well.

After the coast and possible reentry the vehicle jettisons any no longer necessary heat shields and/or parachutes in preparation for the final landing burn. If the atmosphere is thick enough it can be used to slow the vehicle a lot, thus saving on fuel. In this case a gravity turn is not the optimal entry trajectory but it does allow for approximation of the true delta-v required. In the case where there is no atmosphere however the landing vehicle must provide the full delta-v necessary to land safely on the surface.

Landing



The final approach and landing portion of the descent. The vehicle loses horizontal speed while transitioning to a vertical hover, allowing it to settle down on the surface.

If it is not already properly oriented, the vehicle lines up its engines to fire directly opposite its current surface velocity vector, which at this point is either parallel to the ground or only slightly vertical, as shown to the left. The vehicle then fires its landing engine to slow down for landing. As the vehicle loses horizontal velocity the gravity of the body to be landed on will begin pulling the trajectory closer and closer to a vertical descent. In an ideal maneuver on a perfectly spherical body the vehicle could reach zero horizontal velocity, zero vertical velocity, and zero altitude all at the same moment, landing safely on the surface. However due to rocks and uneven surface terrain the vehicle usually picks up a few degrees of angle of attack near the end of the maneuver to zero its horizontal velocity just above the surface. This process is the mirror image if the pitch over maneuver used in the launch procedure and allows the vehicle to hover straight down, landing gently on the surface.

Guidance and control

The steering of a rocket's course during its flight is divided into two separate components; control, the ability to point the rocket in a desired direction, and guidance, the determination of what direction a rocket should be pointed to reach a given target. The desired target can either be a location on the ground, as in the case of a ballistic missile, or a particular orbit, as in the case of a launch vehicle.

Launch

The gravity turn trajectory is most commonly used during early ascent. The guidance program is a precalculated lookup table of pitch vs time. Control is done with engine gimballing and/or aerodynamic control surfaces. The pitch program maintains a zero angle of attack (the definition of a gravity turn) until the vacuum of space is reached, thus minimizing lateral aerodynamic loads on the vehicle. (Excessive aerodynamic loads can quickly destroy the vehicle.) Although the preprogrammed pitch schedule is adequate for some applications, an adaptive inertial guidance system that determines location, orientation and velocity with accelerometers and gyroscopes, is almost always employed on modern rockets. The British satellite launcher Black Arrow was an example of a rocket that flew a preprogrammed pitch schedule, making no attempt to correct for errors in its trajectory, while the Apollo-Saturn rockets used "closed loop" inertial guidance after the gravity turn through the atmosphere.

The initial pitch program is an open-loop system subject to errors from winds, thrust variations, etc. To maintain zero angle of attack during atmospheric flight, these errors are not corrected until reaching space. Then a more sophisticated closed-loop guidance program can take over to correct trajectory deviations and attain the desired orbit. In the Apollo missions, the transition to closed-loop guidance took place early in second stage flight after maintaining a fixed inertial attitude while jettisoning the first stage and interstage ring. Because the upper stages of a rocket operate in a near vacuum, fins are ineffective. Steering relies entirely on engine gimballing and a reaction control system.

Landing

To serve as an example of how the gravity turn can be used for a powered landing, an Apollo type lander on an airless body will be assumed. The lander begins in a circular orbit docked to the command module. After separation from the command module the lander performs a retrograde burn to lower its periapsis to just above the surface. It then coasts to periapsis where the engine is restarted to perform the gravity turn descent. It has been shown that in this situation guidance can be achieved by maintaining a constant angle between the thrust vector and the line of sight to the orbiting command module. This simple guidance algorithm builds on a previous study which investigated the use of various visual guidance cues including the uprange horizon, the downrange horizon, the desired landing site, and the orbiting command module. The study concluded that using the command module provides the best visual reference, as it maintains a near constant visual separation from an ideal gravity turn until the landing is almost complete. Because the vehicle is landing in a vacuum, aerodynamic control surfaces are useless. Therefore a system such as a gimballing main engine, a reaction control system, or possibly a control moment gyroscope must be used for attitude control.

Limitations

Although gravity turn trajectories use minimal steering thrust they are not always the most efficient possible launch or landing procedure. Several things can affect the gravity turn procedure making it less efficient or even impossible due to the design limitations of the launch vehicle. A brief summary of factors affecting the turn is given below.

- Atmosphere — In order to minimize gravity drag the vehicle should begin gaining horizontal speed as soon as possible. On an airless body such as the Moon this presents no problem, however on a planet with a dense atmosphere this is not possible. A trade off exists between flying higher before starting downrange acceleration, thus increasing gravity drag losses; or starting downrange acceleration earlier, reducing gravity drag but increasing the aerodynamic drag experienced during launch.

- Maximum dynamic pressure — Another effect related to the planet's atmosphere is the maximum dynamic pressure exerted on the launch vehicle during the launch. Dynamic pressure is related to both the atmospheric density and the vehicle's speed through the atmosphere. Just after liftoff the vehicle is gaining speed and increasing dynamic pressure faster than the reduction in atmospheric density can decrease the dynamic pressure. This causes the dynamic pressure exerted on the vehicle to increase until the two rates are equal. This is known as the point of maximum dynamic pressure (abbreviated "max Q"), and the launch vehicle must be built to withstand this amount of stress during launch. As before a trade off exists between gravity drag from flying higher first to avoid the thicker atmosphere when accelerating; or accelerating more at lower altitude, resulting in a heavier launch vehicle because of a higher maximum dynamic pressure experienced on launch.

- Maximum engine thrust — The maximum thrust the rocket engine can produce affects several aspects of the gravity turn procedure. First before the pitch over maneuver the vehicle must be capable of not only overcoming the force of gravity but accelerating upwards. The more acceleration the vehicle has beyond the acceleration of gravity the quicker vertical speed can be obtained allowing for lower gravity drag in the initial launch phase. When the pitch over is executed the vehicle begins its downrange acceleration phase; engine thrust affects this phase as well. Higher thrust allows for a faster acceleration to orbital velocity as well. By reducing this time the rocket can level off sooner; further reducing gravity drag losses. Although higher thrust can make the launch more efficient, accelerating too much low in the atmosphere increases the maximum dynamic pressure. This can be alleviated by throttling the engines back during the beginning of downrange acceleration until the vehicle has climbed higher. However, with solid fuel rockets this may not be possible.

- Maximum payload acceleration — Another limitation related to engine thrust is the maximum acceleration that can be safely sustained by the crew and/or the payload. Near main engine cut off (MECO) when the launch vehicle has consumed most of its fuel it will be much lighter than it was at launch. If the engines are still producing the same amount of thrust the acceleration will grow as a result of the decreasing vehicle mass. If this acceleration is not kept in check by throttling back the engines injury to the crew or damage to the payload could occur. This forces the vehicle to spend more time gaining horizontal velocity, increasing gravity drag.

Use in orbital redirection

For spacecraft missions where large changes in the direction of flight are necessary, direct propulsion by the spacecraft may not be feasible due to the large delta-v requirement. In these cases it may be possible to perform a flyby of a nearby planet or moon, using its gravitational attraction to alter the ship's direction of flight. Although this maneuver is very similar to the gravitational slingshot it differs in that a slingshot often implies a change in both speed and direction whereas the gravity turn only changes the direction of flight.

A variant of this maneuver, the free return trajectory allows the spacecraft to depart from a planet, circle another planet once, and return to the starting planet using propulsion only during the initial departure burn. Although in theory it is possible to execute a perfect free return trajectory, in practice small correction burns are often necessary during the flight. Even though it does not require a burn for the return trip, other return trajectory types, such as an aerodynamic turn, can result in a lower total delta-v for the mission.

Use in spaceflight

Many spaceflight missions have utilized the gravity turn, either directly or in a modified form, to carry out their missions. What follows is a short list of various mission that have used this procedure.

- Surveyor program — A precursor to the Apollo Program, the Surveyor Program's primary mission objective was to develop the ability to perform soft landings on the surface of the moon, through the use of an automated descent and landing program built into the lander. Although the landing procedure can be classified as a gravity turn descent, it differs from the technique most commonly employed in that it was shot from the Earth directly to the lunar surface, rather than first orbiting the moon as the Apollo landers did. Because of this the descent path was nearly vertical, although some "turning" was done by gravity during the landing.

- Apollo program — Launches of the Saturn V rocket during the Apollo program were carried out using a gravity turn in order to minimize lateral stress on the rocket. At the other end of their journey, the lunar landers utilized a gravity turn landing and ascent from the moon.

- Mariner 10 — The Mariner 10 mission used a gravity assist from the planet Venus to travel to Mercury. In 1970, three years before its launch, Giuseppe Colombo noticed that because the spacecraft's orbit around the Sun after the encounter with Mercury was very close to twice the orbital period of Mercury. By properly orienting the first flyby of Mercury the spacecraft underwent a gravity turn which allowed it to make a second flyby of the planet.

- Ulysses — The Ulysses probe utilized a gravity turn around Jupiter to change the inclination of its orbit around the sun. This was done because the delta-v required to launch into a polar orbit around the sun was greater than the capability of any existing rocket. The spacecraft left Earth, arriving at Jupiter slightly "below" it; this caused Jupiter's gravity to incline the orbit so the probe would pass over the Sun's "north" pole.

Mathematical description

The simplest case of the gravity turn trajectory is that which describes a point mass vehicle, in a uniform gravitational field, neglecting air resistance. The thrust force $\vec{F}$ is a vector whose magnitude is a function of time and whose direction can be varied at will. Under these assumptions the differential equation of motion is given by:

$$m\frac{d\vec{v}}{dt} = \vec{F} - mg\hat{k} \ .$$

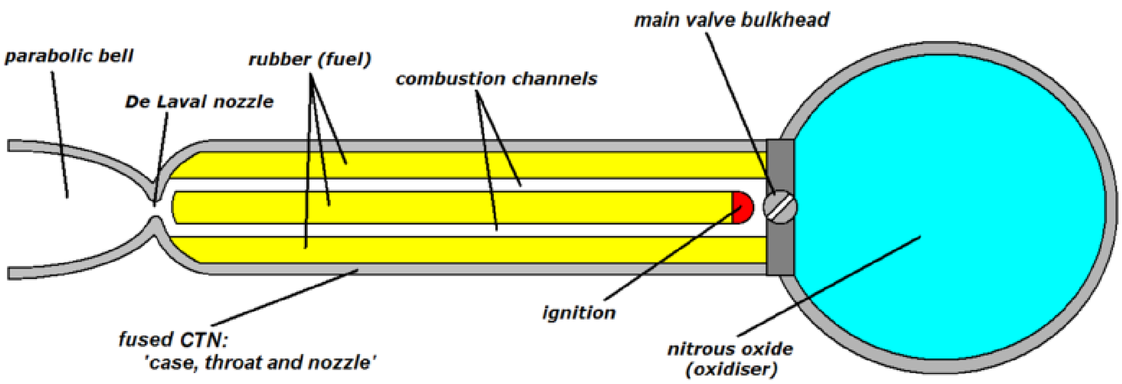
Here $\hat{k}$ is a unit vector in the vertical direction and *m* is the instantaneous vehicle mass. By constraining the thrust vector to point parallel to the velocity and separating the equation of motion into components parallel to $\vec{v}$ and those perpendicular to $\vec{v}$ we arrive at the following system:

$$\dot{v} = g(n - \cos\beta) \ ,$$
$$v\dot{\beta} = g\sin\beta \ .$$

Here the current thrust to weight ratio has been denoted by $n = F / mg$ and the current angle between the velocity vector and the vertical by $\beta = \arccos(\vec{\tau_1} \cdot \hat{k})$. This results in a coupled system of equations which can be integrated to obtain the trajectory. However, for all but the simplest case of constant *n* over the entire flight, the equations cannot be solved analytically and must be integrated numerically.

Chapter 4

Hybrid Rocket

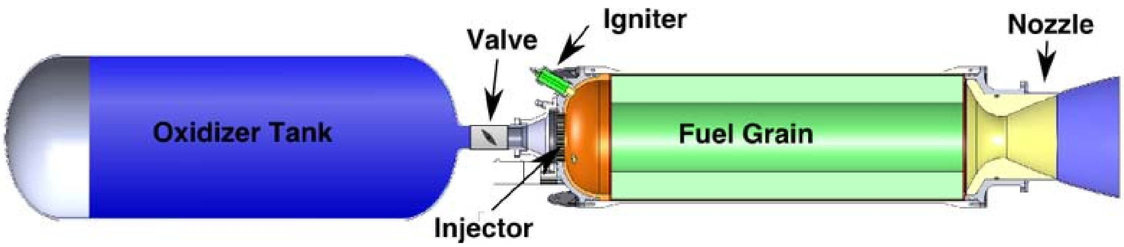


Hybrid rocket motor detail of SpaceShipOne (more information).

A **hybrid rocket** is a rocket with a rocket motor which uses propellants in two different states of matter - one solid and the other either gas or liquid. The Hybrid rocket concept can be traced back at least 75 years.

Hybrid rockets exhibit advantages over both liquid rockets and solid rockets especially in terms of simplicity, safety, and cost. Because it is nearly impossible for the fuel and oxidizer to be mixed intimately (being different states of matter), hybrid rockets tend to fail more benignly than liquids or solids. Like liquid rockets and unlike solid rockets they can be shut down easily and are simply throttle-able. The theoretical specific impulse(I_{sp}) performance of hybrids is generally higher than solids and roughly equivalent to hydrocarbon-based liquids. I_{sp} as high as 400s has been measured in a hybrid rocket using metalized fuels. Hybrid systems are slightly more complex than solids, but the significant hazards of manufacturing, shipping and handling solids offset the system simplicity advantages.

Basic concepts



Hybrid rocket propulsion system conceptual overview

In its simplest form a hybrid rocket consists of a pressure vessel (tank) containing the liquid propellant, the combustion chamber containing the solid propellant, and a valve isolating the two. When thrust is desired, a suitable ignition source is introduced in the combustion chamber and the valve is opened. The liquid propellant (or gas) flows into the combustion chamber where it is vaporized and then reacted with the solid propellant. Combustion occurs in a boundary layer diffusion flame adjacent to the surface of the solid propellant.

Generally the liquid propellant is the oxidizer and the solid propellant is the fuel because solid oxidizers are problematic and lower performing than liquid oxidizers. Furthermore, using a solid fuel such as HTPB or paraffin allows for the incorporation of high-energy fuel additives such as aluminium, lithium, or metal hydrides.

Common oxidizers include gaseous or liquid oxygen or nitrous oxide. Common fuels include polymers such as polyethylene, cross-linked rubber such as HTPB or liquefying fuels such as paraffin.

Advantages of hybrid rockets

Hybrid rocket motors exhibit some obvious as well as some subtle advantages over liquid-fuel rockets and solid rockets. A brief summary of some of these is given below:

Advantages compared with bipropellant liquid rockets

- Mechanically simpler - requires only a single liquid propellant resulting in less plumbing, fewer valves, and simpler operations.
- Denser fuels - fuels in the solid phase generally have higher density than those in the liquid phase
- Metal additives - reactive metals such as aluminum, magnesium, lithium or beryllium can be easily included in the fuel grain increasing specific impulse(I_{sp})

Advantages compared with solid rockets

- Higher theoretical I_{sp} obtainable

- Less explosion hazard - Propellant grain more tolerant of processing errors such as cracks
- More controllable - Start/stop/restart and throttling are all achievable with appropriate oxidizer control
- Safe and non-toxic oxidizers such as liquid oxygen and nitrous oxide can be used
- Can be transported to site in a benign form and loaded with oxidizer remotely immediately before launch, improving safety.

Disadvantages of hybrid rockets

Hybrid rockets also exhibit some disadvantages when compared with liquid and solid rockets. These include:

- Oxidizer-to-fuel ratio shift ("O/F shift") - with a constant oxidizer flow-rate, the ratio of fuel production rate to oxidizer flow rate will change as a grain regresses. This leads to off-peak operation from a chemical performance point of view.
- Low regression-rate (rate at which the solid phase recedes) fuels often drive multi-port fuel grains. Multi-port fuel grains have poor volumetric efficiency and, often, structural deficiencies. High regression-rate liquefying fuels developed in the late 1990s offer a potential solution to this problem.

For a well-designed hybrid, O/F shift has a very small impact on performance because I_{sp} is insensitive to O/F near the peak.

In general, much less development work has been performed with hybrids than liquids or solids and it is likely that some of these disadvantages could be rectified through further investment in research and development.

Hybrid safety

Generally, well designed and carefully constructed hybrids are very safe. The primary hazards associated with hybrids are:

- **Pressure vessel failures** - Chamber insulation failure may allow hot combustion gases near the chamber walls leading to a "burn-through" in which the vessel ruptures.

- **Blow back** - For oxidizers that decompose exothermically such as nitrous oxide or hydrogen peroxide, flame or hot gasses from the combustion chamber can propagate back through the injector, igniting the oxidizer and leading to a tank explosion. Blow-back requires gases to flow back through the injector due to insufficient pressure drop which can occur during periods of unstable combustion. Blow back is inherent to specific oxidizers and is not possible with oxidizers such as oxygen or nitrogen tetroxide unless fuel is present in the oxidizer tank.

- **Hard starts** - An excess of oxidizer in the combustion chamber prior to ignition, particularly for monopropellants such as nitrous oxide, can result in a temporary over-pressure or "spike" at ignition.

Because the fuel in a hybrid does not contain an oxidizer, it will not combust explosively on its own. For this reason, hybrids are classified as having no TNT equivalent explosive power. In contrast, solid rockets often have TNT equivalencies similar in magnitude to the mass of the propellant grain. Liquids typically have TNT equivalencies calculated based on the amount of fuel and oxidizer which could realistically intimately combine before igniting explosively; this is often taken to be 10-20% of the total propellant mass. For hybrids, even filling the combustion chamber with oxidiser prior to ignition will not generally create an explosive with the solid fuel, the explosive equivalence is often quoted as 0%.

Organizations working on hybrids

In 1998 SpaceDev acquired all of the intellectual property, designs, and test results generated by over 200 hybrid rocket motor firings by the American Rocket Company over its eight year life. SpaceDev developed and produced all of the hybrid rocket motors for SpaceShipOne. SpaceDev is currently developing SpaceDev Streaker, an expendable small launch vehicle, and SpaceDev Dream Chaser, capable of both suborbital and orbital human space flight. Both Streaker and Dream Chaser use hybrid rocket motors that burn nitrous oxide and the synthetic rubber HTPB.

SpaceShipOne, the first private manned spacecraft, is powered by a hybrid rocket burning HTPB with nitrous oxide.

Space Propulsion Group was founded in 1999 by Dr. Arif Karabeyoglu, Prof. Brian Cantwell and others from Stanford University to develop high regression-rate liquefying hybrid rocket fuels. They have successfully fired motors as large as 12.5 in. diameter which produce 13,000 lbf. using the technology and are currently developing a 24 in. diameter, 25,000 lbf. motor to be initially fired in 2010.

Orbital Technologies Corporation (Orbitec) has been involved in some US government funded research on hybrid rockets including the "Vortex Hybrid" concept.

Environmental Aerospace Corporation (eAc) was incorporated in 1994 to develop hybrid rocket propulsion systems. It was included in the design competition for the SpaceShipOne motor but lost the contract to SpaceDev.

The Reaction Research Society (RRS), although known primarily for their work with liquid rocket propulsion, has a long history of research and development with hybrid rocket propulsion.

Copenhagen Suborbitals, a Danish rocket group, has designed and test-fired several hybrids using N_2O at first and currently LOX. Their fuel is epoxy, paraffin, or polyurethane.

Several universities have recently experimented with hybrid rockets. BYU, the University of Utah, and Utah State University launched a student-designed rocket called Unity IV in 1995 which burned the solid fuel hydroxyl-terminated polybutadiene (HTPB) with an oxidizer of gaseous oxygen, and in 2003 launched a larger version which burned HTPB with nitrous oxide.

Stanford University is the institution where liquid-layer combustion theory for hybrid rockets was developed. A group at Stanford is currently developing the Peregrine Sounding rocket which will be capable of 100 km altitude.

University of Brasilia's Hybrid Team has extensive research in paraffin/nitrous oxide hybrids having already made more than 50 tests fires. Hybrid Team is currently working liquefied propellant, numeric optimization and rocket design

Many other universities, such as the University of Michigan at Ann Arbor, the University of Arkansas at Little Rock, Hendrix College, the University of Illinois, Portland State University, and Texas A&M University have hybrid motor test stands that allow for student research with hybrid rockets. Boston University's student-run "Rocket Team", which in the past has launched only solid motor rockets, has completed several static tests of motors using paraffin and HTPB solid fuels and nitrous oxide as the oxidizer.

Florida Institute of Technology has successfully tested and evaluated hybrid technologies with their Panthr Project.

A United Kingdom-based team (laffin-gas) is using four N_2O hybrid rockets in a drag-racing style car. Each rocket has an outer diameter of 150mm and is 1.4m long. They use a fuel grain of high-density wound paper soaked in cooking oil. The N_2O supply is provided by Nitrogen-pressurised piston accumulators which provide a higher rate of delivery than N_2O gas alone and also provide damping of any reverse shock.

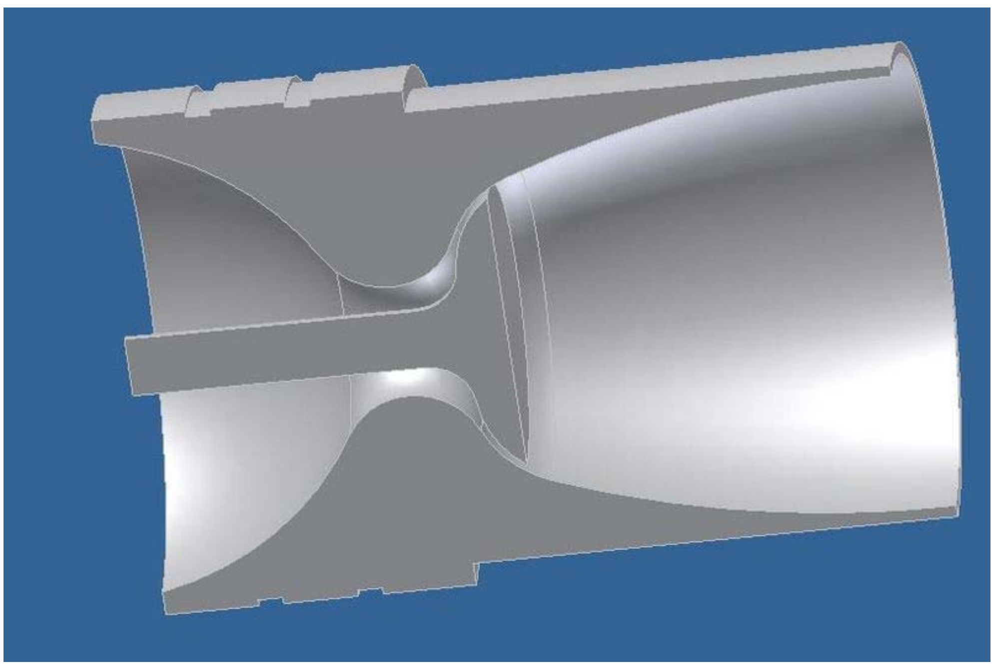
There are a number of hybrid rocket motor systems available for amateur/hobbyist use in high-powered model rocketry. These include the popular HyperTek systems and a number of 'Urbanski-Colburn Valved' (U/C) systems such as RATTWorks, Skyripper Systems, West Coast Hybrids, Contrail Rockets, and Propulsion Polymers. All of these systems use nitrous oxide as the oxidizer and a plastic fuel (such as PVC or PolyPropylene) or a polymer-based fuel such as HTPB. This reduces the cost per flight compared to solid rocket motors, although there is generally more 'GSE' (ground support equipment) required with hybrids.

Chapter 5

Expansion Deflection Nozzle

The **expansion-deflection nozzle** is an advanced rocket nozzle which achieves altitude compensation through interaction of the exhaust gas with the atmosphere, much like the plug and aerospike nozzles.

Description



This section through an ED nozzle clearly shows the pintle. In this example the outer wall appears similar to the internal contour of a bell nozzle.

It appears much like a standard bell nozzle, but at the throat is a 'centrebody' or 'pintle' which deflects the flow towards the walls. The exhaust gas flows past this in a more outward direction than in standard bell nozzles while expanding before being turned towards the exit. This allows for shorter nozzles than the standard design whilst maintaining nozzle expansion ratios. Because of the atmospheric boundary, the atmospheric pressure affects the exit area ratio so that atmospheric compensation can be obtained up to the geometric maximum allowed by the specific nozzle.

The nozzle operates in two distinct modes: open and closed. In closed wake mode, the exhaust gas fills the entire nozzle exit area. The ambient pressure at which the wake changes from open to closed modes is called the design pressure. If the ambient pressure reduces any further, additional expansion will occur outside of the nozzle much like a standard bell nozzle and no altitude compensation effect will be gained. In open wake mode, the exit area is dependant on the ambient pressure and the exhaust gas exits the nozzle in a doughnut shape as it does not fill the entire nozzle. Because the ambient pressure controls the exit area, the area ratio should be perfectly compensating to the altitude up to the design pressure.

If the pintle is designed to move along its axis of rotation, the throat area can be varied. This would allow for effective throttling, whilst maintaining chamber pressure.

Like the aerospike and plug nozzles, if modular combustion chambers were used in place of a single combustion chamber, then thrust vectoring would be achievable by throttling the flow to various chambers.

Developed models

The ED nozzle has been known about since the 1960s and there has been several attempts to develop it, with several reaching the level of static hot-firings. These were attempted by private companies, so no literature exists in the public domain from these efforts, which include the 'Expansion-Deflection 50k' (Rocketdyne), the 'Expansion-Deflection 10k' (Rocketdyne) and the RD-0126 (CADB). Rocketdyne also developed a third, smaller E-D nozzle.

Rocketdyne carried out their work during an initial surge in interest in the 1960s, initially developing the E-D 50k nozzle, which had a chamber pressure of 20.7 bar (2.07 MPa) delivering a thrust of 50,000 lbf (220 kN) and was uncooled, allowing it to be tested for a couple seconds at a time. The E-D 10k nozzle had a chamber pressure of 15.5 bar (1.55 MPa) delivering 10,000 lbf (44.5 kN), a cooled-thrust chamber and was tested in an altitude simulation facility. The smaller E-D nozzle developed 9900 lbf (44 kN) and was also used to test the altitude compensation ability. These tests confirmed a performance advantage over equivalent bell nozzles.

The Chemical Automatics Design Bureau E-D nozzle was fully cooled and used for hot-fire tests in 1998. Its centrebody houses the combustion chamber (much like the Astrium design mentioned below) allowing for a reduction in length, beyond that of the improved contouring.

Wickman Spacecraft & Propulsion Company have developed and static-tested a solid motor in conjunction with an E-D.

The University of Bristol, UK, has recently successfully tested gaseous Hydrogen/Air propellants as part of the STERN project. They are also involved in developing knowledge of the in-flight behaviour of the E-D nozzle using a hybrid rocket motor.

Potential uses

While research into this nozzle is on-going, it could be used before all its advantages are developed. As an upper stage, where it would be used in a low ambient pressure/vacuum environment specifically in closed wake mode, an E-D nozzle would offer weight reductions, length reductions and a potential increase to the specific impulse over bell nozzles (depending on engine cycle) allowing increased payloads. A study suggests it could add an additional 180 kg (400 lb) to the payload of an Ariane 5 over the new Vinci engine provided it is also an expander cycle. Such a nozzle could be brought into service before its altitude compensation abilities are developed.

It is also being investigated for Reaction Engines Skylon spaceplane. Employment on a single-stage-to-orbit (SSTO) rocket would use an E-D nozzles altitude compensating abilities fully, allowing for a substantial increase in payload. Reaction Engines, Airborne Engineering and the University of Bristol are currently involved in the STERN (Static Test Expansion deflection Rocket Nozzle) project to assess the abilities of the E-D nozzle, and to develop the technology.

Chapter 6

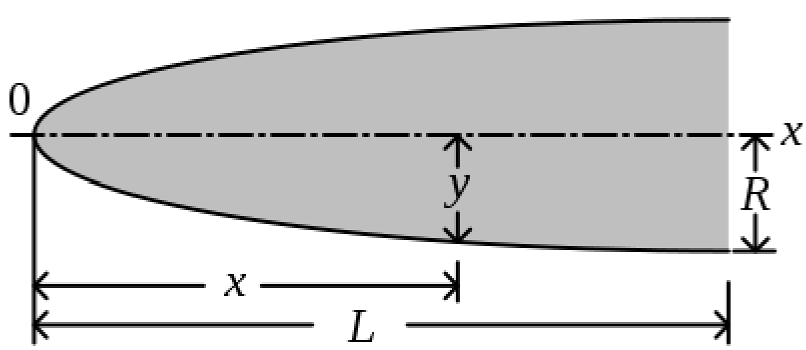
Nose Cone Design

Given the problem of the aerodynamic design of the nose cone section of any vehicle or body meant to travel through a compressible fluid medium (such as a rocket or aircraft, missile or bullet), an important problem is the determination of the nose cone geometrical shape for optimum performance. For many applications, such a task requires the definition of a solid of revolution shape that experiences minimal resistance to rapid motion through such a fluid medium, which consists of elastic particles.

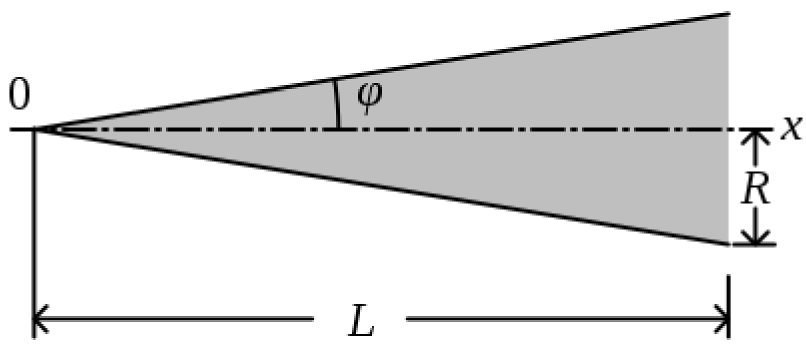
Nose cone shapes and equations

General dimensions

In all of the following nose cone shape equations, L is the overall length of the nose cone and R is the radius of the base of the nose cone. y is the radius at any point x, as x varies from 0, at the tip of the nose cone, to L. The equations define the 2-dimensional profile of the nose shape. The full body of revolution of the nose cone is formed by rotating the profile around the centerline *(C/L)*. Note that the equations describe the 'perfect' shape; practical nose cones are often blunted or truncated for manufacturing or aerodynamic reasons.



Conical



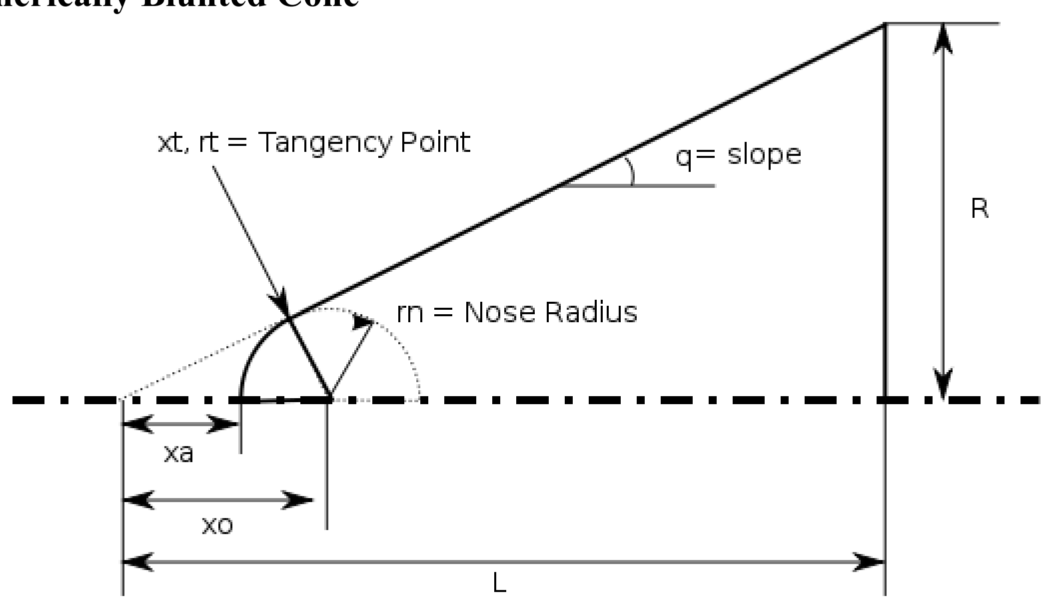
A very common nose cone shape is a simple cone. This shape is often chosen for its ease of manufacture, and is also often (mis)chosen for its drag characteristics. The sides of a conical profile are straight lines, so the diameter equation is simply

$$y = \frac{xR}{L}$$

Cones are sometimes defined by their half angle, ϕ :

$$\phi = \arctan\left(\frac{R}{L}\right) \text{and } y = x\tan(\phi)$$

Spherically Blunted Cone



In practical applications, a conical nose is often blunted by capping it with a segment of a sphere. The tangency point where the sphere meets the cone can be found from:

$$x_t = \frac{L^2}{R}\sqrt{\frac{r_n^2}{R^2+L^2}}$$
$$y_t = \frac{x_t R}{L}$$
where:
r_n is the radius of the spherical nose cap.

The center of the spherical nose cap can be found from:

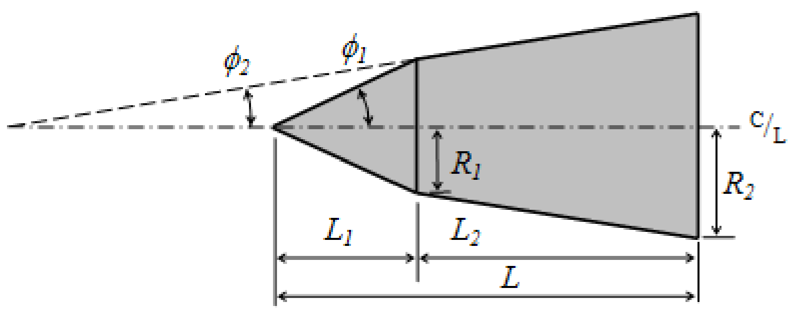
$$x_o = x_t + \sqrt{r_n^2 - y_t^2}$$

And the apex point can be found from:

$$x_a = x_o - r_n$$

Bi-conic

A bi-conic nose cone shape is simply a cone with length L_1 stacked on top of a frustum of a cone (commonly known as a *conical transition section* shape) with length L_2, where the base of the upper cone is equal in radius R_1 to the top radius of the smaller frustum with base radius R_2.



$L = L_1 + L_2$

- for $0 \le x \le L_1$: $y = \dfrac{xR_1}{L_1}$

half angle :

$$\phi_1 = \arctan\left(\frac{R_1}{L_1}\right) \text{ and } y = x\tan(\phi_1)$$

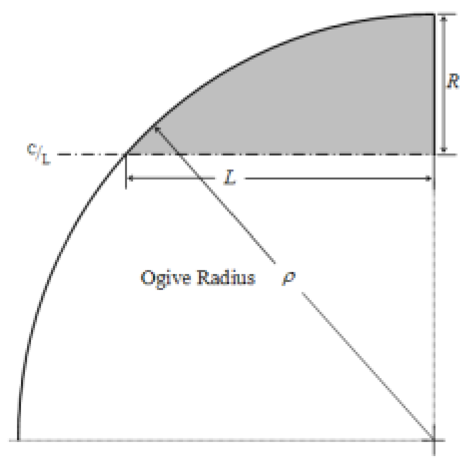
- for $L_1 \leq x \leq L$: $y = R_1 + \frac{(x - L_1)(R_2 - R_1)}{L_2}$

half angle :

$$\phi_2 = \arctan\left(\frac{R_2 - R_1}{L_2}\right) \text{ and } y = R_1 + (x - L_1)\tan(\phi_2)$$

Tangent ogive

Next to a simple cone, the tangent ogive shape is the most familiar in hobby rocketry. The profile of this shape is formed by a segment of a circle such that the rocket body is tangent to the curve of the nose cone at its base; and the base is on the radius of the circle. The popularity of this shape is largely due to the ease of constructing its profile.



The radius of the circle that forms the ogive is called the Ogive Radius ρ and it is related to the length and base radius of the nose cone as expressed by the formula:

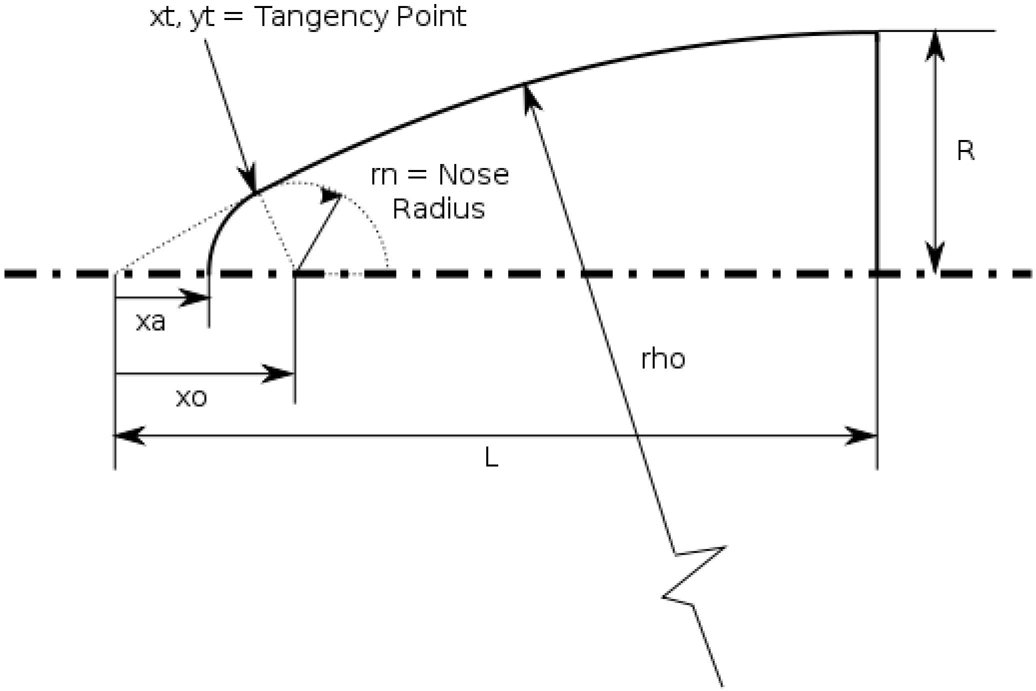
$$\rho = \frac{R^2 + L^2}{2R}$$

The radius *y* at any point *x*, as *x* varies from *0* to *L* is:

$$y = \sqrt{\rho^2 - (L - x)^2} + R - \rho$$

The nose cone length, *L*, must be equal to, or less than the Ogive Radius ρ. If they are equal, then the shape is a hemisphere.

Spherically blunted tangent ogive



A tangent ogive nose is often blunted by capping it with a segment of a sphere. The tangency point where the sphere meets the tangent ogive can be found from:

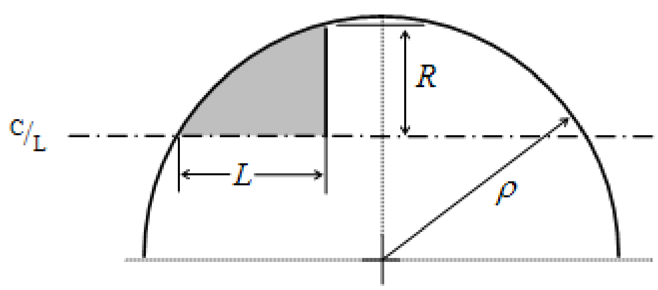
$$x_o = L - \sqrt{(\rho - r_n)^2 - (\rho - R)^2}$$
$$y_t = \frac{r_n(\rho - R)}{\rho - r_n}$$
$$x_t = x_o - \sqrt{r_n^2 - y_t^2}$$

where:
r_n is the radius and x_o is the center of the spherical nose cap.

And the apex point can be found from:

$$x_a = x_o - r_n$$

Secant ogive

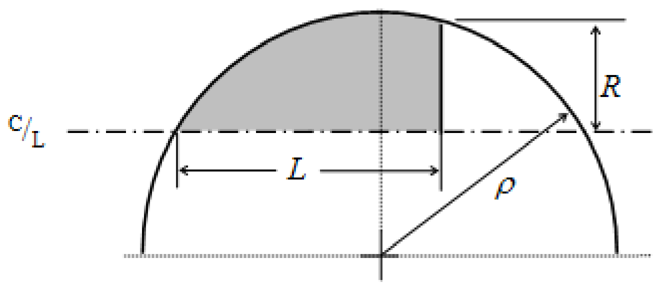


The profile of this shape is also formed by a segment of a circle, but the base of the shape is not on the radius of the circle defined by the ogive radius. The rocket body will **not** be tangent to the curve of the nose at its base. The Ogive Radius ρ is not determined by *R* and *L* (as it is for a tangent ogive), but rather is one of the factors to be chosen to define the nose shape. If the chosen Ogive Radius of a Secant Ogive is greater than the Ogive Radius of a Tangent Ogive with the same *R* and *L*, then the resulting Secant Ogive appears as a Tangent Ogive with a portion of the base truncated.

$$\rho > \frac{R^2 + L^2}{2R} \text{ and } \alpha = \arctan\left(\frac{R}{L}\right) - \arccos\left(\frac{\sqrt{L^2 + R^2}}{2\rho}\right)$$

Then the radius *y* at any point *x* as *x* varies from *0* to *L* is:

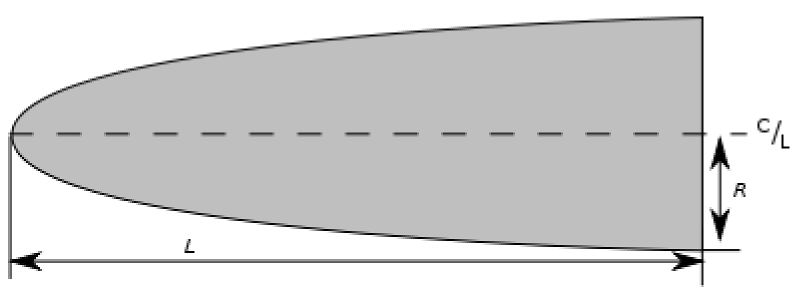
$$y = \sqrt{\rho^2 - (\rho\cos\alpha - x)^2} + \rho\sin\alpha$$



If the chosen ρ is less than the tangent ogive ρ, then the result will be a Secant Ogive that bulges out to a maximum diameter that is greater than the base diameter. The classic example of this shape is the nose cone of the Honest John. Also, the chosen ogive radius must be greater than twice the length of the nose cone.

$$2L < \rho < \frac{R^2 + L^2}{2R}$$

Elliptical



The profile of this shape is one-half of an ellipse, with the major axis being the centerline and the minor axis being the base of the nose cone. A rotation of a full ellipse about its major axis is called a prolate spheroid, so an elliptical nose shape would properly be known as a prolate hemispheroid. This shape is popular in subsonic flight (such as model rocketry) due to the blunt nose and tangent base. This is not a shape normally found in professional rocketry. If *R* equals *L*, this is a hemisphere.

$$y = R\sqrt{1 - \frac{x^2}{L^2}}$$

Parabolic

This nose shape is not the blunt shape that is envisioned when people commonly refer to a 'parabolic' nose cone. The Parabolic Series nose shape is generated by rotating a segment of a parabola around a line parallel to its Latus rectum. This construction is similar to that of the Tangent Ogive, except that a parabola is the defining shape rather than a circle. Just as it does on an Ogive, this construction produces a nose shape with a sharp tip.

For $0 \leq K' \leq 1$: $$y = R\left(\frac{2(\frac{x}{L}) - K'(\frac{x}{L})^2}{2 - K'}\right)$$

K' can vary anywhere between 0 and 1, but the most common values used for nose cone shapes are:

K' = 0 for a cone
K' = 0.5 for a 1/2 parabola
K' = 0.75 for a 3/4 parabola
K' = 1 for a full parabola

For the case of the full Parabola (K'=1) the shape is tangent to the body at its base, and the base is on the axis of the parabola. Values of K' less than one result in a 'slimmer' shape, whose appearance is similar to that of the secant ogive. The shape is no longer tangent at the base, and the base is parallel to, but offset from, the axis of the parabola.

Power series

The Power Series includes the shape commonly referred to as a 'parabolic' nose cone, but the shape correctly known as a parabolic nose cone is a member of the Parabolic Series, and is something completely different. The Power Series shape is characterized by its (usually) blunt tip, and by the fact that its base is not tangent to the body tube. There is always a discontinuity at the nose cone / body joint that looks distinctly non-aerodynamic. The shape can be modified at the base to smooth out this discontinuity. Both a flat-faced cylinder and a cone are shapes that are members of the Power Series.

The Power series nose shape is generated by rotating a parabola about its axis. The base of the nose cone is parallel to the latus rectum of the parabola, and the factor *n* controls the 'bluntness' of the shape. As *n* decreases towards zero, the Power Series nose shape becomes increasingly blunt. At values of *n* above about 0.7, the tip becomes sharp.

For $0 \leq n \leq 1$: $y = R\left(\frac{x}{L}\right)^n$

Where:

n = 1 for a cone
n = 0.75 for a 3/4 power
n = 0.5 for a 1/2 power (parabola)
n = 0 for a cylinder

Haack series

Unlike all of the nose cone shapes above, the Haack Series shapes are not constructed from geometric figures. The shapes are instead mathematically derived for the purpose of minimizing drag. While the series is a continuous set of shapes determined by the value of *C* in the equations below, two values of *C* have particular significance: when *C = 0*, the notation *LD* signifies minimum drag for the given length and diameter, and when *C = 1/3*, *LV* indicates minimum drag for a given length and volume. The Haack series nose cones are not perfectly tangent to the body at their base, however the discontinuity is usually so slight as to be imperceptible. Haack nose tips do not come to a sharp point, but are slightly rounded.

$$\theta = \arccos\left(1 - \frac{2x}{L}\right)$$
$$y = \frac{R\sqrt{\theta - \frac{\sin(2\theta)}{2} + C\sin^3\theta}}{\sqrt{\pi}}$$

Where:

C = 1/3 for LV-Haack
C = 0 for LD-Haack

Von Kármán

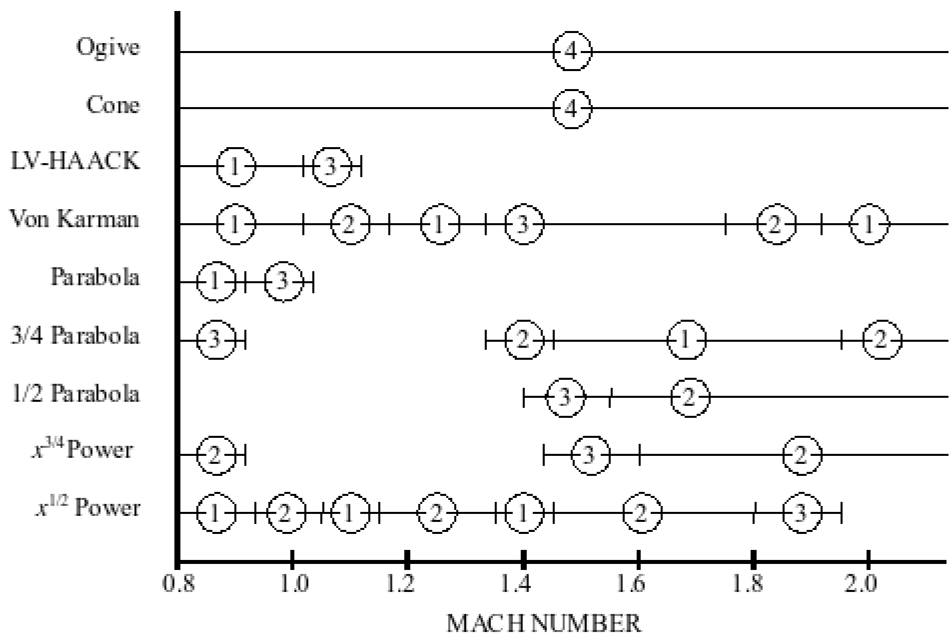
The Haack series giving minimum drag for the given length and diameter, LD-Haack, is commonly referred to as the Von Kármán or the *Von Kármán Ogive*.

Nose cone drag characteristics

For aircraft and rockets, below Mach .8, the nose pressure drag is essentially zero for all shapes. The major significant factor is friction drag, which is largely dependent upon the wetted area, the surface smoothness of that area, and the presence of any discontinuities in the shape. For example, in strictly subsonic rockets a short, blunt, smooth elliptical shape is usually best. In the transonic region and beyond, where the pressure drag increases dramatically, the effect of nose shape on drag becomes highly significant. The factors influencing the pressure drag are the general shape of the nose cone, its fineness ratio, and its bluffness ratio.

Influence of the general shape

Many references on nose cone design contain empirical data comparing the drag characteristics of various nose shapes in different flight regimes. The chart shown here seems to be the most comprehensive and useful compilation of data for the flight regime of greatest interest. This chart generally agrees with more detailed, but less comprehensive data found in other references (most notably the USAF Datcom).



Comparison of drag characteristics of various nose cone shapes in the transonic to low-mach regions. Rankings are: superior (1), good (2), fair (3), inferior (4).

In many nose cone designs, the greatest concern is flight performance in the transonic region from 0.8 to 1.2 Mach. Although data is not available for many shapes in the transonic region, the table clearly suggests that either the Von Kármán shape, or Power

Series shape with n = 1/2, would be preferable to the popular Conical or Ogive shapes, for this purpose.

This observation goes against the often-repeated conventional wisdom that a conical nose is optimum for "Mach-breaking". Fighter aircraft are probably good examples of nose shapes optimized for the transonic region, although their nose shapes are often distorted by other considerations of avionics and inlets. For example, an F-16 nose appears to be a very close match to a Von Karman shape.

Influence of the fineness ratio

The ratio of the length of a nose cone compared to its base diameter is known as the *fineness ratio*. This is sometimes also called the *aspect ratio*, though that term is usually applied to wings and fins. Fineness ratio is often applied to the entire vehicle, considering the overall length and diameter. The length/diameter relation is also often called the *caliber* of a nose cone. At supersonic speeds, the fineness ratio has a significant effect on nose cone wave drag, particularly at low ratios; but there is very little additional gain for ratios increasing beyond 5:1. As the fineness ratio increases, the wetted area, and thus the skin friction component of drag, is also going to increase. Therefore the minimum drag fineness ratio is ultimately going to be a tradeoff between the decreasing wave drag and increasing friction drag.

Chapter 7

Rocket Propellant

Rocket propellant is mass that is stored in some form of propellant tank, prior to being used as the propulsive mass that is ejected from a rocket engine in the form of a fluid jet to produce thrust. A fuel propellant is often burned with an oxidizer propellant to produce large volumes of very hot gas. These gases expand and push on a nozzle, which accelerates them until they rush out of the back of the rocket at extremely high speed, making thrust. Sometimes the propellant is not burned, but can be externally heated for more performance. For smaller attitude control thrusters, a compressed gas escapes the spacecraft through a propelling nozzle.

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.

In ion propulsion, the propellant is made of electrically charged atoms (ions), which are electromagnetically pushed out of the back of the spacecraft. Magnetically accelerated ion drives are not usually considered to be rockets however, but a similar class of thrusters use electrical heating and magnetic nozzles.

Overview

The Space Shuttle Atlantis during ascent.

Rockets create thrust by expelling mass backwards in a high speed jet. Chemical rockets, the subject here, create thrust by reacting propellants within a combustion chamber into a very hot gas at high pressure, which is then expanded and accelerated by passage through a nozzle at the rear of the rocket. The amount of the resulting forward force, known as thrust, that is produced is the mass flow rate of the propellants multiplied by their exhaust velocity (relative to the rocket), as specified by Newton's third law of motion. Thrust is therefore the equal and opposite reaction that moves the rocket, and not by interaction of the exhaust stream with air around the rocket. Equivalently, one can think of a rocket being accelerated upwards by the pressure of the combusting gases against the combustion chamber and nozzle. This operational principle stands in contrast to the commonly-held assumption that a rocket "pushes" against the air behind or below it. Rockets in fact perform better in outer space (where there is nothing behind or beneath them to push against), because there is a reduction in air pressure on the outside of the engine, and because it is possible to fit a longer nozzle without suffering from flow separation.

The maximum velocity that a rocket can attain in the absence of any external forces is primarily a function of its mass ratio and its *exhaust velocity*. The relationship is described by the *rocket equation*: $V_f = V_e \ln(M_0 / M_f)$. The mass ratio is just a way to express what proportion of the rocket is propellant (fuel/oxidizer combination) prior to engine ignition. Typically, a single-stage rocket might have a mass fraction of 90% propellant, 10% structure, and hence a mass ratio of 10:1 . The impulse delivered by the motor to the rocket vehicle per weight of fuel consumed is often reported as the rocket propellant's *specific impulse*. A propellant with a higher specific impulse is said to be more efficient because more thrust is produced while consuming a given amount of propellant.

Lower stages will usually use high-density (low volume) propellants because of their lighter tankage to propellant weight ratios and because higher performance propellants require higher expansion ratios for maximum performance than can be attained in atmosphere. Thus, the Apollo-Saturn V first stage used kerosene-liquid oxygen rather than the liquid hydrogen-liquid oxygen used on its upper stages Similarly, the Space Shuttle uses high-thrust, high-density solid rocket boosters for its lift-off with the liquid hydrogen-liquid oxygen SSMEs used partly for lift-off but primarily for orbital insertion.

Chemical propellants

There are three main types of propellants: solid, liquid, and hybrid.

History

The earliest rockets were created hundreds of years ago by the Chinese, and were used primarily for fireworks displays and as weapons. They were fueled with black powder, a type of gunpowder consisting of a mixture of charcoal, sulfur and potassium nitrate (saltpeter). Rocket propellant technology did not advance until the end of the 19th century, by which time smokeless powder had been developed, originally for use in firearms and artillery pieces. Smokeless powders and related compounds have seen use as double-base propellants.

Description

Solid propellants (and almost all rocket propellants) consist of an oxidizer and a fuel. In the case of gunpowder, the fuel is charcoal, the oxidizer is potassium nitrate, and sulfur serves as a catalyst. (Note: sulfur is not a true catalyst in gunpowder as it is consumed to a great extent into a variety of reaction products such as K_2S. The sulfur acts mainly as a sensitizer lowering threshold of ignition.) During the 1950s and 60s researchers in the United States developed what is now the standard high-energy solid rocket fuel, Ammonium Perchlorate Composite Propellant (APCP). This mixture is primarily ammonium perchlorate powder (an oxidizer), combined with fine aluminium powder (a fuel), held together in a base of PBAN or HTPB (rubber-like fuels). The mixture is formed as a liquid, and then cast into the correct shape and cured into a rubbery solid.

Advantages

Solid-fueled rockets are much easier to store and handle than liquid-fueled rockets, which makes them ideal for military applications. In the 1970s and 1980s the U.S. switched entirely to solid-fueled ICBMs: the LGM-30 Minuteman and LG-118A Peacekeeper (MX). In the 1980s and 1990s, the USSR/Russia also deployed solid-fueled ICBMs (RT-

23, RT-2PM, and RT-2UTTH), but retains two liquid-fueled ICBMs (R-36 and UR-100N). All solid-fueled ICBMs on both sides have three initial solid stages and a precision maneuverable liquid-fueled bus used to fine tune the trajectory of the reentry vehicle.

Their simplicity also makes solid rockets a good choice whenever large amounts of thrust are needed and cost is an issue. The Space Shuttle and many other orbital launch vehicles use solid-fueled rockets in their first stages (solid rocket boosters) for this reason.

Disadvantages

Relative to liquid fuel rockets, solid rockets have a number of disadvantages. Solid rockets have a lower specific impulse than liquid-fueled rockets. It is also difficult to build a large mass ratio solid rocket because almost the entire rocket is the combustion chamber, and must be built to withstand the high combustion pressures. If a solid rocket is used to go all the way to orbit, the payload fraction is very small. (For example, the Orbital Sciences Pegasus rocket is an air-launched three-stage solid rocket orbital booster. Launch mass is 23,130 kg, low earth orbit payload is 443 kg, for a payload fraction of 1.9%. Compare to a Delta IV Medium, 249,500 kg, payload 8600 kg, payload fraction 3.4% without air-launch assistance.)

A drawback to solid rockets is that they cannot be throttled in real time, although a predesigned thrust schedule can be created by altering the interior propellant geometry.

Solid rockets can often be shut down before they run out of fuel. Essentially, the rocket is vented or an extinguishant injected so as to terminate the combustion process. In some cases termination destroys the rocket, and then this is typically only done by a Range Safety Officer if the rocket goes awry. The third stages of the Minuteman and MX rockets have precision shutdown ports which, when opened, reduce the chamber pressure so abruptly that the interior flame is blown out. This allows a more precise trajectory which improves targeting accuracy.

Finally, casting very large single-grain rocket motors has proved to be a very tricky business. Defects in the grain can cause explosions during the burn, and these explosions can increase the burning propellant surface enough to cause a runaway pressure increase, until the case fails.

History

Though early rocket theorists, such as Konstantin Tsiolkovsky, proposed liquid hydrogen and liquid oxygen as propellants, the first liquid-fueled rocket, launched by Robert Goddard on March 16, 1926, used gasoline and liquid oxygen. Liquid hydrogen was first used by the engines designed by Pratt and Whitney for the Lockheed CL-400 Suntan reconnaissance aircraft in the mid-1950s. In the mid-1960s, the Centaur and Saturn upper stages were both using liquid hydrogen and liquid oxygen.

The highest specific impulse chemistry ever test-fired in a rocket engine was lithium and fluorine, with hydrogen added to improve the exhaust thermodynamics (making this a tripropellant). The combination delivered 542 seconds (5.32 kN·s/kg, 5320 m/s) specific impulse in a vacuum. The impracticality of this chemistry highlights why exotic propellants are not actually used: to make all three components liquids, the hydrogen must be kept below -252 °C (just 21 K) and the lithium must be kept above 180 °C (453 K). Lithium and fluorine are both extremely corrosive, liquid lithium ignites on contact with air, fluorine ignites on contact with most fuels, and hydrogen, while not hypergolic, is an explosive hazard. Fluorine and the hydrogen fluoride (HF) in the exhaust are very toxic, which damages the environment, makes work around the launch pad difficult, and makes getting a launch license that much more difficult. The rocket exhaust is also ionized, which would interfere with radio communication with the rocket.

Current Types

The most common liquid propellants in use today:

- LOX and kerosene (RP-1). Used for the lower stages of most Russian and Chinese boosters, the first stages of the Saturn V and Atlas V, and all stages of the developmental Falcon 1 and Falcon 9. Very similar to Robert Goddard's first rocket. This combination is widely regarded as the most practical for boosters that lift off at ground level and therefore must operate at full atmospheric pressure.
- LOX and liquid hydrogen, used in the Space Shuttle orbiter, the Centaur upper stage of the Atlas V, Saturn V upper stages, the newer Delta IV rocket, the H-IIA rocket, and most stages of the European Ariane rockets.

- Nitrogen tetroxide (N_2O_4) and hydrazine (N_2H_4), MMH, or UDMH. Used in military, orbital and deep space rockets, because both liquids are storable for long periods at reasonable temperatures and pressures. N_2O_4/UDMH is the main fuel for the Proton rocket. This combination is hypergolic, making for attractively simple ignition sequences. The major inconvenience is that these propellants are highly toxic, hence they require careful handling.

- Monopropellants such as hydrogen peroxide, hydrazine and nitrous oxide are primarily used for attitude control and spacecraft station-keeping where their long-term storability, simplicity of use and ability to provide the tiny impulses needed, outweighs their lower specific impulse as compared to bipropellants. Hydrogen peroxide is also used to drive the turbopumps on the first stage of the Soyuz launch vehicle.

Historical propellants

These include propellants such as syntin, which is an expensive high energy hydrocarbon fuel which was used on Soyuz U2 until 1995.

Advantages

Liquid fueled rockets have higher specific impulse than solid rockets and are capable of being throttled, shut down, and restarted. Only the combustion chamber of a liquid fueled rocket needs to withstand combustion pressures and temperatures and they can be regeneratively cooled by the liquid propellant. On vehicles employing turbopumps, the propellant tanks are at very much less pressure than the combustion chamber, and thus can be built far more lightly than a solid propellant rocket case, permitting a higher mass ratio. For these reasons, most orbital launch vehicles use liquid propellants.

The primary performance advantage of liquid propellants is due to the oxidizer. Several practical liquid oxidizers (liquid oxygen, nitrogen tetroxide, and hydrogen peroxide) are available which have much better specific impulse than the ammonium perchlorate used in most solid rockets, when paired with comparable fuels. These facts have led to the use of hybrid propellants: a storable oxidizer used with a solid fuel, which retain most virtues of both liquids (high ISP) and solids (simplicity).

While liquid propellants are cheaper than solid propellants, for orbital launchers, the cost savings do not, and historically have not mattered; the cost of propellant is a very small portion of the overall cost of the rocket.

Disadvantages

The main difficulties with liquid propellants are also with the oxidizers. These are generally at least moderately difficult to store and handle due to their high reactivity with common materials, may have extreme toxicity (nitric acids), moderately cryogenic (liquid oxygen), or both (liquid fluorine, FLOX- a fluorine/LOX mix). Several exotic oxidizers have been proposed: liquid ozone (O_3), ClF_3, and ClF_5, all of which are unstable, energetic, and toxic.

Liquid fueled rockets also require potentially troublesome valves and seals and thermally stressed combustion chambers, which increase the cost of the rocket. Many employ specially designed turbopumps which raise the cost enormously due to difficult fluid flow patterns that exist within the casings.

Gas propellants

A gas propellant usually involves some sort of compressed gas. However, due to the low density and high weight of the pressure vessel, gases see little current use, but are sometimes used for vernier engines, particularly with inert propellants.

GOX was used as one of the propellant for the Buran program for the orbital manoeuvring system.

Hybrid propellants

A hybrid rocket usually has a solid fuel and a liquid or gas oxidizer. The fluid oxidizer can make it possible to throttle and restart the motor just like a liquid fueled rocket. Hybrid rockets are also cleaner than solid rockets because practical high-performance solid-phase oxidizers all contain chlorine, versus the more benign liquid oxygen or nitrous oxide used in hybrids. Because just one propellant is a fluid, hybrids are simpler than liquid rockets.

Hybrid motors suffer two major drawbacks. The first, shared with solid rocket motors, is that the casing around the fuel grain must be built to withstand full combustion pressure and often extreme temperatures as well. However, modern composite structures handle this problem well, and when used with nitrous oxide and a solid rubber propellent (HTPB), relatively small percentage of fuel is needed anyway, so the combustion chamber is not especially large.

The primary remaining difficulty with hybrids is with mixing the propellants during the combustion process. In solid propellants, the oxidizer and fuel are mixed in a factory in carefully controlled conditions. Liquid propellants are generally mixed by the injector at the top of the combustion chamber, which directs many small swift-moving streams of fuel and oxidizer into one another. Liquid fueled rocket injector design has been studied at great length and still resists reliable performance prediction. In a hybrid motor, the mixing happens at the melting or evaporating surface of the fuel. The mixing is not a well-controlled process and generally quite a lot of propellant is left unburned, which limits the efficiency and thus the exhaust velocity of the motor. Additionally, as the burn continues, the hole down the center of the grain (the 'port') widens and the mixture ratio tends to become more oxidiser rich.

There has been much less development of hybrid motors than solid and liquid motors. For military use, ease of handling and maintenance have driven the use of solid rockets. For orbital work, liquid fuels are more efficient than hybrids and most development has concentrated there. There has recently been an increase in hybrid motor development for nonmilitary suborbital work:

- The Reaction Research Society, although known primarily for their work with liquid rocket propulsion, has a long history of research and development with hybrid rocket propulsion.

- Several universities have recently experimented with hybrid rockets. Brigham Young University, the University of Utah and Utah State University launched a student-designed rocket called Unity IV in 1995 which burned the solid fuel hydroxy-terminated polybutadiene (HTPB) with an oxidizer of gaseous oxygen, and in 2003 launched a larger version which burned HTPB with nitrous oxide. Stanford University researches nitrous-oxide/paraffin hybrid motors.

- The Rochester Institute of Technology was building a HTPB hybrid rocket to launch small payloads into space and to several near Earth objects. Its first launch was scheduled for Summer 2007.

- Scaled Composites SpaceShipOne, the first private manned spacecraft, is powered by a hybrid rocket burning HTPB with nitrous oxide. The hybrid rocket engine was manufactured by SpaceDev. SpaceDev partially based its motors on experimental data collected from the testing of AMROC's (American Rocket Company) motors at NASA's Stennis Space Center's E1 test stand. Motors ranging from as small as 1000 lbf (4.4 kN) to as large as 250,000 lbf (1.1 MN) thrust were successfully tested. SpaceDev purchased AMROCs assets after the company was shut down for lack of funding.

Inert propellants

Some rocket designs have their propellants obtain their energy from non chemical or even external sources. For example water rockets use the compressed gas, typically air, to force the water out of the rocket.

Solar thermal rockets and Nuclear thermal rockets typically propose to use liquid hydrogen for an I_{sp} (Specific Impulse) of around 600–900 seconds, or in some cases water that is exhausted as steam for an I_{sp} of about 190 seconds.

Additionally for low performance requirements such as attitude jets, inert gases such as nitrogen have been employed.

Mixture ratio

The theoretical exhaust velocity of a given propellant chemistry is a function of the energy released per unit of propellant mass (specific energy). Unburned fuel or oxidizer drags down the specific energy. However, most rockets run fuel-rich.

The usual explanation for fuel-rich mixtures is that fuel-rich mixtures have lower molecular weight exhaust, which by reducing *M* increases the ratio $\frac{\sqrt{T_c}}{M}$ which is approximately equal to the theoretical exhaust velocity. This explanation, though found in some textbooks, is wrong. Fuel-rich mixtures actually have lower theoretical exhaust velocities, because $\sqrt{T_c}$ decreases as fast or faster than *M*.

The nozzle of the rocket converts the thermal energy of the propellants into directed kinetic energy. This conversion happens in a short time, on the order of one millisecond. During the conversion, energy must transfer very quickly from the rotational and vibrational states of the exhaust molecules into translation. Molecules with fewer atoms (like CO and H_2) store less energy in vibration and rotation than molecules with more atoms (like CO_2 and H_2O). These smaller molecules transfer more of their rotational and vibrational energy to translation energy than larger molecules, and the resulting improvement in nozzle efficiency is large enough that real rocket engines improve their actual exhaust velocity by running rich mixtures with somewhat lower theoretical exhaust velocities.

The effect of exhaust molecular weight on nozzle efficiency is most important for nozzles operating near sea level. High expansion rockets operating in a vacuum see a much smaller effect, and so are run less rich. The Saturn-II stage (a LOX/LH_2 rocket) varied its mixture ratio during flight to optimize performance.

LOX/hydrocarbon rockets are run only somewhat rich (O/F mass ratio of 3 rather than stoichiometric of 3.4 to 4), because the energy release per unit mass drops off quickly as the mixture ratio deviates from stoichiometric. LOX/LH_2 rockets are run very rich (O/F mass ratio of 4 rather than stoichiometric 8) because hydrogen is so light that the energy release per unit mass of propellant drops very slowly with extra hydrogen. In fact, LOX/LH_2 rockets are generally limited in how rich they run by the performance penalty of the mass of the extra hydrogen tankage, rather than the mass of the hydrogen itself.

Another reason for running rich is that off-stoichiometric mixtures burn cooler than stoichiometric mixtures, which makes engine cooling easier. And as most engines are made of metal or carbon, hot oxidizer-rich exhaust is extremely corrosive, where fuel-rich exhaust is less so. American engines have all been fuel-rich. Some Soviet engines have been oxidizer-rich.

Additionally, there is a difference between mixture ratios for optimum I_{sp} and optimum thrust. During launch, shortly after takeoff, high thrust is at a premium. This can be achieved at some temporary reduction of I_{sp} by increasing the oxidiser ratio initially, and then transitioning to more fuel-rich mixtures. Since engine size is typically scaled for takeoff thrust this permits reduction of the weight of rocket engine, pipes and pumps and the extra propellant use can be more than compensated by increases of acceleration towards the end of the burn by having a reduced dry mass.

Propellant density

Although liquid hydrogen gives a high I_{sp}, its low density is a significant disadvantage: hydrogen occupies about 7x more volume per kilogram than dense fuels such as kerosene. This not only penalises the tankage, but also the pipes and fuel pumps leading from the tank, which need to be 7x bigger and heavier. (The oxidiser side of the engine and tankage is of course unaffected.) This makes the vehicle's dry mass much higher, so the use of liquid hydrogen is not such a big win as might be expected. Indeed, some dense hydrocarbon/LOX propellant combinations have higher performance when the dry mass penalties are included.

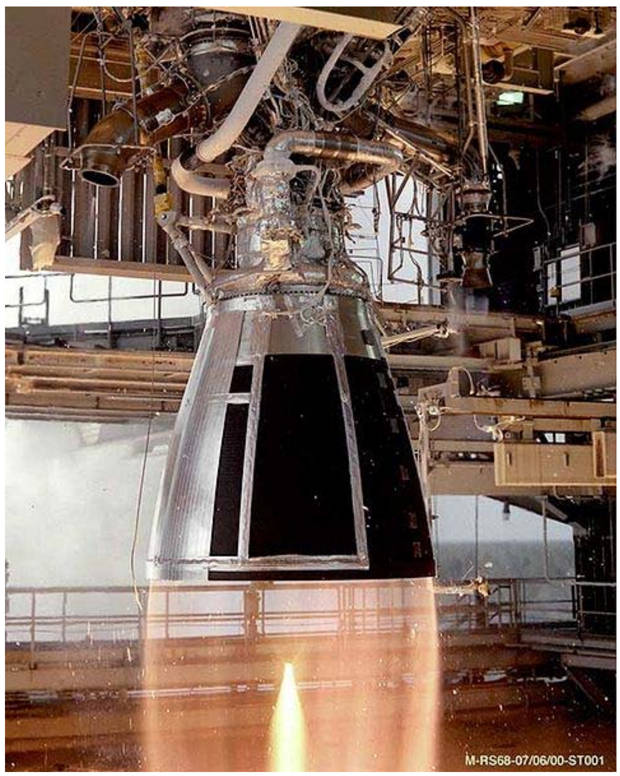
Due to lower I_{sp}, dense propellant launch vehicles have a higher takeoff mass, but this does not mean a proportionately high cost; on the contrary, the vehicle may well end up cheaper. Liquid hydrogen is quite an expensive fuel to produce and store, and causes many practical difficulties with design and manufacture of the vehicle.

Because of the higher overall weight, a dense-fueled launch vehicle necessarily requires higher takeoff thrust, but it carries this thrust capability all the way to orbit. This, in combination with the better thrust/weight ratios, means that dense-fueled vehicles reach orbit earlier, thereby minimizing losses due to gravity drag. Thus, the effective delta-v requirement for these vehicles are reduced.

However, liquid hydrogen does give clear advantages when the overall mass needs to be minimised; for example the Saturn V vehicle used it on the upper stages; this reduced weight meant that the dense-fueled first stage could be made significantly smaller, saving quite a lot of money.

Chapter 8

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.

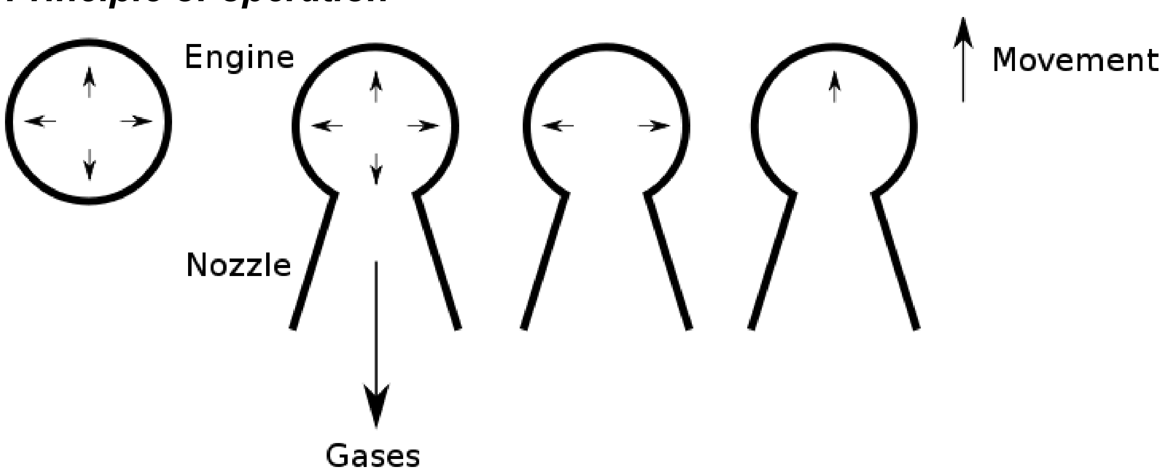
Rocket motor (or **solid-propellant rocket motor**) is a synonymous term with rocket engine that usually refers to solid rocket engines.

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

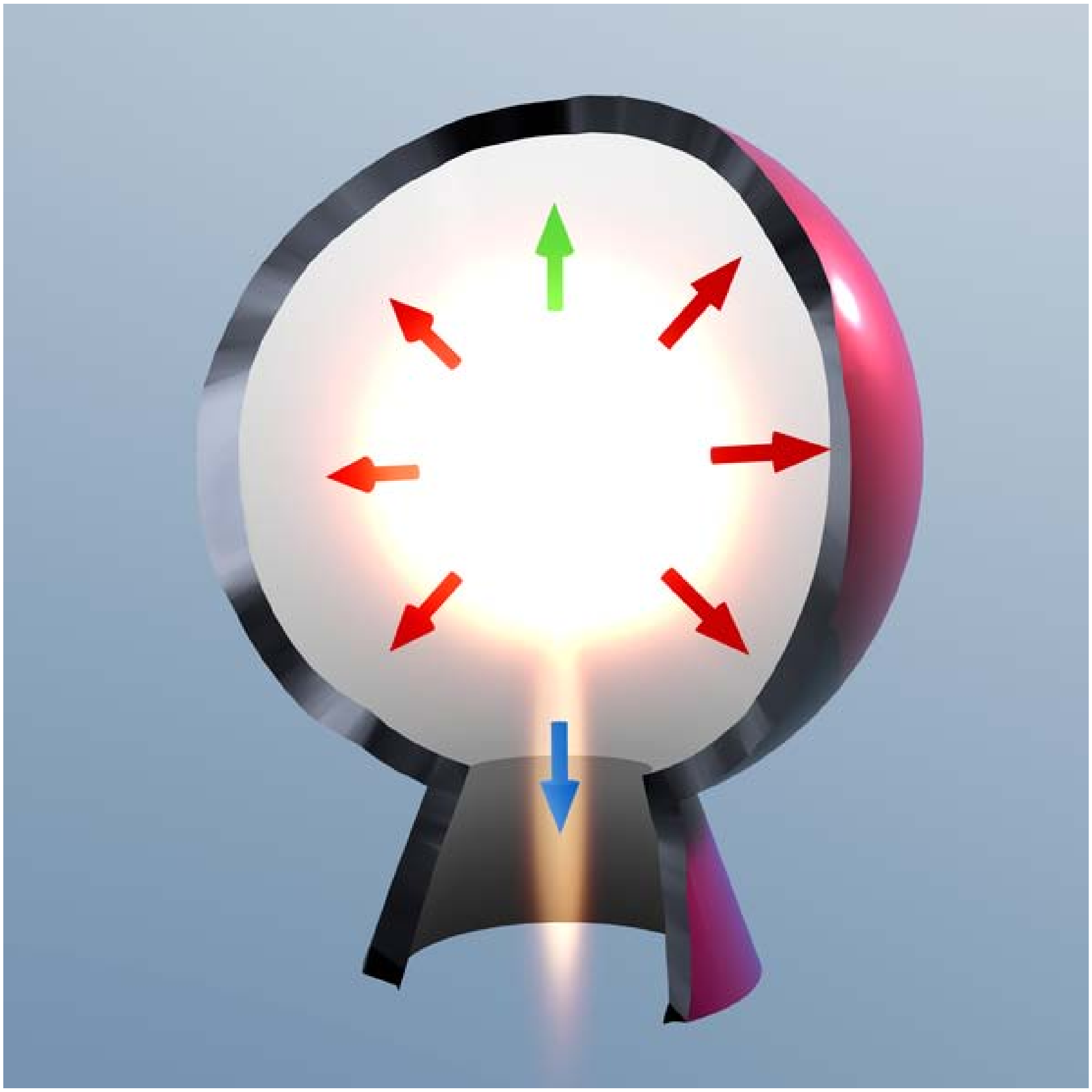
Hybrid rockets have a solid propellant in the combustion chamber and a second liquid or gas propellant is added to permit it to burn.

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.

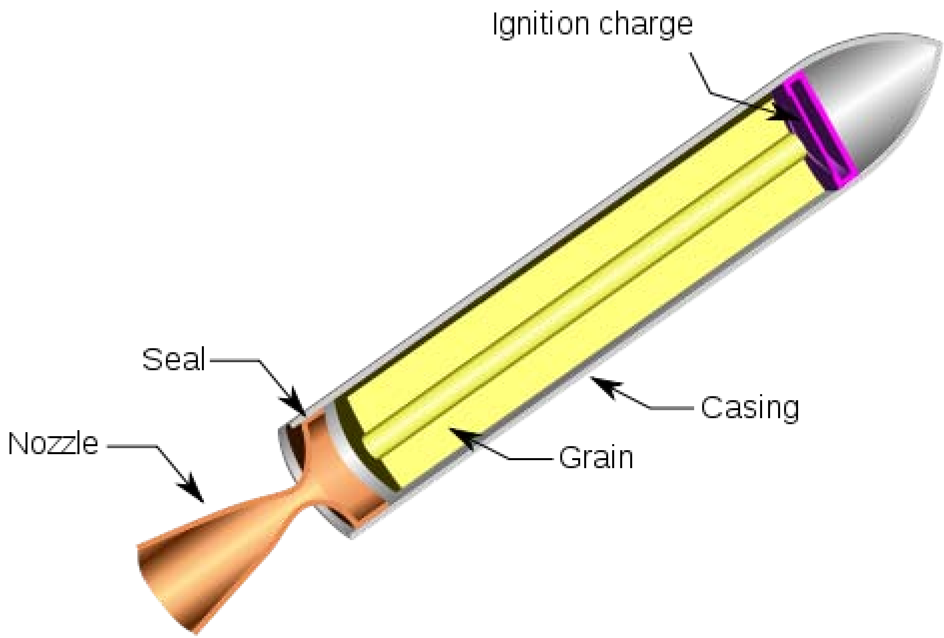
The fluid exhaust is then passed through a supersonic propelling nozzle which uses heat energy of the gas to accelerate the exhaust to very high speed, and the reaction to this pushes the engine in the opposite direction.

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^* :

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

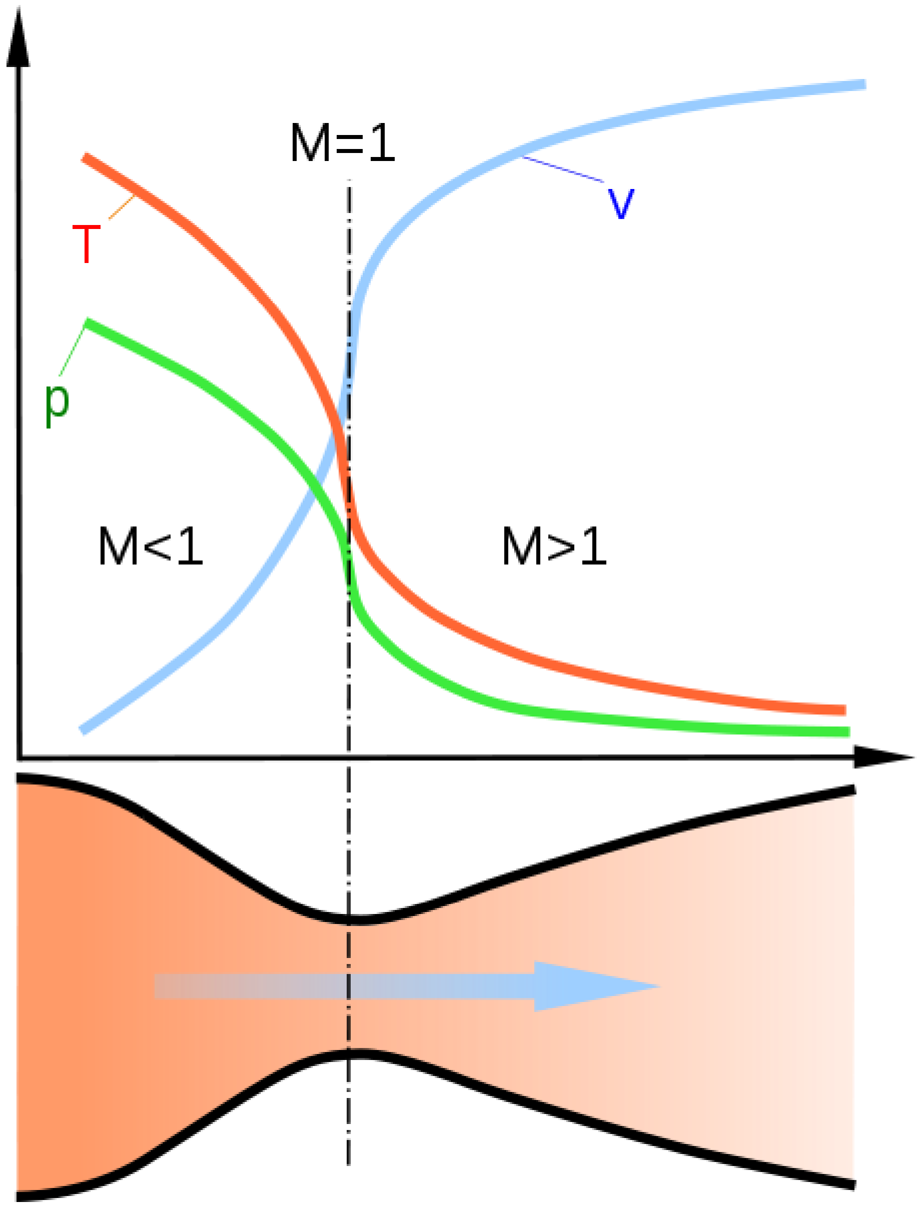
where:

- 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



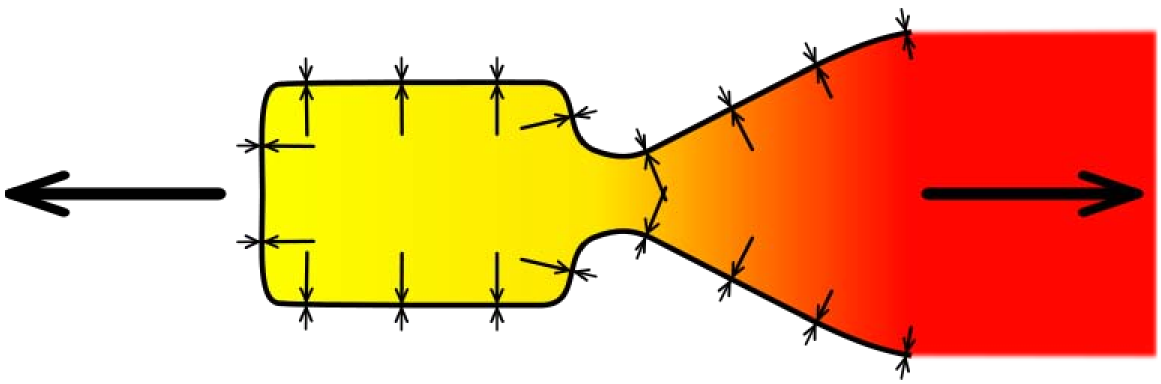
Typical temperatures (T) and pressures (p) and speeds (v) in a De Laval Nozzle

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:

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

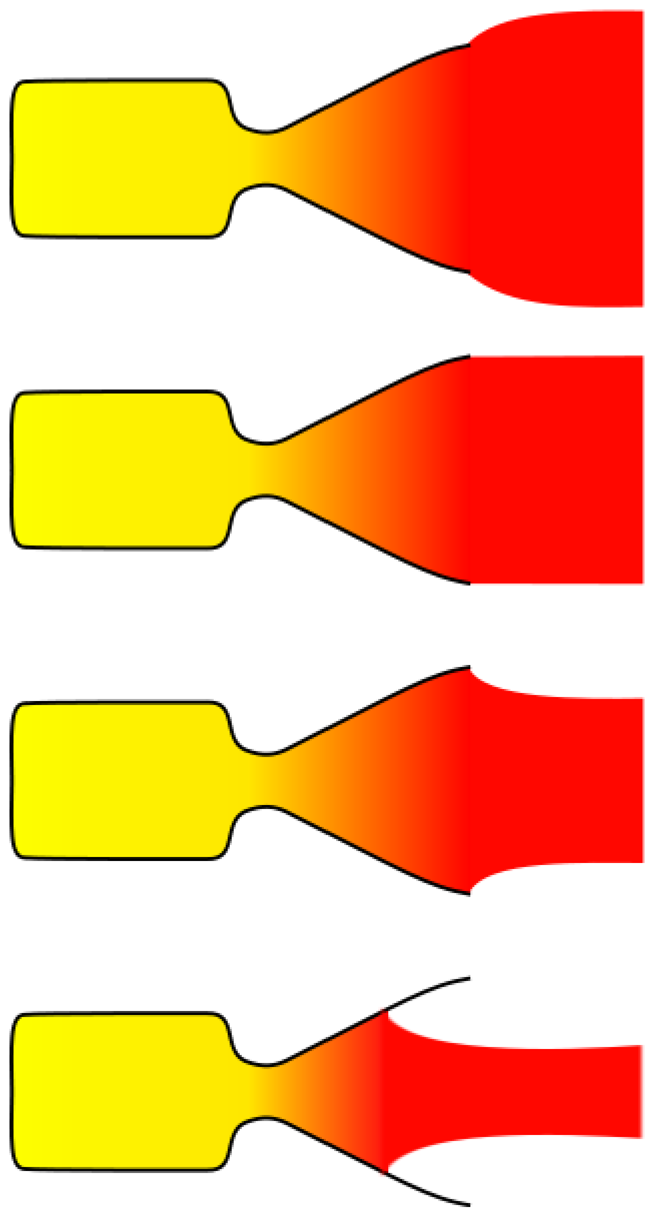
where:

$\dot{m}$ =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:

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

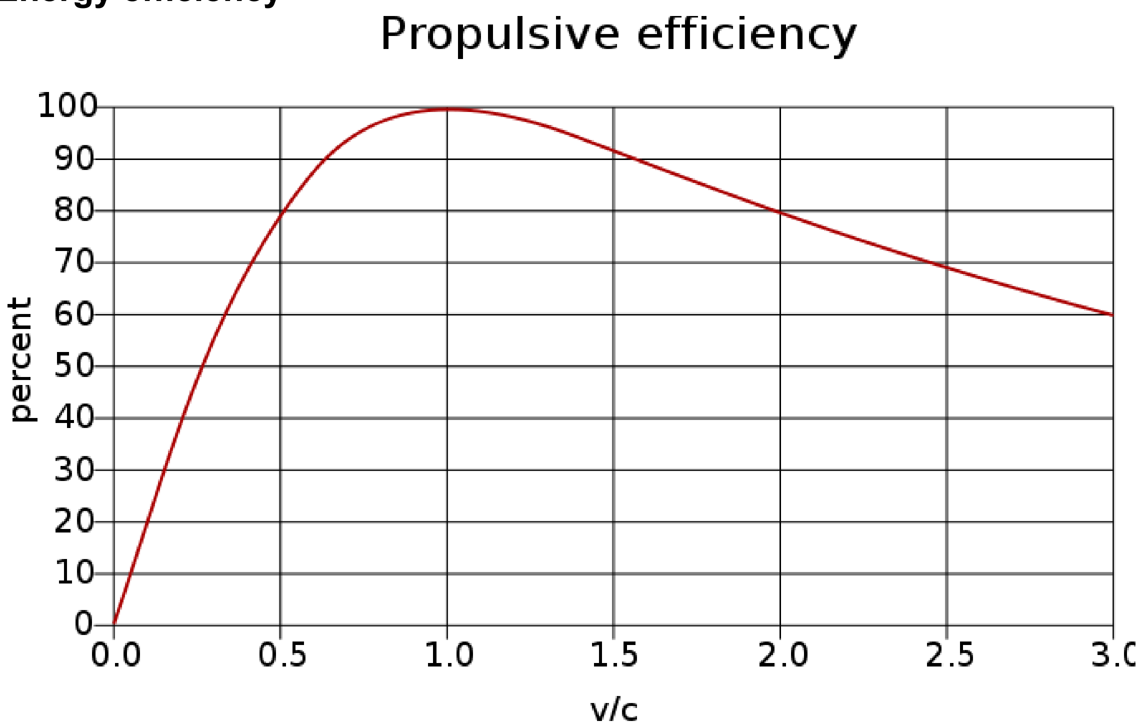
Throttling

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	Some oxidisers are monopropellants, can explode in own right; mechanical failure of solid propellant can block nozzle (very rare with rubberised

		off.	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	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

	propellant mixture ratios and throttled or shut down		
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 which allows higher temperatures	1600 seconds I_{sp}	Very low thrust and high power, performance is similar to Ion drive.
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	Simple design. Using hydrogen propellant, 900 seconds of I_{sp} is	Only useful once in space, as thrust is fairly low, but hydrogen has not been

collector		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.	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 (annihilation energy)	Antimatter annihilation heats propellant	Extremely energetic, very high theoretical exhaust velocity	Problems with antimatter 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 9

Model Rocket

A typical model rocket during launch

A **model rocket** is a small rocket that is commonly advertised as being able to be launched by anybody, to generally low altitudes (usually to around 100–500 m (300–1500 ft) for a 30 g (1 oz.) model) and recovered by a variety of means.

According to the United States National Association of Rocketry (NAR) Safety Code, model rockets are constructed of paper, wood, plastic and other lightweight materials. The code also provides guidelines for motor use, launch site selection, launch methods, launcher placement, recovery system design and deployment and more. Since the early 1960s, a copy of the Model Rocket Safety Code has been provided with most model rocket kits and motors. Despite its inherent association with extremely flammable substances and objects with a pointed tip traveling at high speeds, model rocketry historically has proven to be a very safe hobby and has been credited as the most significant source of inspiration for children who eventually become scientists and engineers.

The launch of a scale model of Saturn V

History of model rocketry

While there were many small rockets produced years of for research and experimentation, the first modern model rocket, and more importantly, the model rocket motor, was designed in 1954 by Orville Carlisle, a licensed pyrotechnics expert, and his brother Robert, a model airplane enthusiast. They originally designed the motor and rocket for Robert to use in lectures on the principles of rocket powered flight. But then Orville read articles written in Popular Mechanics by G. Harry Stine about the safety problems associated with young people trying to make their own rocket engines. With the launch of Sputnik, many young people were trying to build their own rocket motors, often with tragic results. Some of these attempts were dramatized in the fact-based movie

October Sky. The Carlisles realized their motor design could be marketed and provide a safe outlet for a new hobby. They sent samples to Mr. Stine in January 1957. Stine, a range safety officer at White Sands Missile Range, built and flew the models, and then devised a safety handbook for the activity based on his experience at the range.

Companies

The first American model rocket company was Model Missiles Incorporated (MMI), in Denver, Colorado, opened by Stine and others. Stine had model rocket engines made by a local fireworks company recommended by Carlisle, but reliability and delivery problems forced Stine to approach others. Eventually Stine approached Vernon Estes, the son of a local fireworks maker, sadly model missiles closed due to unwise business discussions . Estes founded Estes Industries in 1958 in Denver, Colorado, and developed a high speed automated machine for manufacturing solid model rocket motors for MMI. The machine, nicknamed "Mabel", made low cost motors with great reliability, and did so in quantities much greater than Stine needed. Stine's business faltered and this enabled Estes to market the motors separately. Subsequently, he began marketing model rocket kits in 1960, and eventually, Estes dominated the market. Estes moved his company to Penrose, Colorado in 1961. Estes Industries was acquired by Damon Industries in 1970. It continues to operate in Penrose today.

Competitors like Centuri and Cox came and went in America during the 1960s, 70s and 80s, but Estes continued to control the American market, offering discounts to schools and clubs like Boy Scouts of America to help grow the hobby. In recent years, companies like Quest Aerospace have taken a small portion of the market, but Estes continues to be the main source of rockets, motors, and launch equipment for the low-medium powered rocketry hobby today.

Since the advent of High Power Rocketry, which began in the mid-80s with the availability of G through J class motors (each letter designation has twice the energy of the one before), a number of companies have shared the market for larger and more powerful rockets. By the early 1990s, Aerotech Consumer Aerospace, LOC/Precision, and Public Missiles Limited (PML) had taken up leadership positions, while a host of engine manufacturers provided ever larger motors, and at much higher costs. Companies like Aerotech, Vulcan, and Kosdon were widely popular at launches during this time as high powered rockets routinely broke Mach 1 and reached heights over 3,000 m (10,000 ft). In a span of about five years, the largest regularly made production motors available reached N, which had the equivalent power of over 1,000 D engines combined, and could lift rockets weighing 100 kg (200 lb.) with ease. Custom motor builders continue to operate on the periphery of the market today, often creating propellants which produce colored flame (red, blue, and green being common), black smoke and sparking combinations, as well as occasionally building enormous motors of P, Q, and even R class for special projects such as extreme altitude attempts over 17,000 m (50,000 ft).

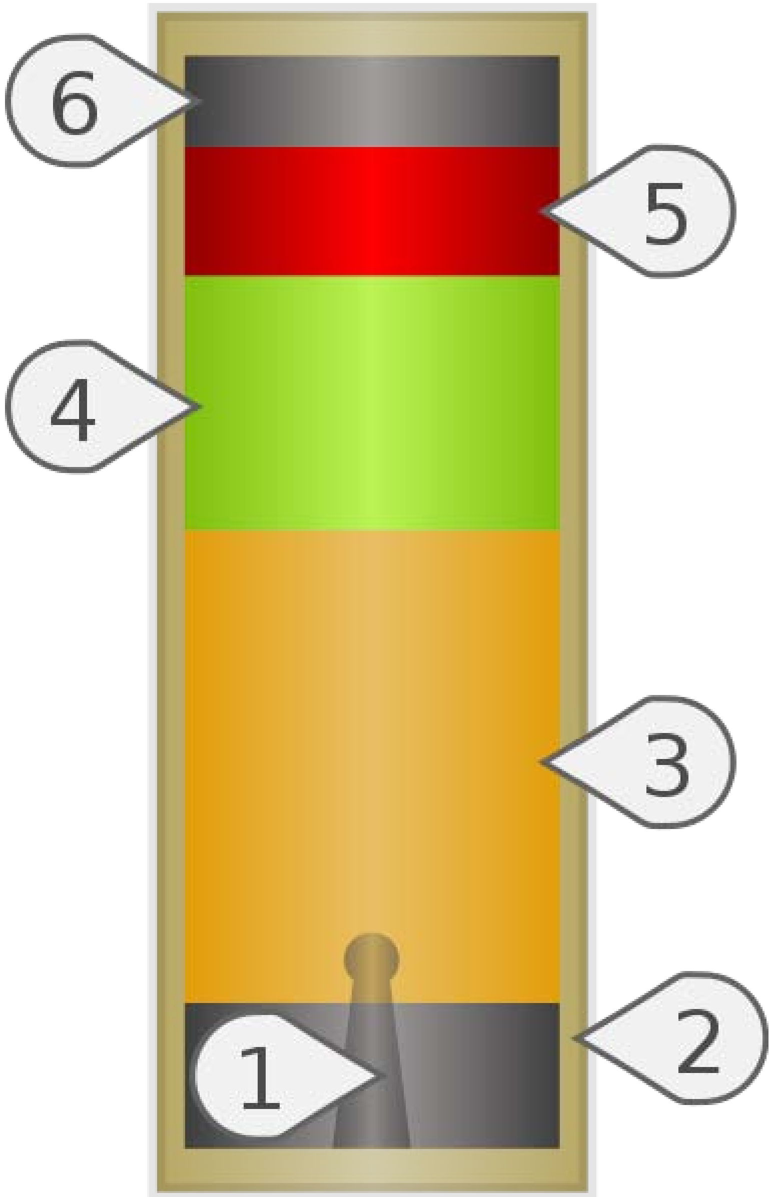
High power engine reliability was a significant issue in the late 80s and early 90s, with catastrophic engine failures occurring relatively frequently (est. 1 in 20) in motors of L class or higher. At costs exceeding $300 per motor, the need to find a cheaper and more reliable alternative was apparent. Reloadable motor designs (metal sleeves with screwed-on end caps and filled with cast propellant slugs) were introduced by Aerotech and became very popular over the span of a few years. These metal containers needed only to be cleaned and refilled with propellant and a few throw-away components after each launch. The cost of a "reload" was typically half of a comparable single use motor. While catastrophes at take-off (CATOs) still occur occasionally with reloadable motors (mostly due to poor assembly techniques by the user), the reliability of launches has risen significantly. In addition, it is possible to change the thrust profile of reloadable motors by selecting different propellant designs. Since thrust is proportional to burning surface area, propellant slugs can be shaped to produce very high thrust for a second or two, or to have a lower thrust which continues for an extended time. Depending on the weight of the rocket and the maximum speed threshold of the airframe and fins, appropriate motor choices can be used to maximize performance and the chance of successful recovery. Aerotech, Pro-38, Rouse-Tech, Loki and others have standardized around a set of common reload sizes such that customers have great flexibility in their hardware and reload selections, while there continues to be an avid group of custom engine builders who create unique designs and occasionally offer them for sale.

Precaution

Model rocketry is a safe and widespread hobby. Individuals such as G. Harry Stine and Vernon Estes helped ensure this by developing and publishing the NAR Model Rocket Safety Codes and by commercially producing safe, professionally-designed and manufactured model rocket motors.

One of the main motivations for the development of the hobby in the 1950s and 1960s was to provide young people the opportunity to construct flying rocket models without having to engage in dangerous construction of motor units and direct handling of explosive propellants.

Model rocket motors

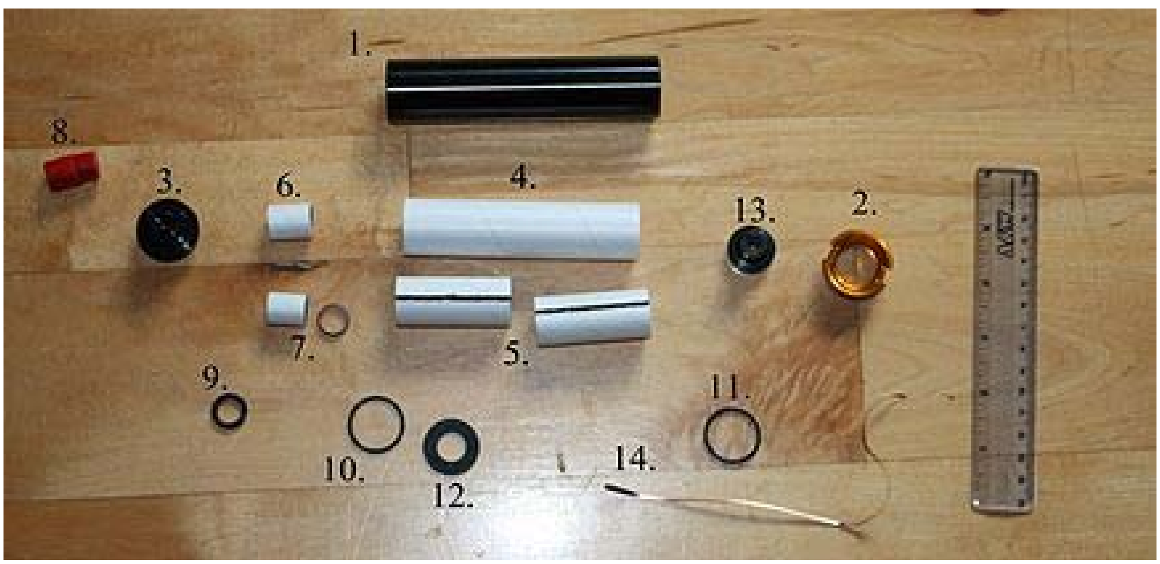


Anatomy of a basic model rocket engine. A typical engine is about 8cm long. 1. Nozzle; 2. Case; 3. Propellant; 4. Delay charge; 5. Ejection charge; 6. End cap

Most small model rocket motors are single-use engines, with cardboard bodies and lightweight molded ceramic nozzles, ranging in impulse class from fractional A to G. Model rockets generally use commercially-manufactured black powder motors. These motors are tested and certified by the National Association of Rocketry, the Tripoli Rocketry Association(TRA) or the Canadian Association of Rocketry(CAR). Blackpowder motors come in impulse ranges from 1/8A to E, although a few F blackpowders motors have been made.

The physically largest blackpowder model rocket motors are typically E-class, for black powder is very brittle. If a large black powder motor is dropped, or is exposed to many heating/cooling cycles (for example, in a closed vehicle exposed to high heat), the propellant charge may develop hairline fractures. These fractures increase the surface area of the propellant, so that when the motor is ignited, the propellant burns much more quickly than it should, producing greater than normal internal chamber pressure inside the engine. This pressure may exceed the strength of the paper case, causing the motor to burst. A bursting motor can cause damage to the model rocket ranging from a simple ruptured motor tube or body tube to the violent ejection (and occasionally ignition) of the recovery system.

Rocket motors with power ratings higher than D to E, therefore, customarily use composite propellants made of ammonium perchlorate, potassium nitrate, aluminium powder, and a rubbery binder substance contained in a hard plastic case. This type of propellant is similar to that used in the solid rocket boosters of the space shuttle and is not as fragile as black powder, increasing motor reliability and resistance to fractures in the propellant. These motors range in impulse from size D to O. Composite motors produce more impulse per unit weight (specific impulse) than do black powder motors.



The components of a commercially produced G64-10W motor made by Aerotech Consumer Aerospace for a 29/40-120 casing. 1. Motor Casing 2. Aft Closure 3. Forward Closure 4. Propellant Liner 5. Propellant Grains (C-Slot Geometry) 6. Delay Insulator 7. Delay Grain and Delay Spacer 8. Black Powder Ejection Charge 9. Delay O-Ring 10 & 11. Forward and Aft O-Rings 12. Forward Insulator 13. Nozzle 14. Electric Igniter

Reloadable composite-propellant motors are also available. These are commercially-produced motors requiring the user to assemble propellant grains, o-rings and washers (to contain the expanding gases), delay grains and ejection charges into special non-shattering aluminum motor casings with screw-on or snap-in ends (closures). The advantage of a reloadable motor is the cost: firstly, because the main casing is reusable, reloads cost significantly less than single-use motors of the same impulse. Secondly, assembly of larger composite engines is labor-intensive and difficult to automate; off-loading this task on the consumer results in a cost savings. Reloadable motors are available from D through O class.

Motors are electrically ignited with an electric match consisting of a short length of pyrogen-coated nichrome, copper, or aluminum bridgewire pushed into the nozzle and held in place with flameproof wadding, a rubber band, a plastic plug or masking tape. On top of the propellant is a tracking delay charge which produces smoke but essentially no thrust as the rocket slows down and arcs over. When the delay charge has burned through, it ignites an ejection charge, which is used to deploy the recovery system.

Motor nomenclature

Rocket motors. From left, 13mm A10-0T, 18mm C6-7, 24mm D12-5, 24mm E9-4, 29mm G40-10.

Model rocket motors produced by companies like Estes Industries and Quest Aerospace are stamped with a code (such as A10-3T or B6-4) that indicates several things about the motor.

The Quest Micro Maxx engines are the smallest at a diameter of 6mm. The company Apogee Components made 10.5mm micro motors, but those were discontinued in 2001. Estes manufactures size "T" (Tiny) motors that are 13 mm in diameter by 45 mm long, while standard A, B and C motors are 18 mm in diameter by 70 mm long. Larger C, D, and E class black powder motors are also available; they are 24 mm in diameter and either 70 (C and D motors) or 95 mm long (E motors). Some motors, such as F and G single-use motors, are 29mm in diameter. High-power motors (usually reloadable) are available in 38mm, 54mm, 75mm, and 98mm diameters.

First letter

The letter at the beginning of the code indicates the motor's total impulse range (commonly measured in newton-seconds). Each letter in successive alphabetical order has up to twice the impulse of the letter preceding it. This does not mean that a given "C" motor has twice the total impulse of a given "B" motor, only that C motors are in the 5.01-10.0 N-s range while "B" motors are in the 2.51-5.0 N-S range. The designations "¼A" and "½A" are also used.

For instance, a B6-4 motor from Estes-Cox Corporation has a total impulse rating of 5.0 N-s. A C6-3 motor from Quest Aerospace has a total impulse of 8.5 N-s.

First number

The number that comes after the letter indicates the motor's average thrust, measured in newtons. A higher thrust will result in higher liftoff acceleration, and can be used to launch a heavier model. Within the same letter class, a higher average thrust also implies a shorter burn time (e.g., a B6 motor will not burn as long as but have more initial thrust than a B4).Motors within the same letter class that have different first numbers are usually for rockets with different weights.For example a heavier rocket would require a engine with more initial thrust to get it off of the launch pad, whereas a lighter rocket would need less initial thrust and would sustain a longer burn reaching higher altitudes.

Last number

The last number is the delay in seconds between the end of the thrust phase and ignition of the ejection charge. Black Powder Motors that end in a zero have no delay or ejection charge. Such motors are typically used as first-stage motors in multistage rockets as the lack of delay element and cap permit burning material to burst forward and ignite an upper-stage motor.

A "P" indicates that the motor is "plugged". In this case, there is no ejection charge, but a cap is in place. A plugged motor is used in rockets which do not need to deploy a standard recovery system such as small rockets which tumble or R/C glider rockets. Plugged motors are also used in larger rockets, where electronic altimeters or timers are used to trigger the deployment of the recovery system.

Composite motors usually have a letter or combination of letters after the delay length indicating which of the manufacturer's different propellant formulations is used in that particular motor.

Reloadable motors

Reloadable motor cases. From left: 24/40, 29/40-120, 29/60, 29/100, 29/180, 29/240

Reloadable motors are specified in the same manner as model rocket single-use motors as described above. However, they have an additional designation which specifies both the diameter and maximum total impulse of the motor casing in the form of diameter/impulse. A reload designed for a 29mm diameter case with a maximum total impulse of 60 newton-seconds carries the designation 29/60 in addition to its impulse specification.

Model rocket recovery methods

Model and high-power rockets are designed to be safely recovered and flown repeatedly. The most common recovery methods are parachute and streamer. The parachute is usually blown out when the engine's recoil creates pressure and pops off the nose cone. The parachute is attached to the nose cone, making it pull the parachute out and make a soft landing.

Featherweight recovery

The simplest approach, which is only appropriate for the tiniest of rockets, is to let the rocket flutter back to earth after ejecting the motor. This is slightly different from tumble recovery, which relies on some system to destabilize the rocket to prevent it from entering a ballistic trajectory on its way back to earth.

Tumble recovery

Another simple approach appropriate for small rockets—or rockets with a large cross-sectional area—is to have the rocket tumble back to earth. Any rocket which will enter a stable, ballistic trajectory as it falls is **not** safe to use with tumble recovery. To prevent this, some such rockets use the ejection charge to slide the engine to the rear of the rocket, moving the center of mass behind the center of pressure and thus making the rocket unstable.

Nose-blow recovery

Another very simple recovery technique, used in very early models in the 1950s and occasionally in modern examples, is nose-blow recovery. This is where the ejection charge of the motor ejects the nose cone of the rocket (usually attached by a shock cord made of rubber, Kevlar string or another type of cord) from the body tube, destroying the rocket's aerodynamic profile, causing highly-increased drag, and reducing the rocket's airspeed to a safe rate for landing. Nose-blow recovery is generally only suitable for very light rockets.

Parachute/Streamer

A typical problem with parachute recovery.

The approach used most often in small model rockets, but can be used with larger rocket models given the size of the parachute greatly increases with the size of the rocket. It uses the ejection charge of the motor to deploy, or push out, the parachute or streamer. Typically, a ball or mass of fireproof paper or material is inserted into the body before the parachute or streamer. This allows the ejection charge to propel the fire-proof material, parachute, and nose cone without damaging the recovery equipment. Air resistance slows the rocket's fall, ending in a smooth, controlled and gentle landing.

Glide recovery

In glide recovery, the ejection charge either deploys an airfoil (wing) or separates a glider from the motor. If properly trimmed, the rocket/glider will enter a spiral glide and return safely. In some cases, radio-controlled rocket gliders are flown back to the earth by a pilot in much the way as R/C model airplanes are flown.

Some rockets (typically long thin rockets) are the proper proportions to safely glide to Earth tail-first. These are termed 'backsliders'.

Helicopter recovery

The ejection charge, through one of several methods, deploys helicopter-style blades and the rocket autorotates back to earth. The helicopter recovery usually happens when the engine's recoil creates pressure, making the nose cone pop out. There are rubber bands connected to the nosecone and three or more blades. The rubber bands pull the blades out and they provide enough drag to soften the landing. Very often the blades will break.

Other model rocketry

Aerial photography

Cameras and video cameras can be launched on model rockets to take photographs in-flight. Model rockets equipped with the Astrocam, Snapshot film camera or the Oracle or newer Astrovision digital cameras (all produced by Estes), or with homebuilt equivalents, can be used to take aerial photographs.

These aerial photographs can be taken in many ways. Mechanized timers can be used or passive methods may be employed, such as strings that are pulled by flaps that respond to wind resistance. Microprocessor controllers can also be used. However, the rocket's speed and motion can lead to blurry photographs, and quickly changing lighting conditions as the rocket points from ground to sky can have an impact on video quality. Video frames can also be stitched together to create panoramas. As parachute systems can be prone to failure or malfunction, model rocket cameras need to be protected from impact with the ground.

There are also rockets that shoot short digital videos. There are two widely used ones used on the market, both produced by Estes: the Astrovision and the Oracle. The Astrocam shoots 4 (advertised as 16, and shown when playing the video, but in real life 4)seconds of video, and can also take three consecutive digital still images in flight, with a higher resolution than the video. It takes from size B-6-3 to C-6-3 Engines. The Oracle is a more costly alternative, but is able to capture all or most of its flight and recovery. It is generally used with "D" motors. The Oracle has been on the market longer than the Astrovision, and has a better general reputation.

There are also experimental homemade rockets that include onboard videocameras, with two methods for shooting the video. One is to radio the signal down to earth, like in the BoosterVision series of cameras. The second method for this is to record it on board and be downloaded after recovery, the method employed by the cameras above (some experimenters use the Aiptek PenCam Mega for this, the lowest power usable with this method is a C or D Motor).

Instrumentation and experimentation

Model rockets with electronic altimeters can report and or record electronic data such as maximum speed, acceleration, and altitude. Two methods of determining these things are to a) have an accelerometer and a timer and work backwards from the acceleration to the speed and then to the height and b) to have a barometer onboard with a timer and to get the height (from the difference of the pressure on the ground to the pressure in the air) and to work forwards with the time of the measurements to the speed and acceleration.

Rocket modelers often experiment with rocket sizes, shapes, payloads, multistage rockets, and recovery methods. Some rocketeers build scale models of larger rockets, space launchers, or missiles.

Some high altitude rockets deploy a smaller 'second stage rocket' during flight. Once the main rocket engine begins to die out, a second stage is fired from the main. This greatly increases altitude as the speed of the second rocket adds to the speed of the first rocket. For example if a rocket is traveling at 250 km/h (150 mph) then the second stage deploys at an additional 100 km/h (60 mph) from the main, the second stage is now at 350 km/h (210 mph). However, this is not perfect as other variables such as weather may influence the flight.

High power rocketry

As with low power model rockets, high power rockets are also constructed from lightweight materials. Unlike model rockets, high power rockets often require stronger materials such as fiberglass, composite materials, and aluminum to withstand the higher stresses during flights which often exceed Mach 1 (340 m/s) and over 3,000 m (10,000 ft.) altitude.

High power rockets are propelled by larger motors ranging from class H to class O. Their motors are almost always reloadable rather than single-use in order to reduce cost. Recovery and/or multi-stage ignition may be initiated by small on-board computers, which use an altimeter or accelerometer for detecting when to ignite engines or deploy parachutes.

High powered model rockets can carry large payloads, including cameras and instrumentation such as GPS units.

The NAR and the TRA successfully sued the US Bureau of Alcohol, Tobacco, Firearms and Explosives(BATFE) over the classification of Ammonium Perchlorate Composite Propellant(APCP), the most commonly used propellant in high power rocket motors, as an explosive. The March 13, 2009 decision by DC District court judge Reggie Walton removed APCP from the list of regulated explosives, essentially eliminating BATFE regulation of hobby rocketry.

Chpater 10

Ammonium Perchlorate Composite Propellant

Ammonium perchlorate composite propellant (APCP) is a modern solid rocket propellant used in both manned and unmanned rocket vehicles. It differs from many traditional solid rocket propellants such as black powder or Zinc-Sulfur, not only in chemical composition and overall performance, but also by the nature of how it is processed. APCP is cast into shape, as opposed to powder pressing as with black-powder. This allows for manufacturing regularity and repeatability which are necessary requirements for use in the aerospace industry.

Uses

Ammonium perchlorate composite propellant is typically used in aerospace propulsion applications, where simplicity and reliability are desired and specific impulses (depending on the composition and operating pressure) of 180-260 seconds are adequate. Because of these performance attributes, APCP is regularly implemented in booster applications such as in the Space Shuttle Solid Rocket Boosters, aircraft ejection seats, and specialty space exploration applications such as NASA's Mars Exploration Rover descent stage retrorockets. In addition, the high power rocketry community regularly uses APCP in the form of commercially available propellant "reloads", as well as single-use motors. Experienced experimental and amateur rocketeers also often work with APCP, processing the APCP themselves.

Composition

Overview

Ammonium perchlorate composite propellant is a composite propellant, meaning that it has both fuel and oxidizer mixed with a rubbery binder, all combined into a homogeneous mixture. The propellant is most often composed of ammonium perchlorate (AP), an elastomer binder such as hydroxyl-terminated polybutadiene (HTPB) or polybutadiene acrylic acid acrylonitrile prepolymer (PBAN), small amounts of powdered metal, typically aluminum (Al), and various burn rate catalysts. In addition, curing additives induce elastomer binder cross-linking to solidify the propellant before use. The AP serves as the oxidizer, while the binder and aluminum serve as the fuel. Burn rate catalysts determine how quickly the mixture burns. The resulting cured propellant is slightly viscoelastic (rubbery) which also helps limit fracturing during accumulated damage (such as shipping, installing, cutting) and high acceleration applications such as hobby or military rocketry.

The composition of APCP can vary significantly depending on the application, intended burn characteristics, and constraints such as nozzle thermal limitations or specific impulse (Isp). Rough mass proportions (in high performance configurations) tend to be about 70/15/15 AP/HTPB/Al, though fairly high performance "low-smoke" can have compositions of roughly 80/18/2 AP/HTPB/Al. While metal fuel is not required in APCP, most formulations include at least a few percent as a combustion stabilizer, propellant opacifier (to limit excessive infrared propellant preheating), and increase the temperature of the combustion gases (increasing Isp).

Common species

Oxidizers:

- Ammonium perchlorate
- Strontium nitrate (for red flame, with or without addition of AP)

High energy fuels:

- Aluminium (high performance)
- Magnesium (medium performance)
- Zinc (low performance)

Low energy fuels:

- HTPB
- PBAN

Special considerations

Though increasing the ratio of metal fuel to oxidizer up to the stoichiometric point increases the combustion temperature, the presence of an increasing molar fraction of metal oxides (particularly Al_2O_3) precipitating from the gaseous solution creates globules of solids or liquids that slow down the flow velocity as the mean molecular mass of the flow increases. In addition, the chemical composition of the gases change, varying the effective heat capacity of the gas. Because of these phenomena, there exists an optimal non-stoichiometric composition for maximizing Isp of roughly 16% by mass, assuming the combustion reaction goes to completion inside the combustion chamber.

The combustion time of the aluminum particles in the hot combustion gas varies depending on aluminum particle size and shape. In small APCP motors with high aluminum content, the residence time of the combustion gases does not allow for full combustion of the aluminum and thus a substantial fraction of the aluminum is burned outside the combustion chamber, leading to decreased performance. This effect is often mitigated by reducing aluminum particle size, inducing turbulence (and therefore a long characteristic path length and residence time), and/or by reducing the aluminum content to ensure a combustion environment with a higher net oxidizing potential, ensuring more complete aluminum combustion.

Particle size

The propellant particle size distribution has a profound impact on APCP rocket motor performance. Smaller AP and Al particles lead to higher combustion efficiency but also lead to increased linear burn rate. The burn rate is heavily dependent on mean AP particle size as the AP absorbs heat to decompose into a gas before it can oxidize the fuel components. This process may be a rate-limiting step in the overall combustion rate of APCP. The phenomenon can be explained by considering the heat flux to mass ratio: As the particle radius increases the volume (and therefore mass and heat capacity) increase as the cube of the radius. However, the surface area increases as the square of the radius, which is roughly proportional to the heat flux into the particle. Therefore, a particle's rate of temperature rise is maximized when the particle size is minimized.

Common APCP formulations call for 30-400 µm AP particles (often spherical), as well as 2-50 µm Al particles (often spherical). Because of the size discrepancy between the AP and Al, Al will often take an interstitial position in a pseudo-lattice of AP particles.

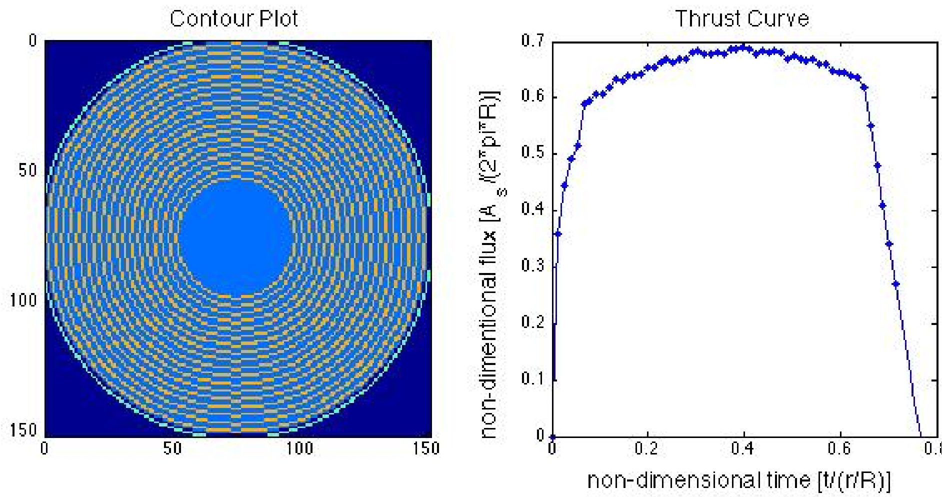
Characteristics

Geometric

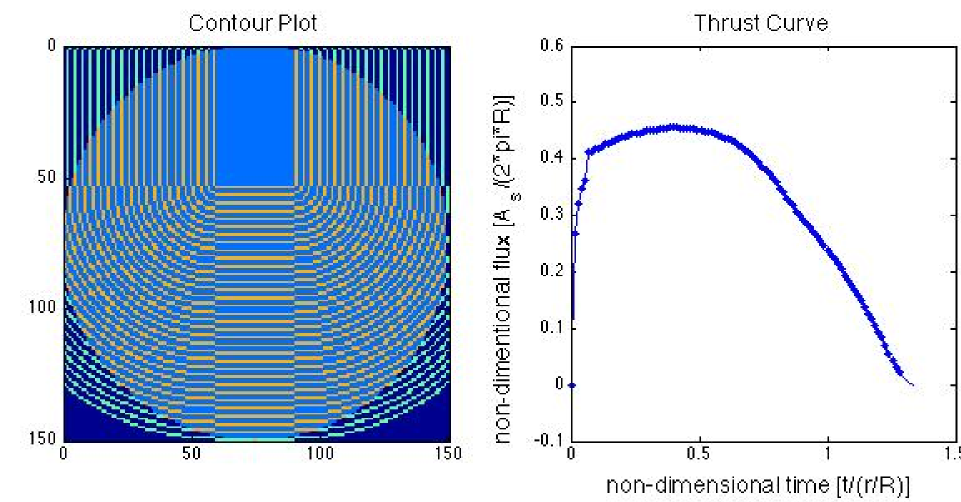
APCP deflagrates from the surface of exposed propellant in the combustion chamber. In this fashion, the geometry of the propellant inside the rocket motor plays an important role in the overall motor performance. As the surface of the propellant burns the shape evolves (a subject of study in internal 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/2):

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

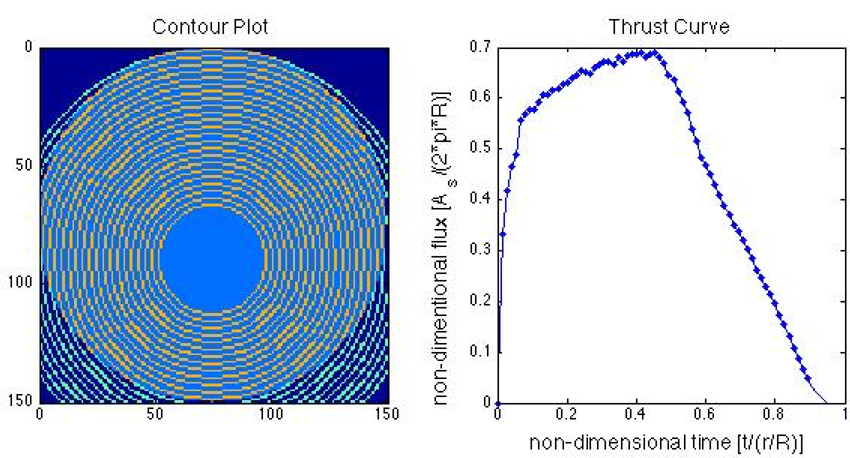
Several geometric configurations are often used depending on the application and desired thrust curve:



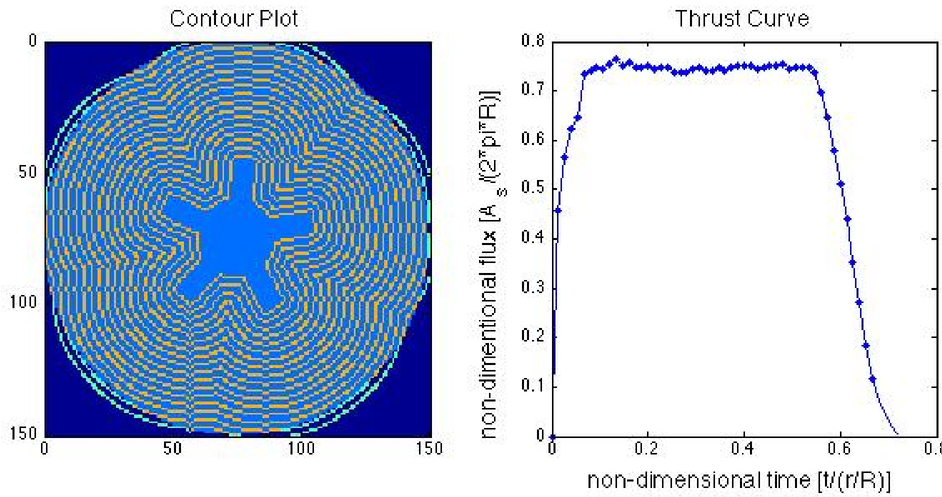
Circular bore simulation



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.

Burn rate

While the surface area can be easily tailored by careful geometric design of the propellant, the burn rate is dependent on several subtle factors:

- Propellant chemical composition.
- AP, Al, additive particle sizes.
- Combustion pressure.
- Heat transfer characteristics.
- Erosive burning (high velocity flow moving past the propellant).
- Initial temperature of propellant.

In summary, however, most formulations have a burn rate between 1–3 mm/sec at STP and 6–12 mm/sec at 68 atm. The burn characteristics (such as linear burn rate) are often determined prior to rocket motor firing using a strand burner test. This test allows the APCP manufacturer to characterize the burn rate as a function of pressure. Empirically, APCP adheres fairly well to the following power-function model:

$$b_r = a \cdot p^n$$

It is worth noting that typically for APCP, 0.3<n<0.5 indicating that APCP is sub-critically pressure sensitive. That is, if surface area were maintained constant during a burn the combustion reaction would not runaway to (theoretically) infinite as the pressure would reach an internal equilibrium. This isn't to say that APCP cannot cause an explosion, but rather that the explosion would be caused by the pressure surpassing the burst pressure of the container (rocket motor).

Model applications

A high-power rocket launch using an APCP motor.

Commercial APCP rocket engines usually come in the form of re-loadable motor systems (RMS) and fully-assembled single use rocket motors. For RMS, the APCP "grains" (cylinders of propellant) are loaded into the reusable motor casing along with a sequence of insulator disks and o-rings and a (graphite or glass-filled phenolic resin) nozzle. The motor casing and closures are typically bought separately from the motor manufacturer and are often precision machined aluminum. The assembled RMS contains both reusable (typically metal) and disposable components.

The major APCP suppliers for hobby use are:

- Aerotech Consumer Aerospace
- Animal Motor Works
- Cesaroni Technology

- Kosdon (by Aerotech)
- Loki Research

To achieve different visual effects and flight characteristics, hobby APCP suppliers offer a variety of different characteristic propellant types. These can range in from fast burning with little smoke and blue flame to classic white smoke and white flame. In addition, colored formulations are available to display reds, greens, blues, and even flameless black smoke.

In medium and high power rocket applications, APCP has largely replaced black powder as a rocket propellant. Compacted black powder slugs become prone to fracture in larger applications, which can result in catastrophic failure in rocket vehicles. APCP's elastic material properties makes it less vulnerable to fracture from accidental shock or high acceleration flights. Due to these attributes, widespread adoption of APCP and related propellant types in the hobby has significantly enhanced the safety of rocketry.

Environmental and other concerns

The exhaust from APCP solid rocket motors contain mostly water, carbon dioxide, hydrogen chloride, and a metal oxide (typically aluminium oxide). The hydrogen chloride can easily dissociate into water and create corrosive hydrochloric acid, damaging launch equipment and biasing the pH of local water and rainfall. Furthermore, for military use, the smoke trail and the infrared radiation from the hot particles make it possible to detect the launch from space. These problems led to the research in smokeless grain which contains nitrogen-containing organic molecules (e.g. ammonium dinitramide).

Regulation and legality

In the United States, APCP for hobby use is regulated indirectly by two non-government agencies: the National Association of Rocketry (NAR), and the Tripoli Rocketry Association (TRA). Both agencies set forth rules regarding the impulse classification of rocket motors and the level of certification required by rocketeers in order to purchase certain impulse (size) motors. The NAR and TRA require motor manufactures to certify their motors for distribution to vendors and ultimately hobbyists. The vendor is charged with the responsibility (by the NAR and TRA) to check hobbyists for high power rocket certification before a sale can be made. The amount of APCP that can be purchased (in the form of a rocket motor reload) correlates to the impulse classification, and therefore the quantity of APCP purchasable by hobbyist is regulated by the NAR and TRA.

The overarching legality concerning the implementation of APCP in rocket motors is outlined in NFPA 1125. Use of APCP outside hobby use is regulated by state and municipal fire codes. On March 16, 2009, it was ruled that APCP is not an explosive and that manufacture and use of APCP no longer requires a license or permit from the ATF.