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Department of Aeronautical Engineering

(R18)

AIR BREATHING PROPULSION

Lecture Notes

B. Tech III YEAR – II SEM

Prepared by

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Dept.Aero

SPACE PROPULSION

B.Tech. III Year AE II Sem.LT/P/DC

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Pre-Requisites: Nil

Course Objectives:

- To know about the propulsion system used in rockets and missiles
- Discuss the working principle of solid and liquid propellant rockets and gain basic knowledgeof hybrid rocket propulsion.

Course Outcomes: To be able to

- Understand about trajectory and orbits
- Illustrate electric propulsion techniques, ion and nuclear rocket and the performances ofdifferent advanced propulsion systems.
- Evaluate various space missions, parameters to be considered for designing trajectories androcket mission profiles
- Understand the fundamentals of chemical rocket propulsion, types of igniters andperformance considerations of rockets.

UNIT - I

Principles of Rocket Propulsion: History of rockets, Newtons third law, orbits and space flight, types of orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO,launch assists.

UNIT - II

Fundamentals of Rocket Propulsion: Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Rocket performance considerations of rockets, types of igniters, preliminary concepts in nozzle less propulsion, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.

UNIT - III

Solid Rocket Propulsion: Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Erosive burning in solid propellant rockets, combustion instability, strand burner and T-burner, applications and advantages of solid propellant rockets.

UNIT - IV

Liquid and Hybrid Rocket Propulsion: Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.

UNIT-V

Advanced Propulsion Techniques: Electric rocket propulsion, types of electric propulsion techniques, Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemicalrocket propulsion systems, future applications of electric propulsion systems, Solar sail.

TEXT BOOKS:

1. Hill, P.G. and Peterson, C.R., —Mechanics and Thermodynamics of Propulsion‖, 2nd Edition, Addison Wesley, 1992.
2. Turner, M. J. L., —Rocket and Spacecraft Propulsion‖, 2nd Edition, MIT Press, 1922.
3. Hieter and Pratt, —Hypersonic Air breathing propulsion‖ 5th Edition, 1993.

REFERENCE BOOKS:

1. Sutton, G.P., —Rocket Propulsion Elements‖ John Wiley & Sons Inc., New York, 5th Edition, 1993.
2. Mathur, M. L., and Sharma, R.P., —Gas Turbine, Jet and Rocket Propulsion‖, Standard Publishers and Distributors, Delhi, 1988, Tajmar, M., Advanced Space Propulsion Systems, Springer 2003

UNIT-1

History and principles of rocket propulsion

Human development has always been closely linked with transportation. The domestication of the horse and the invention of the wheel had a dramatic effect on early civilisation—not always beneficial. Most of the past millennium has been strongly influenced by sailing ship technology, both for war and commerce; in the twentieth century, motor vehicles and aircraft have revolutionised transport. At the beginning of the twenty-first century the rocket may be seen as the emerging revolution in transport. So far, only a few humans have actually travelled in rocket-propelled vehicles, but a surprising amount of commercial and domestic communication is now reliant on satellites. From telephone calls, through news images, to the Internet, most of our information travels from one part of the world to another, through space. The proposed return to the Moon, and new plans to send humans to Mars indicate a resurgence of interest in space exploration for the new millenium.

Rocket propulsion is the essential transportation technology for this rapid growth in human communication and exploration. From its beginnings in ancient China through its rapid development during the Cold War, rocket propulsion has become the essential technology of the late twentieth century. It influences the lives and work of a growing number of people, who may wish to understand at least the principles behind it and its technical limitations. In most cases, users of space transportation are separated from the rocket technology which enables it; this is partly because of the mass of engineering detail and calculation which is essential to make such a complex system work. In what follows, we shall attempt to present the basic principles, and describe some of the engineering detail, in a way that exposes the essential physics and the real limitations to the performance of rocket vehicles.

THE DEVELOPMENT OF THE ROCKET

Hero of Alexandria (*c*. 67 AD) is credited with inventing the rocket principle. He was a mathematician and inventor and devised many machines using water, air pressure, and steam, including a fire engine and a fountain. His *aeolipile* consisted of a metal boiler in which steam was produced, connected by a pipe through a rotating joint, to a pivoted jet system with two opposing jets. The steam issuing from the jets caused the system to rotate. It is not clear if Hero understood the cause of the rotation; but this was the earliest machine to use the reaction principle—the theoretical basis of the rocket. The real inventor of the rocket was certainly Chinese, and is sometimes said to be one Feng Jishen, who lived around 970 AD. Most dictionaries insist on an Italian derivation of the word meaning 'a small distaff '; and this may be correct, although a derivation from the Chinese, meaning 'Fire Arrow', is also plausible. The invention was the practical result of experiments with gunpowder and bamboo tubes; it became, as it still is, a source of beauty and excitement as a firework. There seem to have been two kinds. One was a bamboo tube filled with gunpowder with a small hole in one end; when the gunpowder was ignited, the tube ran along the ground in an erratic fashion, and made the girls scream. The second, like our modern fireworks, had a bamboo stick attached for stability, and took to the sky to make a beautiful display of light and colour.

The rocket was also used as a weapon of oriental war. Kublai Kahn used it during his invasion of Japan in 1275; by the 1300s rockets were used as bombardment weapons as far west as Spain, brought west by the Mongol hordes, and the Arabs. They were also used against the British army in India, by Tipoo Sultan, in the 1770s. Shortly afterwards, Sir William Congreve, an artillery oAcer, realised the rocket's potential, and developed a military rocket which was used into the twentieth century. Congreve rockets were used at sea during the Napoleonic Wars, with some success. They appeared famously at the siege of Fort McHenry in the American War of Independence, and feature in song as 'the rocket's red glare'. At about the same time, rockets came into standard use as signals, to carry lines from ship to ship, and in the rescue of shipwrecked mariners—a role in which they are still used.

The improvements in guns, which came about during the late

1800s, meant that the rocket, with its small payload, was no longer a significant weapon compared with large-calibre shells. The carnage of the First World War was almost exclusively due to high-explosive shells propelled by guns. Military interest in the rocket was limited through the inter-war years, and it was in this period that the amateur adopted a device which had hardly improved over six centuries, and created the modern rocket. The names of the pioneers are well known: Goddard, Oberth, von Braun, Tsiolkovsky, Korolev. Some were practical engineers, some were mathematicians and others were dreamers. There are two strands. The first is the imaginative concept of the rocket as a vehicle for gaining access to space, and the second is the practical development of the rocket. It is not clear to which strand the (possibly apocryphal) seventeenth century pioneer Wan Hu belongs. He is said to have attached 47 rockets to a bamboo chair, with the purpose of ascending into heaven.

Konstantin Tsiolkovsky (1857−1935) (Figure 1.1), a mathematics teacher, wrote about space travel, including weightlessness and escape velocity, in 1883, and he wrote about artificial satellites in 1895. In a paper published in 1903 he derived the rocket equation, and dealt in detail with the use of rocket propulsion for space travel; and in 1924 he described multi-stage rockets. His writings on space travel were

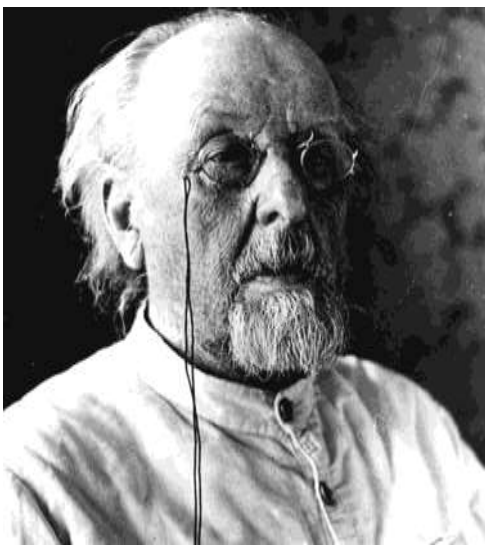


Figure 1.1. Konstantin Eduardovich Tsiolkovsky. Courtesy Kaluga State Museum.soundly based on mathematics—unlike, for example, those of Jules Verne—and he laid the mathematical foundations of spaceflight.

Tsiolkovsky never experimented with rockets; his work was almost purely theoretical. He identified exhaust velocity as the important performance parameter; he realised that the higher temperature and lower molecular weight produced by liquid fuels would be important for achieving high exhaust velocity; and he identified liquid oxygen and hydrogen as suitable propellants for space rockets. He also invented the multi-stage rocket.

Tsiolkovsky‘s counterpart in the German-speaking world was the Rumanian, Herman Oberth (1894−1992) (Figure 1.2). He published his (rejected) doctoral thesis in 1923, as a book in which he examined the use of rockets for space travel, including the design of liquid-fuelled engines using alcohol and liquid oxygen. His analysis was again mathematical, and he himself had not carried out any rocket experiments at the time. His book—which was a best-seller—was very important in that it generated huge amateur interest in rockets in Germany, and was instrumental in the foundation of many amateur rocket societies. The most important of these was the Verein fü r Raumschiffarht, to which Oberth contributed the prize money he won for a later book, in order to buy rocket engines. A later member of the VfR was Werner von Braun.

The people mentioned so far were writers and mathematicians who laid the theoretical foundations for the use of rockets as space vehicles. There were many engineers—both amateur and professional—who tried to make rockets, and who had the usual mixture of success and failure. Most of these remain anonymous, but in the United States, Robert Goddard (1882−1945) (Figure 1.3), a professor of

Figure 1.2. Herman Oberth. Courtesy NASA.

physics at Clark University in Massachusetts, was, as early as 1914, granted patents for the design of liquid-fuelled rocket combustion chambers and nozzles. In 1919 he published a treatise on rocket vehicles called, prosaically, *A Method of Reaching Extreme Altitudes*, which contained not only the theory of rocket vehicles, but also detailed designs and test results from his own experiments. He was eventually granted 214 patents on rocket apparatus.

Goddard's inventions included the use of gyroscopes for guidance, the use of vanes in the jet stream to steer the rocket, the use of valves in the propellant lines to stop and start the engine, the use of turbo-pumps to deliver the propellant to the combustion chamber, and the use of liquid oxygen to cool the exhaust nozzle, all of which were crucial to the development of the modern rocket. He launched his first liquid-fuelled rocket from Auburn, Massachusetts, on 16 March 1926. It weighed 5 kg, was powered by liquid oxygen and petrol, and it reached a height of 12.5 metres. At the end of his 1919 paper Goddard had mentioned the possibility of sending an unmanned rocket to the Moon, and for this he was ridiculed by the Press. Because of his rocket experiments he was later thrown out of Massachusetts by the fire oAcer, but he continued his work until 1940, launching his rockets in New Mexico. In 1960 the US government bought his patents for two million dollars.

The way Goddard was regarded in the United States—and, indeed, the way in which his contemporaries were treated in most other countries—is in marked contrast to the attitude of the German and Russian public and government to rocket amateurs. Tsiolkovsky was honoured by both the Tsarist and the Communist

Figure 1.3. Robert Goddard. Courtesy NASA.

governments, and in Germany the serious public interest in rockets was mirrored by the government. Thus by the time of the Second World War, Russia and Germany were well ahead of other nations in rocketry—Germany most of all. The amateur German rocket societies were noticed by the military, and they soon came under pressure to turn their talents to the military sphere. Von Braun was one of the enthusiastic engineers who took this step, joining the military research station at Peenemunde. Here was developed the A4 liquid-fuelled rocket which became the notorious V2 weapon, and which, from its launch site in Germany, carried a 1,000- pound bomb into the centre of London.

The A4 embodied many concepts similar to those patented by

Goddard, and it was the first practical and reproducible liquid-fuelled rocket. It brought together the ideas of the amateurs and dreamers, and these were developed within the discipline and urgency of a military programme. When the war ended, there was a race to reach Peenemunde, both by the Russians (informed by Churchill) and the Americans. In the end, as is well known, the Russians took Peenemunde and its contents, and the US took von Braun and key members of his team.

The military weapon and space races, which followed, have been well documen- ted. To some extent, the old attitudes to rocketry were reflected in the relative progress of the United States and Russia. The Apollo programme will be seen as the most significant achievement of the twentieth century, and nothing can take that away from the United States. It did, however, represent a concerted effort to catch up with, and overtake the earlier, dramatic Russian successes.

1.1.1 The Russian spase programme

In general terms, the Russian space programme has been the most active and focused in history: the first artificial satellite, the first man in space, the first spacecraft on the Moon, the first docking of two spacecraft, and the first space station. All of these are the achievements of Russia (or, rather, the Soviet Union). In the period from 1957 to 1959, three satellites and two successful lunar probes had been launched by the USSR, ironically, fulfilling Goddard‘s prophesy. In 1961, Yuri Gagarin became the first man in space, and at the same time, several fly-bys of Mars and Venus were accomplished. In all there were 12 successful Russian lunar probes launched before the first Saturn V. Apart from the drive and vision of the Soviet engineers—particularly Sergei Korolev—the reason for this success lay in the fact that the Russian rockets were more powerful, and were better designed. The pre-war Russian attitude to rocketry had found a stimulus in the captured German parts, leading to the development of an already indigenous culture which was to produce the best engines. It is significant that the Saturn V was the brainchild of Werner von Braun, a German, and the Vostok, Soyuz, and Molniya rockets were the brainchildren of Korolev and Glushko, who were Russian.

This Russian inventiveness has continued, and it is interesting to note that, following the end of the Cold War, Russian rocket engines for new launchers are being made under licence in the United States; and that Hall effect electric thrusters, developed in Russia, are one of the key technologies for future exploration of space. The collapse of the Iron Curtain is, in this specific sense, the analogue of the collapse of the walls of Constantinople, in generating a renaissance in space propulsion.

As the epitome of the practical engineer, Sergei Korolev (1906−1966) and his colleague Valentin Glushko (1908−1989) should be credited with much of the Soviet success. Glushko was the engine designer, and Korolev was the rocket designer. Glushko's engines, the RD 100, 200 and 300 series, were and still are used in Russian launchers. It is significant that the 100 series, using liquid oxygen and alcohol, was a Russian *replacement* for the A4 engine. The desire to use a purely Russian engine was already strong. In fact, in the 1930s Glushko had developed liquid-fuelled engines which used regenerative cooling, turbo-pumps, and throttles. Korolev, as chairman of the design bureau, led the space programme through its golden age.

1.1.2 Other national programmes

Before turning to the United States' achievements in rocketry, we should remember that a number of other nations have contributed to the development of the present- day portfolio of launchers and space vehicles. There are active space and launcher programmes in the Far East, where China, Japan, India, and Pakistan all have space programmes. China and Japan both have major launcher portfolios.

In the Middle Kingdom, the invention of the rocket was followed by a long sleep of nearly 1,000 years. It was not until Tsien Hsue-Shen (1911−) was deported from the United States in 1955 that China began the serious development of modern rockets. He had won a scholarship to MIT in 1935, and later became Robert Goddard Professor of Jet Propulsion at CalTech. It is ironic that this supposed communist had been assigned the rank of temporary colonel in the US Air force in 1945, so that he could tour the German rocket sites, and meet Werner von Braun. He became, in effect, the Korolev or Werner von Braun of the Chinese space programme. Work on modern rockets began in China in 1956, and by the end of 1957, through an agreement with the USSR, R-1 and R-2 rocket technology had been transferred to the

Chinese. Understandably these were old Russian rockets, and bore more resemblance to the German A4 than to the then current Russian launchers. Following the breach with the USSR in 1960, the Chinese programme continued, with an indigenous version of the R-2 called Dong Feng, or East Wind. Engulfed by the Cultural Revolution, the programme struggled through, with the support of Zhou Enlai, to the design of a new rocket—the Chang Zheng, or Long March. This was ultimately used to launch China‘s first satellite in 1970, a year after Apollo 11. China has continued to launch satellites for communications and reconnaissance, using versions of the Long March. Since 1990, this vehicle has been available as a commercial launcher. Tsien continued to play a major part in the programme, but fell into disfavour in 1975. Nevertheless he is still considered as the father of the modern Chinese space programme, and was honoured by the government in 1991. The Long March used a variety of engines, all developed in China, including those using liquid hydrogen and liquid oxygen. Despite the setbacks caused by political upheaval, China has succeeded in establishing and maintaining an indigenous modern rocket technology. Recent developments in China have placed the country as the third in the world to have launched a man into space. China is developing a strong manned space programme.

Japan is a modern democracy, and rockets were developed there in an exclusively non-military environment. In fact, Japan‘s first satellite, Osumi, was launched by a rocket designed and built by what was essentially a group of university professors. The heritage of this remarkable success is that Japan had two space agencies: the Institute of Space and Astronautical Science, depending from the Ministry of Culture, or Monbusho; and the National Space Development Agency, depending from the Ministry of Industry. ISAS was founded in the mid-1950s, and has developed a series of indigenous, solid-fuelled launchers used exclusively for scientific missions. These have ranged from small Earth satellites, to missions to the Moon, Mars and to comets and asteroids. ISAS launched Japan‘s first satellite in February 1970, after the US and France, and before China and the United Kingdom. Although small, ISAS has continued to develop advanced rocket technology including liquid hydrogen and liquid oxygen engines; and experiments on electric propulsion and single stage to orbit technology are in progress. NASDA is more closely modelled on

NASA and ESA, and is concerned with the development of heavy launchers and the launching of communication and Earth resources satellites. It has an ambitious space programme, and is a partner in the International space Station. After its foundation in 1964, NASDA began work, using US prototype technology, to produce the N series of heavy launchers. It has now developed entirely Japanese rocket technology for the H series of launchers. Japan was the fourth nation to launch a satellite, and with its two space agencies ranks as a major space-faring nation.[1]

India began space activities in 1972, when its first satellite was launched by the USSR; but development of a native launcher—the SLV rocket, which launched the satellite Rohini in 1980—took longer. There is now a substantial launcher capability with the ASLV and PSLV rockets.

Following the devastation of the Second World War, European nations entered the space age belatedly, with satellite launches by France, and later Britain. The National Centre for Space Studies (CNES) was founded in France in 1962, and retains responsibility for an active and wide-ranging national space programme. Using the Diamant rocket, it launched the first French satellite, Asterix, in 1965. Britain also developed a launcher to launch the Prospero satellite in 1971. Given the size of the US and USSR space programmes, individual nations in Europe could not hope to make a significant impact on space exploration. This was recognised by the creation of the European Space Agency in 1975.

ESA enabled the focusing of the technology programmes of the individual nations into a single space programme, and has been remarkably successful. It has succeeded in the creation of a coherent space programme, in which is combined the co-operative efforts of 17 member-states. This is evident in the many satellites which have been launched, and major participation in the International Space Station; but more so in the development of the Ariane European heavy launcher. Beginning with Ariane 1 in 1979, some 84 launches had been completed by 1996, the versions advancing to Ariane 4. Ariane has continued to develop, and in 1998 the first successful launch of the Ariane 5 vehicle took place. This is all-new technology, with a main-stage engine fuelled by liquid hydrogen and liquid oxygen, solid boosters, and the most modern control and guidance systems. Ariane 5 is now the main launcher for ESA and Ariane 4 has been discontinued.

AThe scale of the Ariane effort can be appreciated from the fact that engine production numbers exceed 1,000 (for the Viking, used on Ariane 4). Thus Europe has the most up-to-date rocket technology, and is in serious contention with the United States for the lucrative commercial satellite launcher market. Amongst the European nations, France and Germany take the lead in the Ariane programme, as in much else in Europe. The Ariane V has recently increased its capacity to 10 tonnes in geostationary transfer orbit. ESA is also about to commission a small launcher VEGA, and the Kourou launch-site is now ready to use Soyuz launchers under an agreement between Russia and Arianespace.

1.1.3 The fnited States spase programme

The achievement of the United States in realising humanity‘s dream of walking on the Moon cannot be overrated. Its origin in the works of Tsiolkovsky and Oberth, its national expression in the dream of Robert Goddard, and its final achievement through the will of an American president and people, is unique in human history. From what has gone before it is clear that the ambition to walk on the Moon was universal amongst those who could see the way, and did not belong to any one nation or hemisphere. Nor was the technology exclusive. In fact, the Soviet Union came within an ace of achieving it. But it rested with one nation to achieve that unity of purpose without which no great endeavour can be achieved. That nation was the United States of America.

After the Second World War, the United States conducted rocket development based on indigenous technology, and the new ideas coming from the German programme, involving von Braun. Inter-service rivalry contributed to the diAculties in achieving the first US satellite launch, which was mirrored by the divisions between different design bureaux in the Soviet Union, although there it was less costly. The Army developed the Redstone rocket, basically improving upon the A4, in the same way that the R series developed in the USSR. The Navy had its own programme based on the indigenous Viking. The Air Force was working on developments that would lead to the Atlas and Titan rockets.

Finally there developed two competing projects to launch a satellite—one involving von Braun and the Redstone, and the other the Naval Research Laboratory with the Viking. The Redstone version, approved in 1955, was empowered to use existing technology to launch a small satellite, but only three months later, the more sophisticated Vanguard project from the NRL, was put in its place. In the event, this rocket—with a Viking first stage, and Aerobee second stage and a third stage, not yet developed—lost the race into space.

After two successful sub-orbital tests, the first satellite launch was set for December 1957—just after the successful Sputnik 1 flight. It exploded 2 seconds after launch. A second attempt in February 1958 also failed, and it was not until March 1958 that a 1-kg satellite was placed in orbit. This was too late, compared with both the Soviet programme and the rival Redstone programme. The latter had been restarted after the first Vanguard loss, and on 31 January 1958 it launched the United States‘ first satellite, which weighed 14 kg. The first stage was a Redstone, burning liquid oxygen and alcohol, and the upper stage was a Jupiter C solid motor. The satellite was both a programmatic and a scientific success: it discovered the Van Allen radiation belts.

After this, the von Braun concept held sway in the US space programme. Although the folly of competing programmes was not completely abandoned, NASA was set up on 1 October 1958 and began looking at plans for a Moon landing. Immediately after the first sub-orbital flight of an American in May 1961, and one month after Yuri Gagarin‘s orbital flight, John F. Kennedy made his famous announcement to Congress: ‘I believe that this nation should ‘

But there were still competing concepts from the different organisations. Von Braun conceived the Saturn under the aegis of the Army. The debate on whether to refuel in orbit, or to complete the mission through a direct launch from Earth, continued for some time; a similar debate took place in Russia a few years later. The latter concept required a huge, yet-to-be-designed rocket called Nova; but finally the concept of Apollo, using the lunar orbiter, emerged as the most practical solution. This was based on an original Russian idea published in the 1930s, and was elaborated by John Houbolt, from NASA Langley. Von Braun‘s support was crucial in the final acceptance of this idea. The launcher needed to be huge, but not as big as the Nova and it eventually emerged as the Saturn V. The lunar lander would be a separate spacecraft, which would need only to journey between the Moon and lunar orbit. The lunar orbiter would be designed to journey between Earth and lunar orbit. This separation of roles is the key to simple and reliable design, and it contributed to the success of the Apollo programme. The concept was fixed in July 1962, after which work began on the Saturn.

In its final form, as the Apollo 11 launcher, the Saturn V (Plate 6) was the largest rocket ever built. It needed to be, in order to send its heavy payload to the Moon in direct flight from the surface of the Earth. It needed powerful high-thrust engines to lift it off the ground, and high exhaust speed to achieve the lunar transfer trajectory. The lower stage was based on the liquid oxygen−kerosine engines, which had emerged, via the Redstone rocket, from the original German A4 engine that used liquid oxygen and alcohol. To achieve suAcient thrust to lift the 3 million-kg rocket off the pad, five F-1 engines—the largest ever built—provided a thrust of 34 MN, using liquid oxygen and kerosene. The exhaust velocity of these engines was 2,650 m s^{-1}, but they had a very high total thrust. The important innovation for the second and third stages was the use of liquid hydrogen. It was the first operational use of this fuel and was vital in achieving the necessary velocity to reach the Moon. The second stage had five J-2 engines (Figure 1.4), burning liquid oxygen and liquid hydrogen, and providing a total thrust of 5.3 MN, with an exhaust velocity of 4,210 m s^{-1}. The third stage had a single J-2 engine, providing a thrust of 1.05 MN. The first manned operational launch took place in December 1968, and the first lunar landing mission was launched on 16 July 1969. The mission took eight days, and the astronauts returned safely, having spent 22 hours on the Moon.

As a milestone in technology, the Saturn V was unique in the twentieth century; and as a human achievement, Apollo 11 was unique in the history of the planet. There was a strong hope that Apollo 11 would be the first step in a concerted effort towards human exploration of space—in particular, the planets. However, the shock of achievement left a sense of anticlimax, and the incentive to continue the programme, in the United States, began to diminish almost at once. The NASA budget fell from around $20 billion in 1966 to $5 billion in 1975, and since then there has been a slow rise to around $13 billion diminished in value by inflation. The planetary programme continued with unmanned probes, which have been very successful, and which have provided us with close-up views of all the planets, and some comets. These probes were launched on Atlas and Titan rockets—considerably smaller than the Saturn V.

The main technical advance since the Saturn V has been the development of the Space Shuttle. The idea of a 'space plane' originates from at least as early as the 1920s, when it was proposed by Friedrich Tsander, and elaborated ten years later by Eugene Sanger. The latter devised A4 propelled rocket planes at Peenemunde. The US Air Force had a design called the Dynosoar at the time Saturn was selected as the lunar vehicle, and, interestingly, this concept had been worked on by Tsien.

J-2 ENGINE FACT SHEET

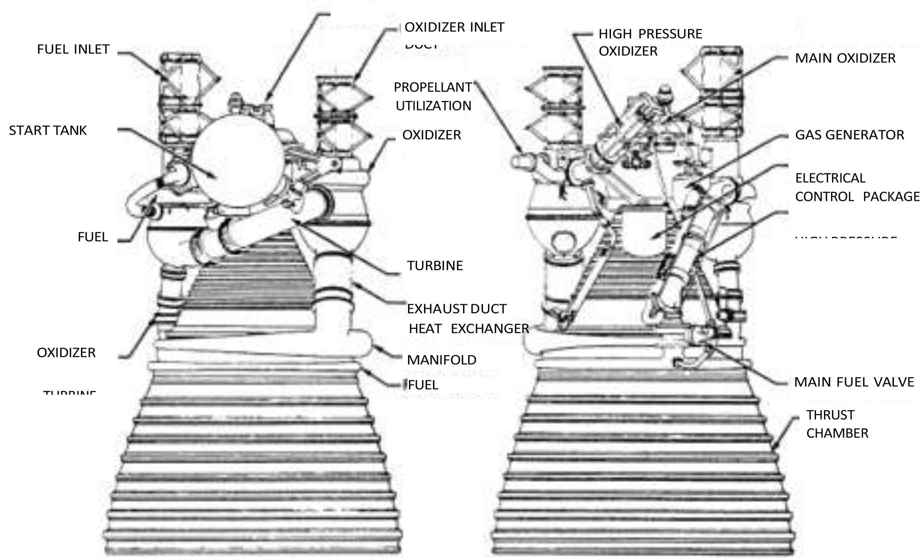


Figure 1.4. The J-2 engine used for the upper stages of Saturn V. It was the first production engine to use liquid hydrogen and liquid oxygen. It was also re-startable, a remarkable development for the 1960s. Note the lower part of the nozzle is not shown here.

It became clear, fairly soon after Apollo 11, that the budget for space could not be sustained at a high level, and that a more cost-effective way of continuing the manned programme was needed. Manned missions need to be very reliable, and this means that the components and construction have to be of the very best quality; multiple subsystems have to be provided, so that there is always a backup if one fails. The Saturn V had all of this, but it was used only once, and then discarded. All the expensive component manufacture and the huge effort to make a reliable vehicle had to be repeated for the next launch. The Shuttle concept was to bring the main vehicle back to Earth so that it could be re-used.

The use of a space plane had obvious advantages, and designs were emanating from the drawing board as early as 1971, two years after Apollo 11. The final selection—a delta-winged vehicle—was made partly on the basis of the Air Force‘s need to launch military payloads into polar orbits. The large wing meant that the Shuttle could return from a polar launch attempt to Vandenburg AFB in the event of an abort. As embodied in the first orbiter, *Columbia*, the Shuttle concept enabled re-use of the engines and control systems; propellant was carried in a disposable drop-tank, and solid boosters were used, which could be re-charged with propellant for re-use. After a ten-year development programme, the first Shuttle flew in April 1981. The total mass was two thousand tonnes, and the thrust at lift-off was 26 MN—both around 60% of the Saturn V. The height was 56 m, about half that of the Saturn V. The orbiter was fitted with three liquid hydrogen−liquid oxygen

Figure 1.5. The launch of the Space Shuttle *Atlantis*, 3 October, 1985. Courtesy NASA via Astro Info Service.

engines (SSME) which provided the high exhaust velocity needed (Figure 1.5). The propellant for these was contained in the external drop tank. At launch, the main engines were ignited, and after a few seconds, when their thrust was stable, the twin solid boosters were fired. The boosters dropped off when exhausted, and the orbiter continued into space under the power of the main engines. When the propellant was exhausted, the external tank was discarded. The orbiter was fitted with smaller storable propellant engines for orbital manoeuvring, and the all- important de-orbiting. Most of the kinetic energy from the orbit was dumped as heat during re-entry to the atmosphere. Once the velocity had dropped to a low enough value for the orbiter to behave like an aircraft, it was flown as a supersonic glider to its landing strip.

There were 24 successful flights before the loss of the *Challenger* due to a failed gas seal on one of the boosters. This underlined the need for continuous vigilance and attention to detail when dealing with such powerful forces. Shuttle flights were resumed in 1988, and have continued to the present day. The Shuttle is the primary means of launching the components of the International Space Station. The payload capability of 24 tonnes to low Earth orbit is not matched by any of the current expendable launchers, although the Ariane 5 has a capability similar to the Shuttle for geosynchronous orbit.

The recent accident with *Columbia*, and the loss of seven astronauts, has underlined the diAculties and dangers of human spaceflight. The Space Shuttle programme, and NASA itself have been subject to major review, and very significant changes to the NASA programme and organisational structure have been imple- mented. The Space Shuttle will be retired, and all non-space station flights have been cancelled except for a mission to re-furbish the Hubble space telescope. New safety requirements make this essential. The Space Shuttle will be replaced with a *Crewed Exploration Vehicle*, launched on an expendable rocket, and using Apollo style re- entry rather than the complex, and—as has been proven so tragically—dangerous, Shuttle system. The programme of NASA has been re-directed towards a return to the Moon and ultimately a human expedition to Mars.

1.1.4 Commentary

This brief summary of the history of rockets brings us to the present day when, after a period of relative stagnation, new rocket concepts are again under active consideration; these will be discussed later. A number of noteworthy points emerge from this survey. The invention of the rocket preceded the theory by 1,000 years, but it was not until the theory had been elaborated that serious interest in the rocket as a space vehicle developed. The theory preceded the first successful vehicle, the A4, by about 50 years. This seems to be because there were serious engineering problems, which required solution before ideas could be put into practice. One of these problems was guidance. A rocket is inherently unstable, and it was not until gyroscopes were used that vehicles could be relied upon to remain on course. Rocket engines are very high-power devices, and this pushes many materials and com- ponents to their limits of stress and temperature. Thus, rocket vehicles could not be realised until these problems were solved.

And this required the materials and engineering advances of the early twentieth century.

It is also noteworthy that the basic ideas were universal. Looking at the developments in different countries, we see parallel activity. Goddard's patents were 'infringed', but not through theft; the basic ideas simply led to the same solutions in different places. It is, however, remarkable that the A4 programme should have played such a seminal role. It seems that the solutions arrived at on Peenemunde provided just that necessary step forward needed to inspire engineers around the world to apply their own knowledge to the problems of rocket vehicles. The use to which the A4 rocket was put, and its means of manufacture, were dreadful, but cannot be denied a position in the history of rocket engineering.

NEWTON'S THIRD LAW AND THE ROCKET EQfATION

As we have seen, the rocket had been a practical device for more than 1,000 years before Tsiolkovsky determined the dynamics that explained its motion. In doing so, he opened the way to the use of the rocket as something other than an artillery weapon of dubious accuracy. In fact, he identified the rocket as the means by which humanity could explore space. This was revolutionary: earlier, fictious journeys to the Moon had made use of birds or guns as the motive force, and rockets had been discounted. By solving the equation of motion of the rocket, Tsiolkovsky was able to show that space travel *was* possible, and that it could be achieved using a device which was readily to hand, and only needed to be scaled up. He even identified the limitations and design issues which would have to be faced in realising a practical space vehicle. The dynamics are so simple that it is surprising that it had not been solved before—but this was probably due to a lack of interest: perusal of dynamics books of the period reveals consistent interest in the flight of unpowered projectiles, immediately applicable to gunnery.

In its basic form, a rocket is a device which propels itself by emitting a jet of matter. The momentum carried away by the jet results in a force, acting so as to accelerate the rocket in the direction opposite to that of the jet. This is familiar to us all—from games with deflating toy balloons, if nothing else. The essential facts are that the rocket accelerates, and its mass decreases; the latter is not so obvious with a toy balloon, but is nevertheless true.

In gunnery, propulsion is very different. All the energy of a cannon ball is given to it in the barrel of the gun by the expansion of the hot gases produced by the explosion of the gunpowder. Once it leaves the barrel, its energy and its velocity begin to decrease, because of air friction or drag. The rocket, on the other hand, experiences a continuous propulsive force, so its flight will be different from that of a cannon ball. In fact, while the cannon ball is a *projectile*, the rocket is really a *vehicle*. The Boston Gun Club cannon, in Jules Verne‘s novel, was in fact the wrong method. To get to the Moon, or indeed into Earth orbit, requires changes in speed and direction, and such changes cannot be realised with a projectile. H. G. Wells‘ *cavorite*-propelled vehicle was closer to the mark.

1.1.1 Tsiolkorsky’s rosket equation

Tsiolkovsky was faced with the dynamics of a vehicle, the mass of which is decreasing as a jet of matter is projected rearwards. As we shall see later, the force that projects the exhaust is the same force that propels the rocket. It partakes in Newton‘s third law—‘action and reaction are equal and opposite‘, where ‘action‘means force. The accelerating force is represented, using Newton‘s law, as

$$F = mv_e$$

In this equation, the thrust of the rocket is expressed in terms of the *mass flow rate*, *m*, and the *effective exhaust velocity*, v_e.

So the energy released by the burning propellant appears as a fast-moving jet of matter, *and* a rocket accelerating in the opposite direction. Newton‘s law can be applied to this dynamical system, and the decreasing mass can be taken into account, using some simple differential calculus (the derivation is given at the beginning of Chapter 5). The resultant formula which Tsiolkovsky obtained for the vehicle velocity *V* is simple and revealing:

$$V=Ve \, Loge \, [Mo/M]$$

Here M_0 is the mass of the rocket at ignition, and *M* is the current mass of the rocket. The only other parameter to enter into the formula is v_e, the effective exhaust velocity. This simple formula is the basis of all rocket propulsion. The velocity increases with time as the propellant is burned. It depends on the natural logarithm of the ratio of initial to current mass; that is, on how much of the propellant has been burned. For a fixed amount of propellant burned, it also depends on the exhaust velocity—how fast the mass is being expelled.

This is shown in Figure 1.б, where the rocket velocity is plotted as a function of the *mass ratio*. The mass ratio, often written as *R*, or fl, is just the ratio of the initial to the current mass:

$$R=Mo/M$$

Elliptical transfer orbits

This horizontal acceleration is also used to move from one orbit to another, using an elliptical transfer orbit. When the Space Shuttle has to rendezvous from a low orbit, with, for instance, the Space Station, it accelerates *along* its orbit. Instead of travelling faster around the orbit, the orbit rises; it has in fact become a slightly elliptical orbit, the perigee being where the thrust was turned on. This elliptical orbit intersects with the Space Station orbit at the point where the Station is located when the Shuttle arrives there—if the burn was actuated at the correct time. Once at the Station, the Shuttle has to accelerate, again horizontally, in order to match its speed to that of the Station. This acceleration effectively puts the Shuttle into a circular orbit at the new altitude.

It should be clarified that the velocity of a spacecraft in a circular orbit decreases with the square root of the radius. So the Space Station is moving more slowly in its orbit than is the Shuttle in its lower orbit. However, for an elliptical orbit the velocity at apogee is lower than the intersecting circular orbit, because of the exchange of kinetic energy for potential energy as the spacecraft rises. At the same time, the velocity of the elliptical orbit is greater than that of the intersecting circular orbit at perigee. So to transfer from a low orbit to a higher orbit, two velocity increments are necessary: one at the perigee of the elliptical transfer orbit, and another at the apogee.

For convenient reference, some of the equations for the velocity of tangential circular and elliptic orbits are presented here.

$$V_0 = \sqrt{\frac{GM_\oplus}{r_0}}$$

for a circular orbit

$$V_{Escape} = \sqrt{\frac{2GM_\oplus}{r_0}}$$

for a parabolic escape orbit

$$\frac{V_1}{V_0} = \sqrt{1 + \frac{r_2 - r_0}{r_2 + r_0}}, \quad \frac{V_2}{V_1} = \frac{r_0}{r_2}$$

for elliptic orbits

where r_0 is the perigee radius, r_2 is the apogee radius, V_1 is the elliptic orbit velocity at perigee, and V_2 is the elliptic orbit velocity at apogee. These are shown in Figure 1.9. It will perhaps be obvious from the foregoing that the manoeuvre to ‘catch up‘ with a spacecraft in the *same* orbit is quite complicated. Simply accelerating will not cause the Shuttle to move faster round the orbit; it will put it into an elliptical orbit, which will pass above the target. The Shuttle needs to decelerate and drop to a lower and faster orbit; and then, at the correct point in the lower orbit it should accelerate again to bring it up to the target. Both of these manoeuvres are elliptical orbit transfers. A further acceleration is then needed to match the target speed; that is, to circularise the orbit of the shuttle. These orbits need only be separated from one another by a few tens of kilometres, and the velocity changes are small.

A decrease in velocity when a spacecraft is in a circular orbit causes it to enter an elliptical orbit. The apogee is at the same altitude as the circular orbit, and the perigee is determined by the velocity decrease. At perigee, the spacecraft velocity is greater than the corresponding new circular orbit, so a further decrease is needed. This is the initial manoeuvre when a spacecraft returns to Earth: atmospheric drag takes over before perigee is reached, and the resultant deceleration causes the trajectory to steepen continuously until it intersects with the Earth‘s surface.

On an airless body like the Moon, the elliptical orbit taken up should have its perilune close to the surface at the desired landing point, and the vehicle has to be brought to rest using thrust from the motors. The Apollo 11 descent ellipse did not pass through the correct landing point, because the non-spherical components of the Moon‘s gravitational field were not known accurately. Armstrong had to take over control, and by using lateral thrust from the motors he guided it to the correct point.

Launch tvajectovies

Having discussed how a spacecraft can move from one orbit to a higher orbit, it is possible to see how a spacecraft can leave the surface of a planet and enter into a low orbit around that planet. For the Moon it is a reversal of the descent just described. For minimum energy expenditure, it would, in theory, be better to launch in a horizontal direction, regarding the launch point as being on a circular orbit at zero altitude. The velocity of the spacecraft, at rest on the surface, is wrong for the circular orbit, otherwise it would be hovering above the ground; so, during the burn, a significant vertical component of velocity is needed, while horizontal velocity is gained. There is also the question of mountains. The Apollo lunar module took off with a short vertical segment, before moving into a horizontal trajectory. Once in anelliptical orbit with an apogee at the required altitude, the spacecraft coasts towards it. At apogee, a further burn circularises the orbit.

For launch from Earth, the atmosphere is a significant problem. Although the density of air drops rapidly with height, the velocity of a spacecraft is large, and drag is proportional to the square of the velocity. Below 200 km there is suAcient drag to make an orbit unstable. In fact, to have a lifetime measured in years, an orbit needs to be above 500 km. This means that a significant proportion of the stored chemical energy in a rocket has to be used to raise the spacecraft above the atmosphere. Atmospheric drag also slows the rocket in the lower atmosphere. For this reason, spacecraft are launched vertically; height is gained, and the drag in the dense lower atmosphere is minimised, because the rocket is not yet moving very fast. Once the densest part of the atmosphere is passed—at about 30 km—a more inclined trajectory can be followed. It cannot be horizontal because the velocity is not yet high enough. Ultimately, suAcient velocity is reached for an elliptical trajectory to the desired altitude to be followed, and the spacecraft can then coast. At apogee a further horizontal acceleration is needed to enter the circular orbit. During a launch, the velocity is never high enough to attain a circular orbit at an intermediate altitude; this occurs only after the final injection has taken place. Intermediate orbits would require a still larger horizontal velocity, because of the inverse dependence of velocity on radius. Thus the motion of the rocket before final injection is along a trajectory which is always steeper than the free space ellipse, and it intersects the Earth‘s surface. When a launch fails, this becomes obvious. The non- optimum nature of the launch trajectory, compared with a transfer ellipse, means that a good deal of the energy of the rocket is lost to the Earth‘s gravitational field

THERMAL ROCKET ENGINE

The rocket principle is the basis of all propulsion in space, and all launch vehicles. The twin properties of needing no external medium for the propulsion system to act upon, and no external oxidant for the fuel, enable rockets to work in any ambient conditions, including the vacuum of space. The thermal rocket is the basis of all launchers, and almost all space propulsion (although some electric propulsion uses a different principle). In this chapter we shall treat the rocket motor as a heat engine, and examine the physical principles of its operation. From these physical principles the strengths and limitations of rocket motors can be understood and appreciated. The thermal rocket motor is a heat engine: it converts the heat, generated by burning the propellants—fuel and oxidiser, in the combustion chamber—into kinetic energy of the emerging exhaust gas. The momentum carried away by the exhaust gas provides the thrust, which accelerates the rocket. As a heat engine, the rocket is no different in principle from other heat engines, such as the steam engine or the internal combustion engine. The conversion of heat into work is the same, whether the work is done on a piston, or on a stream of exhaust gas. It will be helpful if we look first at the basic form of the thermal rocket.

THE BASIC CONFIGURATION

A liquid-fuelled rocket engine (see Figure 2.1) consists of a combustion chamber into which fuel and oxidant are pumped, and an expansion nozzle which converts the high-pressure hot gas, produced by the combustion, into a high velocity exhaust stream. It is the expansion of the hot gas against the walls of the nozzle which does work and accelerates the rocket.

A solid-fuelled motor (Figure 2.2) operates in the same way, except that the fuel and oxidant are pre-mixed in solid form, and are contained within the combustion chamber. Normally the combustion takes place on the inner surface of the propellant

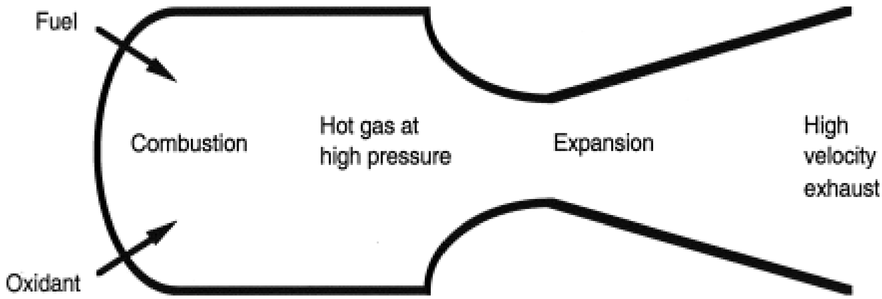


Figure 2.1. A liquid-fuelled rocket engine.

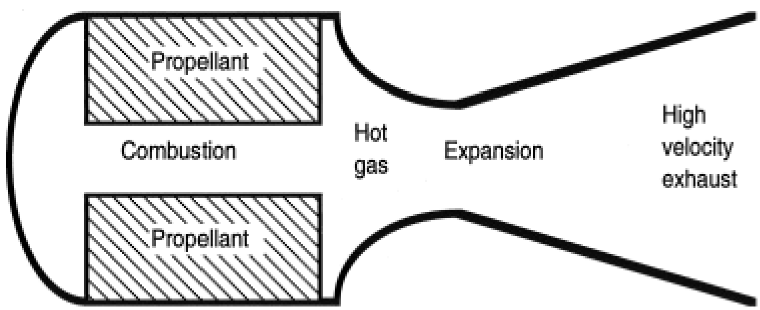


Figure 2.2. A solid-fuelled rocket motor.

charge. The exhaust nozzle is identical in form to that in the liquid-fuelled motor, and the principles of operation are the same. In this chapter we shall make little distinction between the solid- and liquid-fuelled variants of the thermal rocket motor.

The combustion, which takes place in the chamber, can be any chemical reaction which produces heat. It may be simple oxidation of a fuel (hydrocarbon, or pure hydrogen, for example) by liquid oxygen; or it may be one of a number of other kinds of exothermic chemical reaction, as, for example, between fuming nitric acid and hydrazine. Solid propellants may contain an oxidiser such as ammonium perchlorate together with finely divided aluminium and carbon, all bound together in a rubber-like material. Gunpowder is the classical solid propellant, and was used in the first Chinese rockets. A mixture of aluminium powder and sulphur is an example of a solid propellant with no oxidiser involved: the exothermic reaction produces aluminium sulphide. The main requirement for all propellant combinations is to maximise the energy release per kilogramme; as with any other rocket component, the lower the mass for a given energy release, the higher will be the ultimate velocity of the vehicle. The principles of the thermal rocket do not depend on specific types of propellant, so this aspect will only impact in a minor way on

DEVELOPMENT OF THRfST AND THE EFFECT OF THE ATMOSPHERE

In Chapter 1 the discussion of the rocket equation and the application of Newton‘s third law to rocket propulsion ignored the effects of atmospheric pressure and the actual forces involved in producing the propulsive thrust. The concept of *effective* exhaust velocity enabled this simplification. The effective exhaust velocity is that velocity which, when combined with the actual mass flow in the exhaust stream, produces the measured thrust, F mv_e, where m is the mass flow rate, and v_e is the effective exhaust velocity; v_e combines the true exhaust velocity with the effects of atmospheric pressure and the pressure in the exhaust stream, into one parameter. The true exhaust velocity, however, is a function of these parameters, as well as the conditions of temperature and pressure in the combustion chamber. Here we shall look in more detail at the functioning of the rocket engine, and the development of exhaust velocity and thrust.

In the middle of the combustion chamber, the hot gas containing the energy released in the chemical reaction is virtually stationary. The energy—at this moment represented by the temperature and pressure of the gas—has to be converted into velocity. This occurs as the gas expands and cools while it passes through the nozzle. The velocity rises very rapidly, passing the speed of sound (for the local conditions) as it crosses the ‘throat‘ or narrowest part of the nozzle. Thereafter it continues to accelerate until it leaves the nozzle. The accelerating force on the gas stream is the reaction of the nozzle wall to the gas pressure, as the gas expands against it. Thus the thrust is mostly developed by the nozzle itself, and is then transferred to the vehicle through the mounting struts. The accelerating force on the rocket is thus linked into the structure holding the rocket engine, and thereby to the base of the rocket itself. The development of thrust, and the effect of atmosphere, can be examined through the derivation of the *thrust equation*, which relates the thrust of the rocket to the actual exhaust velocity, the pressure in the combustion chamber, and the atmospheric pressure. It allows insight into some of the main issues in rocket motor design. The equation is derived by considering two separate applications of Newton‘s third law: once to the exhaust gases and once to the rocket motor, and the vehicle to which it is attached. It is important to recognise that the processes in a rocket engine result in two motions: the forward motion of the rocket and the backward motion of the exhaust stream, both of which require application of Newton‘s third law. There are two forces involved: the reaction of the internal surfaces of the rocket engine, which accelerates the gas; and the pressure force of the gas on those internal surfaces, which accelerates the rocket.

Figure 2.3 represents the action of the gas pressure on the combustion chamber and the exhaust nozzle; this is the force which accelerates the rocket. It also shows the reaction of the walls of the combustion chamber and of the exhaust nozzle,

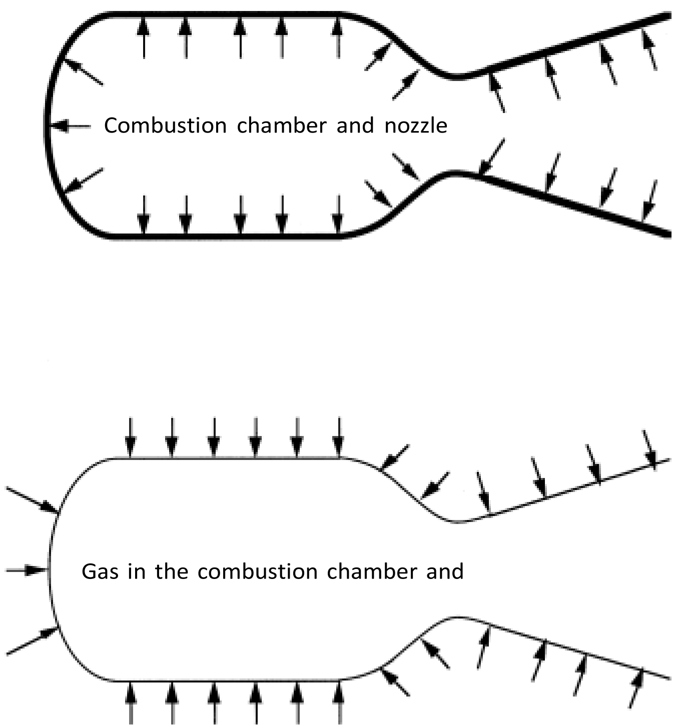


Figure 2.3. Forces in the combustion chamber and exhaust nozzle.

acting on the gas contained by them, which is the force that accelerates the exhaust gas.

The force accelerating the exhaust gas, the reaction of the walls, is

$$=$$

equal to the surface integral of the pressure, taken over the whole inner surface of the chamber and nozzle: $F \quad p\, dA$. This is not the only force acting on the gas: there is also a retarding force, which can best be appreciated by referring to Figure 2.4.

The gas flowing through the nozzle is impelled by the pressure gradient from the combustion chamber to the exit. At any point in the nozzle, the pressure upstream is greater than the pressure downstream. Considering the shaded portion of the exhaust stream represented in Figure 2.4, the net accelerating force acting on the shaded portion is

$$dF = pA — (p — dp)A$$

where A is the cross-sectional area at any given point, and the pressure gradient is dp/dx. This is the force that accelerates the gas through the nozzle. This formula applies at any point in the nozzle. For an element at the extreme end of the nozzle— the exit point shown in Figure 2.4—the outward force is pA, but the retarding force is the pressure at the exit plane, which can be denoted by p_e, multiplied by the area at the exit plane, A_e.

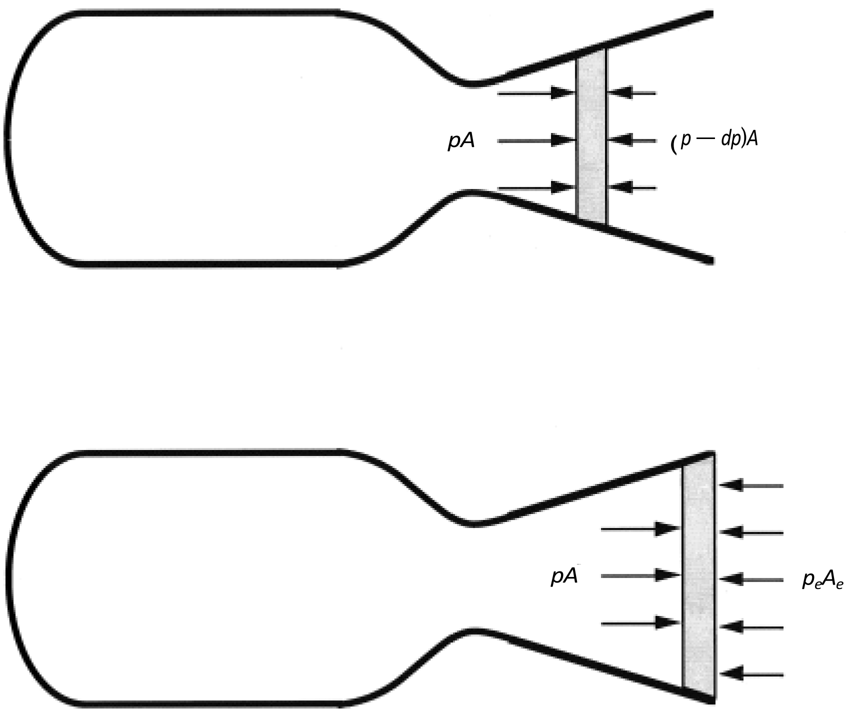


Figure 2.4. Gas flow through the nozzle.

It is important to realise that the exhaust stream immediately beyond the end of the nozzle is not affected by ambient pressure: it is travelling at supersonic velocity, and hydrostatic effects can only travel at sound speed. Further downstream, effects of turbulence at the boundary between the exhaust and the atmosphere will make themselves felt, and under certain conditions shock waves can develop. But immedi- ately beyond the exit plane the flow is undisturbed unless extreme conditions prevail. Thus for our purposes the above analysis holds.

Considering now the application of Newton‘s law to the exhaust gases, the accelerating force is represented by

$$F_G = \quad p\, dA - p_e A_e = m u_e$$

where m is the mass flow rate through the nozzle, and u_e is the exhaust velocity. This is the force that accelerates the exhaust stream in the nozzle; beyond the end of the nozzle the stream ceases to accelerate, and until turbulence starts to slow the stream down the exhaust velocity is a constant.

Turning now to the accelerating force on the rocket, this is represented by the surface integral of the pressure over the walls of the combustion chamber and

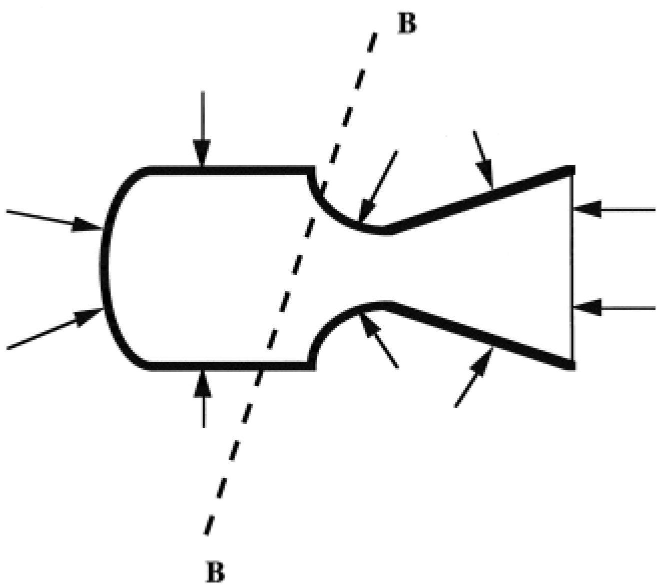


Figure 2.5. Static force due to atmospheric pressure.

nozzle:

$$F_R = \quad p\,dA$$

which is the force tending to accelerate the rocket.

Again there is a retarding force acting on the rocket due to the atmospheric pressure. This is a static force which operates whether or not the rocket is moving through the atmosphere. (There are of course, in addition, aerodynamic forces of drag and lift, developed through the motion of the rocket through the atmosphere; these are considered in Chapter 5.) To evaluate this static force, consider the cold rocket motor—not firing—shown in Figure 2.5.

As the rocket is stationary under the atmospheric pressure forces, they must balance across any arbitrarily chosen plane cutting the rocket, BB. When the rocket is active, and the supersonic exhaust stream occupies the region to the right of the nozzle exit plane, there is no longer a force due to atmospheric pressure acting on the exit plane (see the argument given above). Since the plane across which the atmospheric forces balance, in the cold case, can be chosen arbitrarily, BB can be moved to coincide with the exit plane of the nozzle without violating any physical principle. The unbalanced atmospheric force is then seen to be a retarding force, equal in magnitude to the atmospheric pressure integrated over the exit plane: p_aA_e, where p_a is the atmospheric pressure and A_e is the area of the exit plane. This is of course equal to the atmospheric pressure force integrated over the whole surface of the rocket engine, but the former is much easier to calculate.

So the net force accelerating the rocket is represented by:

$$F_R = \quad p\,dA — p_aA_e$$

This is the net thrust of the rocket. The surface integral, which appears in both equations, would be diAcult to evaluate, but fortunately we have two expressions involving the same integral, and it can be cancelled. This arises because the magnitude of the force acting on the combustion chamber and nozzle is identical to that acting on the exhaust gases. Substituting for $p\,dA$, from the equation for the acceleration of the exhaust gases, we find:

$$F_R = mu_e + p_eA_e — p_aA_e$$

This is the *thrust equation.*

The difference between this equation and the version given in Chapter 1 is that the true exhaust velocity u_e is used, together with the exit plane area of the nozzle A_e and the two pressures p_e and p_a. By using the real exhaust velocity, the various forces acting on the rocket are separated out. Using this equation, we can begin to examine performance parameters of a rocket, taking into account the ambient conditions.

An expression for the effective exhaust velocity may easily be derived from the above:

$$m \qquad v_e = u_e$$

with the thrust written $F_R = \dot{m} v_e$ (as in Chapter 1).

As formulated above, the thrus equation is incomplete: for a given true exhaust velocity the thrust can be derived, taking into account the ambient conditions; however, the true exhaust velocity ue is not itself independent of the ambient conditions. Later in this chapter we shall derive an expression for ue which includes the ambient

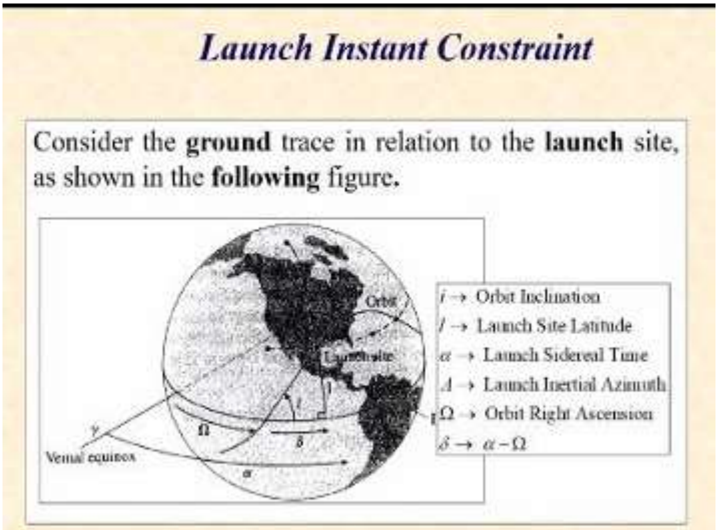
LAUNCH ORBIT TO SSTO

We know that time taken to complete a space mission is generally quite large because of the conventional launch methodology in which the larger spacecraft were designated point in space. And then there are a large number of maneuvers which are performed over period of days before the spacecraft is put into the final orbit that is as per the mission. Of course, in case of a spacecraft which is unmanned, this is not a problem, because time is really of no great essence. But you will immediately notice that in respect of missions that involve humans or other time critical requirements such as a space-based rescue operation, there is a need for faster mechanisms to complete space missions. Of course, in case of a spacecraft which is unmanned, this is not a problem, because time is really of no great essence. But you will immediately notice that in respect of missions that involve humans or other time critical requirements such as a space-based rescue operation, there is a need for faster mechanisms to complete space missions.

It has been found that in such cases where there is a time criticality attached to the mission, it is better to directly launch into the desired orbit in order to minimize the time taken to reach the orbit. These are also called launch to orbit or single-stage-toorbit concept. However, as we will see shortly, such missions need special conditions to be satisfied to create efficient transfers. One of the points which is commonly recognized is the fact that fast transfer is energy efficient if the spacecraft is launched, in a typically called a chase orbit, with destination vertically overhead. In this case, we typically wait for the destination to be overhead at the launch point and then launch the spacecraft so that it intersects the desired orbit behind the destination. In this context, it is worth noting that all spacecraft have a designated point in the space at which they must be placed in relation to other spacecraft and similar objects in similar orbits.

Of course, it is not very difficult to realize that this would require a higher energy launch in comparison to the energy of the destination orbit, because you are in a chase orbit, and you must move faster in order to ensure that your destination is reached in the desired time slot. And in order to do that, you must spend extra energy to accelerate the vehicle. SSTO that is single-stage-to-orbit or even two-stage-to-orbit that is TSTO missions are common examples of such fast transfers. Both SSTO and TSTO aim to reduce the staging related complexities. That is another benefit that you get from these technologies so that we can move towards a fully reusable rocket and significantly reduce operational cost. In this context, let me make a mention of the various efforts that are going on with regard to the reusable technology, which started with the development of space shuttle in the late 80s. But in recent times, the efforts are being made to fully reuse the rocket as the Space Shuttle Program was only partially reusable. The implication is that if you can fully reuse a launch vehicle, then the operational cost reduces significantly.

An important application of the launch to orbit technology or the SSTO technology is for missions that involve docking with another spacecraft. And this has been very successfully used in the context of space station missions of space shuttle where the space shuttle is launched in such a manner so as to dock with space station as per the required time slots. Now in order to understand this, we need to note that all spacecraft, which are visible from earth create a path on the surface of the earth which is called the ground trace. Ground trace is nothing but the locus of all points which is a result of the intersection of the radius vector of earth’s surface over which the spacecraft completes one orbital cycle. What it essentially means is that as spacecraft orbits around earth, its radius vector also sweeps one orbital cycle around the earth. And if you take the intersection of that radius vector with earth's surface, then locus of all those points is nothing but what we call the ground trace. It is almost like hypothetical shadow that the spacecraft would make on the surface of the earth as it moves around the earth. Typically, this ground trace appears like a sine wave as the spacecraft moves in its orbit. We further find that if it is visible about the launch point which has a specific latitude then it will be visible overhead at the launch point only at two-time instants in 24 hours leading to launch time restrictions or the concept of launch window.



This launch window or the launch instant constraint can be visualized through a simple schematic as shown below. So here we have the spherical earth representation and we have a launch site determined by a latitude, the launch latitude, given by this one and a launch longitude with respect to the vernal equinox in terms of the right ascension and the angle δ. The spacecraft also has an inclination with respect to the equatorial plane of earth. And with this we define the launch site. We can use simple trigonometric relations to show that the following parameters will directly decide the two-time instants at which the spacecraft will be overhead at this launch point or the launch latitude. Let us look at this relation.

It is found that satellite orbiting earth appears overhead twice a day when α, the angle as shown in the figure previously, is either $\omega + \delta$ or it is $\omega + 180 - \delta$.

That is, either it is on one side or it is on the other side. This is because both, earth is revolving around its axis and the spacecraft is revolving around this earth. So that there will be two-time instances where there will be a coincidence of motion in such a manner that the spacecraft would be overhead at the launch point. The resulting solution for time window for launch can be obtained as follows. Here, we have taken the course to what is commonly termed as spherical trigonometry, where we use spherical triangles to establish the triangle relations. And from that, it is possible to show that these two-time instants will be related to the parameters of the launch site as well as the rotation of earth.

So how does one design this orbit or the launch trajectory? In this case, it is not an orbit but it is essentially the launch trajectory, which directly is decided by the requirement of creating a chase orbit for a specific destination. However, you must note that the orbiting satellite typically has a dwell time of only a few seconds over the launch site. Please note that the spacecraft is continuously moving in the orbit. So, its visibility from the launch point will be only for a few seconds before it moves away due to the angular separation. So obviously, you cannot actually capture that moment because you cannot really have impulsive launches. While impulsive launches have been seen earlier, which reduce the gravity loss, they are highly costly in terms of the drag loss and also the other technological issues would preclude launching instantaneously.

It obviously means that you have to have a fast transfer on a trajectory which will generate a very large velocity in lower atmosphere in order to reach the destination with not very large gap. In this context, there are certain technologies which are used which permit design of such orbits which are capable of creating larger velocities in relation to the velocity of the destination point so that the desired objective of the mission can be achieved. Of course, the launch mechanism or the launch trajectory generally leave the spacecraft some distance away from the final destination point primarily due to safety concerns.

And the final step is completed by performing small magnitude orbital maneuvers which then complete the mission as desired. launch to orbit and SSTO type of missions provide a way to reduce the overall mission time in situations that require it. However, such missions are not energy efficient due to trajectories that give importance to time rather than cost. There is another drawback attached to such missions is that in case your payload fractions are high, the launch mass or the liftoff mass of the rocket is going to be large and it is going to be a huge rocket.

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AIR BREATHING PROPULSION

Lecture Notes

B. Tech III YEAR – II SEM

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Dept.Aero

SPACE PROPULSION

B.Tech. III Year AE II Sem.LT/P/DC

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Pre-Requisites: Nil

Course Objectives:

- To know about the propulsion system used in rockets and missiles
- Discuss the working principle of solid and liquid propellant rockets and gain basic knowledge of hybrid rocket propulsion.

Course Outcomes: To be able to

- Understand about trajectory and orbits
- Illustrate electric propulsion techniques, ion and nuclear rocket and the performances of different advanced propulsion systems.
- Evaluate various space missions, parameters to be considered for designing trajectories and rocket mission profiles
- Understand the fundamentals of chemical rocket propulsion, types of igniters and performance considerations of rockets.

UNIT - I

Principles of Rocket Propulsion: History of rockets, Newtons third law, orbits and space flight, typesof orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO, launch assists.

UNIT - II

Fundamentals of Rocket Propulsion: Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Rocket performance considerations of rockets, types of igniters, preliminary concepts in nozzle less propulsion, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.

UNIT - III

Solid Rocket Propulsion: Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Erosive burning in solid propellant rockets, combustion instability, strand burner and T-burner, applications and advantages of solid propellant rockets.

UNIT - IV

Liquid and Hybrid Rocket Propulsion: Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.

UNIT-V

Advanced Propulsion Techniques: Electric rocket propulsion, types of electric propulsion techniques,Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemicalrocket propulsion systems, future applications of electric propulsion systems, Solar sail.

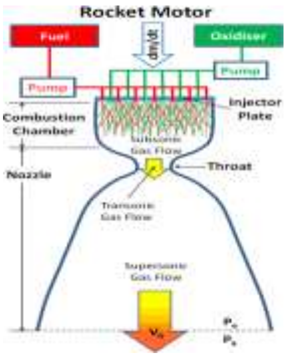
TEXT BOOKS:

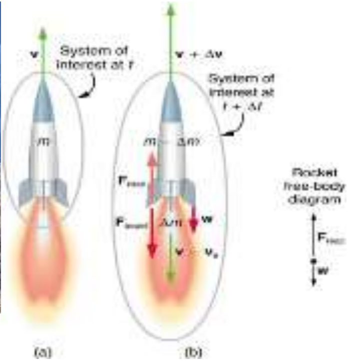
1. Hill, P.G. and Peterson, C.R., ―Mechanics and Thermodynamics of Propulsion‖, 2nd Edition, Addison Wesley, 1992.
2. Turner, M. J. L., ―Rocket and Spacecraft Propulsion‖, 2nd Edition, MIT Press, 1922.
3. Hieter and Pratt, ―Hypersonic Air breathing propulsion‖ 5th Edition, 1993.

REFERENCE BOOKS:

1. Sutton, G.P., ―Rocket Propulsion Elements‖ John Wiley & Sons Inc., New York, 5th Edition, 1993.
2. Mathur, M. L., and Sharma, R.P., ―Gas Turbine, Jet and Rocket Propulsion‖, Standard Publishers and Distributors, Delhi, 1988, Tajmar, M., Advanced Space Propulsion Systems, Springer 2003

UNIT-2





Rocket engine: A vehicle or device propelled by one or more rocket engines, especially such a vehicle designed to travel through space A projectile weapon carrying a warhead that is powered and propelledby rockets.

A projectile firework having a cylindrical shape and a fuse that is lit fromthe rear.

19-02-2019

Missile: An object or weapon that is fired, thrown, dropped, or otherwise projected at a target; a projectile.

PROPELLANT

Propellant is the chemical mixture burned to produce thrust in rockets andconsists of a fuel and an oxidizer.

A fuel is a substance that burns when combined with oxygen producing gas forpropulsion.

An oxidizer is an agent that releases oxygen for combination with a fuel.

The ratio of oxidizer to fuel is called the mixture ratio. Propellants are classifiedaccording to their state - liquid, solid, or hybrid.

The gauge for rating the efficiency of rocket propellants is specific impulse, stated in seconds.

Liquid Propellants :

- In a liquid propellant rocket, the fuel and oxidizer are stored in separate tanks, and are fed through a system of pipes, valves, and turbo pumps to a combustion chamber where they are combined and burned to produce thrust.
- Liquid propellant engines are more complex than their solid propellantcounter parts, however, they offer several advantages.
- By controlling the flow of propellant to the combustion chamber, the engine can be throttled, stopped, or restarted.

Solid Propellants :

- Solid propellant motors are the simplest of all rocket designs. They consist ofa casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust.
- When ignited, a solid propellant burns from the center out towards thesides of the casing. The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust.

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- Unlike liquid propellant engines, solid propellant motors cannot be shutdown.
- Once ignited, they will burn until all the propellant is exhausted.

Hybrid Propellants:

- Hybrid propellant engines represent an intermediate group between solid and liquid propellant engines.
- One of the substances is solid, usually the fuel, while the other, usually the oxidizer, is liquid.
- The liquid is injected into the solid, whose fuel reservoir also serves as the combustion chamber.
- The main advantage of such engines is that they have high performance, similarto that of solid propellants, but the combustion can be moderated, stopped, oreven estarted.

Operating principle

- A rocket is a machine that develops thrust by the rapid expulsion of matter.
- The major components of a chemical rocket assembly are a rocket motor or engine, propellant consisting of fuel and an oxidizer, a frame to hold the components, control systems and a cargo such as a satellite.
- A rocket is called a launch vehicle when it is used to launch a satellite or other payload into space.
- A rocket becomes a missile when the payload is a warhead and it is used as a weapon.
- At present, rockets are the only means capable of achieving the altitude and velocity necessary to put a payload into orbit.

Rocket Power

- There are a number of terms used to describe the power generated by a rocket.
- Thrust is the force generated, measured in pounds or kilograms. Thrust generated by the first stage must be greater than the weight of the complete launch vehicle while standing on the launch pad in order to get it moving.
- The impulse, sometimes called total impulse, is the product of thrust and the effective firing duration.
- The efficiency of a rocket engine is measured by its specific impulse (Isp). Specific impulse is defined as the thrust divided by the weight of the propellant consumed per second. The result is expressed in seconds.

Mass ratio

- A rocket's mass ratio is defined as the total mass at lift-off divided by the massremaining after all the propellant has been consumed.
- A high mass ratio means that more propellant is pushing less launch vehicle andpayload mass, resulting in higher velocity.
- A high mass ratio is necessary to achieve the high velocities needed to put apayload into orbit.

Thrust

Rocket thrust can be explained using Newton's 2^{nd} and 3^{rd} laws ofmotion.

2^{nd} Law: a force applied to a body is equal to the mass of the body and its acceleration in the direction of the force.

$$F = ma$$

3^{rd} Law: For every action, there is an equal and opposite reaction.

$$F_a = -F_r$$

In rocket propulsion, a mass of propellant (m) is accelerated (via the combustion process) from initial velocity (V_o) to an exit velocity (V_e).The acceleration of this mass is written as:

Thrust

Another component of thrust (*pressure thrust*, F_2) comes from the force exerted by external pressure differences on the system.This isdescribed by the difference of the pressure of the flow leaving the engine (P_e)through the exit area (A_e) compared to the external (ambient) pressure (P_a).

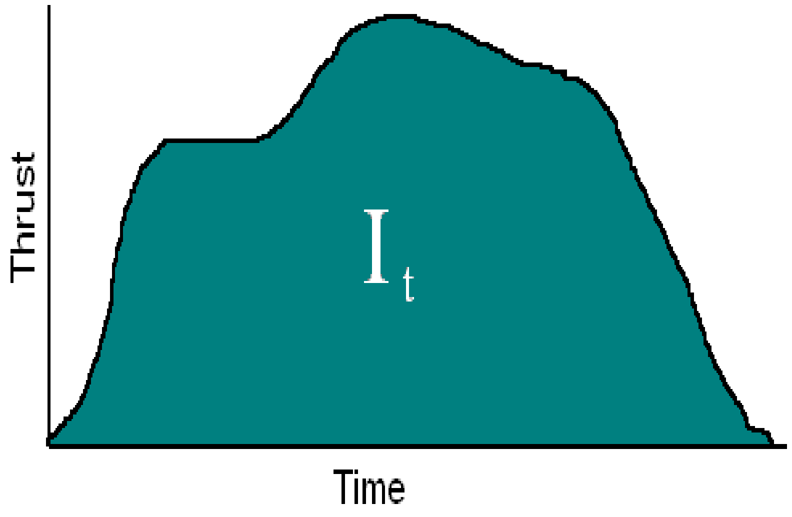
$$F_2 = (P_e - P_a)A_e$$

In space, P_a is assumed to be zero (which explains why thrust rated at vacuum is higher than at sea level).

Combining the two thrust components gives

$$\boxed{F = \frac{\dot{m}}{g}(V)_e + (P_e - P_a)A_e}$$

Specific Impulse

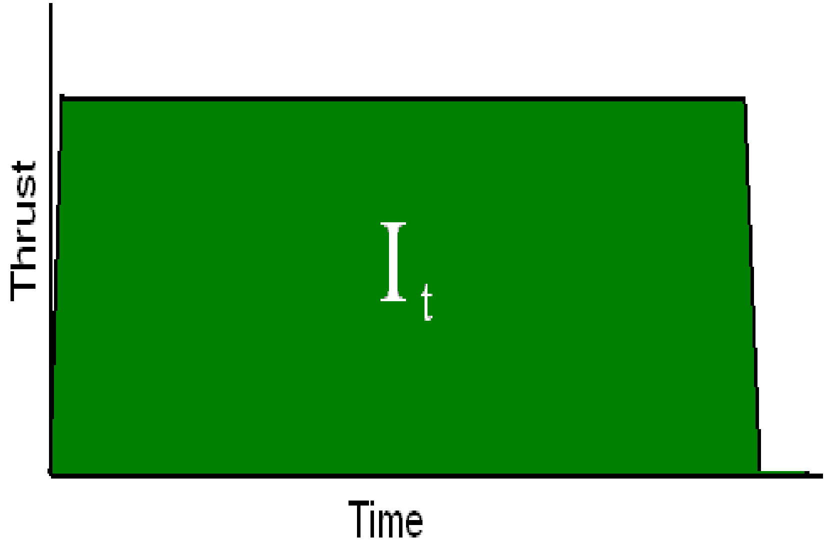


The *total impulse* (I_t) is the thrust integrated overthe run duration (time, t)

$$I_t = \int_0^t F\,dt$$

Assuming constant thrust and negligible transients(i.e., start and shutdown), this becomes

$$I_t = Ft$$



The *specific impulse*, (I_{sp}) is the total impulsegenerated per weight of propellant

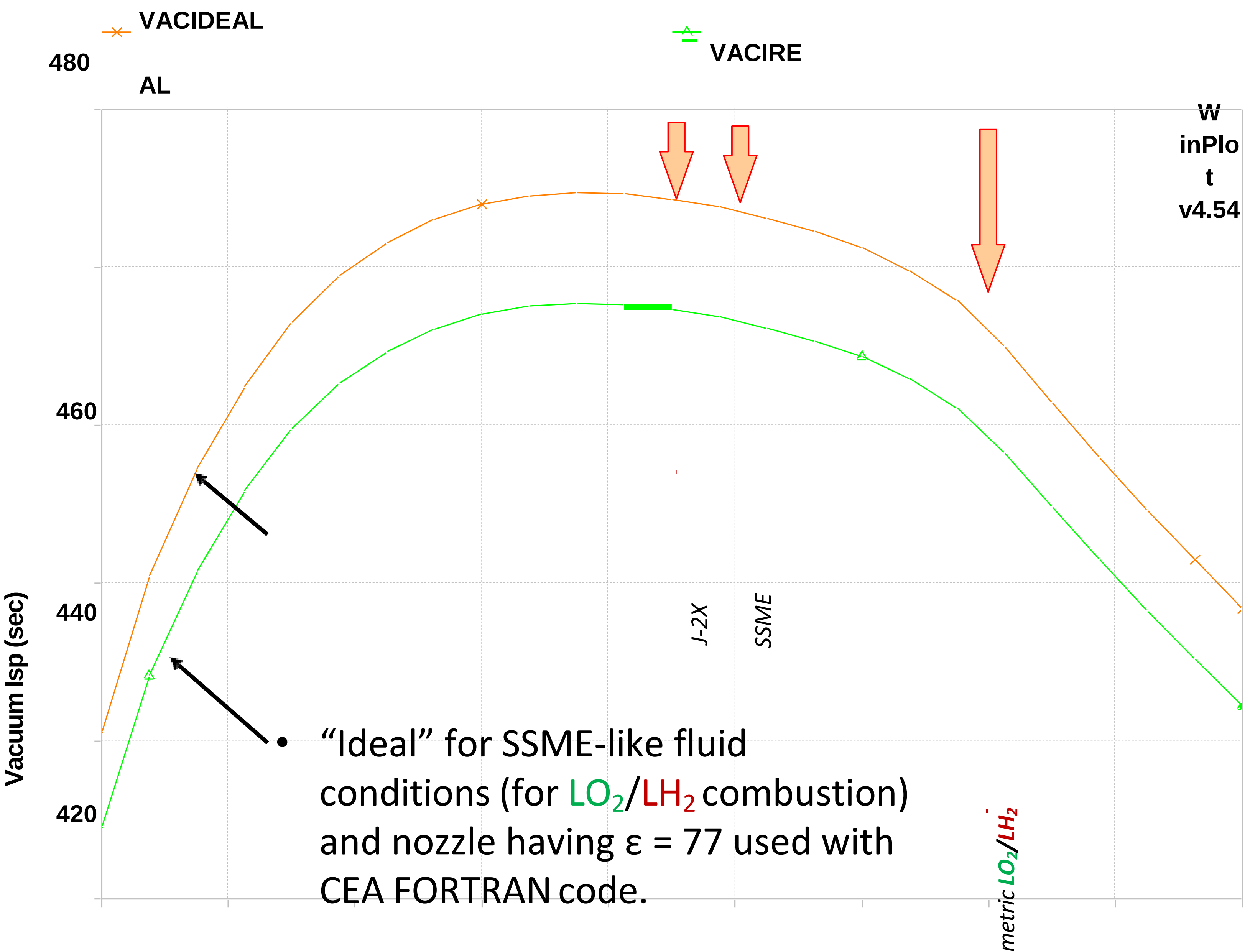
$$I_{sp} = \frac{\int_0^t F\,dt}{g_o \int_0^t \dot{m}\,dt} = \boxed{\frac{F}{\dot{m}}}$$

Rocket propellants are mixed in relative quantities to produce the highest possible system I_{sp}. This ratio of propellant consumption is called mixture ratio, MR.

$$MR = \frac{M_o}{M_f}$$

In most cases, MR is selected for maximum energy release per weight of propellant. This can be achieved by mixing the propellants in a stoichiometric reaction in the combustion chamber, where all the propellants are thoroughly combusted. <u>However, a stoichiometric MR</u> <u>does not necessarily provide an</u> <u>optimized I_{sp}.</u>

☞ The SSME uses a MR of ~6 (stoichiometric for LO_2/ LH_2 combustion is 8) to reduce the internal and plume temperatures, but also to allow a small amount of H_2 to remain in the exhaust. The lighter molecule is able to accelerate to a higher velocity and generate higher kinetic energy ($KE = \frac{1}{2} mV^2$) than a H_2O steam exhaust.



- "Ideal" for SSME-like fluid conditions (for LO_2/LH_2 combustion) and nozzle having ε = 77 used with CEA FORTRAN code.

400

380
1 2 3 5 7 8 9 10
4
6

Mixture
Ratio

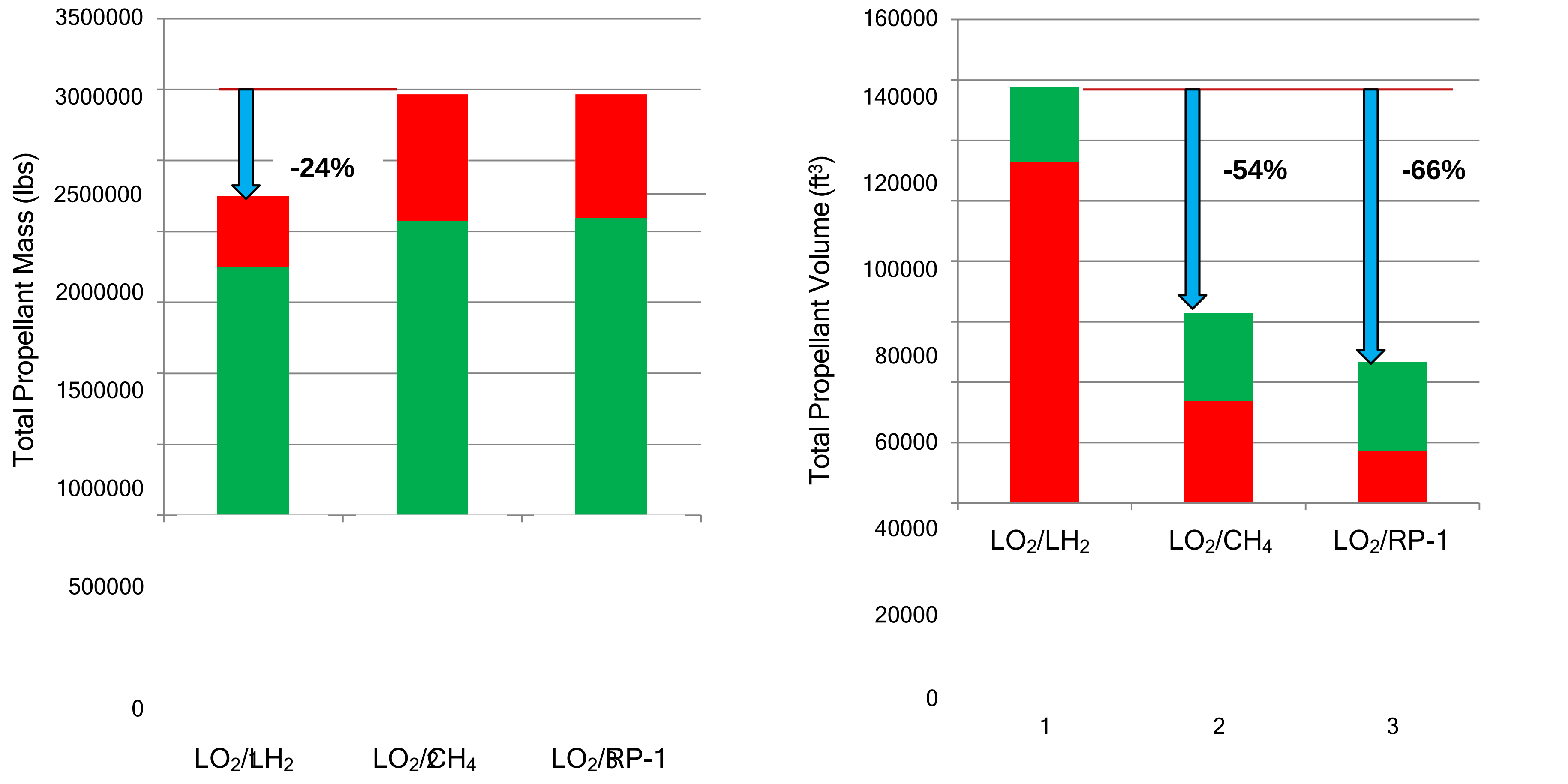
- Liquid bipropellant combinations offer a wide range of performance capabilities.
- Each combination has multiple factors that should be weighed when selecting one for a vehicle.
  - Performance (I_{sp})
  - Density (higher is better)
  - Storability (venting?)
  - Ground Ops (hazards?)
  - Etc.
- One of the more critical trades is that of performance versus density.
- LO_2/LH_2 offers the highest I_{sp} performance, but at the cost of poor density (thus increasing tank size).
- Trading I_{sp} versus density is sometimes referred to as comparing “bulk impulse” or “density impulse”.

- As an example, the densities and I_{sp} performance of the following propellant combinations will be compared.

	Density (g/ml)	Density (lb/ft^3)
Hydrogen	0.07	4.4
Methane	0.42	26.4
RP-1	0.81	50.6
Oxygen	1.14	71.2

P_c = 300 psia expanded to 14.7 psia	MR (O/F)	I_{sp} (sec)
LO_2/LH_2	3.5	347 (1)
LO_2/CH_4	2.33	263 (2)
LO_2/RP-1	2.4	263 (2)

(1) SC *(2) FC*



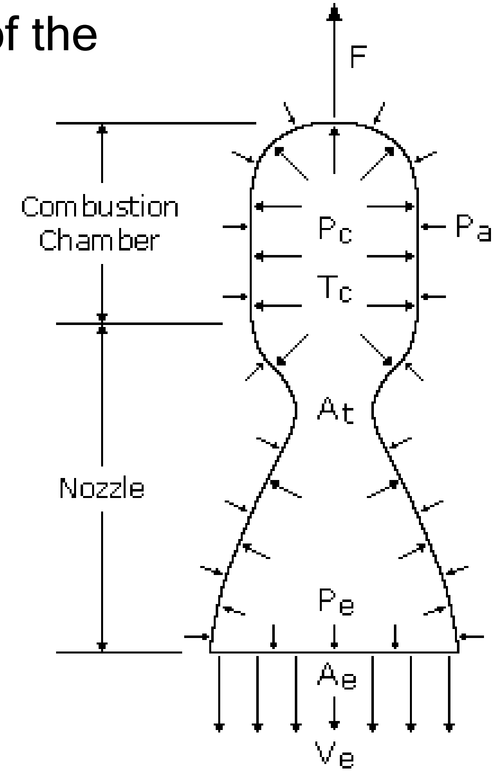
- For an impulse requirement similar to the 3 SSME's used on the Shuttle (~1.5 Mlbf for 520 seconds), the required propellant masses are calculated.

- LO_2/LH_2 requires 24% less propellant mass than the others.
- However…

- When the propellant mass is compared against the tank volume,there is a significant disparity from the low hydrogen density that can adversely impact the size (and totalweight) of the vehicle.
- Lesson: I_{sp} *isn't everything – especially with boost stages.*

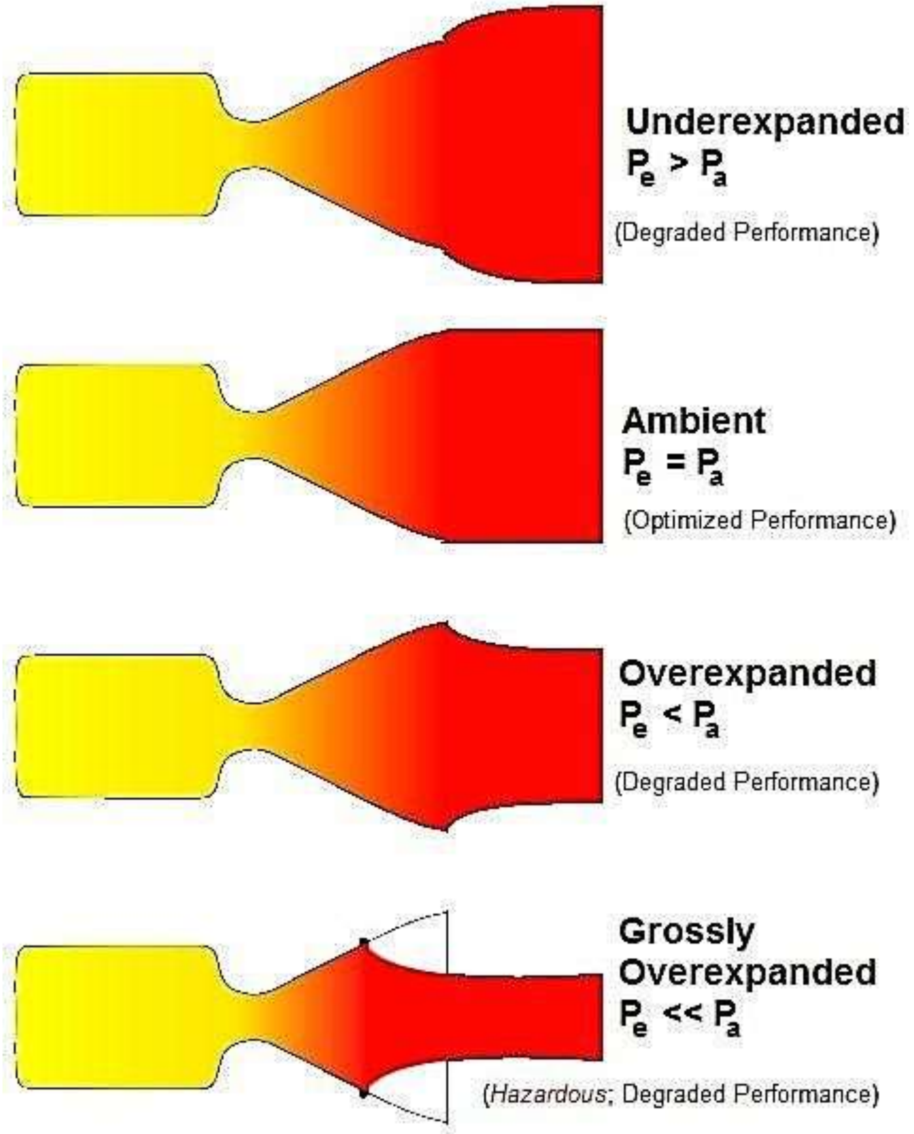
The parameter that determines exit velocity and pressure of the exhaust gases is *area ratio* or *nozzle expansion ratio*, MR

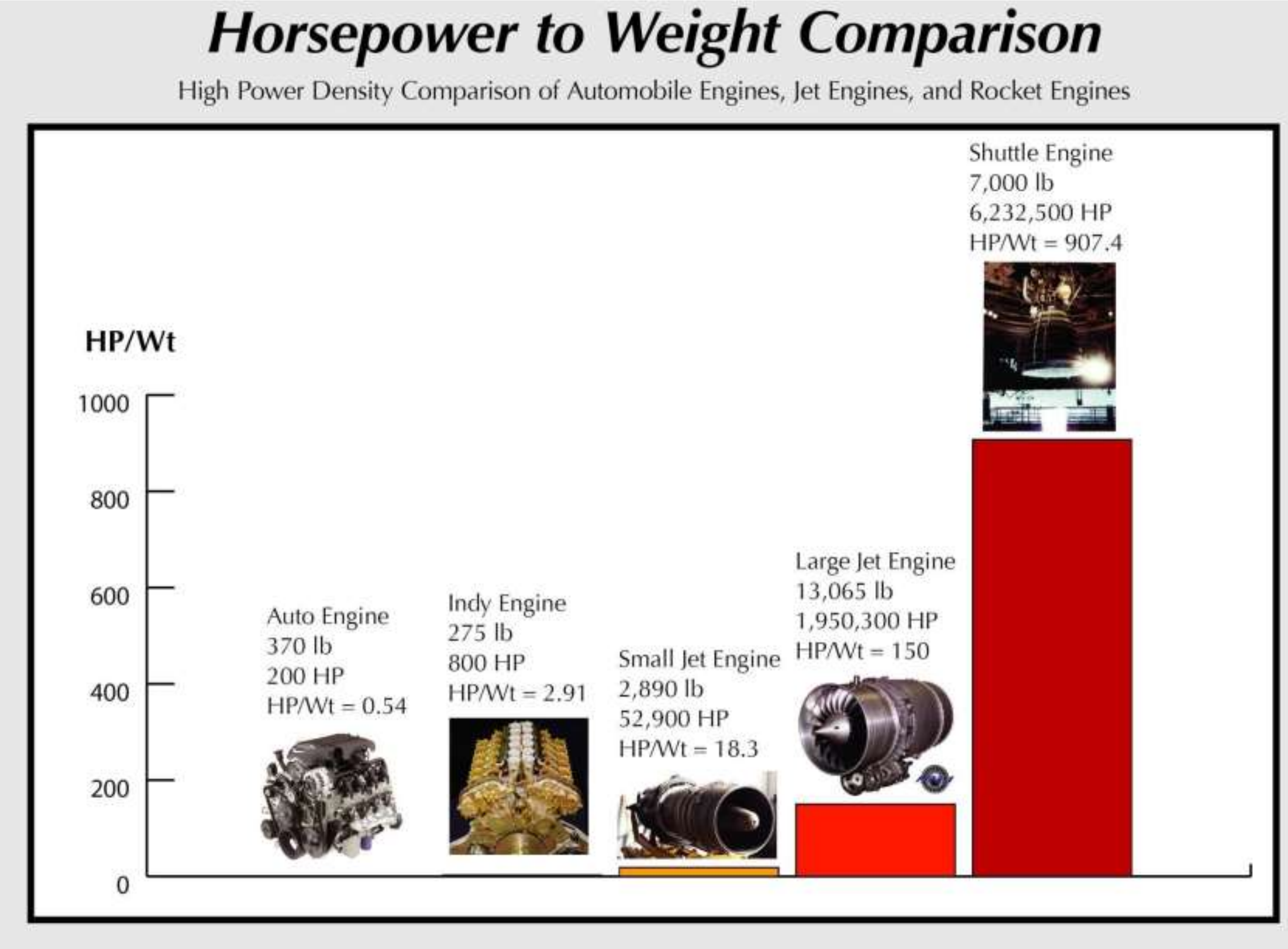
$$\varepsilon = \frac{A_e}{A_t}$$



- As ε increases, the exit velocity increases and the exit pressure decreases (higher I_{sp}).
- When possible, ε is selected so that $P_e = P_a$ and the engine operates at optimum thrust.
- For in-space propulsion (i.e., J-2X), the ε is made as large as the weight requirements or volume limitations permit.
- If $P_e > P_a$, then the nozzle is identified as underexpanded and will not provide optimal performance as the plume will continue to expand after exiting the nozzle.
- If $P_e < P_a$, then the nozzle is identified as overexpanded and will not provide optimal performance as the exit shock will migrate inside the nozzle. This can be hazardous from thrust imbalances and damage to the nozzle.

- Criteria to be considered for selecting a design ε can include the following:
  - ε provides the optimal integrated performance over the engine operating period. Trajectory analysis is used to determine the altitudes (and P_a) that the engine will operate, which can be used to provide an integrated I_{sp} based on ε.
  - ε is optimized to provide the maximum performance during a critical time of the engine operating period. Example: the ε for SSME is optimized at the altitude where the SRBs are staged to provide a needed performance boost at that critical time



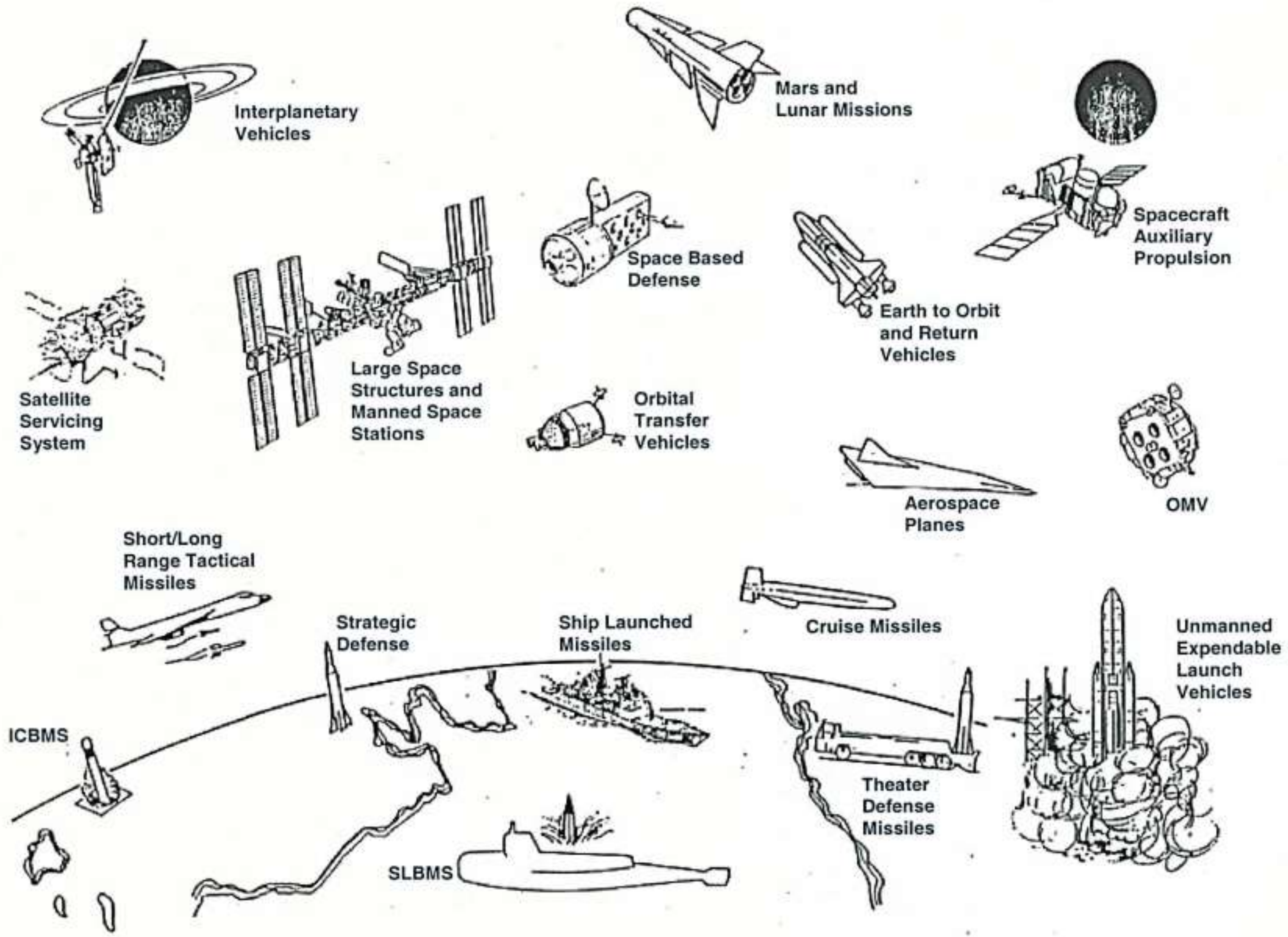

Horsepower to Weight Comparison
High Power Density Comparison of Automobile Engines, Jet Engines, and Rocket Engines
HP/Wt
1000
800
600
400
200
0
Auto Engine
370 lb
200 HP
HP/Wt = 0.54
Indy Engine
275 lb
800 HP
HP/Wt = 2.91
Small Jet Engine
2,890 lb
52,900 HP
HP/Wt = 18.3
Large Jet Engine
13,065 lb
1,950,300 HP
HP/Wt = 150
Shuttle Engine
7,000 lb
6,232,500 HP
HP/Wt = 907.4

- Problem: Calculate the mass ratio needed to escape Earth's gravity starting from rest, given that the escape velocity from Earth is about

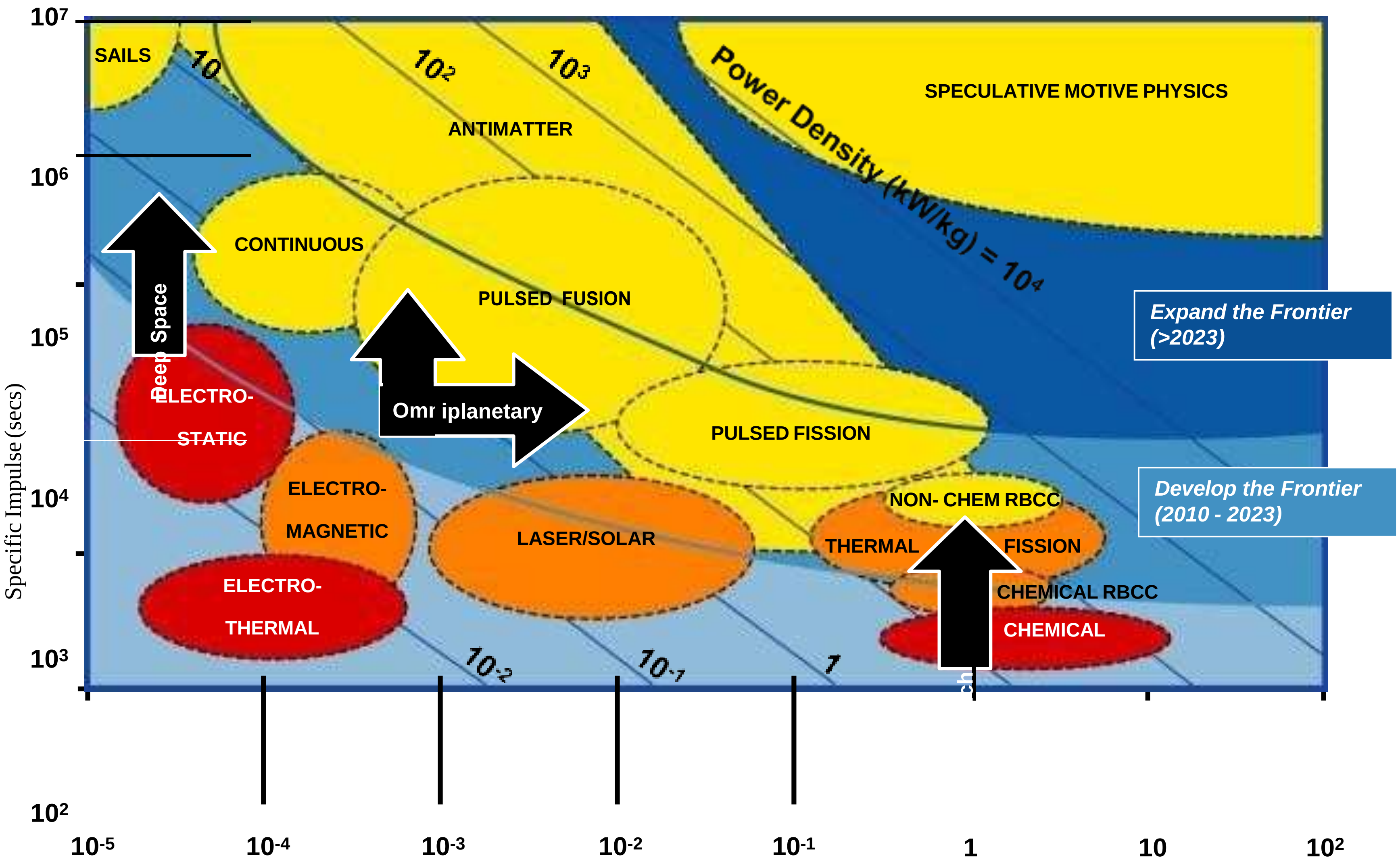
  11.2 10^3 m/s and assuming an exhaust velocity V_e= 2.5x10^3 m/s.

Ans : 1/88

Rocket Propulsion Applications



In-Space Propulsion Performance



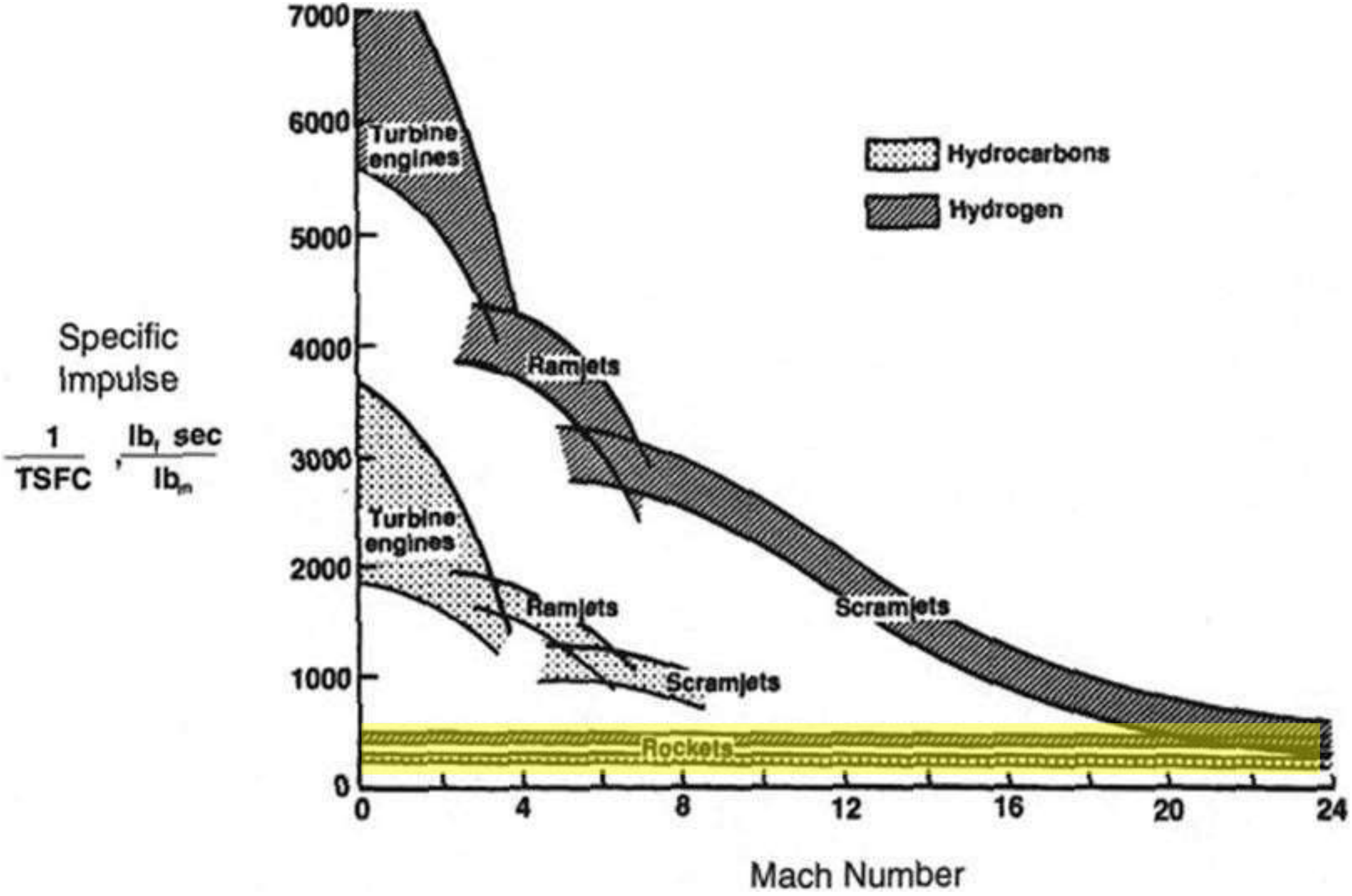
Vehicle Acceleration or T/W Ratio (g’s)

Unproven Technology (TRL 1-3)

Demonstrated Technology (TRL 4-6)

Operational Systems (TRL 7-9)

Jet Propulsion Options



BALLISTICS

- Knowledge of physical forces acting on a projectile and missile is called Ballistics

OR

- Ballistics is the simply science of motion of a projectile

Types

- Exterior or External Ballistics
- Interior or Internal Ballistics
- Terminal or Wound Ballistics

✓ **Exterior Ballistics** deals with the study of motion of a projectile after it leaves the barrel of a firearm.

✓ **Terminal Ballistics** is the study of effect of impact of a projectile on the target leading to wound formation (Also called Wound Ballistics).

INTERIOR / INTERNAL BALLISTICS

Interior Ballistics is the study of physio-chemical phenomenon within thefirearm from the movement of the detonation of primer to the time the projectile leaves the barrel.

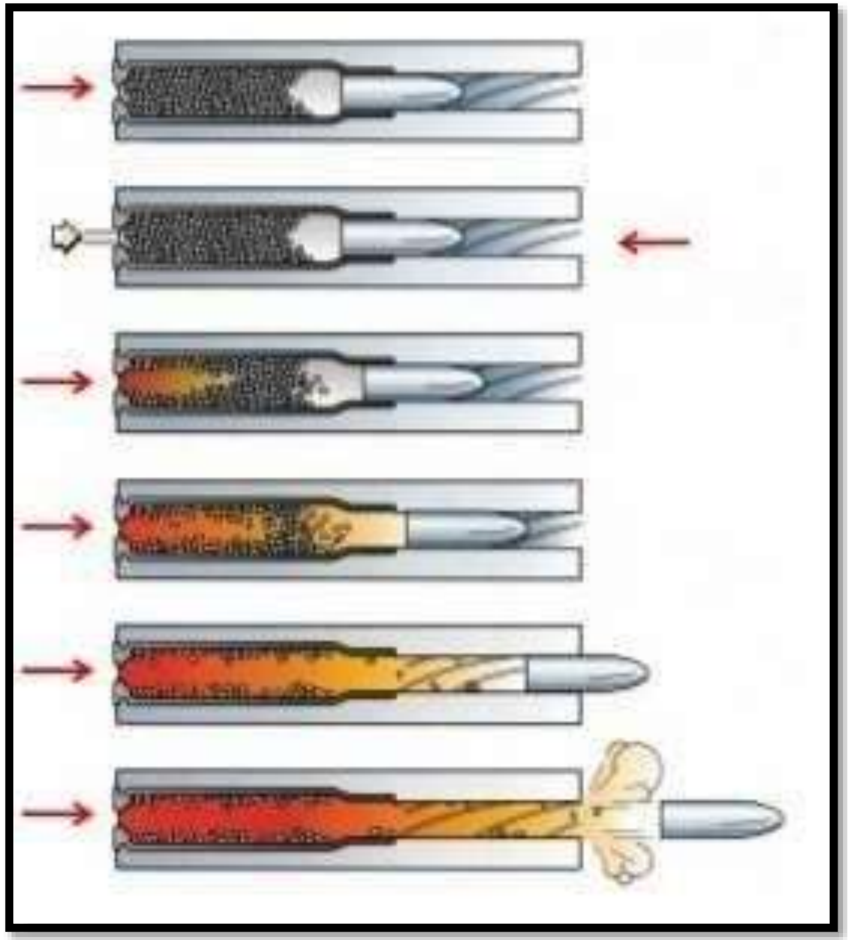
OR

The study of the process originally accelerating the principles is called Interior Ballistics, for example the passage of a bullet through the barrel ofa rifle.

Internal Ballistics

- Internal Ballistics, a subfield of ballistics is the studyof projectile's behavior from the time its propellant's ignites is initiated until it exits the gunbarrel.
- The study of internal ballistics is important to designers and users of firearms of all types.


EXTERIOR BALLISTICS?
I'M OUT OF THE GUN ACCORDING TO PLAN -NOW WHAT?
Initial Speed and Direction
INTERIOR BALLISTICS
Bore Erosion
Propelling Charge and Burning Rate
Powder Temperature
Density and Manner of Loading



The three main factors are-

- Lock Time
- Ignition Time
- Barrel Time

Lock Time

- Lock time is the time interval between release of the sear and the impact of the striker on the percussion cap.
- A short time internal is advantageous in rapidfire.
- The lock time can be measured in a number of ways, in one such system the use of linear Motion sensors and an oscilloscope is made.

Ignition Time

- Ignition time is the duration or interval between the striking of the firing pin to blow the ignition of the first grain of powder.
- The ignition, under the normal conditions, takes place at an interval of about 0.002 seconds.

Barrel Time

- Barrel time is the time interval from the pressing of the trigger to the exit of the bullet from the muzzle end.
- In case of most of the weapons Lock time + Ignition time + barrel time varies from 0.003 to 0.007 seconds.

INTERIOR /INTERNAL BALLISTICS

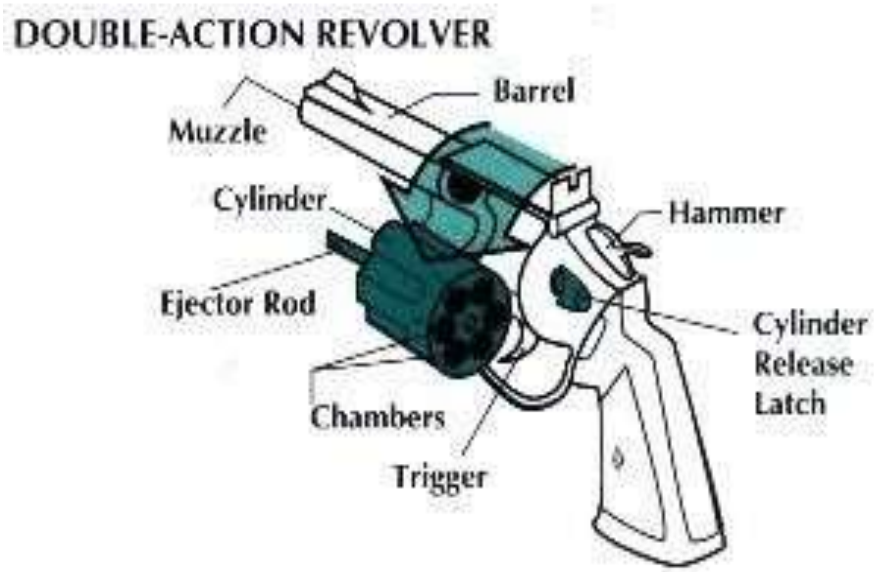
It includes……

I. Structure of Firearm.
II. Design of Ammunition.
III. Chain/Sequence of events.

i. Structure of firearm

Every firearm is basically divided into three parts.....

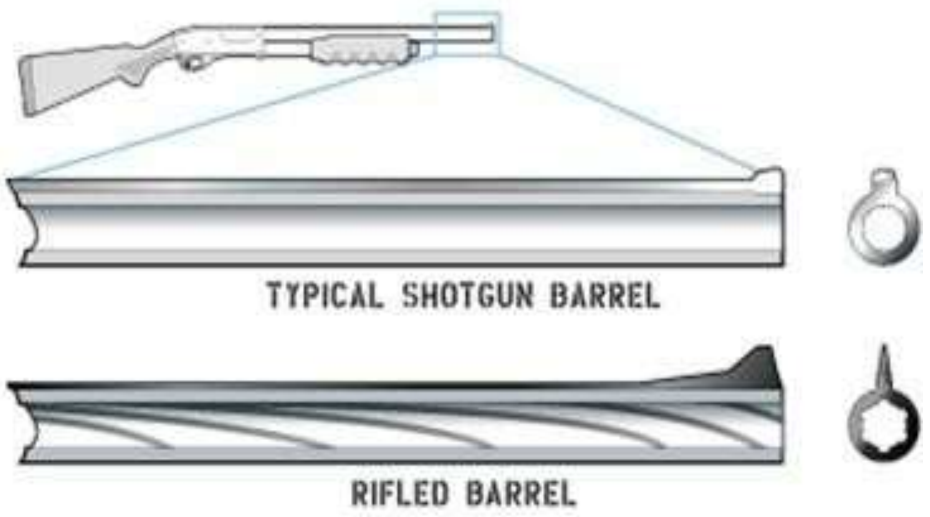
- Grip portion
- Action portion having thr trigger
- Front portion called the barrel



Structure of firearm

- Barrel….. Steel tube for jetting of the projectile
  - Breach end
  - Muzzle end

- Bore/Callibre….. Internal diameter of the barrel.
  - Smooth
  - Choked
  - Non-Choked
- Rifled
  - Short barrel
  - Long barrel



Rifling

- Itconsists of grooves or cuts formed in a spiral nature lengthwise down the barrel of a firearm.

- Because bullets are oblong objects, they must spin in their fight like a thrown football, to be accurate

Lands….. Raised areas between two grooves

Grooves….. Depressed areas

- Rifling pattern of eight grooves will also have eight lands.

II. AMMUNITION DESIGN

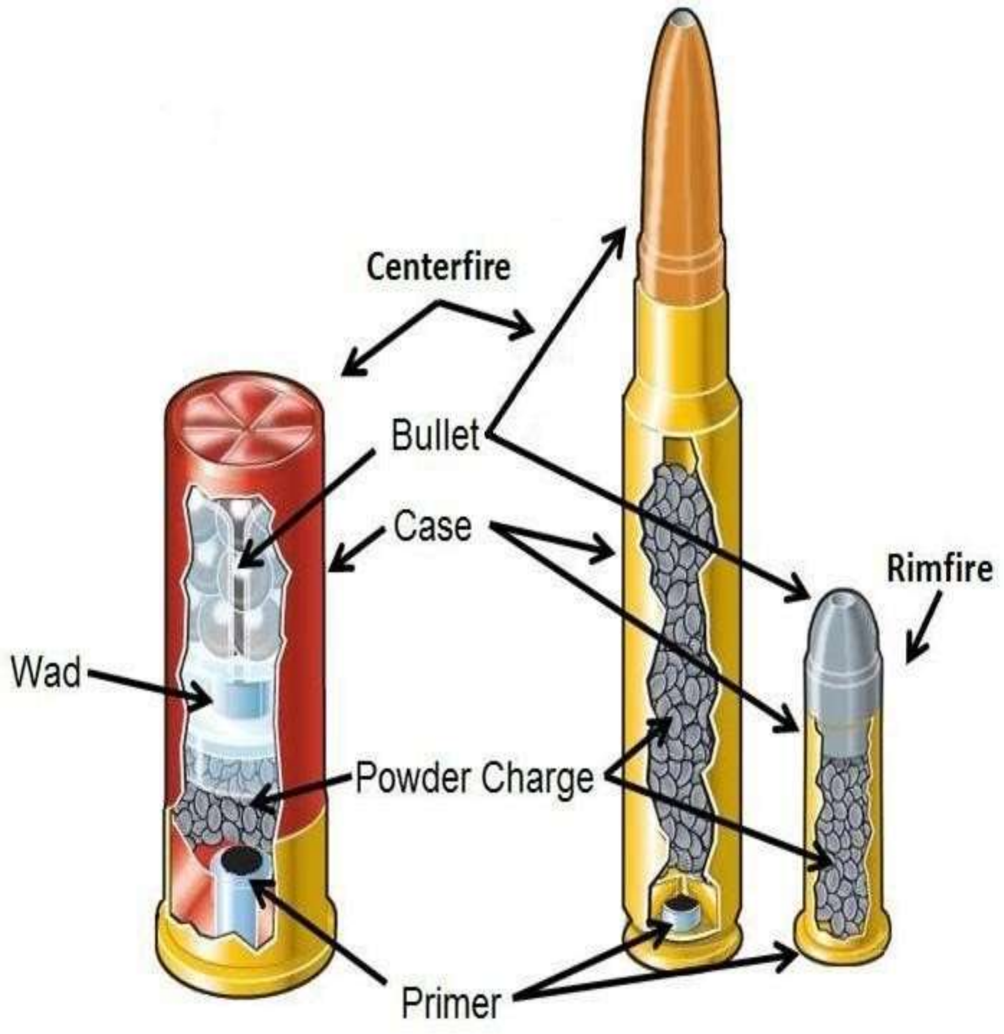
- Cartridge Case
- Primer

Powder Charge

I. Black
II. Smokeless

- Plastic wad
- Short charge

I. Bullet
II. Pellets



AMMUNITION CASES

- **Cartridge cases** are made up of plastic and card board.
- **Bullet cases** are made up of brass(70% copper & 30% Zinc, some have nickel coating).
- **Primer cases** are of similar composition (Copper & Zinc).
- **Bullet cores** are most often lead and antimony with a very few have ferrous alloy core.
- **Bullet jackets** are usually brass, 90% copper with 10% Zinc but some are a ferrous alloy & some are aluminum. Some bullet coating may also contain nickel.

Types of Ammunition

Primer

- The major primer elements are lead, barium or Antimony. Usually all three are present.
- Less common elements include aluminium, Sulphur, Tin, Calcium, Potassium, Chlorine and Silicon.
- Primer elements may be easier to detect in residue because they don't get as hot as powder and compounds may be detectable.

Powder Charge

- Modern gun powder or smokeless powder can contain upto 23 organic compounds.
- Nitrocellulose is virtually always present along with other compounds containing nitrates or nitrogen.

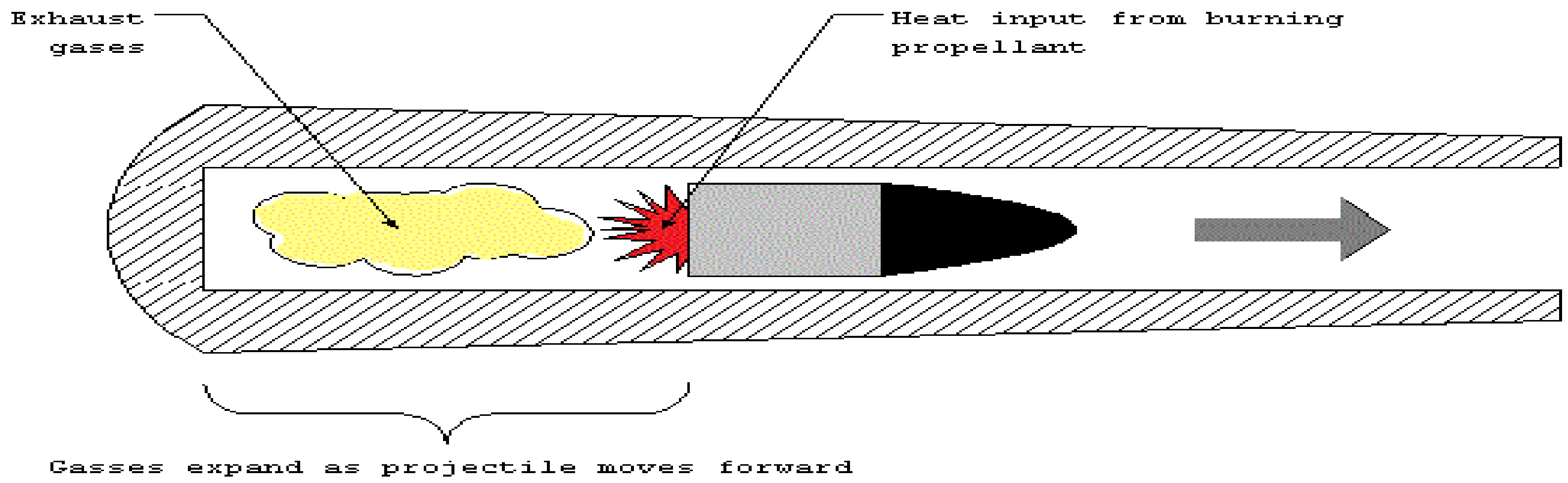
I. **Single base…..** The basic ingredient is nitrocellulose.
II. **Double base…..** When there is added 40% Nitroglycerine to nitrocellulose.

Powder Charge

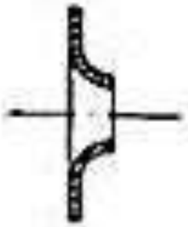
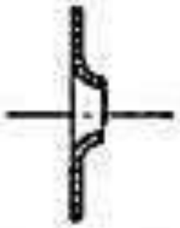
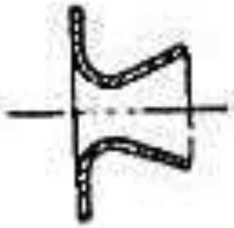
Chain Sequence of Events

Hammer Strikes → Primer Ignites → Powder charge burns → Temperature increases → Gases produced

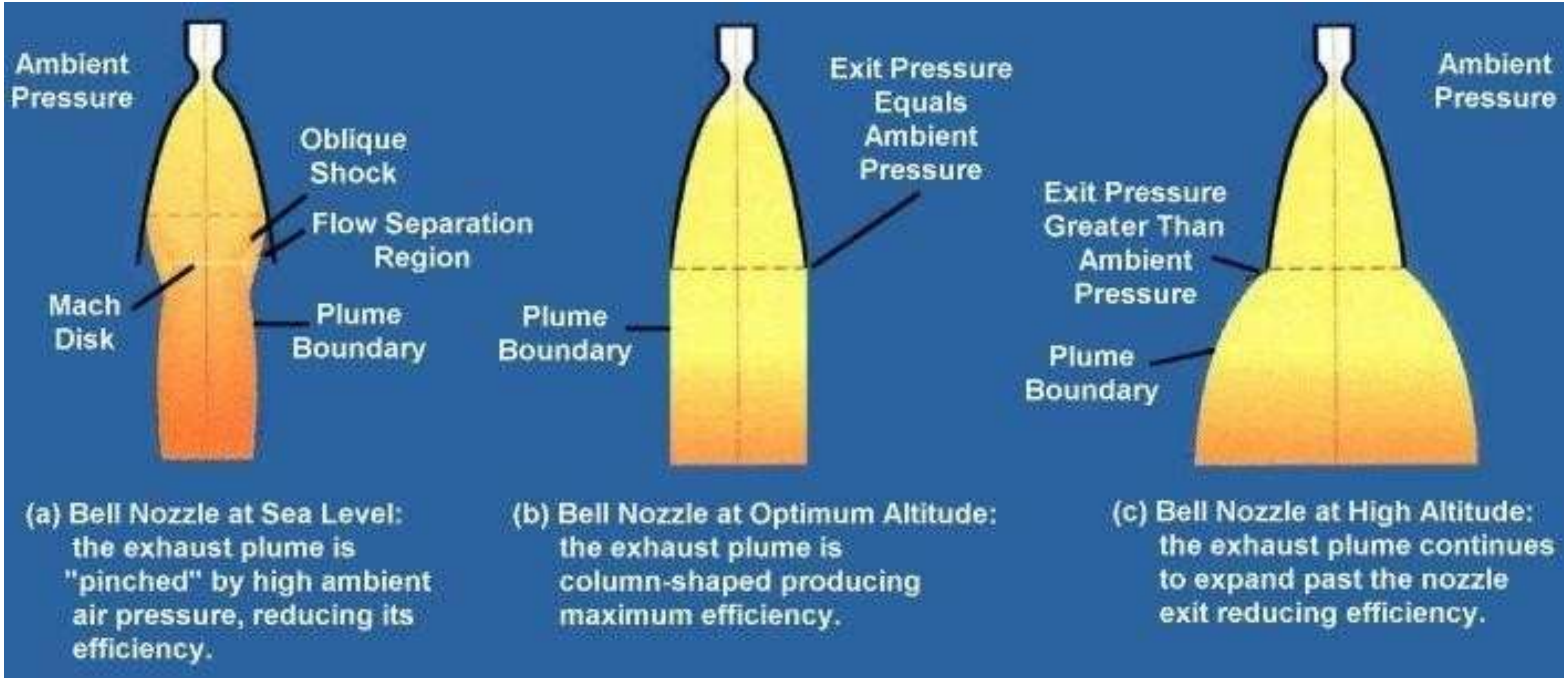
Chamber pressure increased → Bullet charge moves → Exit of bullet → Chamber pressure = Zero



Nozzle Types

	Subsonic	Sonic	Supersonic
Throat velocity	$v_t < a_t$	$v_t = a_t$	$v_t = a_t$
Exit velocity	$v_2 < a_2$	$v_2 = v_t$	$v_2 > v_t$
Mach number	$M_2 < 1$	$M_2 = M_t = 1.0$	$M_2 > 1$
Pressure ratio	$\frac{p_1}{p_2} < \left(\frac{k+1}{2}\right)^{k/(k-1)}$	$\frac{p_1}{p_2} = \frac{p_1}{p_t} = \left(\frac{k+1}{2}\right)^{k/(k-1)}$	$\frac{p_1}{p_2} > \left(\frac{k+1}{2}\right)^{k/(k-1)}$
Shape			

Rocket Nozzle Classification



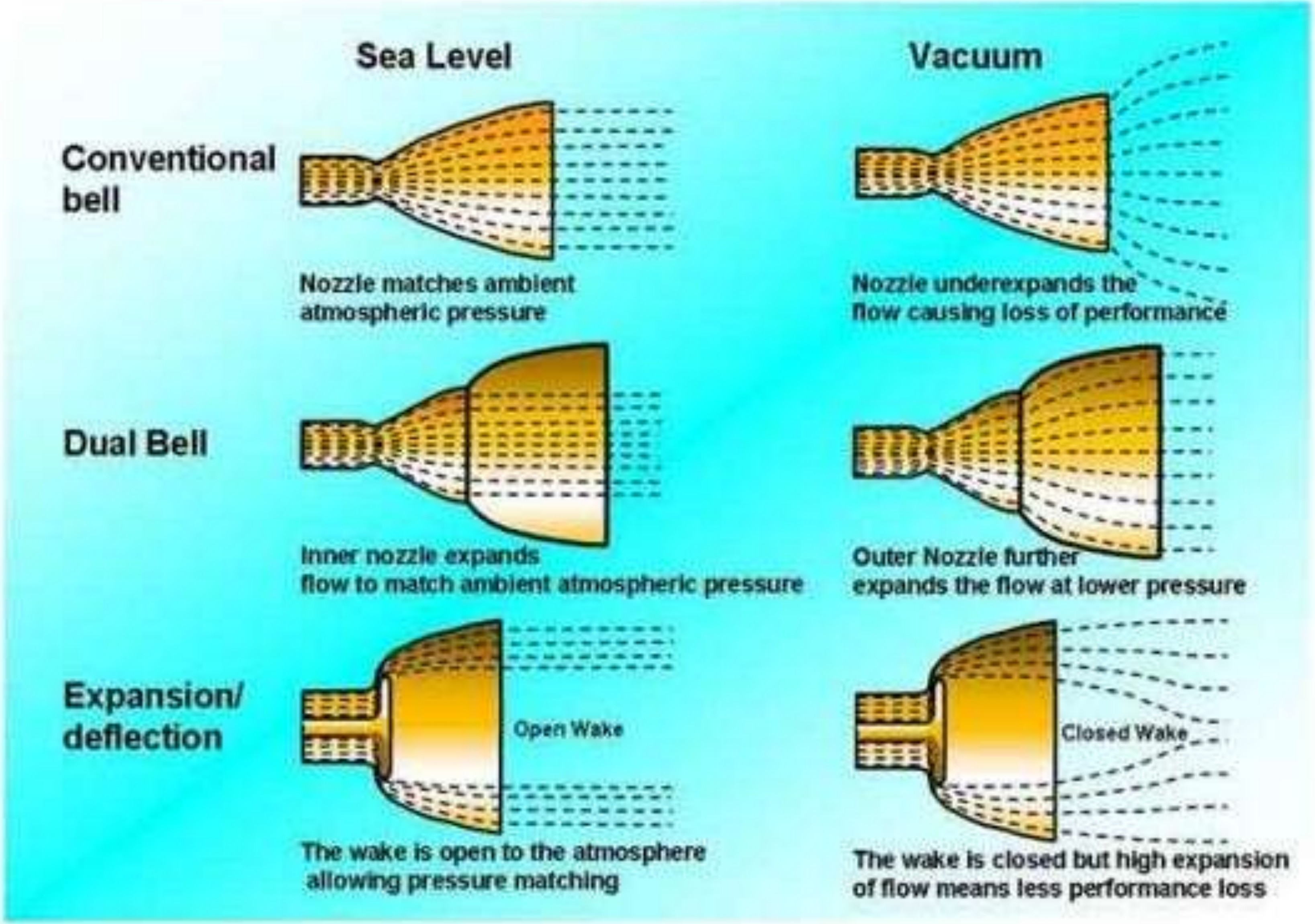
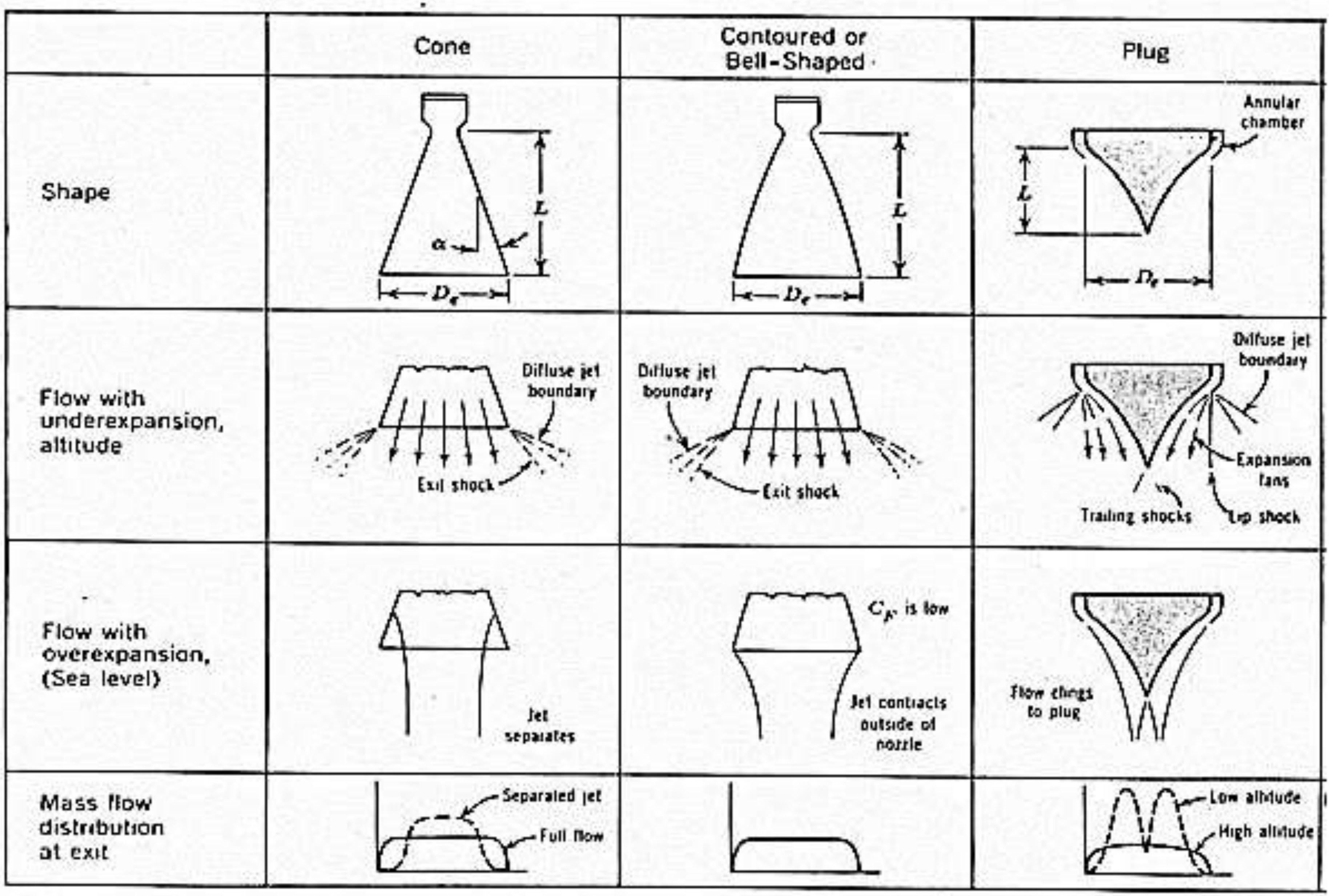
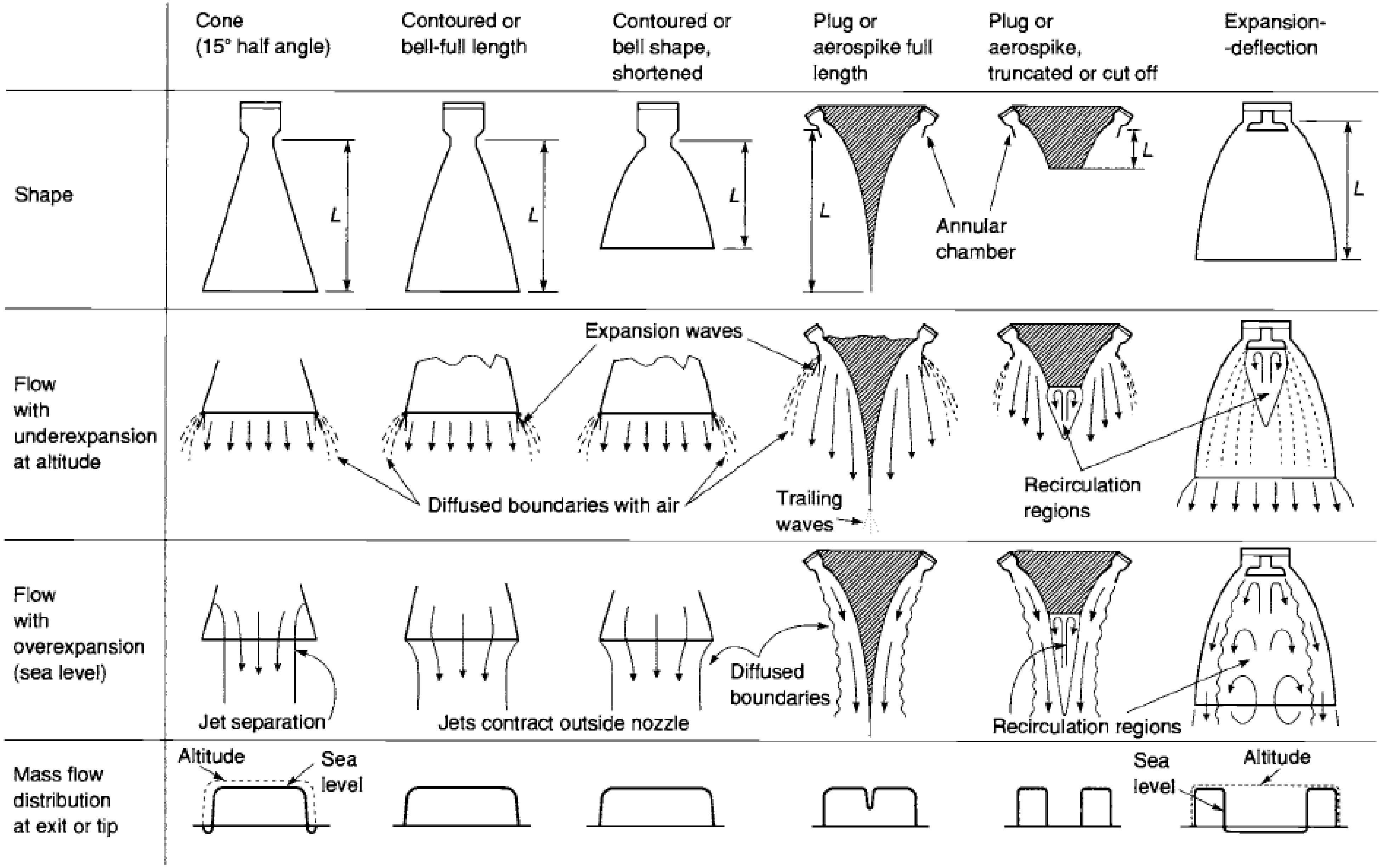


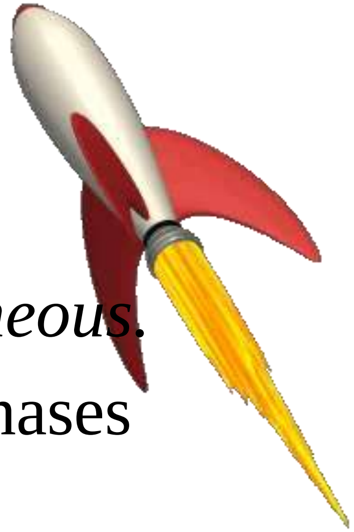
Figure 1. Nozzle types



Cone
Contoured or Bell-Shaped
Plug
Shape
Annular chamber
L
α
D_e
Flow with underexpansion, altitude
Diffuse jet boundary
Exit shock
Expansion fans
Trailing shocks
Lip shock
Flow with overexpansion, (Sea level)
Jet separates
C_F is low
Jet contracts outside of nozzle
Flow clings to plug
Mass flow distribution at exit
Separated jet
Full flow
Low altitude
High altitude


Cone
(15° half angle)
Contoured or
bell-full length
Contoured or
bell shape,
shortened
Plug or
aerospike full
length
Plug or
aerospike,
truncated or cut off
Expansion-
-deflection
Shape
L
Annular
chamber
Flow
with
underexpansion
at altitude
Expansion waves
Diffused boundaries with air
Trailing
waves
Recirculation
regions
Flow
with
overexpansion
(sea level)
Jet separation
Jets contract outside nozzle
Diffused
boundaries
Recirculation regions
Mass flow
distribution
at exit or tip
Altitude
Sea
level

Rocket performance considerations



1. The working substance (or chemical reaction products) is *homogeneous*.

2. All the species of the working fluid are *gaseous*. Any condensed phases (liquid or solid) add a negligible amount to the total mass.

3. The working substance obeys the *perfect gas law*.

4. There is no *heat transfer* across the rocket walls; therefore, the flow is adiabatic.

5. There is no appreciable *friction* and all *boundary layer* effects are neglected.

Contd..

6. There are no *shock waves* or *discontinuities* in the nozzle flow.

7. The *propellant flow* is *steady* and *constant*. The expansion of the working fluid is uniform and steady, without vibration. Transient effects (i.e., start up and shut down) are of very short duration and may be neglected.

8. All exhaust gases leaving the rocket have an *axially directed velocity*.

9. The gas velocity, pressure, temperature, and density are all uniform across any section normal to the nozzle axis.

10. *Chemical equilibrium* is established within the rocket chamber and the gas composition does not change in the nozzle (frozen flow).

11. Stored propellants are at room temperature. Cryogenic propellants are at their boiling points.

- Phase I, Ignition time lag: the period from the moment the igniter receives a signal until the first bit of grain surface burns.
- Phase II, Flame-spreading interval: the time from first ignition of the grain surface until the complete grain burning area has been ignited.
- Phase III, Chamber-filling interval: the time for completing the chamber filling process and for reaching equilibrium chamber pressure and flow

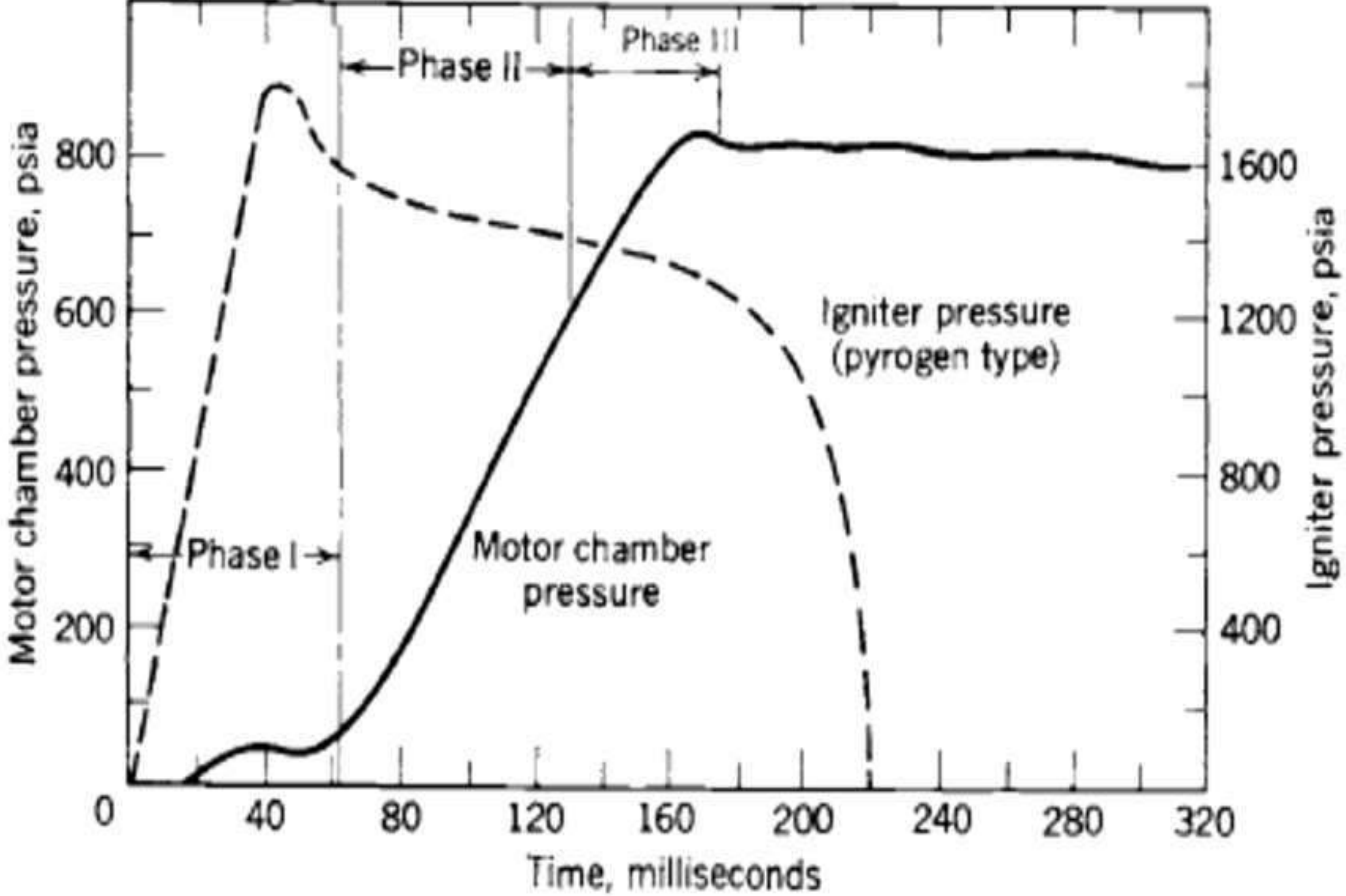
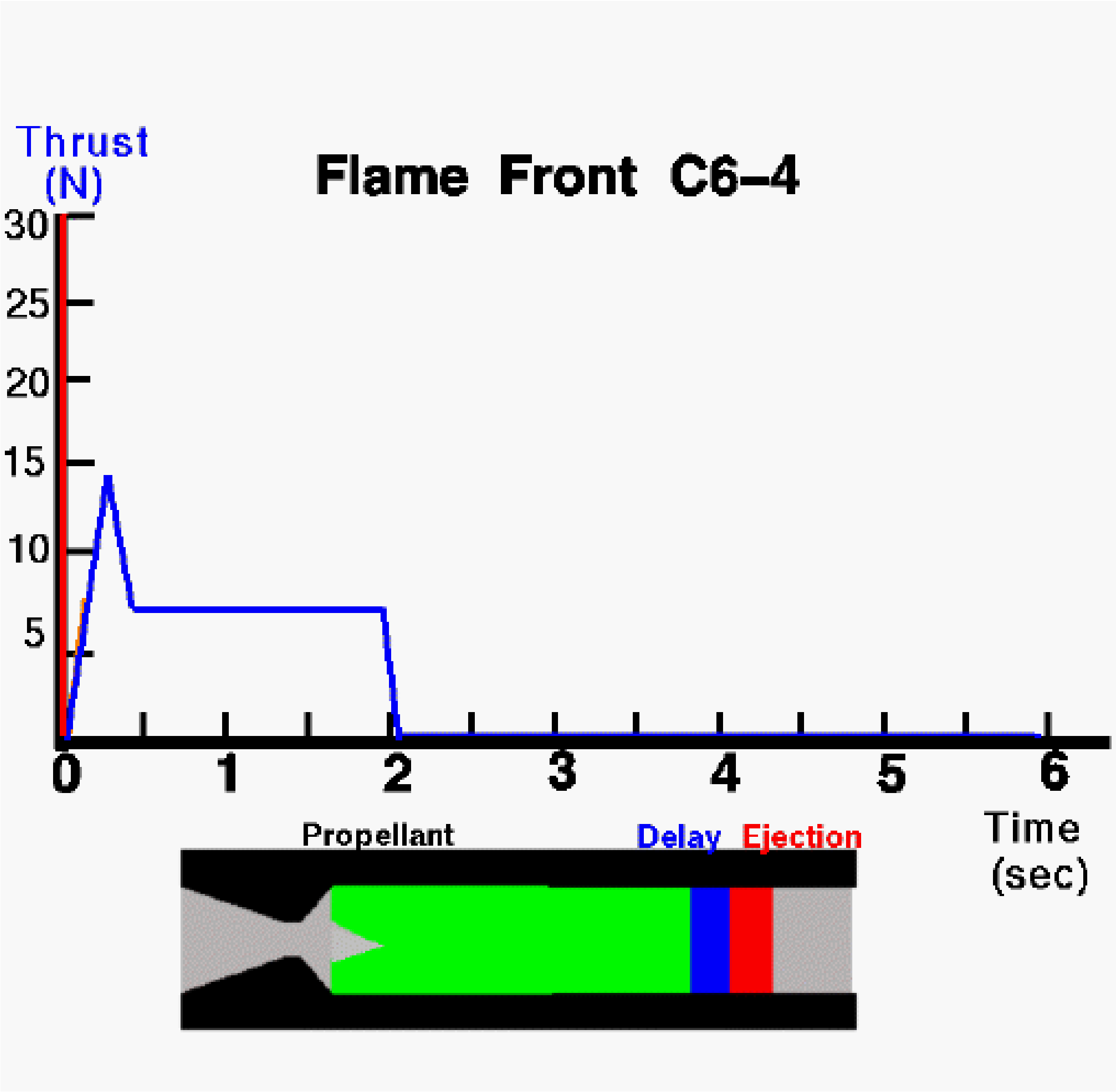
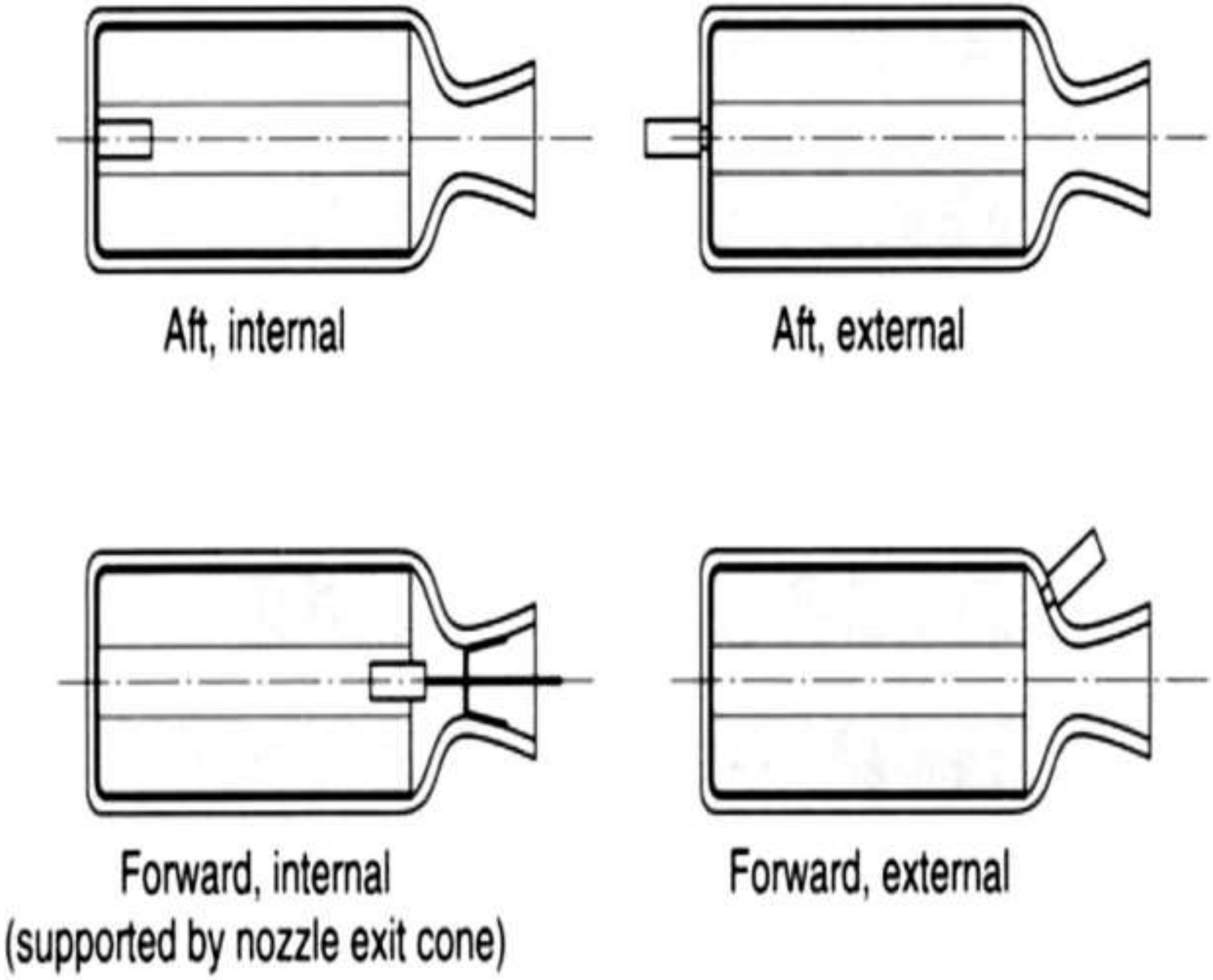


Fig. Typical ignition pressure transient portion of motor chamber pressure time trace with igniter pressure trace and ignition process phases


Thrust
(N)
Flame Front C6–4
30
25
20
15
10
5
0
1
2
3
4
5
6
Time
(sec)
Propellant
Delay
Ejection

Types of igniters



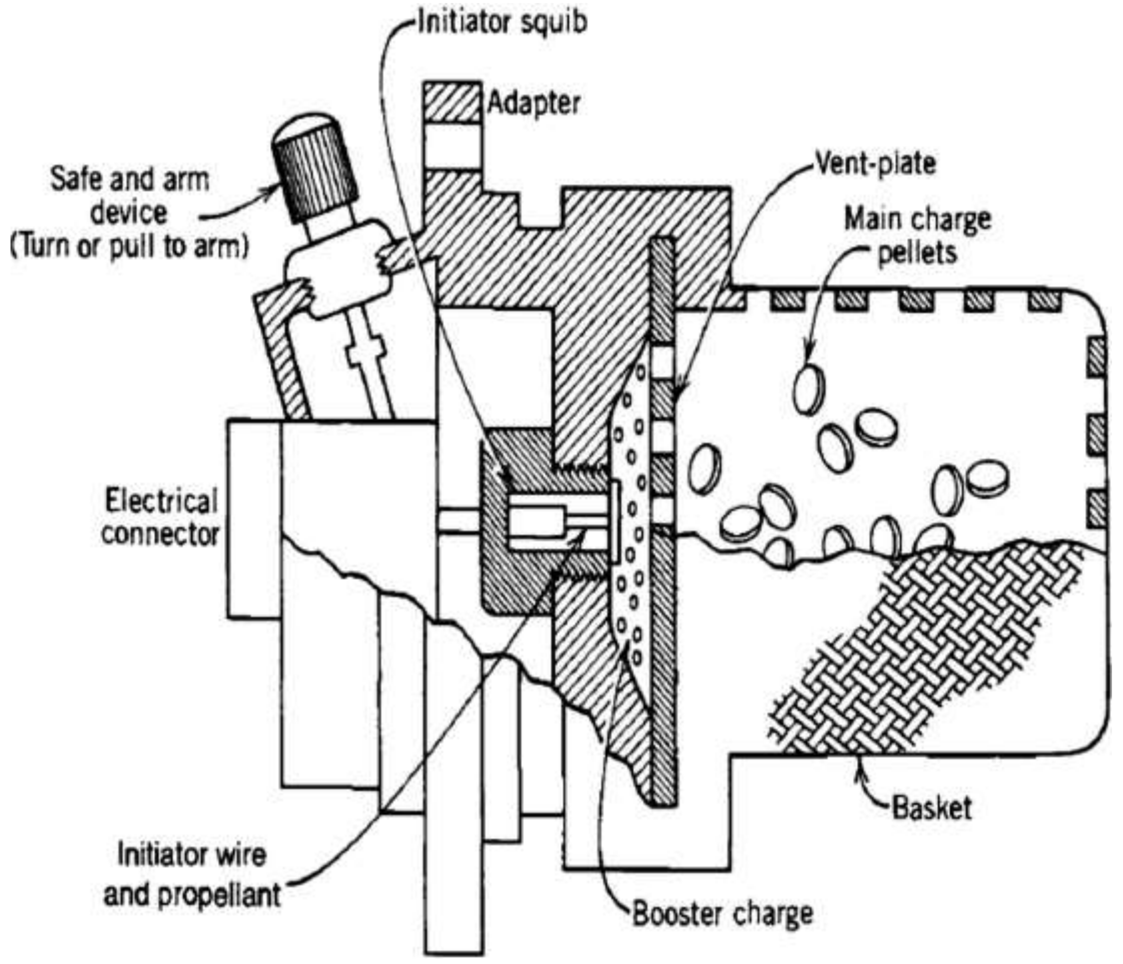
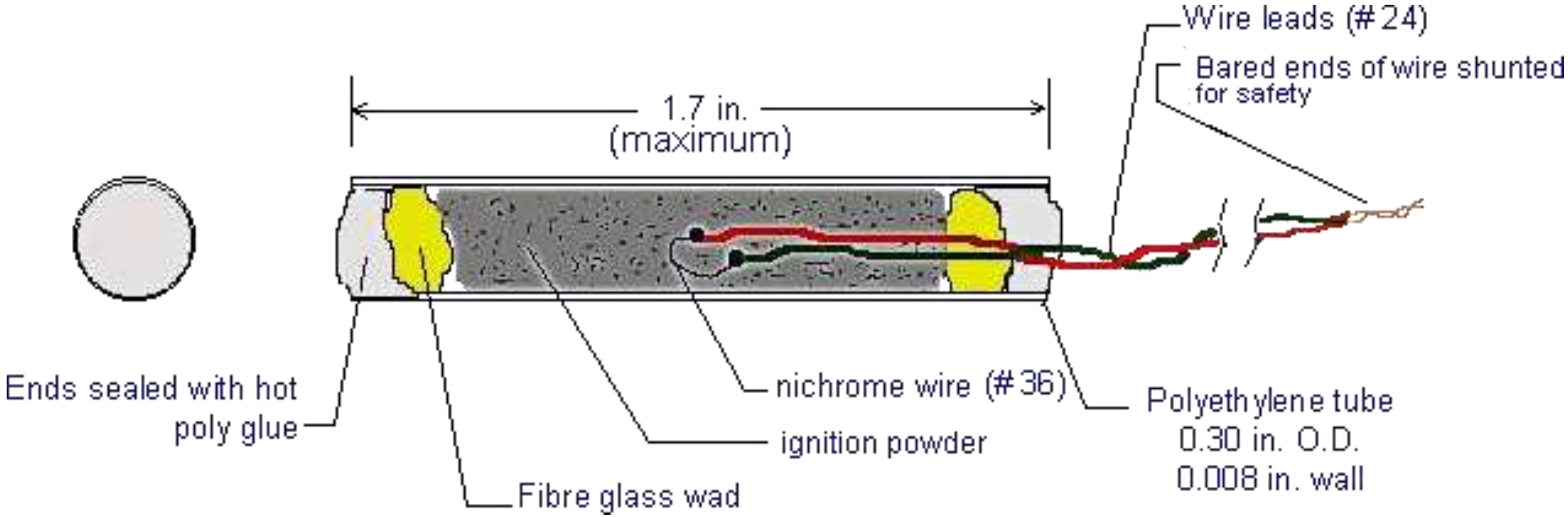


Fig. Typical pyrotechnic igniter with three different propellant charges that ignite in sequence

Pyrotechnic Igniters

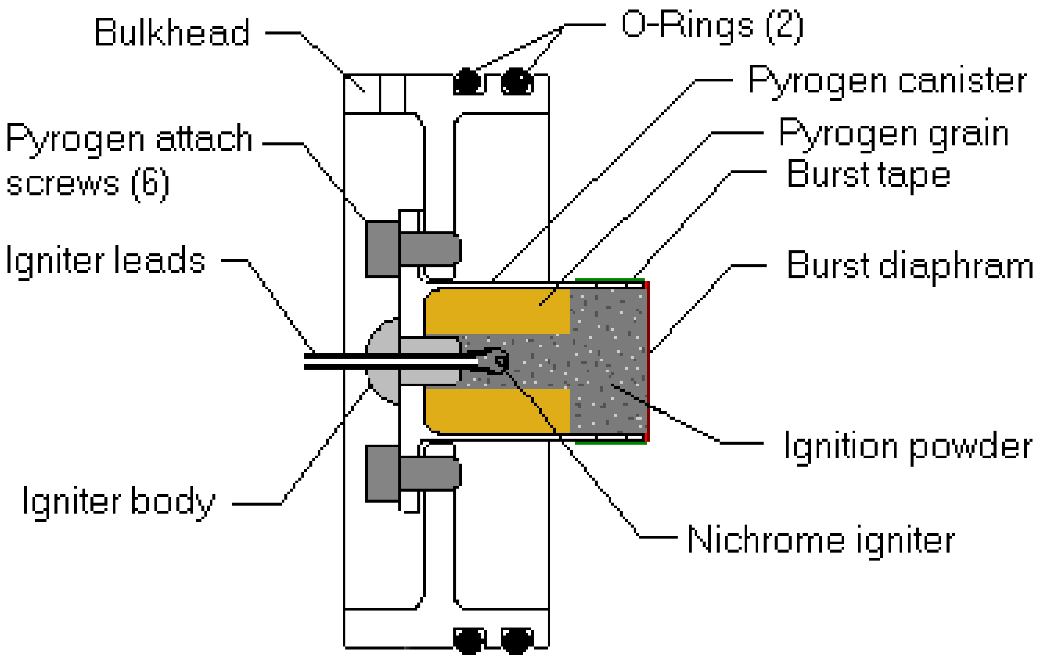
- In industrial practice, pyrotechnic igniters are defined as igniters (other than pyrogen-type igniters as defined further on) using solid explosives or energetic propellant-like chemical formulations (usually small pellets of propellant which give a large burning surface and a short burning time) as the heat-producing material.
- This definition fits a wide variety of designs, known as bag and carbon igniters, powder can, plastic case, pellet basket, perforated tube, combustible case, jellyroll, string, or sheet igniters.

Pyrotechnic Igniter

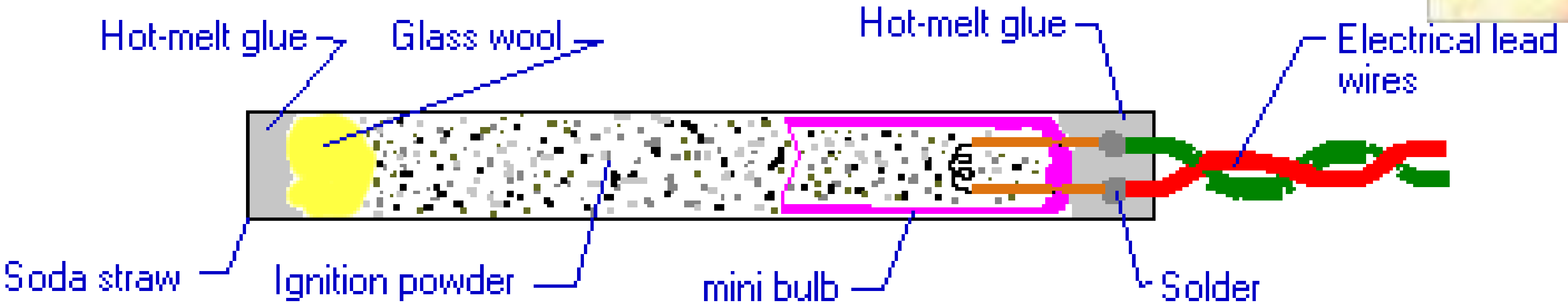


Pyrogen Ignition

- A pyrogen is essentially a small rocket motor mounted at the bulkhead. Nearly instantaneous ignition of the motor grain is assured by the high velocity, particle-laden flame that emanates from the pyrogen. The pyrogen used for the Kappa rocket motor is shown in Figure.



Mini-bulb Igniter



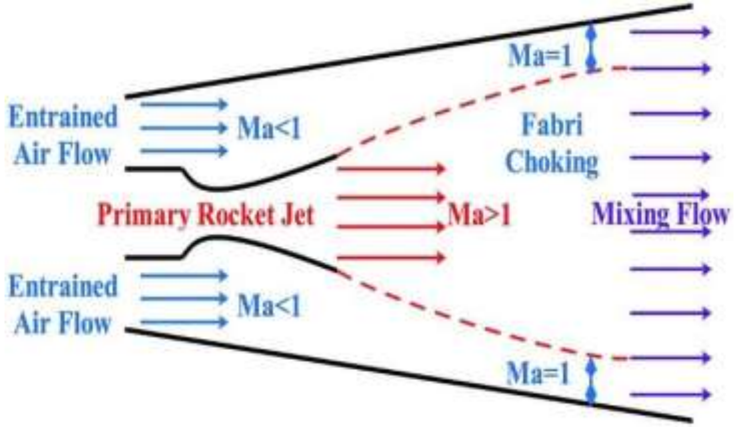
- The igniter described here may be used for either *motor ignition* or for firing a *parachute ejection charge*.
- To make this igniter (shown in Figure), the plastic base of the mini-bulb is first removed (by pulling) and discarded. This exposes the two copper wire leads, which are then scraped clean of oxide, The glass bulb is then carefully broken open. The simplest means is to slowly squeeze the upper half of the bulb in a bench vise. The bulb is first covered with a cloth rag to catch the tiny shards of glass the erupt once the vacuum seal is broken. Safely glasses must be worn during this operation as a redundant safety measure. Care must be taken to prevent damage to the filament bridge wire or to break the lower portion of the bulb.

Ultra-low Current Igniter

An igniter that requires very low electrical power, requiring only 20 mA at 1.2V (= 25mW) to fire. As such, this design is exceptionally reliable and especially useful in cold weather operation, which greatly reduces a typical battery's available power. This igniter may be used for either motor ignition or for firing a parachute ejection charge.
The "Ultra-low Current Igniter" was developed for EARco (Experimental Aerospace Research) by Ken Tucker to increase the safety of Rocketry.

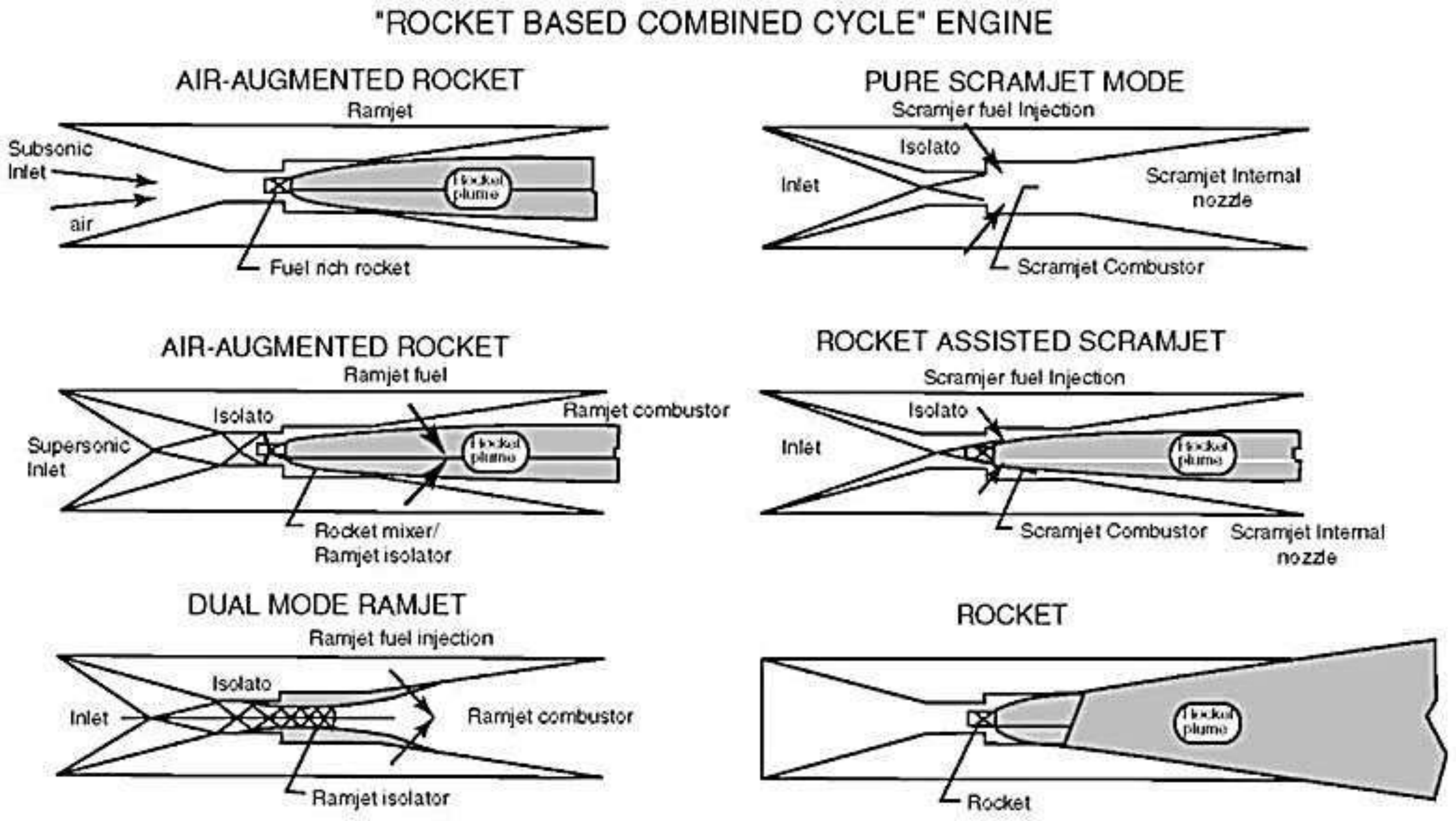
Preliminary concepts in nozzle less propulsion

Air augmented rockets



Air-augmented rockets (also known as rocket-ejector, ramrocket, ducted rocket, integral rocket/ramjets, or ejector ramjets) use the supersonic exhaust of some kind of rocket engine to further compress air collected by ram effect during flight to use as additional working mass, leading to greater effective thrust for any given amount of fuel than either the rocket or a ramjet alone.

It represents a hybrid class of rocket/ramjet engines, similar to a ramjet, but able to give useful thrust from zero speed, and is also able in some cases to operate outside the atmosphere, with fuel efficiency not worse than both a comparable ramjet or rocket at every point.

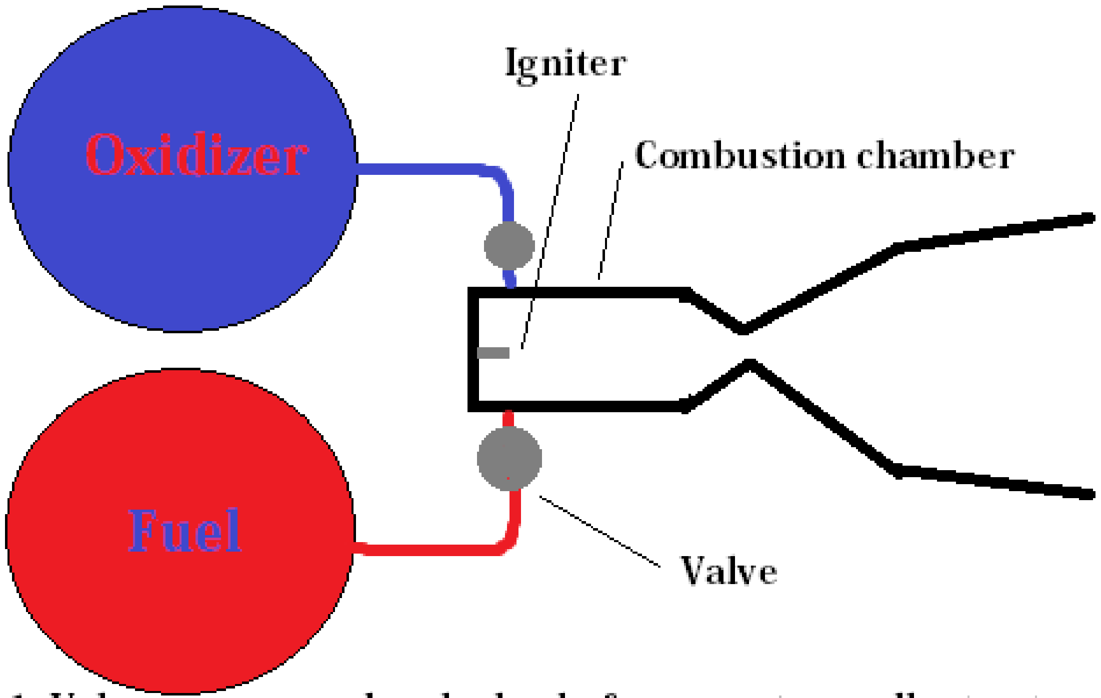

"ROCKET BASED COMBINED CYCLE" ENGINE
AIR-AUGMENTED ROCKET
Ramjet
Subsonic Inlet
air
Rocket plume
Fuel rich rocket
PURE SCRAMJET MODE
Scramjer fuel Injection
Isolato
Inlet
Scramjet Internal nozzle
Scramjet Combustor
AIR-AUGMENTED ROCKET
Ramjet fuel
Isolato
Ramjet combustor
Supersonic Inlet
Rocket plume
Rocket mixer/ Ramjet isolator
ROCKET ASSISTED SCRAMJET
Scramjer fuel Injection
Isolato
Inlet
Rocket plume
Scramjet Combustor
Scramjet Internal nozzle
DUAL MODE RAMJET
Ramjet fuel injection
Isolato
Inlet
Ramjet combustor
Ramjet isolator
ROCKET
Rocket plume
Rocket

Pulse rocket motors

- A pulsed rocket motor is typically defined as a multiple pulse solid-fuel rocket motor. This design overcomes the limitation of solid propellant motors that they cannot be easily shut down and reignited. The pulse rocket motor allows the motor to be burned in segments (or pulses) that burn until completion of that segment. The next segment (or pulse) can be ignited on command by either an onboard algorithm or in pre-planned phase. All of the segments are contained in a single rocket motor case as opposed to staged rocket motors.[1]

- The pulsed rocket motor is made by pouring each segment of propellant separately. Between each segment is a barrier that prevents the other segments from burning until ignited. At ignition of a second pulse the burning of the propellant generally destroys the barrier.

- The benefit of the pulse rocket motor is that by the command ignition of the subsequent pulses, near optimal energy management of the propellant burn can be accomplished. Each pulse can have different thrust level, burn time, and achieved specific impulse depending on the type of propellant used, its burn rate, its grain design, and the current nozzle throat diameter.[2]



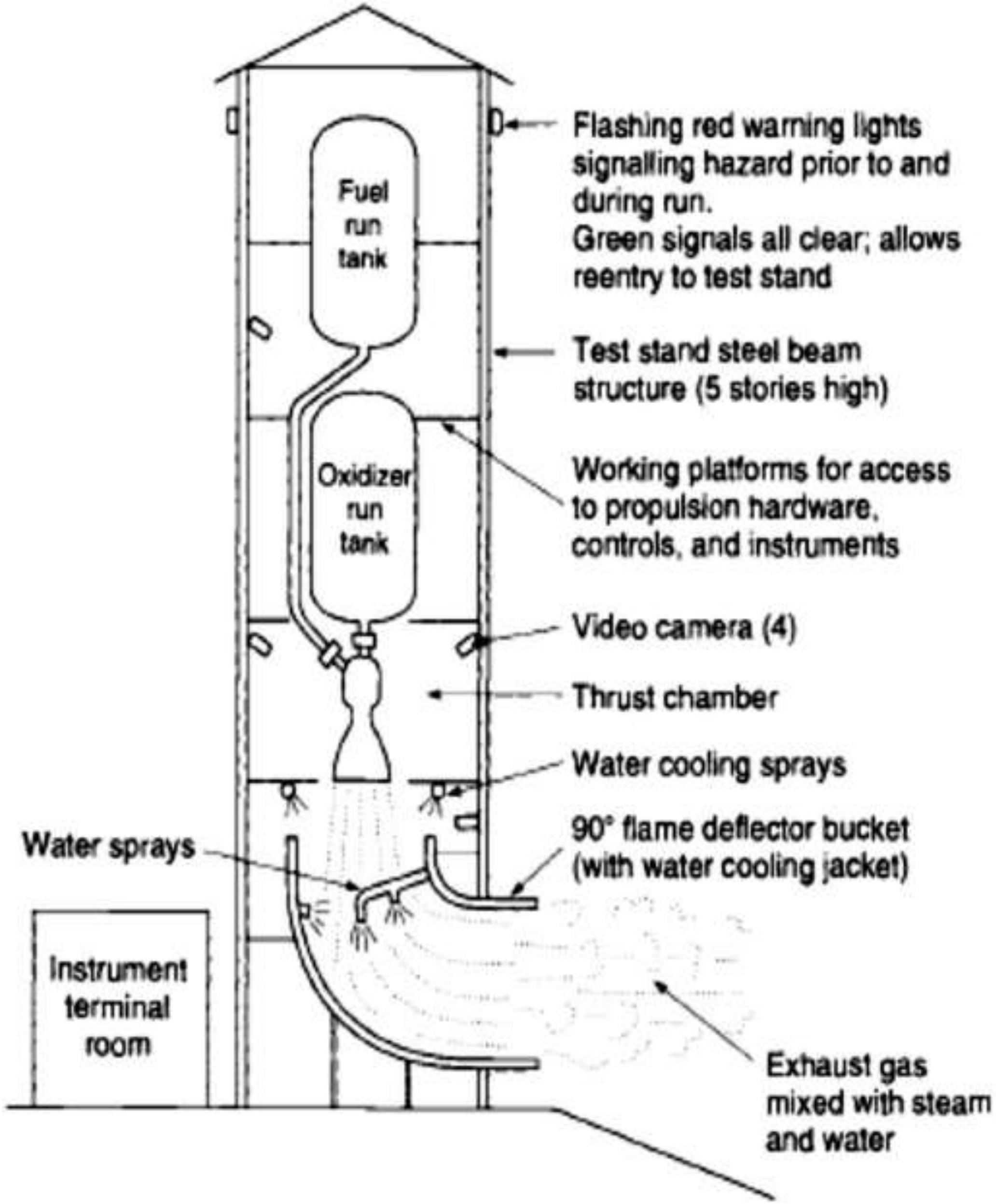
ANIMATION OF A PULSE JET ENGINE

1. Valves are opened and a load of gaseous propellent enters the chamber at a low pressure ~ 3 atm
2. Valves are closed and the propellent is ignited in a high pressue pulse.
3. Valves are opened and a new load of propellent fills the chamber.
4. Valves are closed and the propellent is ignited in a high pressue pulse.
5. The cycle continues.

Static testing of rockets and instrumentation,

For chemical rocket propulsion systems, each test facility usually has the following major systems or components:

1. A test cell or test bay where the article to be tested is mounted, usually in a special test fixture. If the test is hazardous, the test facility must have provisions to protect operating personnel and to limit damage in case of an accident.
2. An instrumentation system with associated computers for sensing, maintaining, measuring, analyzing, correcting, and recording various physical and chemical parameters. It usually includes calibration systems and timers to accurately synchronize the measurements.
3. A control system for starting, stopping, and changing the operating conditions.
4. Systems for handling heavy or awkward assemblies, supplying liquid propellant, and providing maintenance, security, and safety.
5. For highly toxic propellants and toxic plume gases it has been required to capture the hazardous gas.


Fuel
run
tank
Oxidizer
run
tank
Flashing red warning lights
signalling hazard prior to and
during run.
Green signals all clear; allows
reentry to test stand
Test stand steel beam
structure (5 stories high)
Working platforms for access
to propulsion hardware,
controls, and instruments
Video camera (4)
Thrust chamber
Water cooling sprays
90° flame deflector bucket
(with water cooling jacket)
Water sprays
Instrument
terminal
room
Exhaust gas
mixed with steam
and water

Safety considerations

1. Concrete-walled blockhouse or control stations for the protection of personnel and instruments remote from the actual rocket propulsion location.
2. Remote control, indication, and recording of all hazardous operations and measurements; isolation of propellants from the instrumentation and control room.
3. Automatic or manual water deluge and fire-extinguishing systems.
4. Closed circuit television systems for remotely viewing the test.
5. Warning signals (siren, bells, horns, lights, speakers) to notify personnel to clear the test area prior to a test, and an all-clear signal when the conditions are no longer hazardous.
6. Quantity and distance restrictions on liquid propellant tankage and solid propellant storage to minimize damage in the event of explosions; separation of liquid fuels and oxidizers.
7. Barricades around hazardous test articles to reduce shrapnel damage in the event of a blast.
8. Explosion-proof electrical systems, spark-proof shoes, and non-spark hand tools to prevent ignition of flammable materials.
9. For certain propellants also safety clothing , including propellant- and fire-resistant suits, face masks and shields, gloves, special shoes, and hard hats.
10. Rigid enforcement of rules governing area access, smoking, safety inspections, and so forth.
11. Limitations on the number of personnel that may be in a hazardous area at any time.

Instrumentation

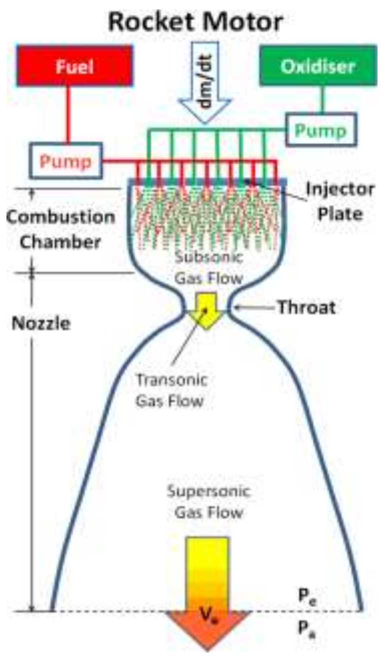
1. Forces (thrust, thrust vector control side forces, short thrust pulses).
2. Flows (hot and cold gases, liquid fuel, liquid oxidizer, leakage).
3. Pressures (chamber, propellant, pump, tank, etc.).
4. Temperatures (chamber walls, propellant, structure, nozzle).
5. Timing and command sequencing of valves, switches, igniters, etc.
6. Stresses, strains, and vibrations (combustion chamber, structures, propellant lines, accelerations of vibrating parts).
7. Time sequence of events (ignition, attainments of full pressure).
8. Movement and position of parts (valve stems, gimbal position, deflection of parts under load or heat).Voltages, frequencies, and currents in electrical or control subsystems.
9. Visual observations (flame configuration, test article failures, explosions) using high-speed cameras or video cameras.

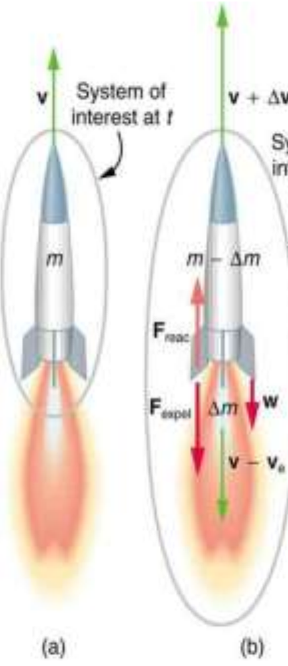
THANK YOU

Answer These Questions ?

- What is the meaning of thrust, total impulse and specific impulse?
- What are the advantages and disadvantages of solid fueled rockets?
- What are the advantages and disadvantages of liquid fueled rockets?
- What are the advantages and disadvantages of hypergolic fueled rockets?
- What was the main contribution of Tsiolkovsky to rocket science?
- What was the main contribution of Oberth to rocket science?
- What was the main contribution of Goddard to rocket science?
- What is the V-2 program and how was it such a technological leap over previous rocket research?

- Explain the difference between how a jet engine, like that described in Theory of Flight, and a rocket engine function. Why don't we use jet engineson rockets?
- Using what you know about forces, explain whether the rocket in the following situations is balanced or unbalanced. If it is unbalanced, describewhich force is greater than the others. Use a free-body diagram to help you.
- a. Rocket during launch.
- b. Rocket during re-entry.
- c. Rocket in orbit at constant velocity.
- d. Rocket accelerating in orbit.
- e. A Lunar Excursion Module (LEM) sitting on the moon.
- Why don't we use ailerons, rudders and elevators to control the direction of flight in space?
- Using what you have learned about Mass Fraction (MF) describe the characteristics of rockets with the following MF's. Will they fly? If so, how muchpayload can they carry? On what types of missions can they be used?
- a. 0.0
- b. 0.27
- c. 0.49
- d. 0.77
- e. 0.96





Rocket engine: A vehicle or device propelled by one or more rocket engines,especially such a vehicle designed to travel through space.

- A projectile weapon carrying a warhead that is powered and propelledby rockets.

- A projectile firework having a cylindrical shape and a fuse that is lit fromthe rear.

Missile: An object or weapon that is fired, thrown, dropped, or otherwiseprojected at a target; a projectile.

PROPELLANT

- Propellant is the chemical mixture burned to produce thrust in rockets andconsists of a fuel and an oxidizer.
- A fuel is a substance that burns when combined with oxygen producing gas forpropulsion.
- An oxidizer is an agent that releases oxygen for combination with a fuel.
- The ratio of oxidizer to fuel is called the mixture ratio. Propellants are classifiedaccording to their state - liquid, solid, or hybrid.
- The gauge for rating the efficiency of rocket propellants is specific impulse,
stated in seconds.

19-02-2019

Liquid Propellants :

- In a liquid propellant rocket, the fuel and oxidizer are stored in separate tanks, and are fed through a system of pipes, valves, and turbo pumps to a combustion chamber where they are combined and burned to produce thrust.
- Liquid propellant engines are more complex than their solid propellantcounter parts, however, they offer several advantages.
- By controlling the flow of propellant to the combustion chamber, the engine can be throttled, stopped, or restarted.

- **Solid Propellants :**

- Solid propellant motors are the simplest of all rocket designs.

They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust.

- When ignited, a solid propellant burns from the center out towards thesides of the casing. The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust.
- Unlike liquid propellant engines, solid propellant motors cannot be shutdown.
- Once ignited, they will burn until all the propellant is exhausted.

Hybrid Propellants:

- Hybrid propellant engines represent an intermediate group between solid andliquid propellant engines.
- One of the substances is solid, usually the fuel, while the other, usually theoxidizer, is liquid.
- The liquid is injected into the solid, whose fuel reservoir also serves as thecombustion chamber.
- The main advantage of such engines is that they have high performance, similarto that of solid propellants, but the combustion can be moderated, stopped, or even estarted.

Operating principle

- A rocket is a machine that develops thrust by the rapid expulsion of matter.
- The major components of a chemical rocket assembly are a rocket motor or engine, propellant consisting of fuel and an oxidizer, a frame to hold the components, control systems and a cargo such as a satellite.
- A rocket is called a launch vehicle when it is used to launch a satellite or other payload into space.
- A rocket becomes a missile when the payload is a warhead and it is used as a weapon.

- At present, rockets are the only means capable of achieving the altitude and velocity necessary to put a payload into orbit.

Rocket Power

- There are a number of terms used to describe the power generated by a rocket.
- Thrust is the force generated, measured in pounds or kilograms. Thrust generatedby the first stage must be greater than the weight of the complete launch vehicle while standing on the launch pad in order to get it moving.
- The impulse, sometimes called total impulse, is the product of thrust and the effective firing duration.
- The efficiency of a rocket engine is measured by its specific impulse (Isp). Specificimpulse is defined as the thrust divided by the weight of the propellant consumed per second. The result is expressed in seconds.

Mass ratio

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- A rocket's mass ratio is defined as the total mass at lift-off divided by the massremaining after all the propellant has been consumed.
- A high mass ratio means that more propellant is pushing less launch vehicle andpayload mass, resulting in higher velocity.
- A high mass ratio is necessary to achieve the high velocities needed to put apayload into orbit.

Thrust

Rocket thrust can be explained using Newton's 2^{nd} and 3^{rd} laws of motion.

2^{nd} Law: a force applied to a body is equal to the mass of the body and its acceleration in the direction of the force.

$$F = ma$$

3^{rd} Law: For every action, there is an equal and opposite reaction.

$F_a \quad F_r$

In rocket propulsion, a mass of propellant (m) is accelerated (via the combustion process) from initial velocity (V_o) to an exit velocity (V_e). The acceleration of this mass is written as:

Thrust

Another component of thrust (*pressure thrust*, F_2) comes from the force exerted by external pressure differences on the system.This is described by the difference of the pressure of the flow leaving the engine (P_e) through the exit area (A_e) compared to the external (ambient) pressure (P_a).

$$F_2 = (P_e - P_a)A_e$$

In space, P_a is assumed to be zero (which explains why thrust rated at vacuum is higher than at sea level).

Combining the two thrust components gives

$$\boxed{F = \frac{\dot{m}}{g}(V)_e + (P_e - P_a)A_e}$$

Specific Impulse

The *total impulse* (I_t) is the thrust integrated over the run duration (time, t)

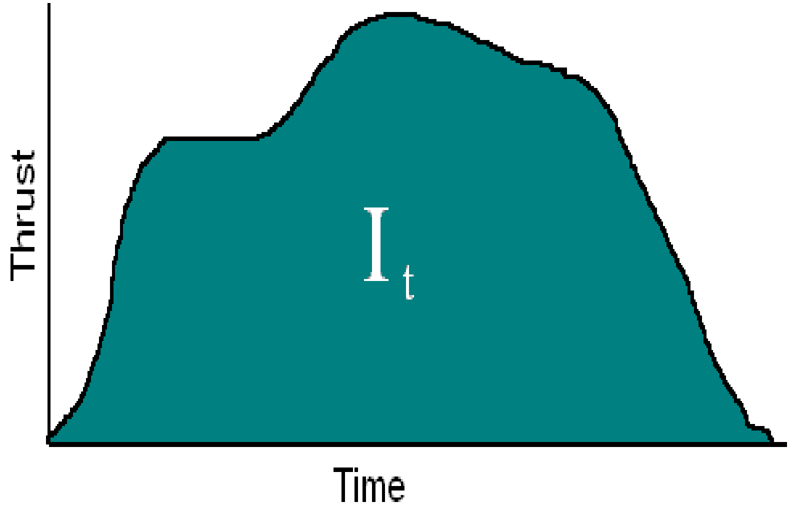
$$I_t = \int_0^t F\,dt$$

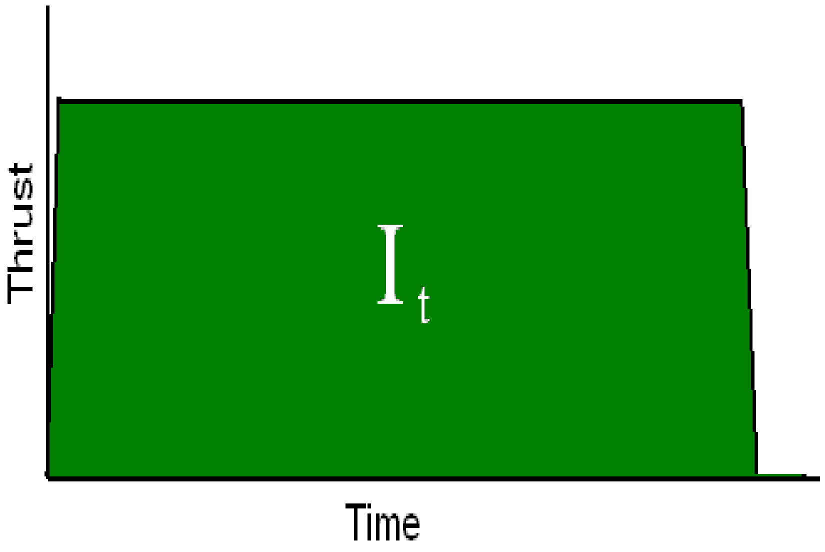
Assuming constant thrust and negligible transients (i.e., start and shutdown), this becomes

$$I_t = Ft$$

The *specific impulse*, (I_{sp}) is the total impulse generated per weight of propellant

$$I_{sp} = \frac{\int_0^t F\,dt}{g_o \int_0^t \dot{m}\,dt} = \boxed{\frac{F}{\dot{m}}}$$





Mixture Ratio

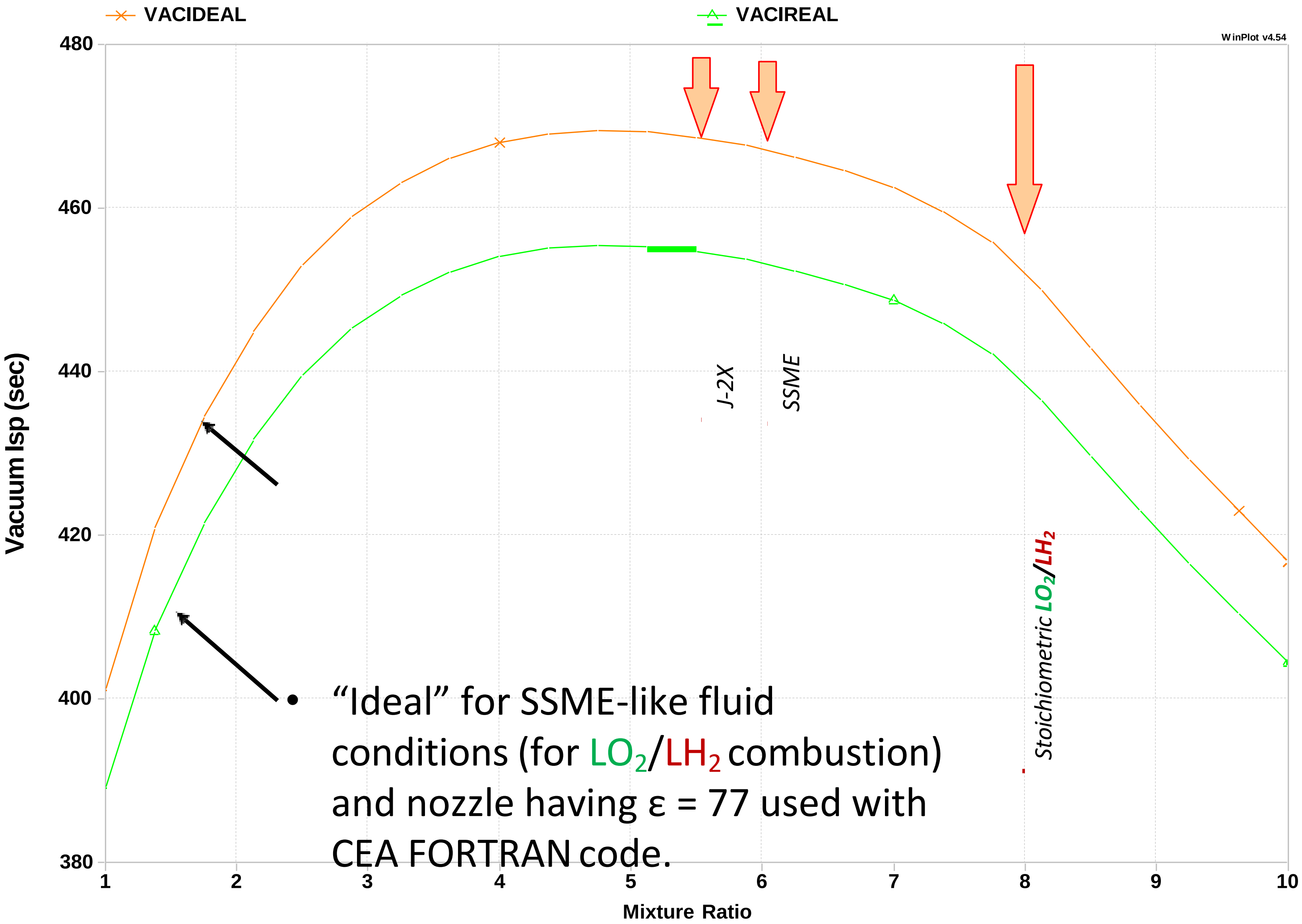
Rocket propellants are mixed in relative quantities to produce the highest possible system I_{sp}. This ratio of propellant consumption is called mixture ratio, MR.

$$MR = \frac{M_o}{M_f}$$

In most cases, MR is selected for maximum energy release per weight of propellant. This can be achieved by mixing the propellants in a stoichiometric reaction in the combustion chamber, where all the propellants are thoroughly combusted. <u>However, a stoichiometric MR</u> <u>does not necessarily provide an optimized I_{sp}.</u>

☞ The SSME uses a MR of ~6 (stoichiometric for LO_2/ LH_2 combustion is 8) to reduce the internal and plume temperatures, but also to allow a small amount of H_2 to remain in the exhaust. The lighter molecule is able to accelerate to a higher velocity and generate higher kinetic energy ($KE = \frac{1}{2} mV^2$) than a H_2O steam exhaust.

I_{sp} vs. MR



- "Ideal" for SSME-like fluid conditions (for LO_2/LH_2 combustion) and nozzle having ε = 77 used with CEA FORTRAN code.

Density vs. I_{sp}

- Liquid bipropellant combinations offer a wide range of performance capabilities.
- Each combination has multiple factors that should be weighed when selecting one for a vehicle.
  - Performance (I_{sp})
  - Density (higher is better)
  - Storability (venting?)
  - Ground Ops (hazards?)
  - Etc.
- One of the more critical trades is that of performance versus density.
- LO_2/LH_2 offers the highest I_{sp} performance, but at the cost of poor density (thus increasing tank size).
- Trading I_{sp} versus density is sometimes referred to as comparing "bulk impulse" or "density impulse".

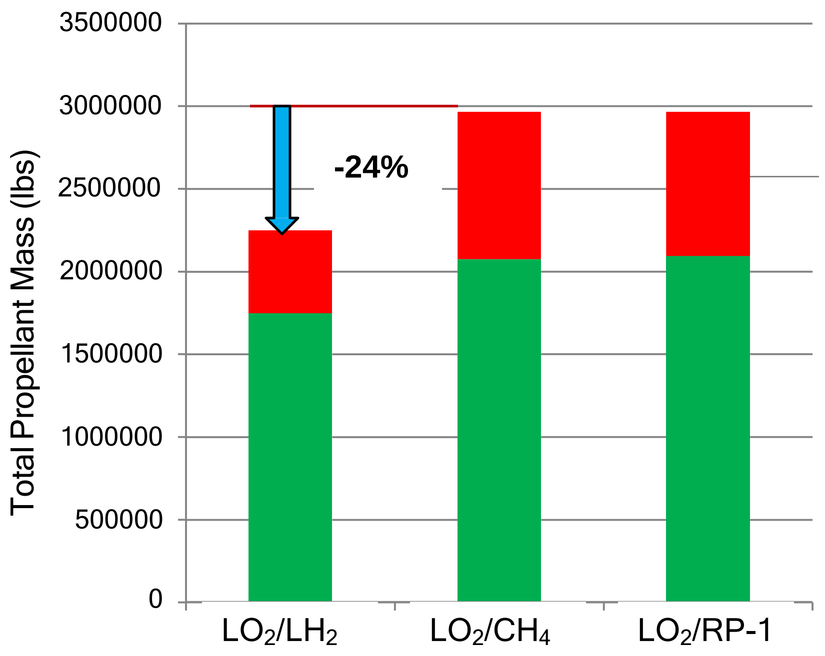
- As an example, the densities and I_{sp} performance of the following propellant combinations will be compared.

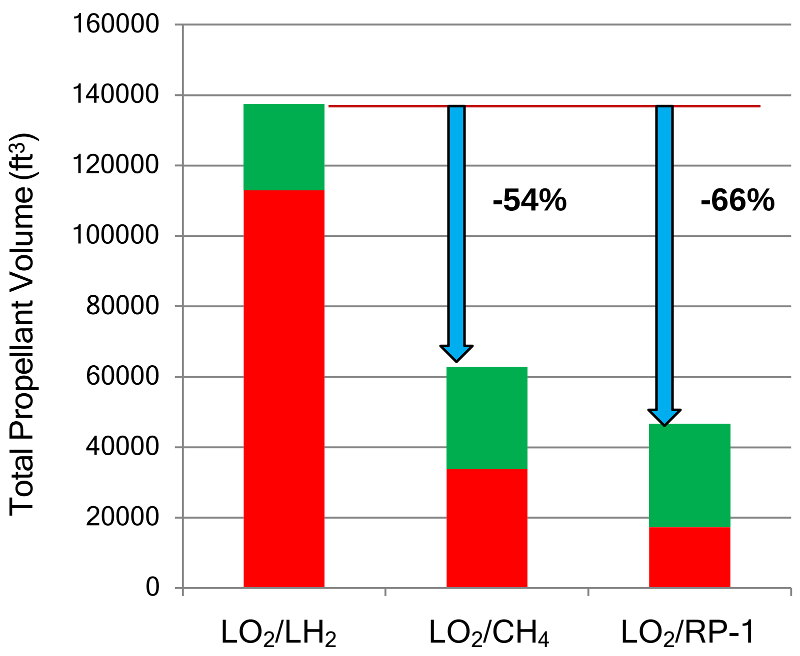
	Density (g/ml)	**Density (lb/ft^3)**
Hydrogen	0.07	4.4
Methane	0.42	26.4
RP-1	0.81	50.6
Oxygen	1.14	71.2

P_c = 300 psia expanded to 14.7 psia	**MR (O/F)**	**I_{sp} (sec)**
LO_2/LH_2	3.5	347 (1)
LO_2/CH_4	2.33	263 (2)
LO_2/RP-1	2.4	263 (2)

(1) SC (2) FC

Propellant Mass vs. Volume





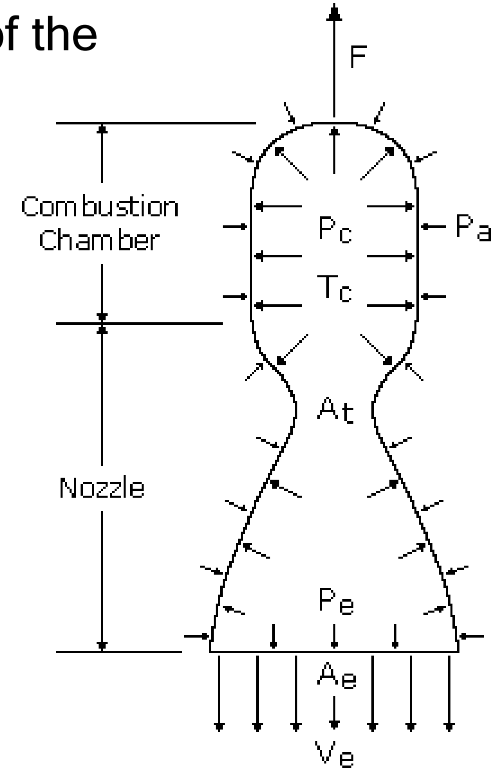
- For an impulse requirement similar to the 3 SSME's used on the Shuttle (~1.5 Mlbf for 520 seconds), the required propellant masses are calculated.
- LO_2/LH_2 requires 24% less propellant mass than the others.
- However…

- When the propellant mass is compared against the tank volume, there is a significant disparity from the low hydrogen density that can adversely impact the size (and total weight) of the vehicle.
- Lesson: *I_{sp} isn't everything – especially with boost stages.*

Area Ratio

The parameter that determines exit velocity and pressure of the exhaust gases is *area ratio* or *nozzle expansion ratio*, MR

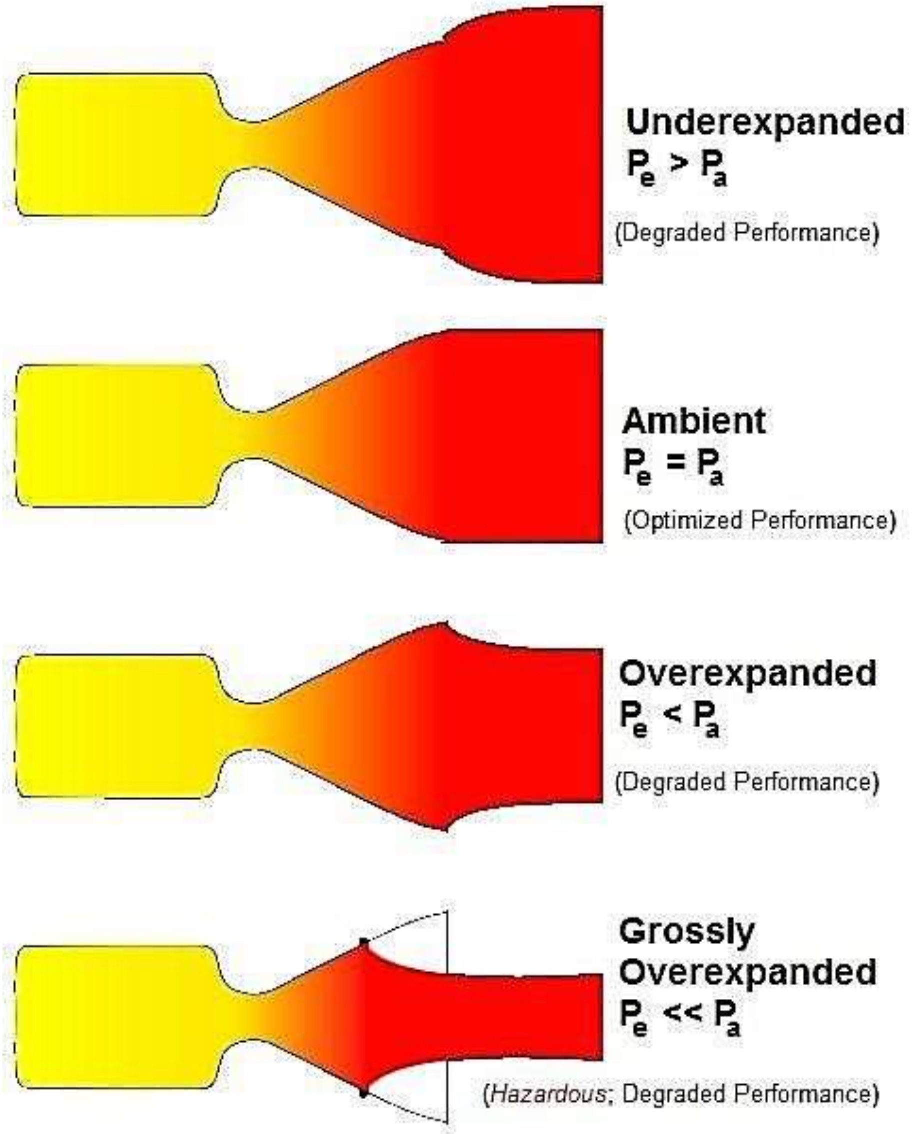
$$\varepsilon = \frac{A_e}{A_t}$$



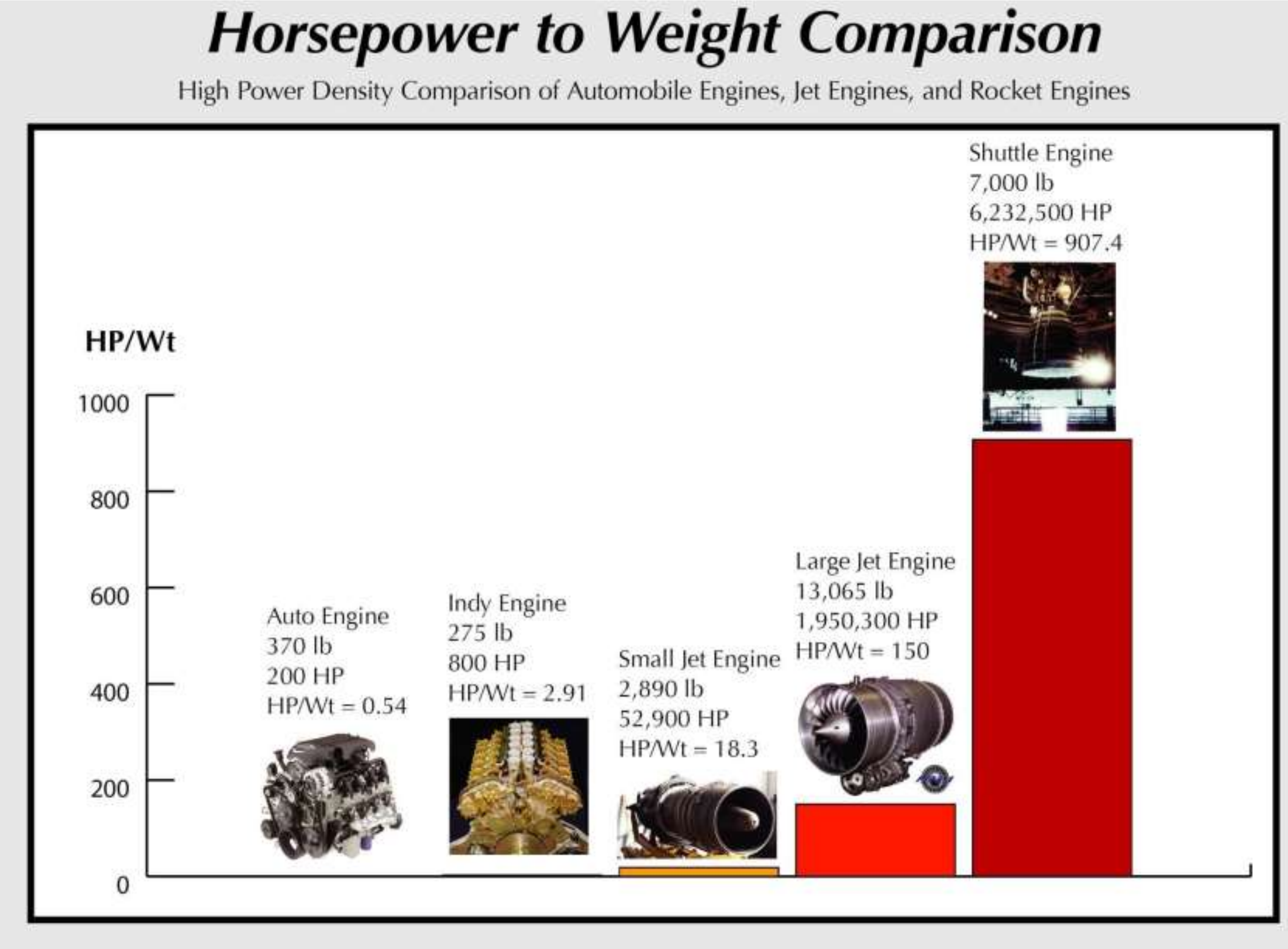
- As ε increases, the exit velocity increases and the exit pressure decreases (higher I_{sp}).
- When possible, ε is selected so that $P_e = P_a$ and the engine operates at optimum thrust.
- For in-space propulsion (i.e., J-2X), the ε is made as large as the weight requirements or volume limitations permit.
- If $P_e > P_a$, then the nozzle is identified as underexpanded and will not provide optimal performance as the plume will continue to expand after exiting the nozzle.
- If $P_e < P_a$, then the nozzle is identified as overexpanded and will not provide optimal performance as the exit shock will migrate inside the nozzle. This can be hazardous from thrust imbalances and damage to the nozzle.

Area Ratio

- Criteria to be considered for selecting a design ε can include the following:
  - ε provides the optimal integrated performance over the engine operating period. Trajectory analysis is used to determine the altitudes (and P_a) that the engine will operate, which can be used to provide an integrated I_{sp} based on ε.
  - ε is optimized to provide the maximum performance during a critical time of the engine operating period. Example: the ε for SSME is optimized at the altitude where the SRBs are staged to provide a needed performance boost at that critical time



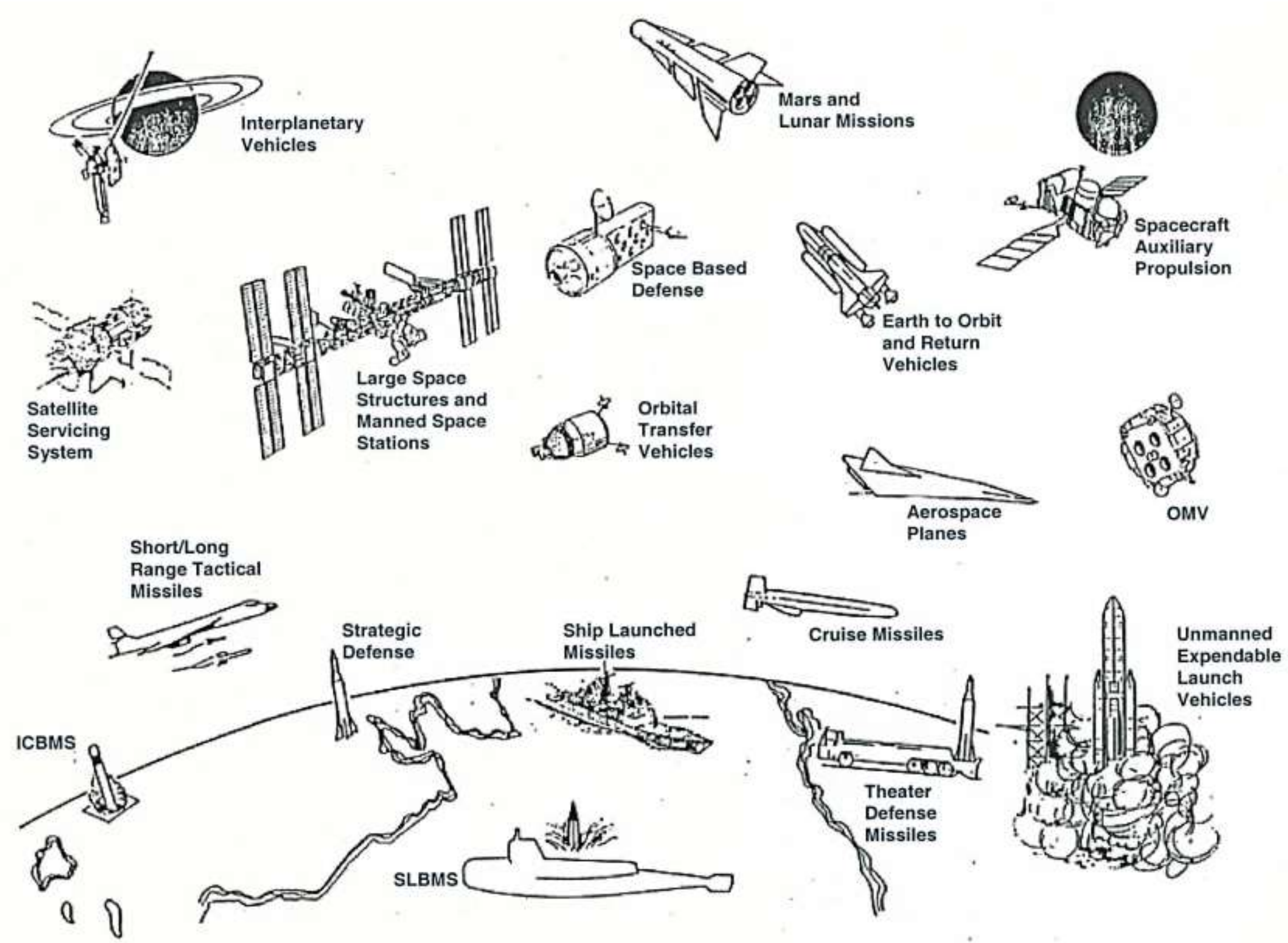
Power Density



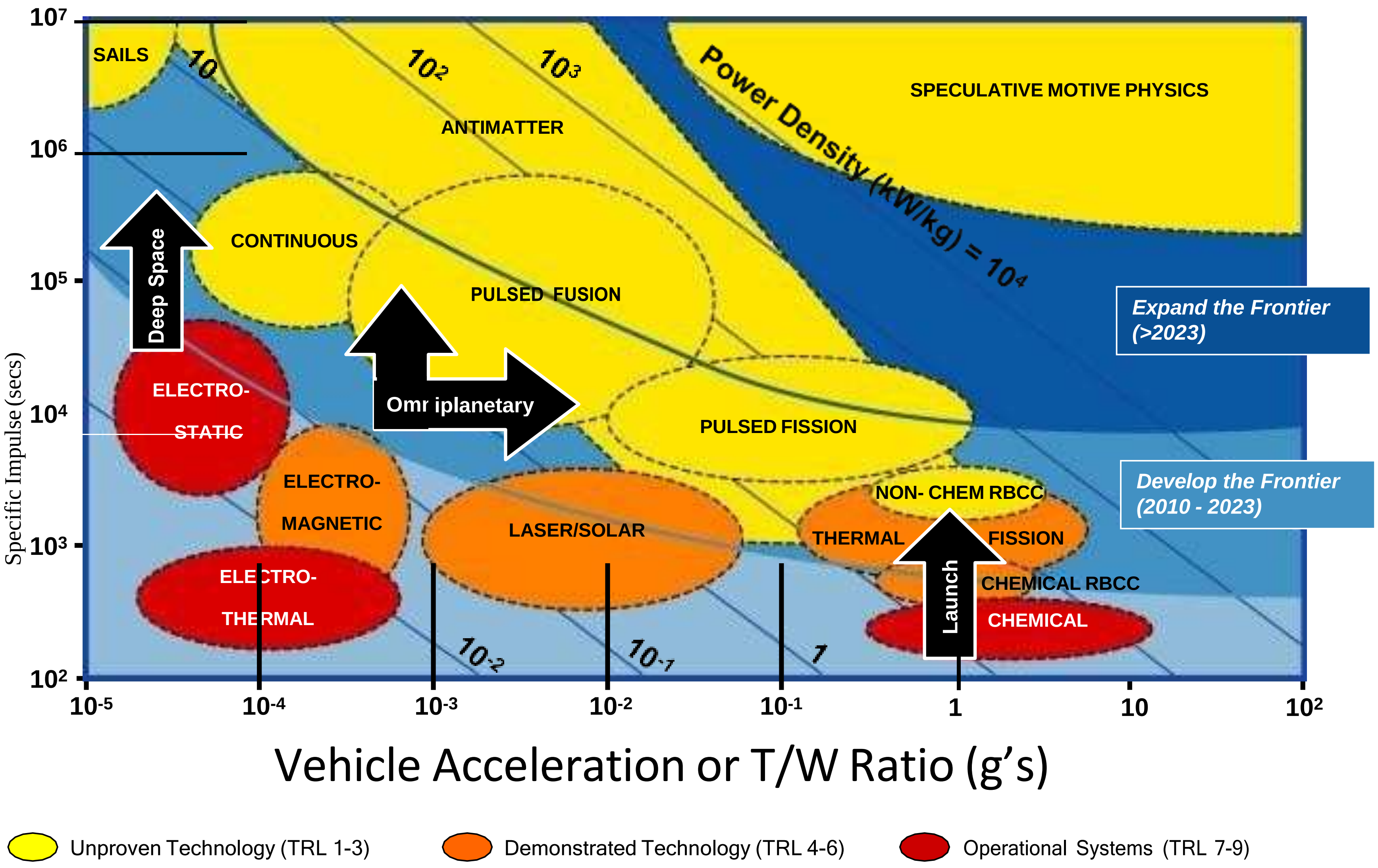
- Problem: Calculate the mass ratio needed to escape Earth's gravity starting from rest, given that the escape velocity from Earth is about 11.3 10^3 m/s and assuming an exhaust velocity V_e= 2.5x10^3 m/s.

Ans : 1/88

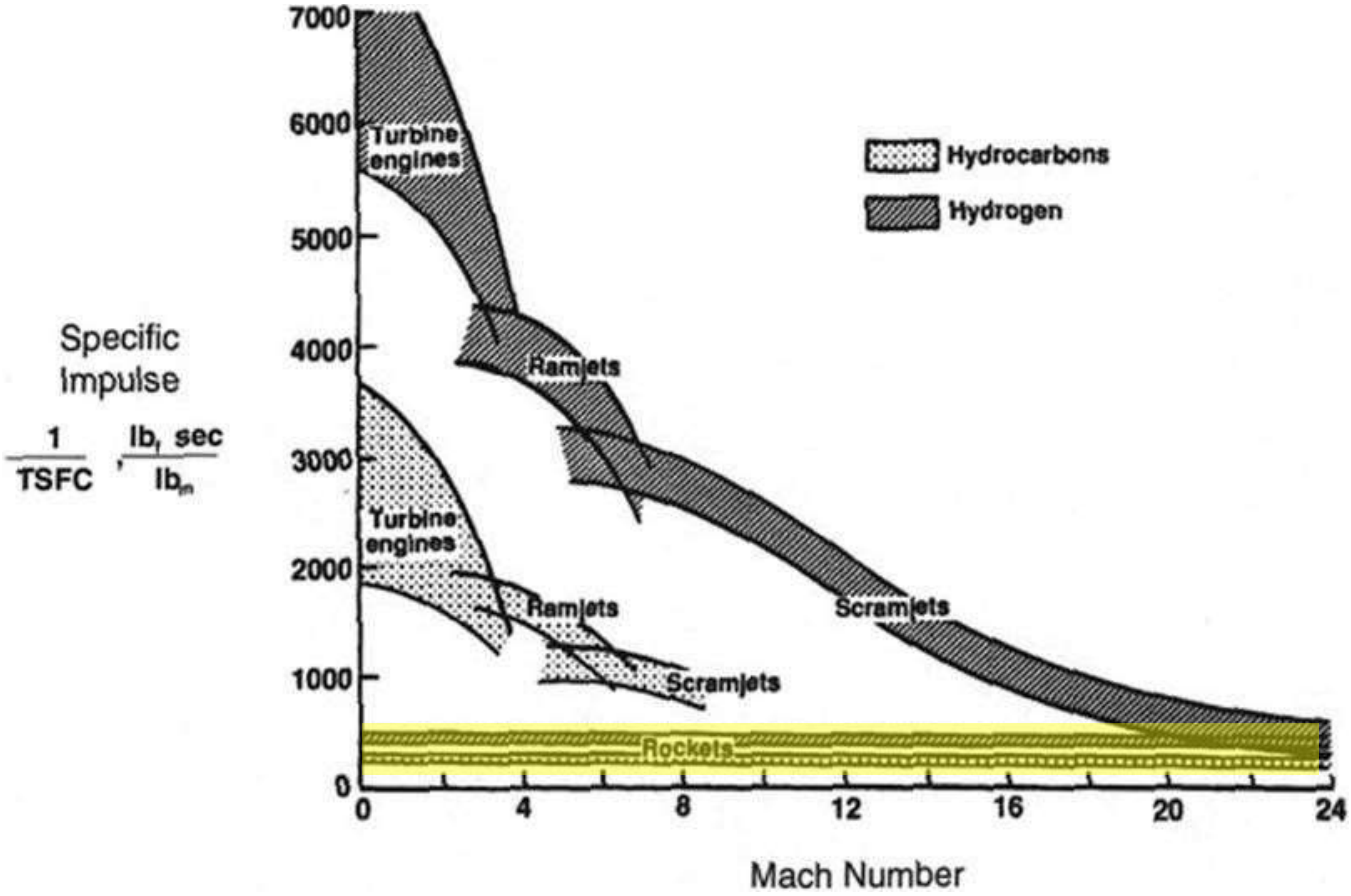
Rocket Propulsion Applications



In-Space Propulsion Performance



Jet Propulsion Options



BALLISTICS

- Knowledge of physical forces acting on a projectile and missile is called Ballistics

OR

- Ballistics is the simply science of motion of a projectile

Types

- Exterior or External Ballistics
- Interior or Internal Ballistics
- Terminal or Wound Ballistics

✓ **Exterior Ballistics** deals with the study of motion of a projectile after it leaves the barrel of a firearm.

✓ **Terminal Ballistics** is the study of effect of impact of a projectile on the target leading to wound formation (Also called Wound Ballistics).

INTERIOR / INTERNAL BALLISTICS

Interior Ballistics is the study of physio-chemical phenomenon within the firearm from the movement of the detonation of primer to the time the projectile leaves the barrel.

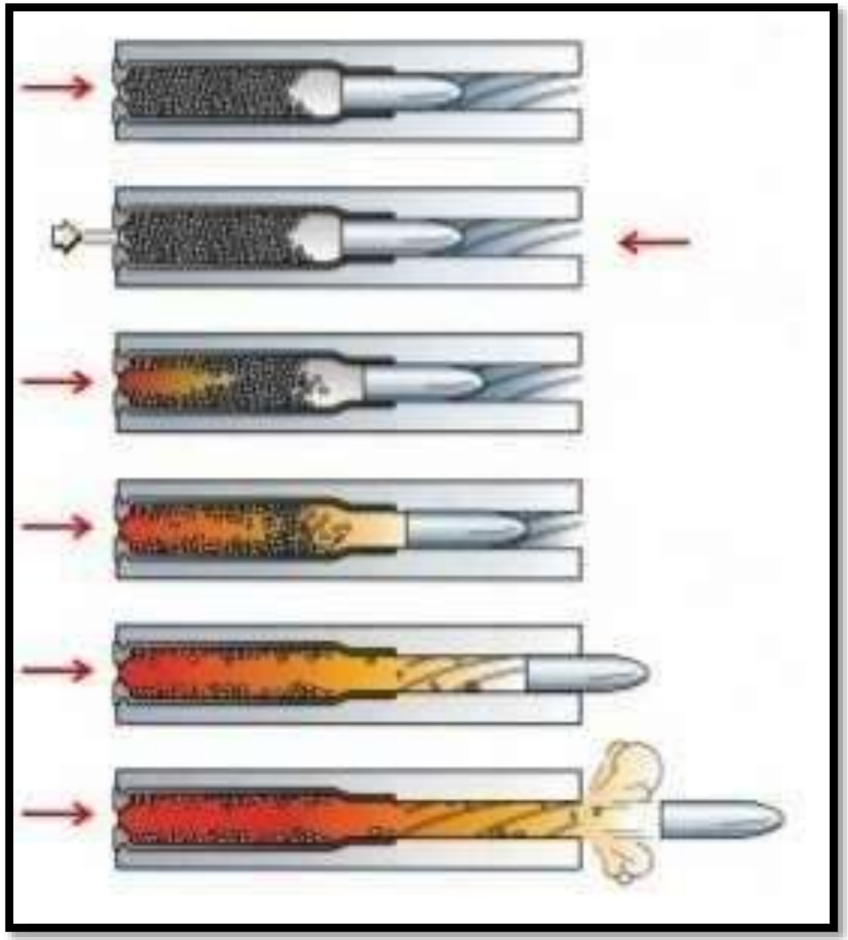
OR

The study of the process originally accelerating the principles is called Interior Ballistics, for example the passage of a bullet through the barrel of a rifle.

Internal Ballistics

- Internal Ballistics, a subfield of ballistics is the studyof projectile's behavior from the time its propellant's ignites is initiated until it exits the gunbarrel.
- The study of internal ballistics is important to designers and users of firearms of alltypes.


EXTERIOR BALLISTICS ?
I'M OUT OF THE GUN ACCORDING TO PLAN —NOW WHAT?
Initial Speed and Direction
INTERIOR BALLISTICS
Bore Erosion
Propelling Charge and Burning Rate
Powder Temperature
Density and Manner of Loading



The three main factors are-

- Lock Time
- Ignition Time
- Barrel Time

Lock Time

- Lock time is the time interval between release of the sear and the impact of the striker on the percussion cap.
- A short time internal is advantageous in rapidfire.
- The lock time can be measured in a number of ways, in one such system the use of linear Motion sensors and an oscilloscope is made.

Ignition Time

- Ignition time is the duration or interval between the striking of the firing pin to blow the ignition of the first grain of powder.
- The ignition, under the normal conditions, takes place at an interval of about 0.002 seconds.

Barrel Time

- Barrel time is the time interval from the pressing of the trigger to the exit of the bullet from the muzzle end.
- In case of most of the weapons Lock time + Ignition time + barrel time varies from 0.003 to 0.007 seconds.

INTERIOR / INTERNAL BALLISTICS

It includes……

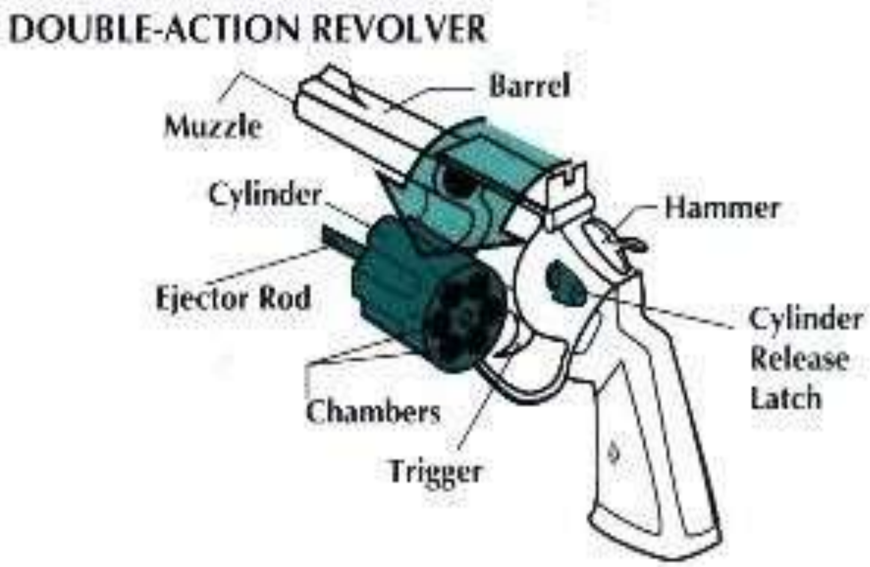
I. Structure of Firearm.
II. Design of Ammunition.
III. Chain/Sequence of events.

ii. Structure of firearm

Every firearm is basically divided into three parts…..

- Grip portion
- Action portion having thr trigger
- Front portion called

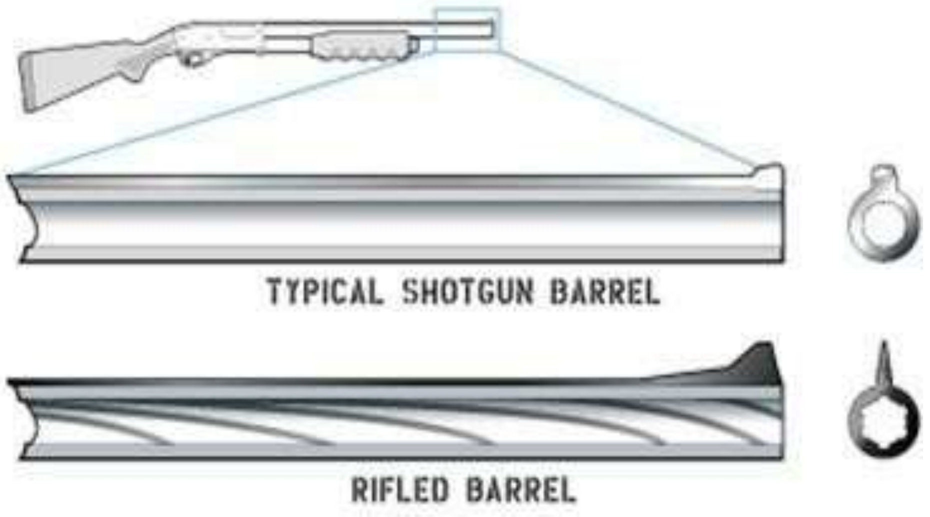
  the barrel



Structure of firearm

- Barrel….. Steel tube for jetting of the projectile
  - Breach end
  - Muzzle end

- Bore/Callibre….. Internal diameter of the barrel.
  - Smooth
  - Choked
  - Non-Choked
- Rifled
  - Short barrel
  - Long barrel



Rifling

- Itconsists of grooves or cuts formed in a spiral nature lengthwise down the barrel of a firearm.

- Because bullets are oblong objects, they must spin in their fight like a thrown football, to be accurate

Lands….. Raised areas between two grooves

Grooves….. Depressed areas

- Rifling pattern of eight grooves will also have eight lands.

III. AMMUNITION DESIGN

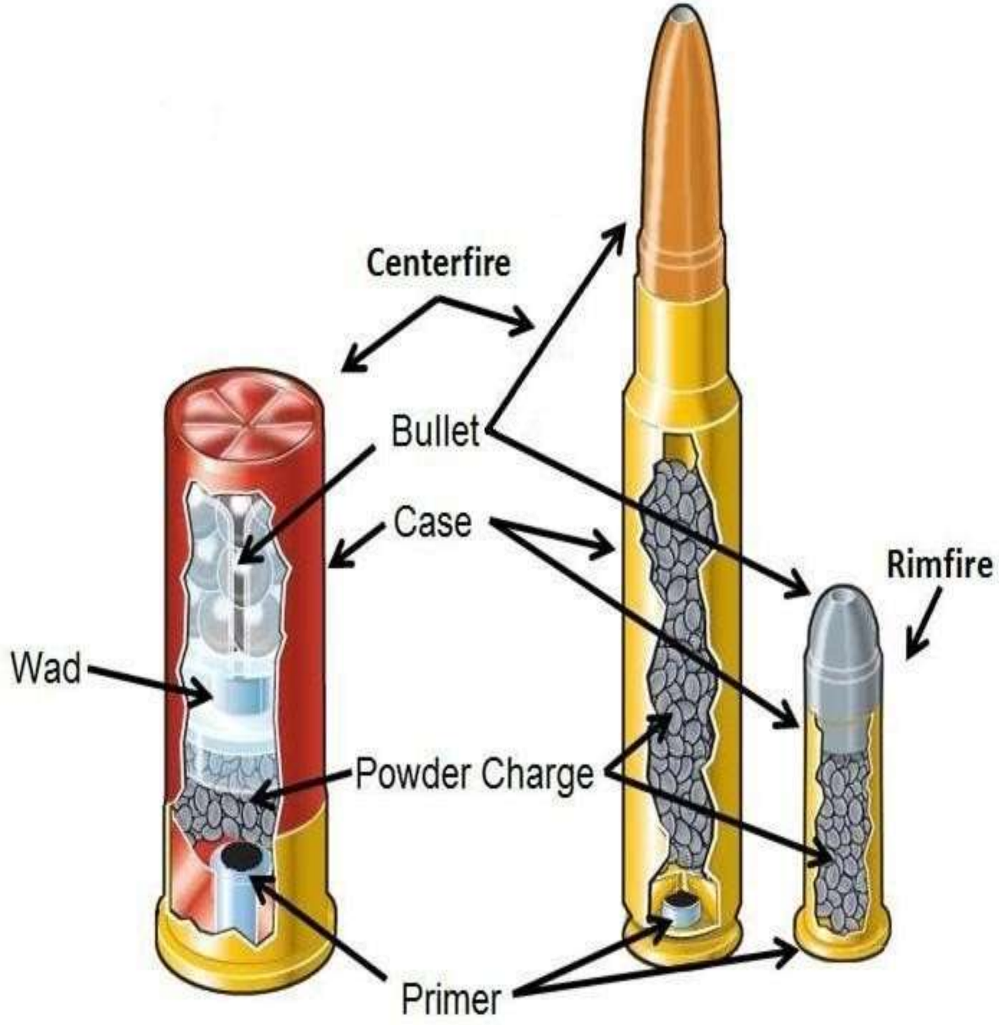
- Cartridge Case
- Primer

Powder Charge

III. Black
IV. Smokeless

- Plastic wad
- Short charge

III. Bullet
IV. Pellets



AMMUNITION CASES

- **Cartridge cases** are made up of plastic and card board.
- **Bullet cases** are made up of brass(70% copper & 30% Zinc, some have nickel coating).
- **Primer cases** are of similar composition (Copper & Zinc).
- **Bullet cores** are most often lead and antimony with a very few have ferrous alloy core.
- **Bullet jackets** are usually brass, 90% copper with 10% Zinc but some are a ferrous alloy & some are aluminum. Some bullet coating may also contain nickel.

Types of Ammunition

Primer

- The major primer elements are lead, barium or Antimony. Usually all three are present.
- Less common elements include aluminium, Sulphur, Tin, Calcium, Potassium, Chlorine and Silicon.
- Primer elements may be easier to detect in residue because they don’t get as hot as powder and compounds may be detectable.

Powder Charge

- Modern gun powder or smokeless powder can contain upto 23 organic compounds.
- Nitrocellulose is virtually always present along with other compounds containing nitrates or nitrogen.

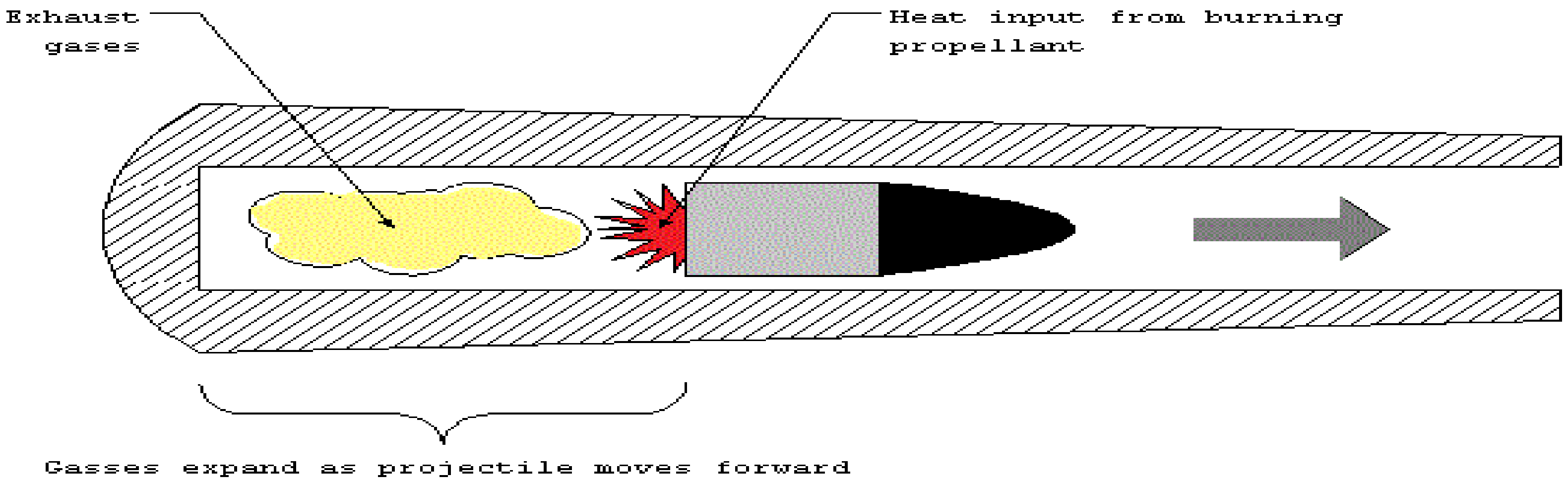
III. **Single base…..** The basic ingredient is nitrocellulose.

IV. **Double base…..** When there is added 40% Nitroglycerineto nitrocellulose.

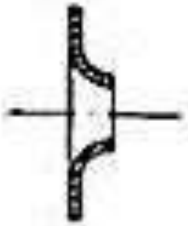
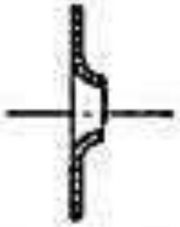
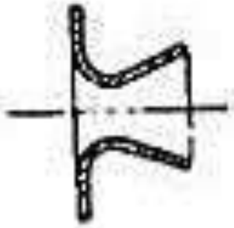
Powder Charge

Chain Sequence of Events

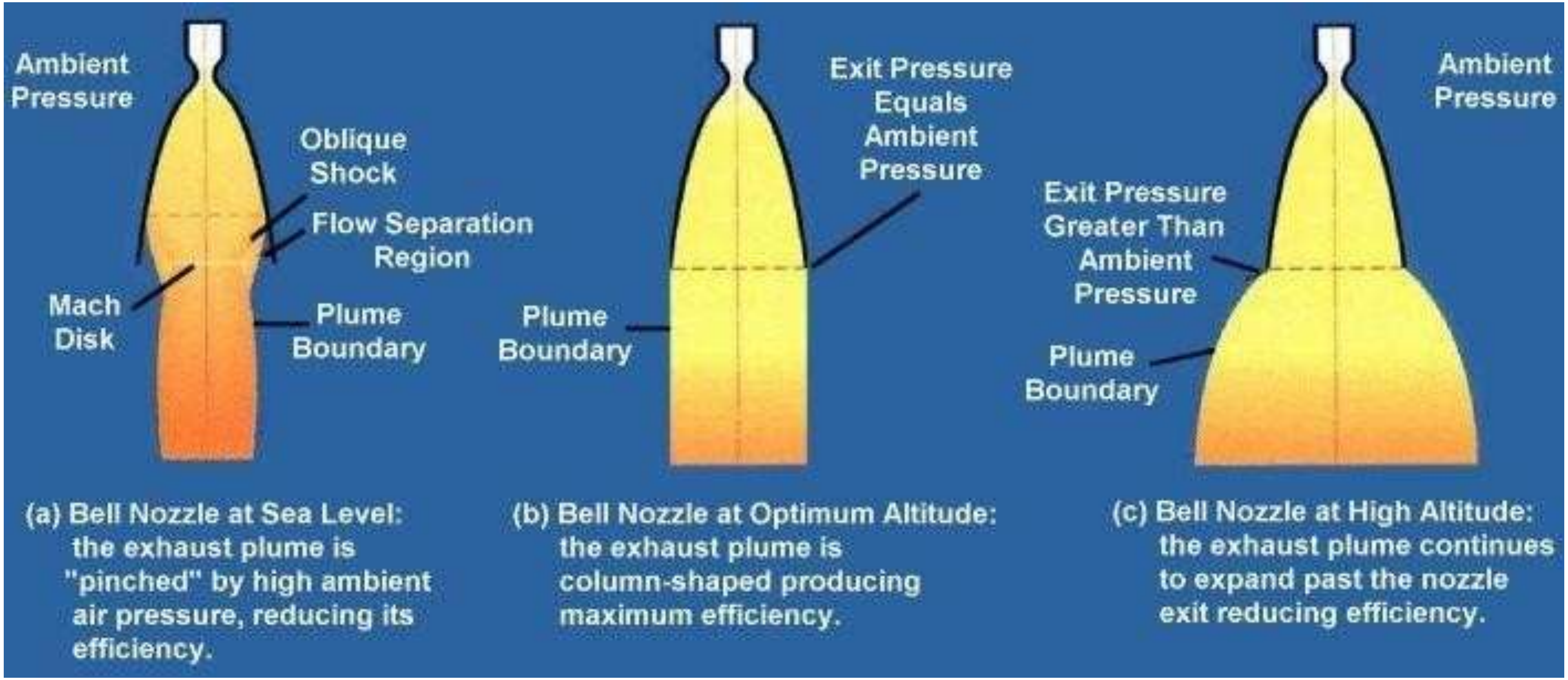
- Hammer Strikes
- Primer Ignites
- Powder charge burns
- Temperature increases
- Gases produced
- Chamber pressure increased
- Bullet charge moves
- Exit of bullet
- Chamber pressure = Zero



Nozzle Types

	Subsonic	Sonic	Supersonic
Throat velocity	$v_t < a_t$	$v_t = a_t$	$v_t = a_t$
Exit velocity	$v_2 < a_2$	$v_2 = v_t$	$v_2 > v_t$
Mach number	$M_2 < 1$	$M_2 = M_t = 1.0$	$M_2 > 1$
Pressure ratio	$\frac{p_1}{p_2} < \left(\frac{k+1}{2}\right)^{k/(k-1)}$	$\frac{p_1}{p_2} = \frac{p_1}{p_t} = \left(\frac{k+1}{2}\right)^{k/(k-1)}$	$\frac{p_1}{p_2} > \left(\frac{k+1}{2}\right)^{k/(k-1)}$
Shape			

Rocket Nozzle Classification



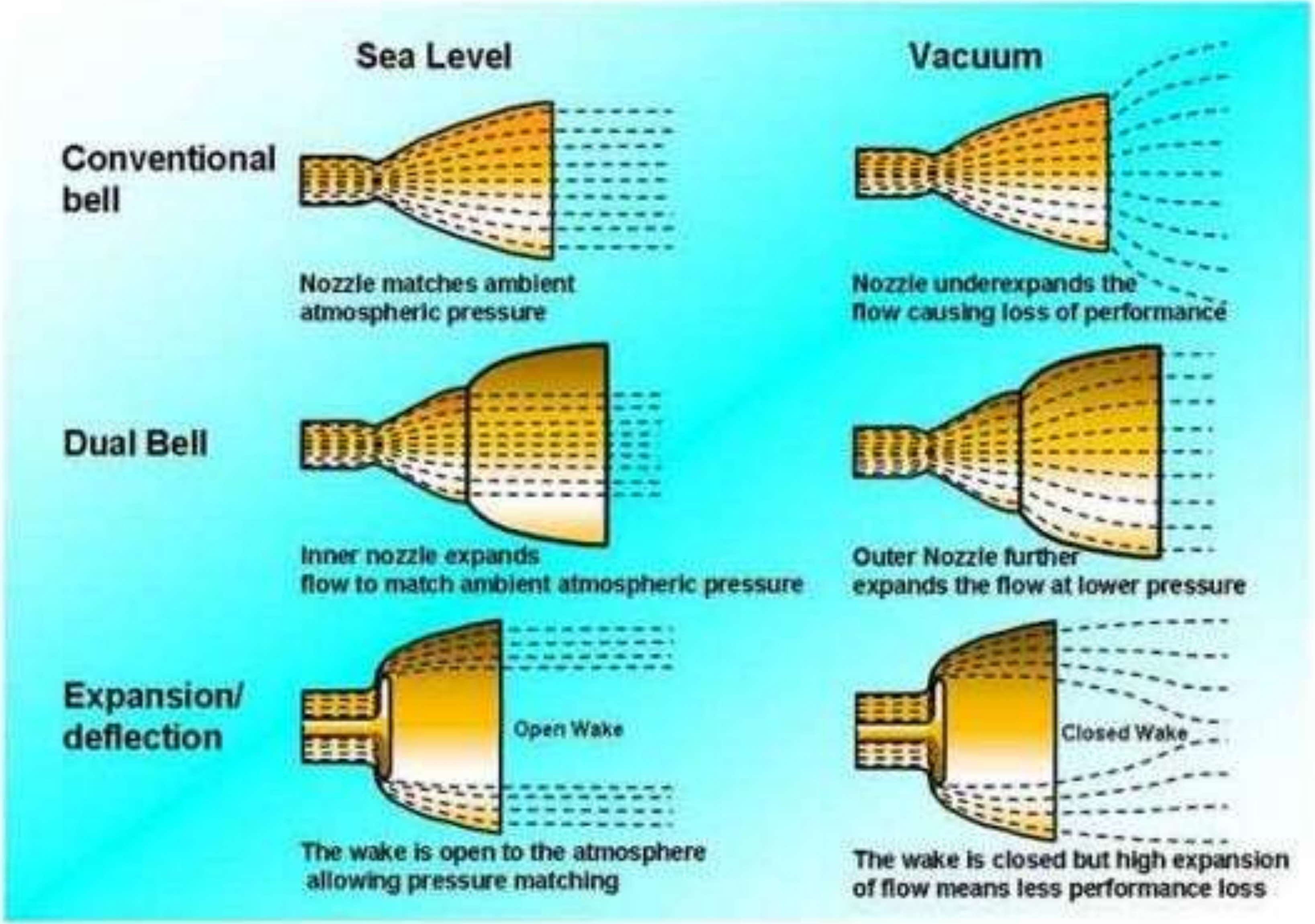
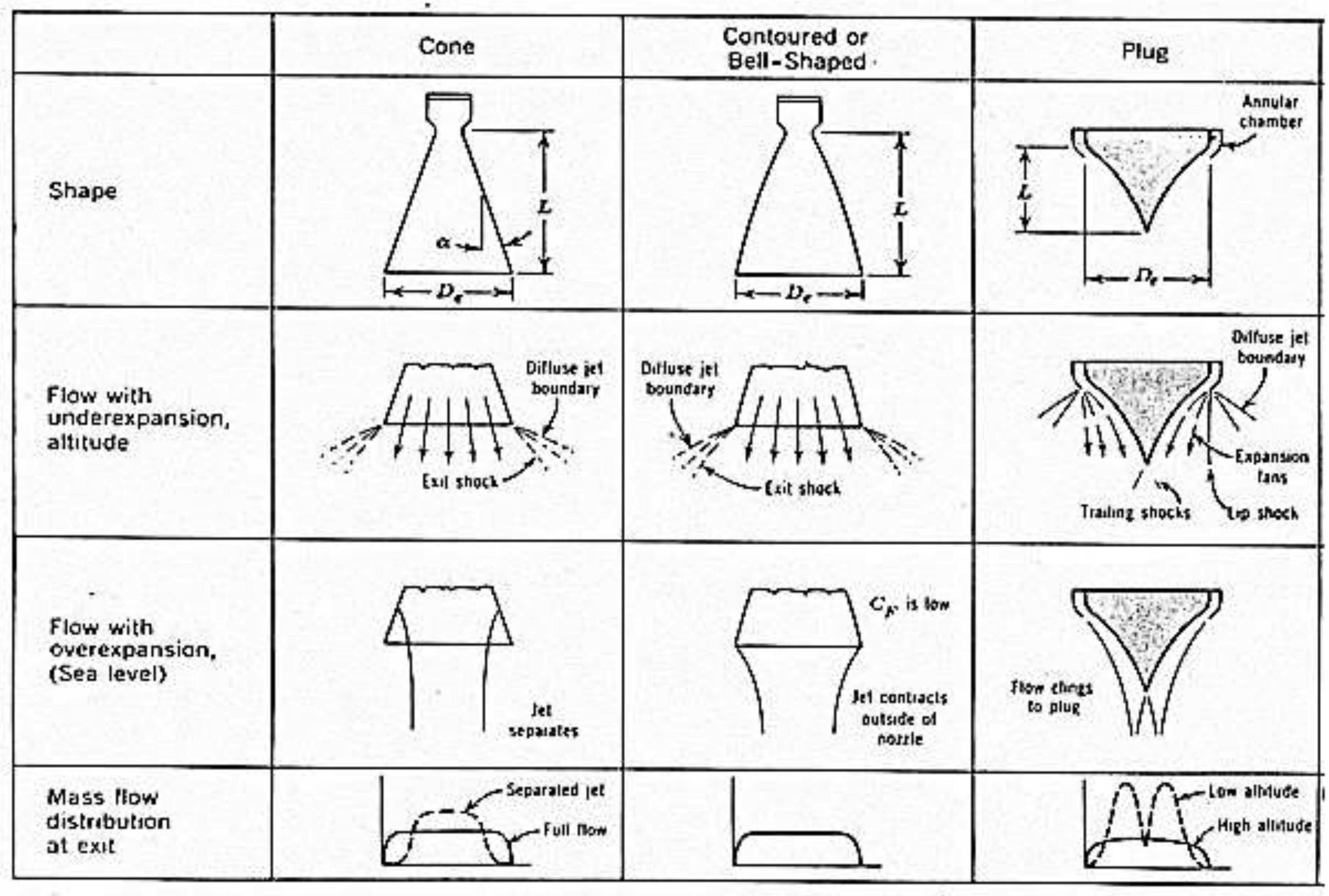
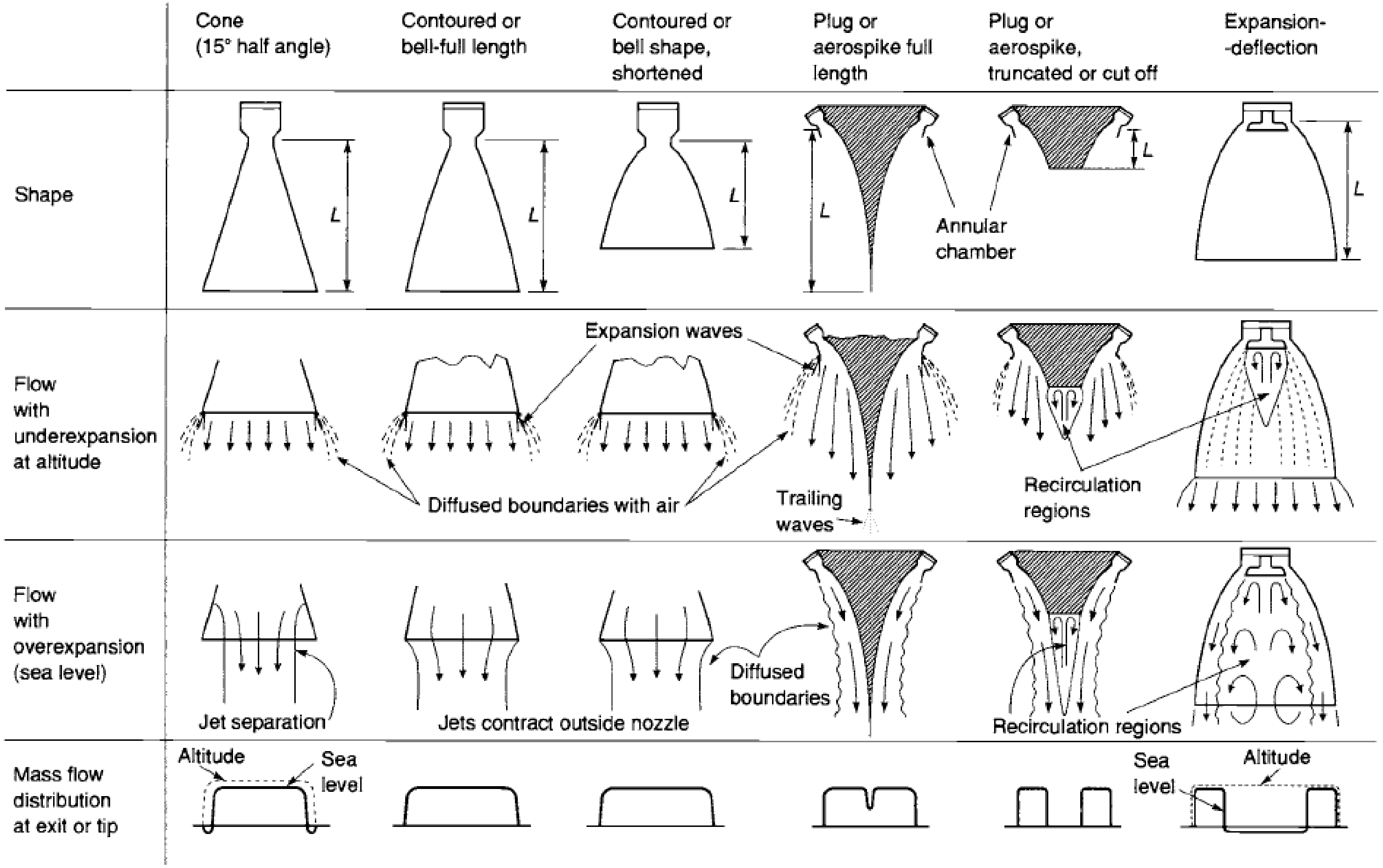
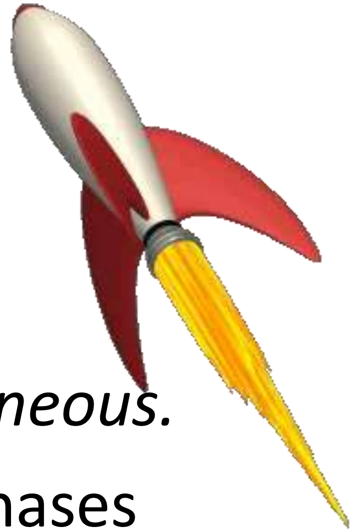


Figure 1. Nozzle types


Cone
Contoured or Bell-Shaped
Plug
Shape
Annular chamber
L
α
D_e
Flow with underexpansion, altitude
Diffuse jet boundary
Exit shock
Expansion fans
Trailing shocks
Lip shock
Flow with overexpansion, (Sea level)
Jet separates
C_F is low
Jet contracts outside of nozzle
Flow clings to plug
Mass flow distribution at exit
Separated jet
Full flow
Low altitude
High altitude


Cone (15° half angle)
Contoured or bell-full length
Contoured or bell shape, shortened
Plug or aerospike full length
Plug or aerospike, truncated or cut off
Expansion--deflection
Shape
L
Annular chamber
Flow with underexpansion at altitude
Expansion waves
Diffused boundaries with air
Trailing waves
Recirculation regions
Flow with overexpansion (sea level)
Jet separation
Jets contract outside nozzle
Diffused boundaries
Recirculation regions
Mass flow distribution at exit or tip
Altitude
Sea level

Rocket performance considerations



1. The working substance (or chemical reaction products) is *homogeneous.*

2. All the species of the working fluid are *gaseous.* Any condensed phases (liquid or solid) add a negligible amount to the total mass.

3. The working substance obeys the *perfect gas law.*

4. There is no *heat transfer* across the rocket walls; therefore, the flow is adiabatic.

5. There is no appreciable *friction* and all *boundary layer* effects are neglected.

Contd..

6. There are no *shock waves* or *discontinuities* in the nozzle flow.

7. The *propellant flow* is *steady* and *constant.* The expansion of the working fluid is uniform and steady, without vibration. Transient effects (i.e., start up and shut down) are of very short duration and may be neglected.

8. All exhaust gases leaving the rocket have an *axially directed velocity.*

9. The gas velocity, pressure, temperature, and density are all uniform across any section normal to the nozzle axis.

10. *Chemical equilibrium* is established within the rocket chamber and the gas composition does not change in the nozzle (frozen flow).

11. Stored propellants are at room temperature. Cryogenic propellants are at their boiling points.

IGNITION SYSTEM IN ROCKETS

- Phase I, Ignition time lag: the period from the moment the igniter receives a signal until the first bit of grain surface burns.
- Phase II, Flame-spreading interval: the time from first ignition of the grain surface until the complete grain burning area has been ignited.
- Phase III, Chamber-filling interval: the time for completing the chamber filling process and for reaching equilibrium chamber pressure and flow

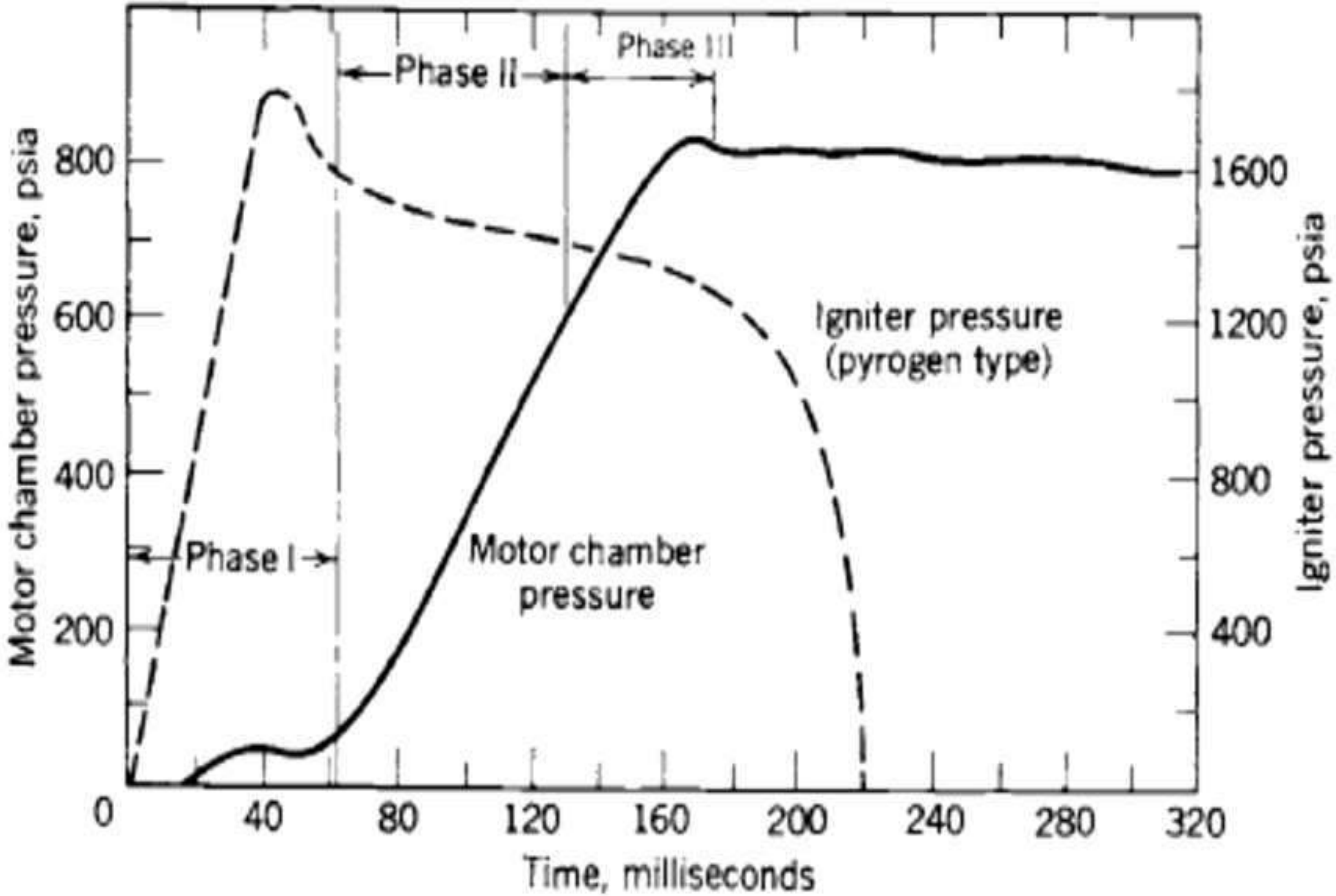
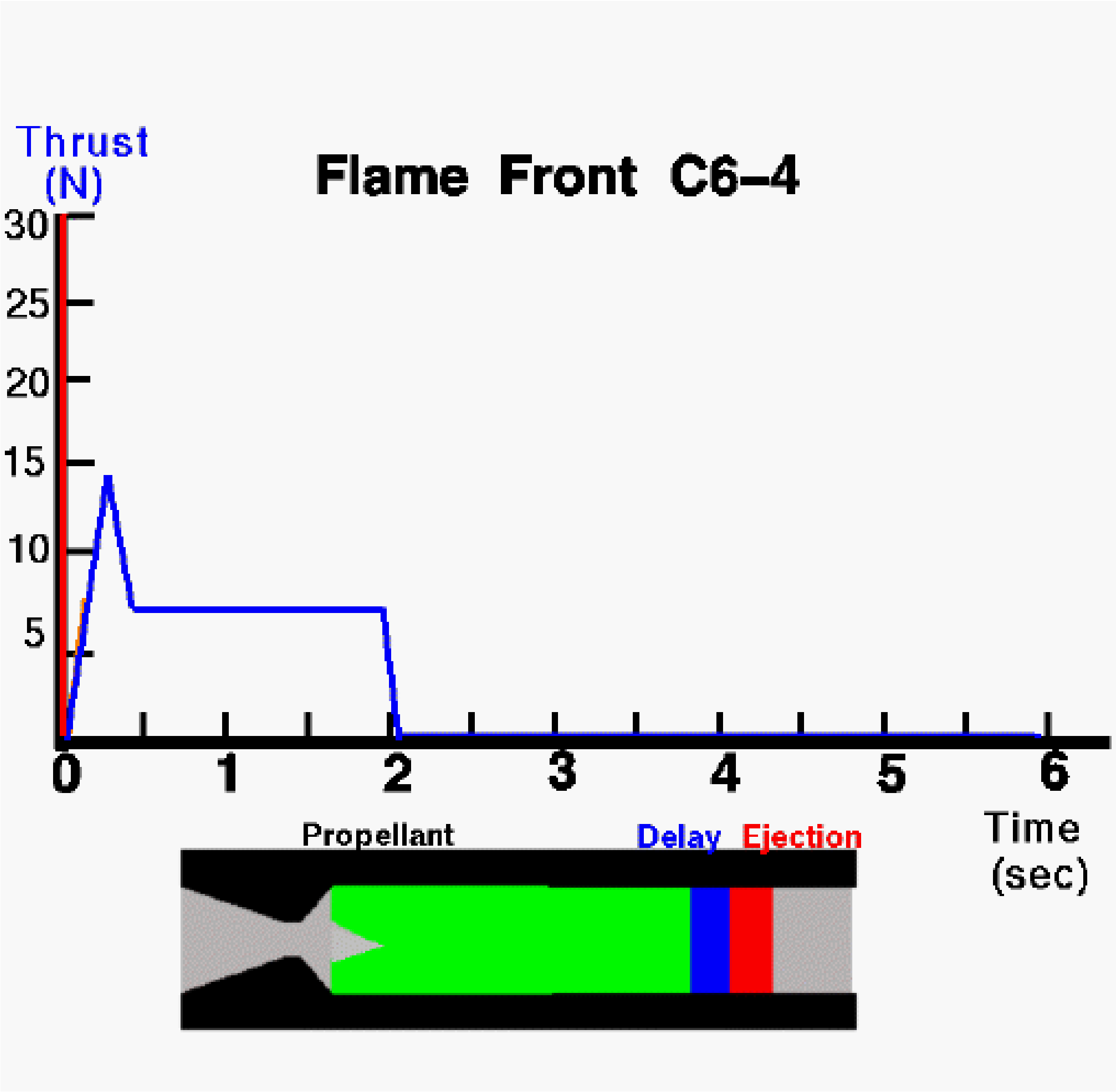
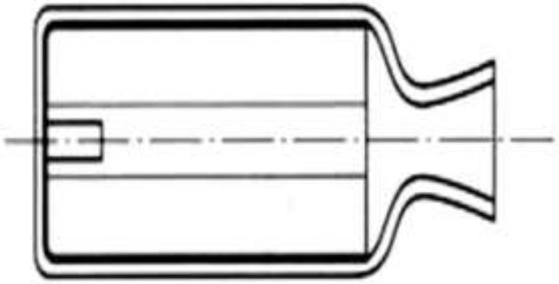


Fig. Typical ignition pressure transient portion of motor chamber pressure time trace with igniter pressure trace and ignition process phases

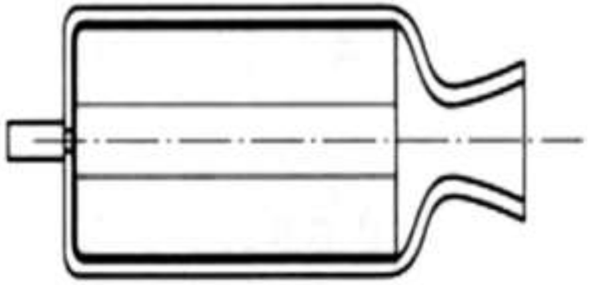


Thrust
(N)
Flame Front C6–4
30
25
20
15
10
5
0
1
2
3
4
5
6
Time
(sec)
Propellant
Delay
Ejection

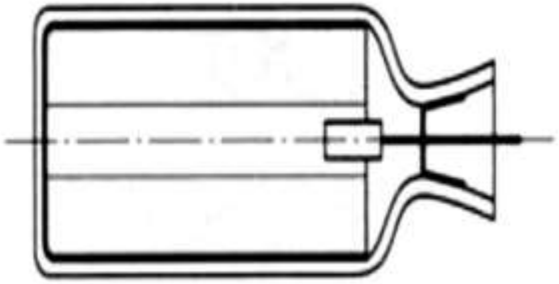
Types of igniters



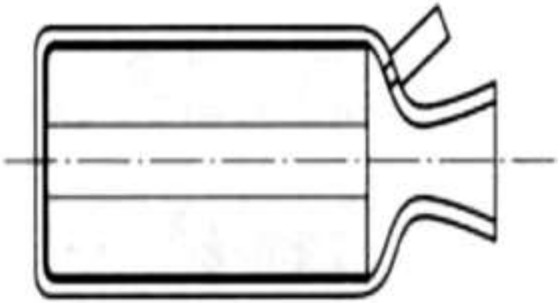
Aft, internal



Aft, external



Forward, internal
(supported by nozzle exit cone)



Forward, external

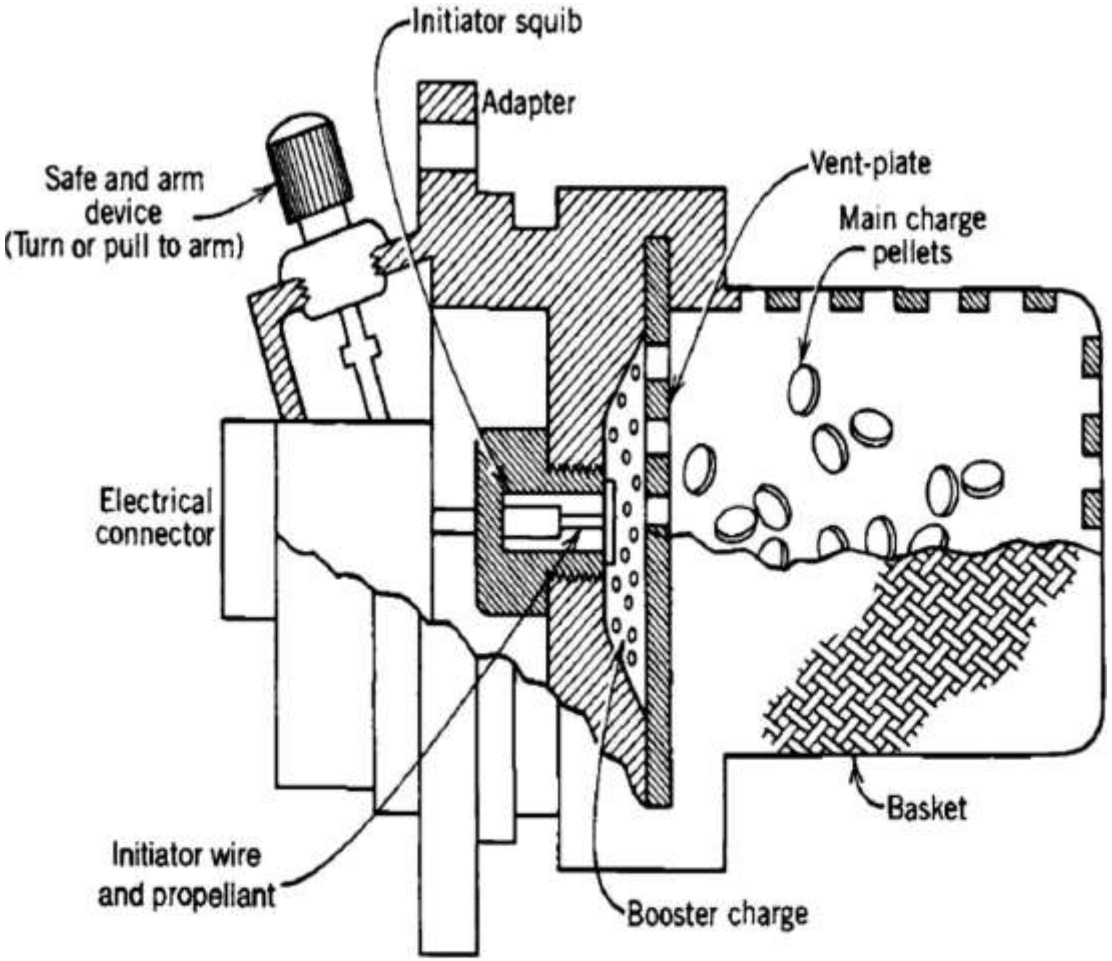
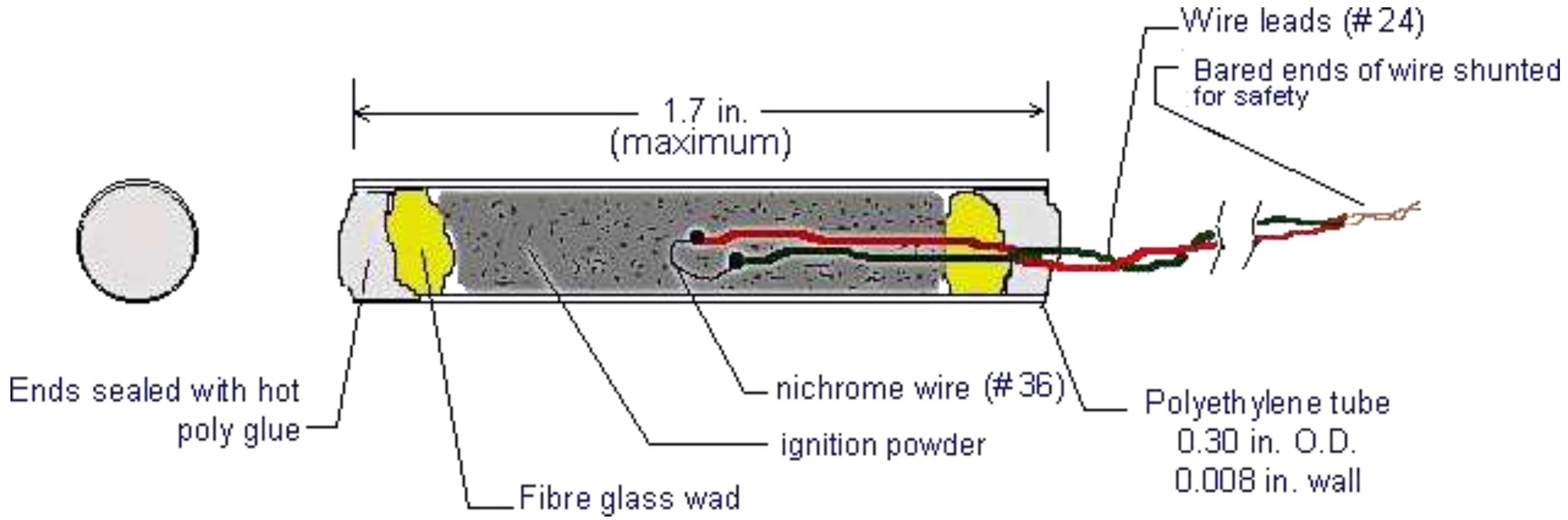


Fig. Typical pyrotechnic igniter with three different propellant charges that ignite in sequence

Pyrotechnic Igniters

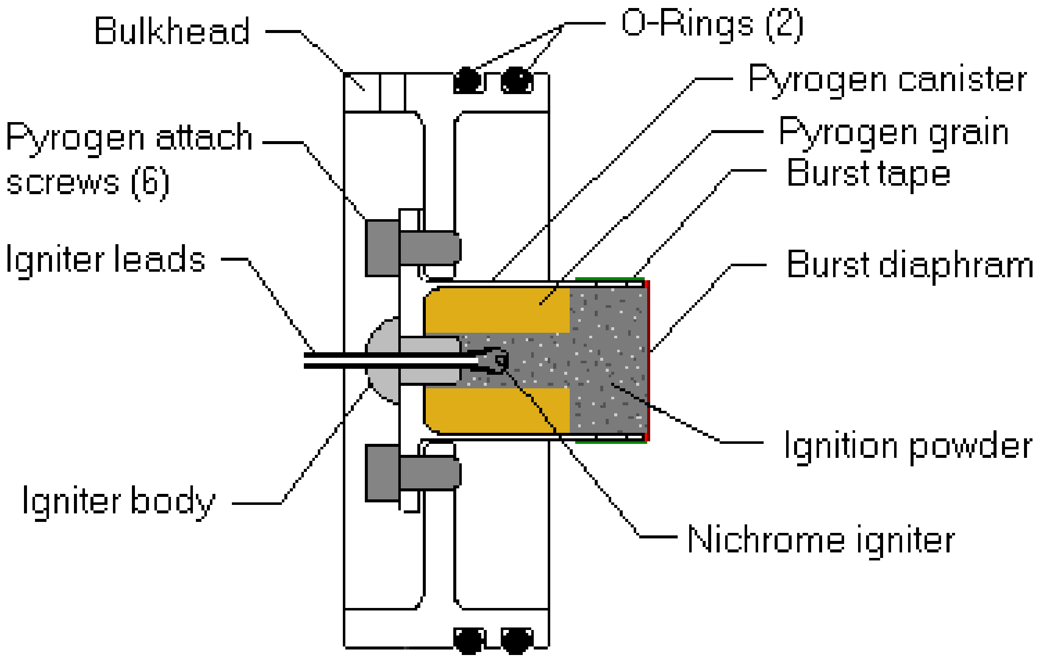
- In industrial practice, pyrotechnic igniters are defined as igniters (other than pyrogen-type igniters as defined further on) using solid explosives or energetic propellant-like chemical formulations (usually small pellets of propellant which give a large burning surface and a short burning time) as the heat-producing material.
- This definition fits a wide variety of designs, known as bag and carbon igniters, powder can, plastic case, pellet basket, perforated tube, combustible case, jellyroll, string, or sheet igniters.

Pyrotechnic Igniter

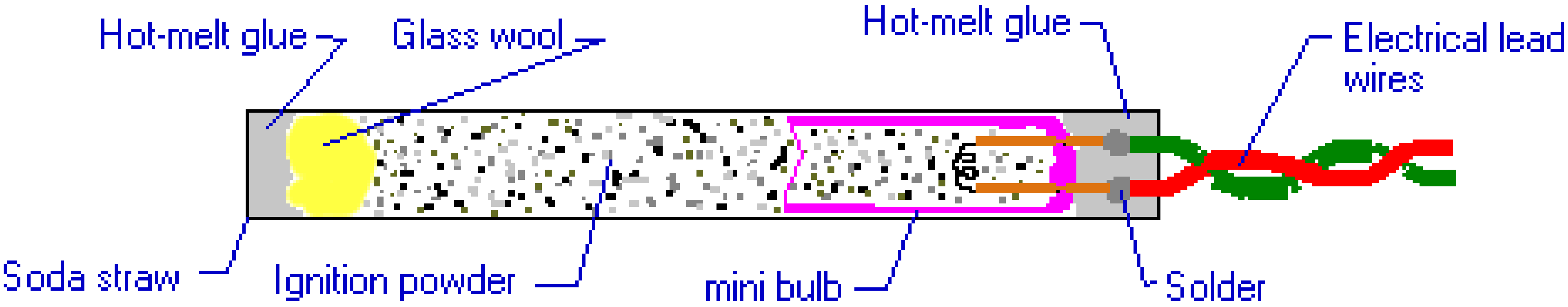


Pyrogen Ignition

- A pyrogen is essentially a small rocket motor mounted at the bulkhead. Nearly instantaneous ignition of the motor grain is assured by the high velocity, particle-laden flame that emanates from the pyrogen. The pyrogen used for the Kappa rocket motor is shown in Figure.



Mini-bulb Igniter



- The igniter described here may be used for either *motor ignition* or for firing a *parachute ejection charge*.
- To make this igniter (shown in Figure), the plastic base of the mini-bulb is first removed (by pulling) and discarded. This exposes the two copper wire leads, which are then scraped clean of oxide, The glass bulb is then carefully broken open. The simplest means is to slowly squeeze the upper half of the bulb in a bench vise. The bulb is first covered with a cloth rag to catch the tiny shards of glass the erupt once the vacuum seal is broken. Safely glasses must be worn during this operation as a redundant safety measure. Care must be taken to prevent damage to the filament bridge wire or to break the lower portion of the bulb.

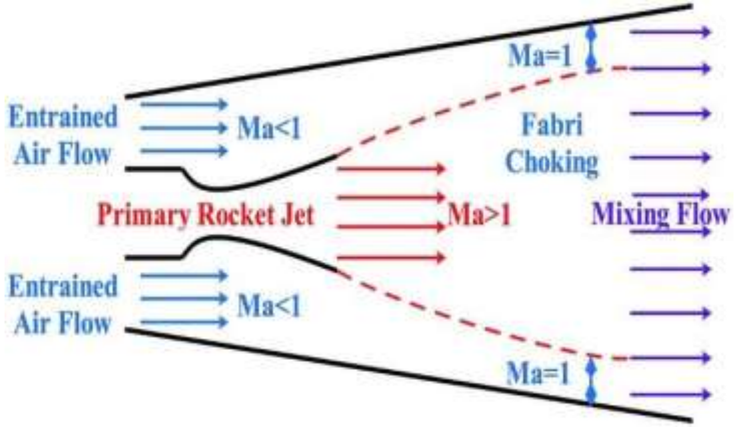
Ultra-low Current Igniter

An igniter that requires very low electrical power, requiring only 20 mA at 1.2V (= 25mW) to fire. As such, this design is exceptionally reliable and especially useful in cold weather operation, which greatly reduces a typical battery's available power. This igniter may be used for either motor ignition or for firing a parachute ejection charge.

The "Ultra-low Current Igniter" was developed for EARco (Experimental Aerospace Research) by Ken Tucker to increase the safety of Rocketry.

Preliminary concepts in nozzle less propulsion

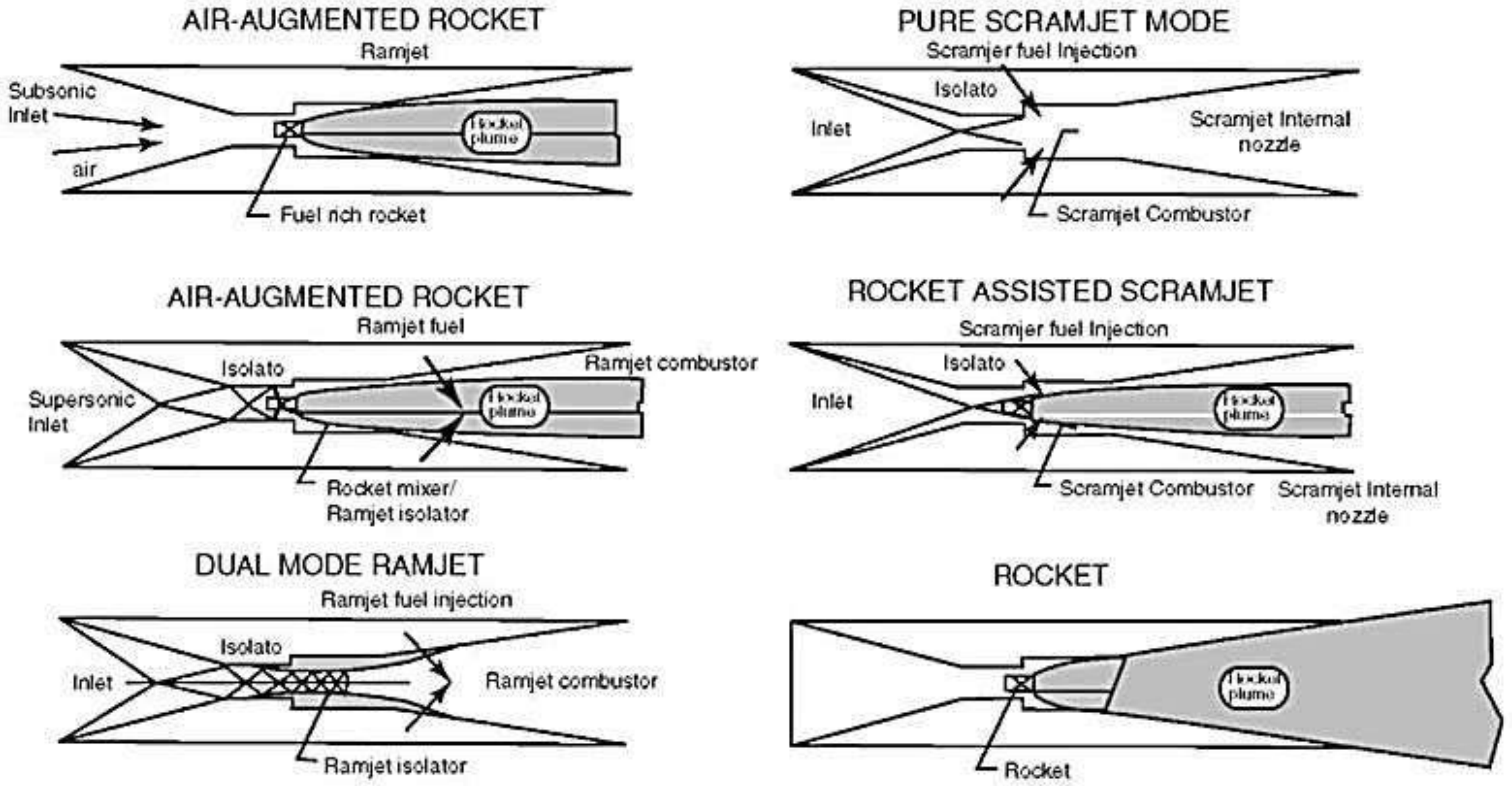
Air augmented rockets



Air-augmented rockets (also known as rocket-ejector, ramrocket, ducted rocket, integral rocket/ramjets, or ejector ramjets) use the supersonic exhaust of some kind of rocket engine to further compress air collected by ram effect during flight to use as additional working mass, leading to greater effective thrust for any given amount of fuel than either the rocket or a ramjet alone.

It represents a hybrid class of rocket/ramjet engines, similar to a ramjet, but able to give useful thrust from zero speed, and is also able in some cases to operate outside the atmosphere, with fuel efficiency not worse than both a comparable ramjet or rocket at every point.

"ROCKET BASED COMBINED CYCLE" ENGINE



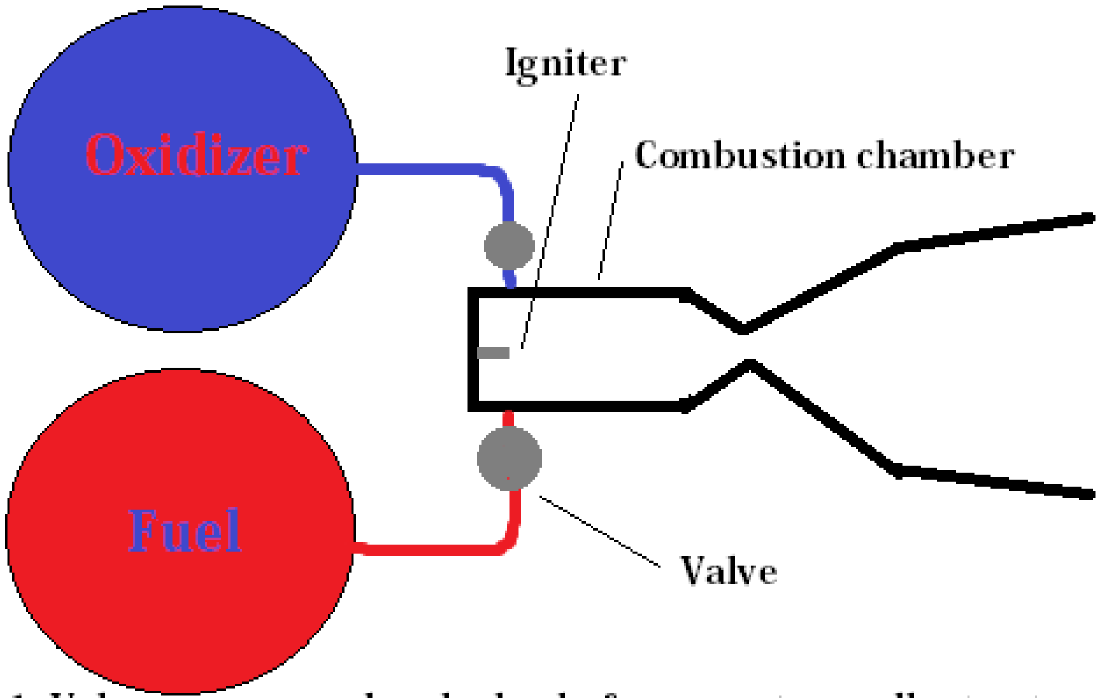
Pulse rocket motors

- A pulsed rocket motor is typically defined as a multiple pulse solid-fuel rocket motor. This design overcomes the limitation of solid propellant motors that they cannot be easily shut down and reignited. The pulse rocket motor allows the motor to be burned in segments (or pulses) that burn until completion of that segment. The next segment (or pulse) can be ignited on command by either an onboard algorithm or in pre-planned phase. All of the segments are contained in a single rocket motor case as opposed to staged rocket motors.[1]

- The pulsed rocket motor is made by pouring each segment of propellant separately. Between each segment is a barrier that prevents the other segments from burning until ignited. At ignition of a second pulse the burning of the propellant generally destroys the barrier.

- The benefit of the pulse rocket motor is that by the command ignition of the subsequent pulses, near optimal energy management of the propellant burn can be accomplished. Each pulse can have different thrust level, burn time, and achieved specific impulse depending on the type of propellant used, its burn rate, its grain design, and the current nozzle throat diameter.[2]

ANIMATION OF A PULSE JET ENGINE

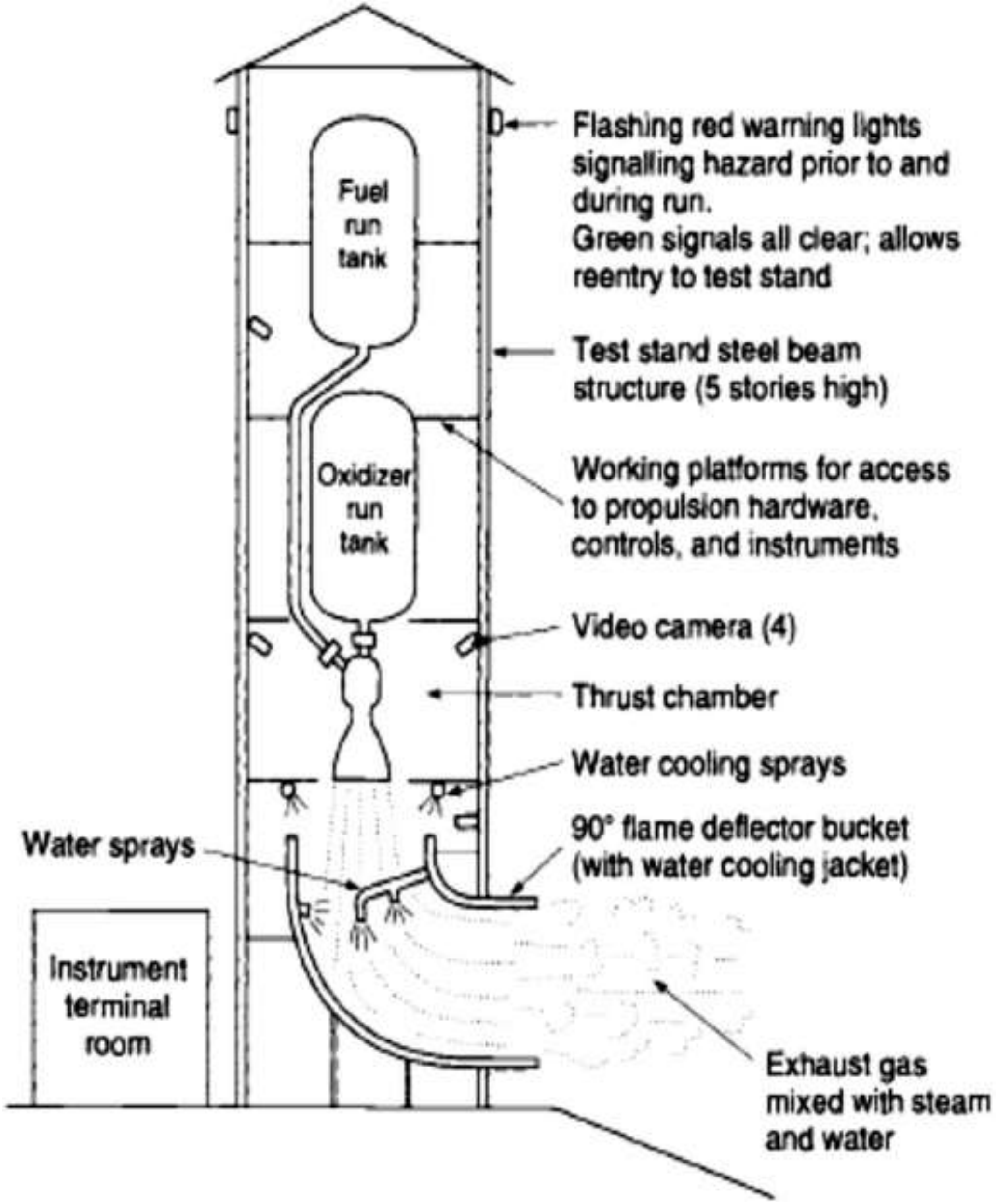


1. Valves are opened and a load of gaseous propellent enters the chamber at a low pressure ~ 3 atm
2. Valves are closed and the propellent is ignited in a high pressue pulse.
3. Valves are opened and a new load of propellent fills the chamber.
4. Valves are closed and the propellent is ignited in a high pressue pulse.
5. The cycle continues.

Static testing of rockets and instrumentation,

For chemical rocket propulsion systems, each test facility usually has the following major systems or components:

6. A test cell or test bay where the article to be tested is mounted, usually in a special test fixture. If the test is hazardous, the test facility must have provisions to protect operating personnel and to limit damage in case of an accident.
7. An instrumentation system with associated computers for sensing, maintaining, measuring, analyzing, correcting, and recording various physical and chemical parameters. It usually includes calibration systems and timers to accurately synchronize the measurements.
8. A control system for starting, stopping, and changing the operating conditions.
9. Systems for handling heavy or awkward assemblies, supplying liquid propellant, and providing maintenance, security, and safety.
10. For highly toxic propellants and toxic plume gases it has been required to capture the hazardous gas.


Fuel run tank
Oxidizer run tank
Flashing red warning lights signalling hazard prior to and during run.
Green signals all clear; allows reentry to test stand
Test stand steel beam structure (5 stories high)
Working platforms for access to propulsion hardware, controls, and instruments
Video camera (4)
Thrust chamber
Water cooling sprays
90° flame deflector bucket (with water cooling jacket)
Water sprays
Instrument terminal room
Exhaust gas mixed with steam and water

Safety considerations

12. Concrete-walled blockhouse or control stations for the protection of personnel and instruments remote from the actual rocket propulsion location.
13. Remote control, indication, and recording of all hazardous operations and measurements; isolation of propellants from the instrumentation and control room.
14. Automatic or manual water deluge and fire-extinguishing systems.
15. Closed circuit television systems for remotely viewing the test.
16. Warning signals (siren, bells, horns, lights, speakers) to notify personnel to clear the test area prior to a test, and an all-clear signal when the conditions are no longer hazardous.
17. Quantity and distance restrictions on liquid propellant tankage and solid propellant storage to minimize damage in the event of explosions; separation of liquid fuels and oxidizers.
18. Barricades around hazardous test articles to reduce shrapnel damage in the event of a blast.
19. Explosion-proof electrical systems, spark-proof shoes, and non-spark hand tools to prevent ignition of flammable materials.
20. For certain propellants also safety clothing , including propellant- and fire-resistant suits, face masks and shields, gloves, special shoes, and hard hats.
21. Rigid enforcement of rules governing area access, smoking, safety inspections, and so forth.
22. Limitations on the number of personnel that may be in a hazardous area at any time.

Instrumentation

1. Forces (thrust, thrust vector control side forces, short thrust pulses).

2. Flows (hot and cold gases, liquid fuel, liquid oxidizer, leakage).

3. Pressures (chamber, propellant, pump, tank, etc.).

4. Temperatures (chamber walls, propellant, structure, nozzle).

5. Timing and command sequencing of valves, switches, igniters, etc.

6. Stresses, strains, and vibrations (combustion chamber, structures, propellant lines, accelerations of vibrating parts).

7. Time sequence of events (ignition, attainments of full pressure).

8. Movement and position of parts (valve stems, gimbal position, deflection of parts under load or heat).Voltages, frequencies, and currents in electrical or control subsystems.

9. Visual observations (flame configuration, test article failures, explosions) using high-speed cameras or video cameras.

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Department of Aeronautical Engineering

(R18)

AIR BREATHING PROPULSION

Lecture Notes

B. Tech III YEAR – II SEM

Prepared by

Dr.S.R.Dhineshkumar
(Professor)
Dept.Aero

SPACE PROPULSION

B.Tech. III Year AE II Sem.LT/P/DC

3 0/0/0 3

Pre-Requisites: Nil

Course Objectives:

- To know about the propulsion system used in rockets and missiles
- Discuss the working principle of solid and liquid propellant rockets and gain basic knowledgeof hybrid rocket propulsion.

Course Outcomes: To be able to

- Understand about trajectory and orbits
- Illustrate electric propulsion techniques, ion and nuclear rocket and the performances ofdifferent advanced propulsion systems.
- Evaluate various space missions, parameters to be considered for designing trajectories androcket mission profiles
- Understand the fundamentals of chemical rocket propulsion, types of igniters and performance considerations of rockets.

UNIT - I

Principles of Rocket Propulsion: History of rockets, Newtons third law, orbits and space flight, types of orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO, launch assists.

UNIT - II

Fundamentals of Rocket Propulsion: Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Rocket performance considerations of rockets, types of igniters, preliminary concepts in nozzle less propulsion, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.

UNIT - III

Solid Rocket Propulsion: Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Erosive burning in solid propellant rockets, combustion instability, strand burner and T-burner, applications and advantages of solid propellant rockets.

UNIT - IV

Liquid and Hybrid Rocket Propulsion: Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.

UNIT-V

Advanced Propulsion Techniques: Electric rocket propulsion, types of electric propulsion techniques, Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemical rocket propulsion systems, future applications of electric propulsion systems, Solar sail.

TEXT BOOKS:

1. Hill, P.G. and Peterson, C.R., ―Mechanics and Thermodynamics of Propulsion‖, 2nd Edition, Addison Wesley, 1992.
2. Turner, M. J. L., ―Rocket and Spacecraft Propulsion‖, 2nd Edition, MIT Press, 1922.
3. Hieter and Pratt, ―Hypersonic Air breathing propulsion‖ 5th Edition, 1993.

REFERENCE BOOKS:

1. Sutton, G.P., ―Rocket Propulsion Elements‖ John Wiley & Sons Inc., New York, 5th Edition, 1993.
2. Mathur, M. L., and Sharma, R.P., ―Gas Turbine, Jet and Rocket Propulsion‖, Standard Publishers and Distributors, Delhi, 1988, Tajmar, M., Advanced Space Propulsion Systems, Springer 2003

UNIT-3

INTRODUCTION

- Chemical propellants come in two forms, solid and liquid. The solid propellant systems are usually referred to as motors and the liquid propellant systems are called engines. Determining what propellants to use is based on the particular application for its use, cost consideration and safety aspects.

- Solid-propellant motors have faster reaction times but are limited, after construction, to only minor adjustments to the rate of thrust and cannot be reignited after shutdown.

- Liquid-propellant engines produce greater thrust per unit mass. They can be easily stopped, restarted, and adjusted during flight. Most liquidpropellant rockets are very hazardous, difficult to store, and are difficult to use safely.

Solid Rocket Motors

A solid rocket motor is a system that uses
solid propellants to produce thrust

Advantages

- High thrust
- Simple
- Storability
- High density Isp

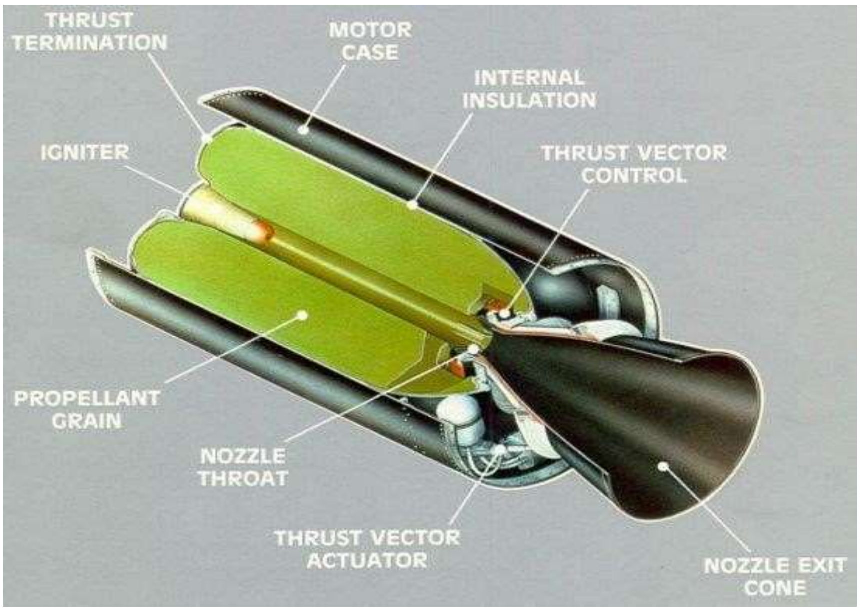
Disadvantages

- Low Isp (compared to liquids)
- Complex throttling
- Difficult to stop and restart
- Safety

Solid Rocket Motors

Solid rocket motors are used for

- Launch vehicles
- High thrust (high F/W ratio)
- High storage density
- Ballistic Missiles
- Propellant storability
- Excellent aging
- Quick response
- storability
- high F/W ratio)



Solid Rocket Motor ComponentsThermal Insulation

- Design involves:
- Analysis of combustion chamber environment
- Stagnation temperature
- Stagnation pressure
- Propellant gases (material compatibility)
- Selection of insulation material
- Material thickness determination for various areas of the motor case
- For the cylindrical part of the case, the walls are only exposed to hot combustion gases at the end of the burn

The Nozzle

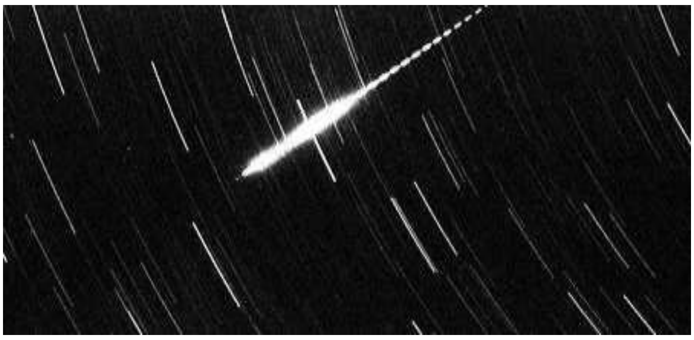
The design of the nozzle follows similar steps as for other thermodynamic rockets

- Throat area determined by desired stagnation pressure and thrust level
- Expansion ratio determined by ambient pressure or pressure range to allow maximum efficiency
- Major difference for solid propellant nozzles is the technique used for cooling
- Ablation
- Fiber reinforced material used in and near the nozzle throat (carbon, graphite, silica, phenolic)

Ablation

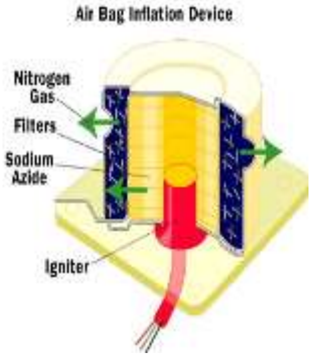
Vyasapuri, Bandlaguda, Post:Keshavgiri
Hyderabad-500005,Telangana, India
Tel:040-29880079, 86,8978380692, 9642703342
9652216001, 9550544411, Website:www.mist.ac.in
Email:principal@mist.ac.in
principal.mahaveer@gmail.com
Counseling code:MHVR, University Code:E3

- Meteorite
- Re-entry speed of 10 - 20 km/sec
- Extreme heating in the atmosphere
- Ablation and internal energy modes cooled the meteorite through its fall
- Ablation gas cloud
- Dissociation
- Internal energy deposition
- Stony-Iron Classification
- (95% of all meteorites)



Ignition System

- Large solid motors typically use a three-stage ignition system
- Initiator: Pyrotechnic element that converts electrical impulse into a chemical reaction (primer)
- Booster charge
- Main charge: A charge (usually a small solid motor) that ignites the propellant grain. Burns for tenths of a second with a mass flow about 1/10 of the initial propellant grain mass flow.
-



Propellant Grain

Two main categories

- Double Base: A homogeneous propellant grain, usually nitrocellulose dissolved in nitroglycerin. Both ingredients are explosive and act as a combined fuel, oxidizer and binder
- Composite: A heterogeneous propellant grain with oxidizer crystals and powdered fuel held together in a matrix of synthetic rubber binder.

Less hazardous to manufacture and handle

Conventional Composite

- Fuel
- 5-22% Powdered Aluminum
- Oxidizer
- 65-70% Ammonium Perchlorate (NH_4ClO_4 or AP)
- Binder
- 8-14% Hydroxyl- Terminated Polybutadiene (HTPB)

Fuels

- Aluminum (Al)
- Molecular Weight: 26.98 kg/kmol
- Density: 2700 kg/m^3
- Most commonly used
- Magnesium (Mg)
- Molecular Weight: 24.32 kg/kmol
- Density: 1750 kg/m^3
- Clean burning (green)
- Beryllium (Be)
- Molecular Weight: 9.01 kg/kmol
- Density: 2300 kg/m^3
- Most energetic, but extremely toxic exhaust products

Oxidizers

- Ammonium Perchlorate (AP)
- Most commonly used
- Cl combining with H can form HCl
    - Toxic
- Depletion of ozone
- Ammonium Nitrate (AN)

- Next most commonly used
- Less expensive than AP
- Less energetic
- No hazardous exhaust products

Binders

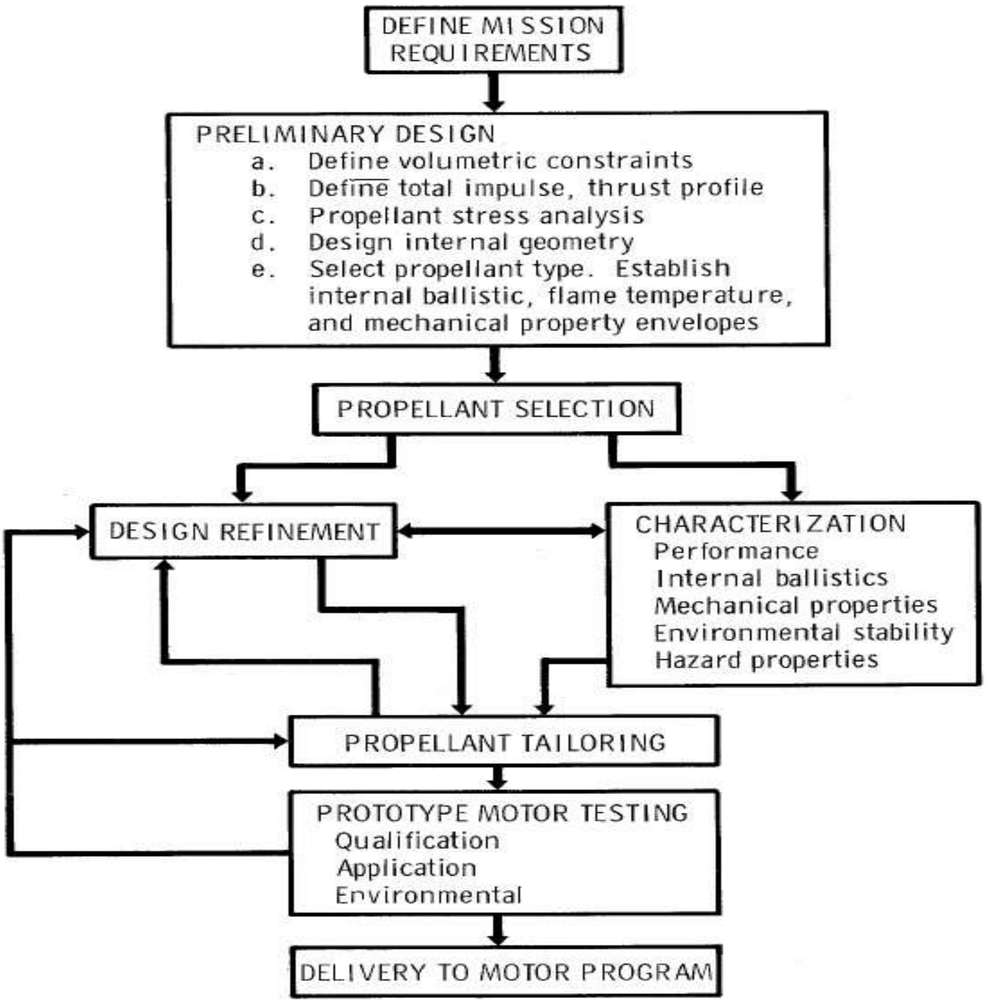
- Hydroxyl Terminated Polybutadiene (HTPB)
- Most commonly used
- Consistency of tire rubber
- Polybutadiene Acrylonitrile (PBAN)
- Nitrocellulose (PNC)
- Double base agent

Additives

- Used to promote
- Curing
- Enhanced burn rate (HMX)
- Bonding
- Reduced radiation through the grain (darkening) • Satisfactory aging
- Reduced cracking

Solid Propellant Motors

- Solid propellant motors are the simplest of all rocket designs. They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust.
- When ignited, a solid propellant burns from the center out towards the sides of the casing.
- The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust.
- Unlike liquid propellant engines, solid propellant motors cannot be shut down.
- Once ignited, they will burn until all the propellant is exhausted.



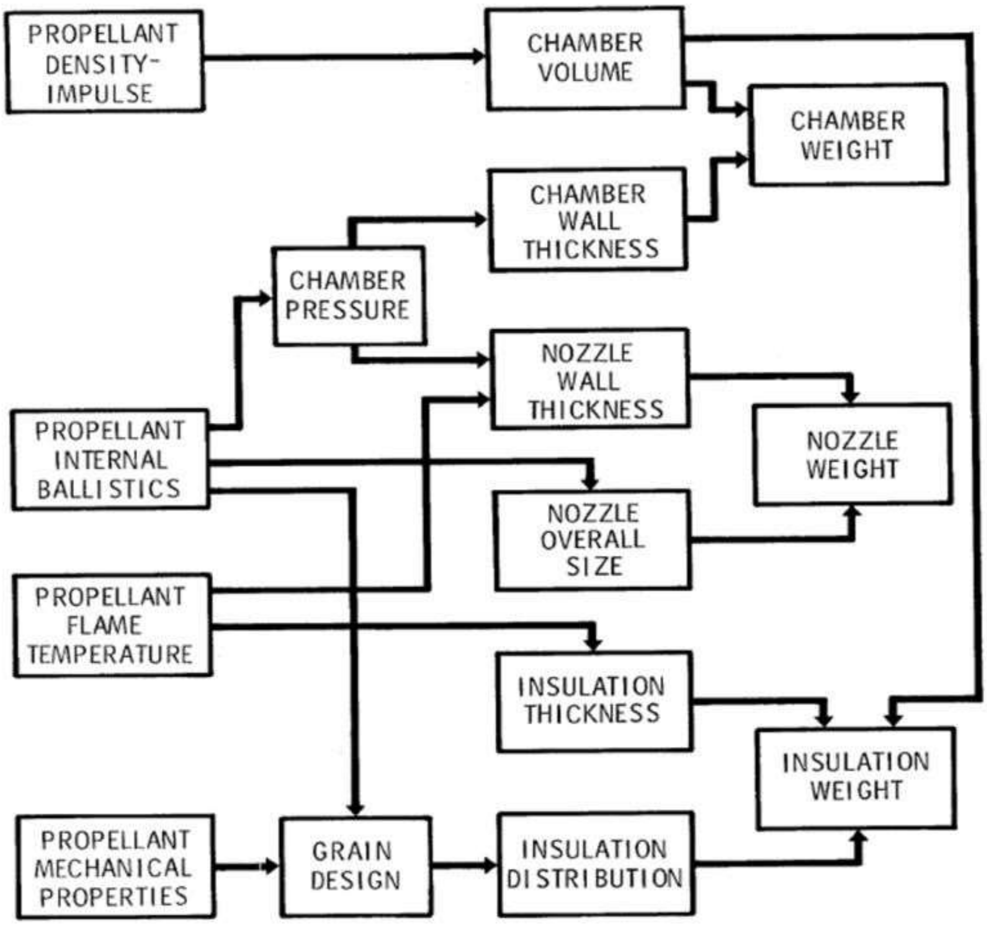


Figure 3.2 influence of propellant properties

ESTIMATION OF PROPELLANT ADIABATIC FLAME TEMPERATURE

- For a combustion process that takes place adiabatically with no shaft work, the temperature of the products is referred to as the adiabatic flame temperature. This is the maximum temperature that can be achieved for given reactants.

- Heat transfer, incomplete combustion, and dissociation all result in lower temperature. The maximum adiabatic flame temperature for a given fuel and oxidizer combination occurs with a stoichiometric mixture (correct proportions such that all fuel and all oxidizer are consumed).

- The amount of excess air can be tailored as part of the design to control the adiabatic flame temperature. The considerable distance between present temperatures in a gas turbine engine and the maximum adiabatic flame temperature at stoichiometric conditions is shown in Figure, based on a compressor exit temperature of (922 K).

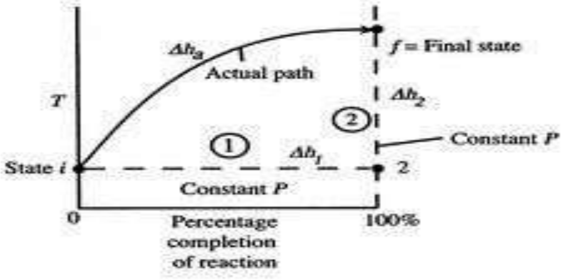


Fig. Schematic of adiabatic flame temperature

Propellant grain design considerations:

- Add a burning rate catalyst, often called burning rate modifier (0.1 to 3.0% of propellant) or increase percentage of existing catalyst.

- Decrease the oxidizer particle size.

- Increase oxidizer percentage.

- Increase the heat of combustion of the binder and/or the plasticizer.

- Imbed wires or metal staples in the propellant.

PROPELLANT GRAIN DESIGN CONSIDERATIONS

- The rocket motor's operation and design depend on the combustion characteristics of the propellant, its burning rate, burning surface, and grain geometry. The branch of applied science describing these is known as internal ballistics;

- The burning surface of a propellant grain recedes in a direction essentially perpendicular to the surface. The rate of regression, usually expressed in cm/sec, mm/sec, or in/sec, is the burning rate r.

- In Fig. we can visualize the change of the grain geometry by drawing successive burning surfaces with a constant time interval between adjacent surface contours. Figure. shows this for a two-dimensional grain with a central cylindrical cavity with five slots.

- Success in rocket motor design and development depends significantly on knowledge of burning rate behavior of the selected propellant under all motor operating conditions and design limit conditions.

Burning rate is a function of the propellant composition. For composite propellants it can be increased by changing the propellant characteristics:

Add a burning rate catalyst, often called burning rate modifier (0.1 to 3.0% of propellant) or increase percentage of existing catalyst.

Decrease the oxidizer particle size.

Increase oxidizer percentage.

Increase the heat of combustion of the binder and/or the plasticizer.

Imbed wires or metal staples in the propellant.

- Aside from the propellant formulation and propellant manufacturing process, burning rate in a full-scale motor can be increased by the following:

  Combustion chamber pressure.

  Initial temperature of the solid propellant prior to start.

  Combustion gas temperature.

  Velocity of the gas flow parallel to the burning surface.

  Motor motion (acceleration and spin-induced grain stress).

- The grain is the shaped mass of processed solid propellant inside the rocket motor. The propellant material and geometrical configuration of the grain determine the motor performance characteristics. The propellant grain is a cast, molded, or extruded body and its appearance and feel is similar to that of hard rubber or plastic.

- There are two methods of holding the grain in the case, as seen in Fig. Cartridgeloaded or freestanding grains are manufactured separately from the case (by extrusion or by casting into a cylindrical mold or cartridge) and then loaded into or assembled into the case. In case-bonded grains the case is used as a mold and the propellant is cast directly into the case and is bonded to the case or case insulation. Free-standing grains can more easily be replaced loaded) and a casebonded grain.

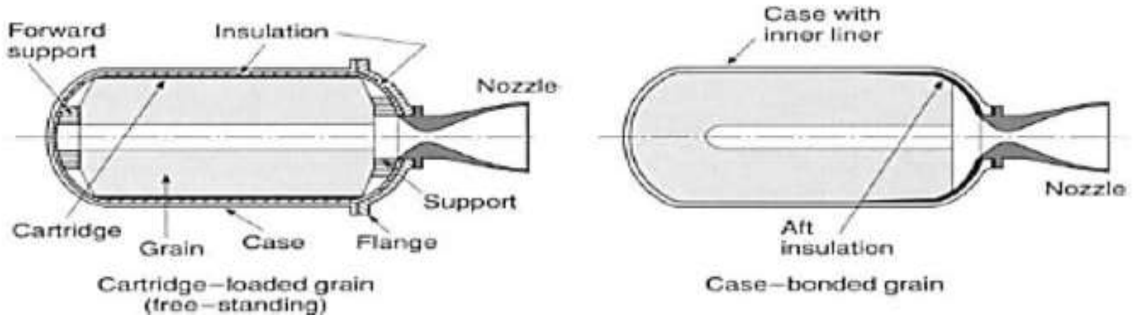


Figure 3.3 Simplified schematic diagrams of a free-standing (or cartridge-loaded) and a case-bonded grain.

EROSIVE BURNING IN SOLID PROPELLANT ROCKETS

Definitions and terminology important to grains include:

- Configuration: The shape or geometry of the initial burning surfaces of a grain as it is intended to operate in a motor.

- Cylindrical Grain: A grain in which the internal cross section is constant along the axis regardless of perforation shape.

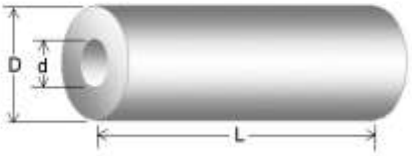
- Neutral Burning: Motor burn time during which thrust, pressure, and burning surface area remain approximately constant, typically within about +15%. Many grains are neutral burning.

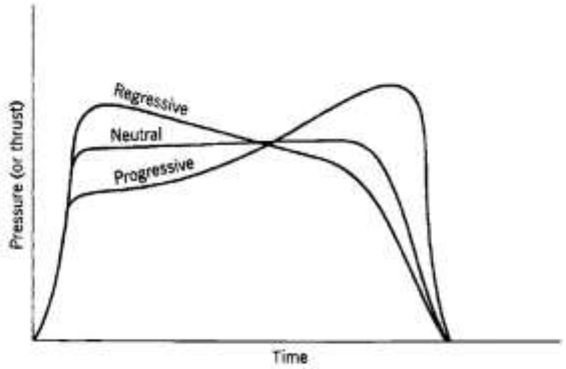
- Perforation: The central cavity port or flow passage of a propellant grain; its cross section may be a cylinder, a star shape, etc.

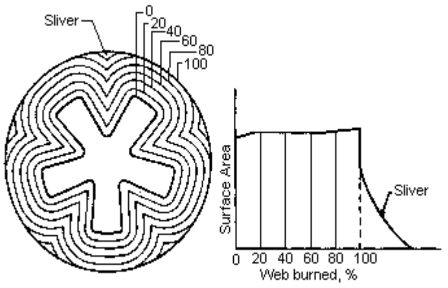
- Progressive Burning: Burn time during which thrust, pressure, and burning surface area increase.

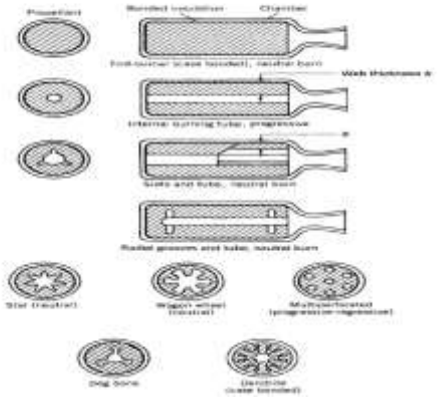
- Regressive Burning: Burn time during which thrust, pressure, and burning surface area decrease

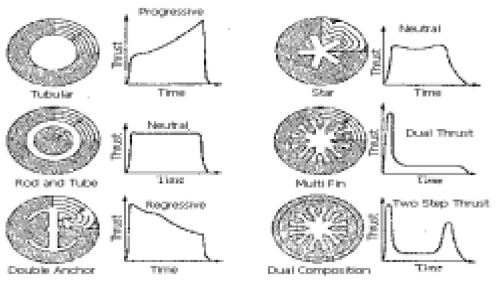
- Sliver: Unburned propellant remaining (or lost--that is, expelled through the nozzle) at the time of web burnout.











- The shape of the fuel block for a rocket is chosen for the particular type of mission it will perform. Since the combustion of the block progresses from its free surface, as this surface grows, geometrical considerations determine whether the thrust increases, decreases or stays constant.

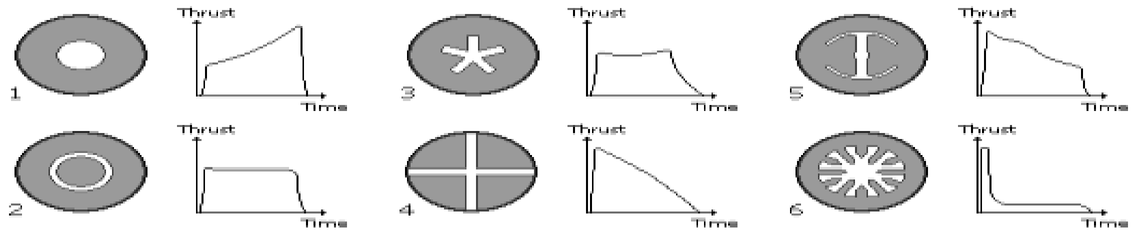


Figure 1.14

- Fuel blocks with a cylindrical channel (1) develop their thrust progressively. Those with a channel and also a central cylinder of fuel (2) produce a relatively constant thrust, which reduces to zero very quickly when the fuel is used up. The five pointed star profile (3) develops a relatively constant thrust which decreases slowly to zero as the last of the fuel is consumed. The 'cruciform' profile (4) produces progressively less thrust. Fuel in a block with a 'double anchor' profile (5) produces a decreasing thrust which drops off quickly near the end of the burn. The 'cog' profile (6) produces a strong inital thrust, followed by an almost constant lower thrust.

A grain has to satisfy several interrelated requirements:

From the flight mission one can determine the rocket motor requirements. They have to be defined and known before the grain can be designed. They are usually established by the vehicle designers. This can include total impulse, a desired thrust-time curve and a tolerance thereon, motor mass, ambient temperature limits during storage and operation, available vehicle volume or envelope, and vehicle accelerations caused by vehicle forces (vibration, bending, aerodynamic loads, etc.).

The grain geometry is selected to fit these requirements; it should be compact and use the available volume efficiently, have an appropriate burn surface versus time profile to match the desired thrust-time curve, and avoid or predictably control possible erosive burning. The remaining unburned propellant slivers, and often also the shift of the center of gravity during burning, should be minimized.

This selection of the geometry can be complex.

The propellant is usually selected on the basis of its performance capability (e.g., characteristic velocity), mechanical properties (e.g., strength), ballistic properties (e.g., burning rate), manufacturing characteristics, exhaust plume characteristics, and aging properties. If necessary, the propellant formulation may be slightly altered or "tailored" to fit exactly the required burning time or grain geometry.

The structural integrity of the grain, including its liner and/or insulator, must be analyzed to assure that the grain will not fail in stress or strain under all conditions of loading, acceleration, or thermal stress. The grain geometry can be changed to reduce excessive stresses.

The complex internal cavity volume of perforations, slots, ports, and fins increases with burning time. These cavities need to be checked for resonance, damping, and combustion stability.

The processing of the grain and the fabrication of the propellant should be simple and low cost.

COMBUSTION INSTABILITY

There seem to be two types of combustion instability:

- A set of acoustic resonances or pressure oscillations, which can occur with any rocket motor, and a vortex shedding phenomenon, which occurs only with particular types of grains.

STRAND BURNER AND T-BURNER

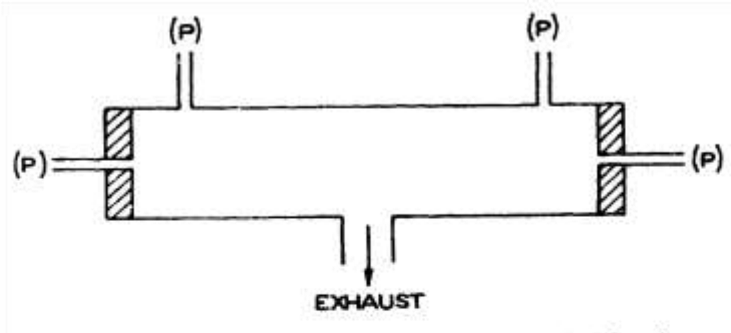
- In contrast with liquid rocket technology, an accepted combustion stability rating procedure does not now exist for full-scale solid rockets.

- Undertaking stability tests on large full-scale flight-hardware rocket motors is expensive, and therefore lower-cost methods, such as subscale motors, T-burners, and other test equipment, have been used to assess motor stability.

- The best known and most widely used method of gaining combustion stability- related data is the use of a T-burner, an indirect, limited method that does not use a full-scale motor. Standard T-burner has a 1.5-in. internal diameter double-ended cylindrical burner vented at its midpoint.

T Burner

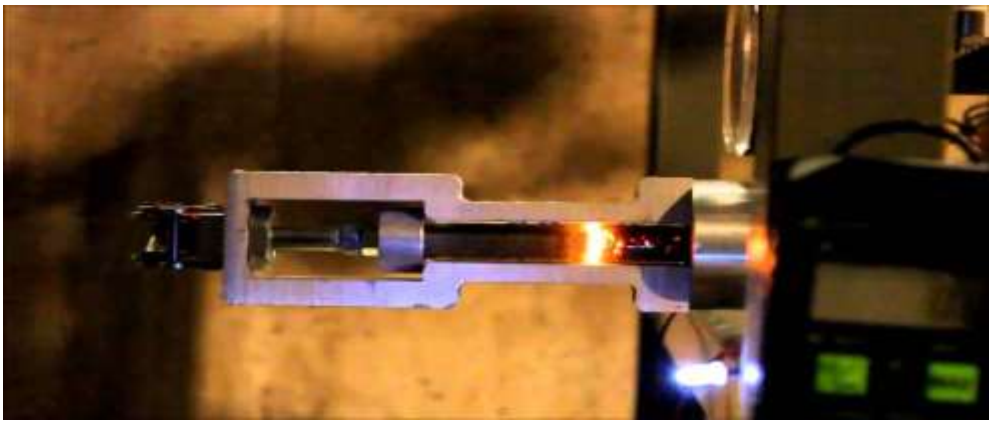
• Sketches of Typical T-Burners



(a) Simple end burner for pulse testing; (P) denotes typical locations of pulsers or pressure transducers.

Once instability has been observed or predicted in a given motor, the motor design has to fix the problem. There is no sure method for selecting the right remedy, and none of the cures suggested below may work. The usual alternatives are:

- Changing the grain geometry to shift the frequencies away from the undesirable values. Sometimes, changing fin locations, port cross-section profile, or number of slots has been successful.
- Changing the propellant composition. Using aluminum as an additive has been most effective in curing transverse instabilities, provided that the particle-size distribution of the aluminum oxide is favorable to optimum damping at the distributed frequency. Changing size distribution and using other particulates (Zr, A1203, or carbon particles) has been effective in some cases. Sometimes changes in the binder have worked.
- Adding some mechanical device for attenuating the unsteady gas motions or changing the natural frequency of cavities. Various inert resonance rods, baffles, or paddles have been added, mostly as a fix to an existing motor with observed instability. They can change the resonance frequencies of the cavities, introduce additional viscous surface losses, but also cause extra inert mass and potential problems with heat transfer or erosion.



APPLICATIONS AND ADVANTAGES OF SOLID PROPELLANT ROCKETS

Category	Application	Typical Characteristics
Large booster and second-stage motors	Space launch vehicles; lower stages of long-range ballistic missiles (see Figs. 11–2 and 14–2)	Large diameter (above 48 in.); L/D of case = 2 to 7; burn time t = 60 to 120 sec; low-altitude operations with low nozzle area ratios (6 to 16)
High-altitude motors	Upper stages of multistage ballistic missiles, space launch vehicles; space maneuvers	High-performance propellant; large nozzle area ratio (20 to 200); L/D of case = 1 to 2; burn time t = 40 to 120 sec (see Fig. 11–3)
Tactical missiles	1. High acceleration: short-range bombardment, antitank missile	Tube launched, L/D = 4 to 13; very short burn time (0.25 to 1 sec); small diameter (2.75 to 18 in.); some are spin stabilized
	2. Modest acceleration: air-to-surface, surface-to-air, short-range guided surface-to-surface, and air-to-air missiles	Small diameter (5 to 18 in.); L/D of case = 5 to 10; usually has fins and/or wings; thrust is high at launch and then is reduced (boost-sustain); many have blast tubes (see Fig. 11–4); wide ambient temperature limits: sometimes minimum temperature −65° F or −53°C, maximum temperature +160°F or +71°C; usually high acceleration; often low-smoke or smokeless propellant
Ballistic missile defense	Defense against long- and medium-range ballistic missiles	Booster rocket and a small upper maneuverable stage with multiple attitude control nozzles and one or more side or divert nozzles
Gas generator	Pilot emergency escape; push missiles from submarine launch tubes or land mobile cannisters; actuators and valves; short-term power supply; jet engine starter; munition dispersion; rocket turbine drive starter; automotive air bags	Usually low gas temperature (< 1300°C); many different configurations, designs, and propellants; purpose is to create high-pressure, energetic gas rather than thrust

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ESTD : 2001

Department of Aeronautical Engineering

(R18)

AIR BREATHING PROPULSION

Lecture Notes

B. Tech III YEAR – II SEM

Prepared by

Dr.S.R.Dhineshkumar
(Professor)Dept.Aero

SPACE PROPULSION

B.Tech. III Year AE II Sem.LT/P/DC

3 0/0/0 3

Pre-Requisites: Nil

Course Objectives:

- To know about the propulsion system used in rockets and missiles
- Discuss the working principle of solid and liquid propellant rockets and gain basic knowledgeof hybrid rocket propulsion.

Course Outcomes: To be able to

- Understand about trajectory and orbits
- Illustrate electric propulsion techniques, ion and nuclear rocket and the performances ofdifferent advanced propulsion systems.
- Evaluate various space missions, parameters to be considered for designing trajectories androcket mission profiles
- Understand the fundamentals of chemical rocket propulsion, types of igniters andperformance considerations of rockets.

UNIT - I

Principles of Rocket Propulsion: History of rockets, Newtons third law, orbits and space flight, typesof orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO,launch assists.

UNIT - II

Fundamentals of Rocket Propulsion: Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Rocket performance considerations of rockets, types of igniters, preliminary concepts in nozzle less propulsion, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.

UNIT - III

Solid Rocket Propulsion: Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Erosive burning in solid propellant rockets, combustion instability, strand burner and T- burner, applications and advantages of solid propellant rockets.

UNIT - IV

Liquid and Hybrid Rocket Propulsion: Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.

UNIT-V

Advanced Propulsion Techniques: Electric rocket propulsion, types of electric propulsion techniques, Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemical rocket propulsion systems, future applications of electric propulsion systems, Solar sail.

TEXT BOOKS:

1. Hill, P.G. and Peterson, C.R., ―Mechanics and Thermodynamics of Propulsion‖, 2nd Edition, Addison Wesley, 1992.
2. Turner, M. J. L., ―Rocket and Spacecraft Propulsion‖, 2nd Edition, MIT Press, 1922.
3. Hieter and Pratt, ―Hypersonic Air breathing propulsion‖ 5th Edition, 1993.

REFERENCE BOOKS:

1. Sutton, G.P., ―Rocket Propulsion Elements‖ John Wiley & Sons Inc., New York, 5th Edition, 1993.
2. Mathur, M. L., and Sharma, R.P., ―Gas Turbine, Jet and Rocket Propulsion‖, Standard Publishers and Distributors, Delhi, 1988, Tajmar, M., Advanced Space Propulsion Systems, Springer 2003

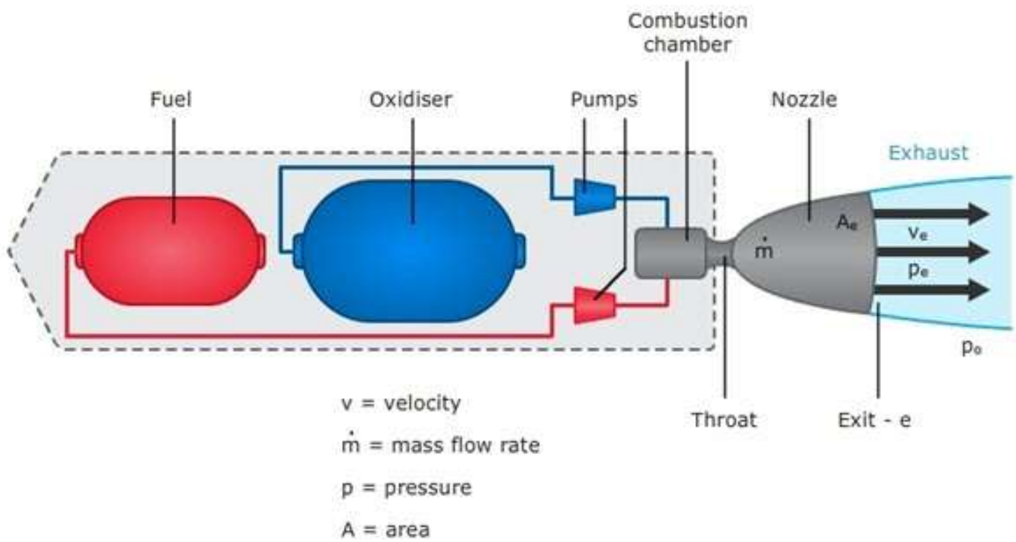
UNIT-1V

Liquid and Hybrid Rocket Propulsion

What Are Liquid Propellant Rockets?

A liquid propellant rocket propulsion system is commonly called a rocket engine. It has all the hardware components and propellants necessary for its operation, that is, for producing thrust. It consists of one or more thrust chambers, one or more tanks to store the propellants, a feed mechanism to force the propellants from the tanks into the thrust chamber(s), a power source to furnish the energy for the feed mechanism, suitable plumbing or piping to transfer the liquids, a structure to transmit the thrust force, and control devices to initiate and regulate the propellant flow and thus the thrust.

In some applications an engine may also include a thrust vector control system, various instrumentation and residual propellant (trapped in pipes, valves, or wetting tank walls). It does not include hardware for non-propulsive purposes, such as aerodynamic surfaces, guidance, or navigation equipment, or the useful payload, such as a scientific space exploration package or a missile warhead.



Salient Features Of Liquid Propellant Rockets:

The design of any propulsion system is tailored to fit a specific application or mission requirement. These requirements are usually stated in terms of the application (anti-aircraft rocket, upper stage launch vehicle propulsion, or projectile assist), mission velocity, the desired flight trajectories (surface launch, orbit transfer, altitude-performance profile), vulnerability, attitude control torques and duty cycle, minimum life (during storage or in orbit), or number of units to be built and delivered. They include constraints on cost, schedule, operating conditions (such as temperature limits), storage conditions, or safety rules. The **mission requirements** can be translated into rocket engine requirements in terms of thrust-time profile, propellants, number of thrust chambers, total impulse, number of restarts, minimum reliability, likely propellant, and engine masses and their sizes or envelopes. We can do this only if we select several of the key engine features, such as the feed system, chamber pressure, the method of cooling the thrust chambers, thrust modulation (restart, throttle, thrust vector control), engine cycle (if using turbopump feed), and other key design features

Many different types of rocket engines have been built and flown, ranging in thrust size from less than 0.01 lbf to over 1.75 million pounds, with one-time operation or multiple starts some have over 150,000 restarts), with or without thrust modulation (called throttling), single use or reusable, arranged as single engines or in clusters of multiple units. One way to categorize liquid propellant rocket engines is described in the below table. There are two categories, namely those used for boosting a payload and imparting a significant velocity increase to a payload, and auxiliary propulsion for trajectory adjustments and attitude control.

Liquid propellant rocket engine systems can be classified in several other ways. They can be reusable (like the Space Shuttle main engine or a booster rocket engine for quick ascent or maneuvers of fighter aircraft) or suitable for a single flight only (as the engines in the Atlas or Titan launch vehicles) and they can be restartable, like a reaction control engine, or single firing, as in a space launch vehicle. They can also be categorized by their propellants, application, or stage, such as an upper stage or booster stage, their thrust level, and by the feed system type (pressurized or turbopump) .

The **thrust chamber** or thruster is the combustion device where the liquid propellants are metered, injected, atomized, mixed, and burned to form hot gaseous reaction products, which in turn are accelerated and ejected at a high velocity to impart a thrust force. A thrust chamber has three major parts: an injector, a combustion chamber, and a nozzle.

In a cooled thrust chamber, one of the propellants (usually the fuel) is circulated through cooling jackets or a special cooling passage to absorb the heat that is transferred from the hot reaction gases to the thrust chamber walls. A radiation- cooled thrust chamber uses a special high-temperature material, such as niobium metal, which can radiate away its excess heat. There are uncooled or heat-absorbing thrust chambers, such as those using ablative materials.

There are two types of **feed systems** used for liquid propellant rocket engines: those that use pumps for moving the propellants from their flight vehicle tanks to the thrust chamber, and those that use high-pressure gas for expelling or displacing their propellants from their tanks.

- Pressure-fed liquid propellant rocket engine
- Turbopump-fed liquid propellant rocket engine

The **propellants**, which are the working substance of rocket engines, constitute the fluid that undergoes chemical and thermodynamic changes. The term liquid propellant embraces all the various liquids used and may be one of the following:

1. Oxidizer (liquid oxygen, nitric acid, etc.)
2. Fuel (gasoline, alcohol, liquid hydrogen, etc.)
3. Chemical compound or mixture of oxidizer and fuel ingredients, capable of self-decomposition.
4. Any of the above, but with a gelling agent.

A **bipropellant rocket** unit has two separate liquid propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of liquid propellant rockets have been manufactured for bipropellant applications.

A **monopropellant** contains an oxidizing agent and combustible matter in a single substance. It may be a mixture of several compounds or it may be a homogeneous material, such as hydrogen peroxide or hydrazine. Monopropellants are stable at ordinary atmospheric conditions but decompose and yield hot combustion gases when heated or catalyzed.

A **cold gas propellant** (e.g., nitrogen) is stored at very high pressure, gives a low performance, allows a simple system and is usually very reliable. It has been used for roll control and attitude control.

A **cryogenic propellant** is liquified gas at low temperature, such as liquid oxygen (-183°C) or liquid hydrogen (-253°C). Provisions for venting the storage tank and minimizing vaporization losses are necessary with this type.

Storable propellants (e.g., nitric acid or gasoline) are liquid at ambient temperature and can be stored for long periods in sealed tanks. Space storable propellants are liquid in the environment of space; this storability depends on the specific tank design, thermal conditions, and tank pressure. An example is ammonia.

A **gelled propellant** is a thixotropic liquid with a gelling additive. It behaves like a jelly or thick paint. It will not spill or leak readily, can flow under pressure, will burn, and is safer in some respects.

What Are The Selection Criteria Of Rocket Propulsion System?

Many criteria used in selecting a particular rocket propulsion system are peculiar to the particular mission or vehicle application. However, some of these selection factors apply to a number of applications which we are going to discuss in this article. Again, this list is incomplete and not all the criteria in this table apply to every application. Below explanation can be used as a checklist to see that none of the criteria listed here are omitted.

Here are some examples of important criteria in a few specific applications. For a spacecraft that contains optical instruments (e.g., telescope, horizon seeker, star tracker, or infrared radiation seeker), the exhaust plume must be free of possible contaminants that may deposit or condense on photovoltaic cells, radiators, optical windows, mirrors, or lenses and degrade their performance, and free of particulates that could scatter sunlight into the instrument aperture, which could cause erroneous signals.

Conventional composite solid propellants and pulsing storable bipropellants are usually not satisfactory, but cold or heated clean gas jets (H2, Ar, N2, etc.) and monopropellant hydrazine reaction gases are usually acceptable. Another example is an emphasis on smokeless propellant exhaust plumes, so as to make visual detection of a smoke or vapor trail very difficult. This applies particularly to tactical missile applications. Only a few solid propellants and several liquid propellants would be truly smokeless and free of a vapor trail under all weather conditions.

Several selection criteria may be in conflict with each other. For example, some propellants with a very high specific impulse are more likely to experience combustion instabilities. In liquid propellant systems, where the oxidizer tank is pressurized by a solid propellant gas generator and where the fuel-rich hot gases are separated by a thin flexible diaphragm from the oxidizer liquid, there is a trade-off between a very compact system and the potential for a damaging system failure (fire, possible explosion, and malfunction of system) if the diaphragm leaks or tears. In electric propulsion, high specific impulse is usually accompanied by heavy power generating and conditioning equipment.

Actual selection will depend on the balancing of the various selection factors in accordance with their importance, benefits, or potential impact on the system, and on quantifying as many of these selection factors as possible through analysis, extrapolation of prior experience/data, cost estimates, weights, and/or separate tests. Design philosophies such as the Taguchi methodology and TQM (total quality management) can be inferred. Layouts, weight estimates, center-of-gravity analyses, vendor cost estimates, preliminary stress or thermal analysis, and other preliminary design efforts are usually necessary to put numerical values on some of the selection parameters.

A comparative examination of the interfaces of alternate propulsion systems is also a part of the process. Some propulsion requirements are incompatible with each other and a compromise has to be made. For example, the monitoring of extra sensors can prevent the occurrence of certain types of failure and thus enhance the propulsion system reliability, yet the extra sensors and control components contribute to the system complexity and their possible failures will reduce the overall reliability. The selection process may also include feedback when the stated propulsion requirements cannot be met or do not make sense, and this can lead to a revision of the initial mission requirements or definition.

Once the cost, performance, and reliability drivers have been identified and quantified, the selection of the best propulsion system for a specified mission proceeds. The final propulsion requirement may come as a result of several iterations and will usually be documented, for example in a propulsion requirement specification. A substantial number of records is required (such as engine or motor acceptance documents, CAD (computer-aided design) images, parts lists, inspection records, laboratory test data, etc.).

There are many specifications associated with design and manufacturing as well as with vendors, models, and so on. There must also be a disciplined procedure for approving and making design and manufacturing changes. This now becomes the starting point for the design and development of the propulsion system.

Typical Criteria Used In The Selection Of A Particular Rocket Propulsion System:

1. Mission Definition (Selection Criteria Of Rocket Propulsion System):

Purpose, function, and final objective of the mission of an overall system are well defined and their implications well understood. There is an expressed need for the mission, and the benefits are evident. The mission requirements are well defined. The payload, flight regime, vehicle, launch environment, and operating conditions are established. The risks, as perceived, appear acceptable. The project implementing the mission must have political, economic, and institutional support with assured funding. The propulsion system requirements, which are derived from mission definition, must be reasonable and must result in a viable propulsion system.

2. Affordability (Cost) (Selection Criteria Of Rocket Propulsion System):
Life cycle costs are low. They are the sum of R&D costs, production costs, facility costs, operating costs, and decommissioning costs, from inception to the retirement of the system. Benefits of achieving the mission should appear to justify costs. Investment in new facilities should be low. Few, if any, components should require expensive materials. For commercial applications, such as communications satellites, the return on investment must look attractive. No need to hire new, inexperienced personnel, who need to be trained and are more likely to make expensive errors.

3. System Performance (Selection Criteria Of Rocket Propulsion System):
The propulsion system is designed to optimize vehicle and system performance, using the most appropriate and proven technology. Inert mass is reduced to a practical minimum, using improved materials and better understanding of loads and stresses. Residual (unused) propellant is minimal. Propellants have the highest practical specific impulse without undue hazards, without excessive inert propulsion system mass, and with simple loading, storing, and handling.
Thrust-time profiles and number of restarts must be selected to optimize the vehicle mission. Vehicles must operate with adequate performance for all the possible conditions (pulsing, throttling, temperature excursions, etc.). Vehicles should be storable over a specified lifetime. Will meet or exceed operational life. Performance parameters (e.g., chamber pressure, ignition time, or nozzle area ratio) should be near optimum for the selected mission. Vehicle should have adequate TVC. Plume characteristics are satisfactory.

4. Survivability (Safety) (Selection Criteria Of Rocket Propulsion System):
All hazards are well understood and known in detail. If failure occurs, the risk of personnel injury, damage to equipment, facilities, or the environment is minimal. Certain mishaps or failures will result in a change in the operating condition or the safe shutdown of the propulsion system. Applicable safety standards must be obeyed. Inadvertent energy input to the propulsion system (e.g., bullet impact, external fire) should not result in a detonation.
The probability for any such drastic failures should be very low. Safety monitoring and inspections must have proven effective in identifying and preventing a significant share of possible incipient failures. Adequate safety factors must be included in the design. Spilled liquid propellants should cause no undue hazards. All systems and procedures must conform to the safety standards. Launch test range has accepted the system as being safe enough to launch.

5. Reliability (Selection Criteria Of Rocket Propulsion System):
Statistical analyses of test results indicate a satisfactory high-reliability level. Technical risks, manufacturing risks, and failure risks are very low, well understood, and the impact on the

overall system is known. There are few complex components. Adequate storage and operating life of components (including propellants) have been demonstrated. Proven ability to check out major part of propulsion system prior to use or launch. If certain likely failures occur, the system must shut down safely. Redundancy of key components should be provided, where effective. High probability that all propulsion functions must be performed within the desired tolerances. Risk of combustion vibration or mechanical vibration should be minimal.

8. Geometric Constraints (Selection Criteria Of Rocket Propulsion System):

Propulsion system fits into vehicle, can meet available volume, specified length, or vehicle diameter. There is usually an advantage for the propulsion system that has the smallest volume or the highest average density. If the travel of the center of gravity has to be controlled, as is necessary in some missions, the propulsion system that can do so with minimum weight and complexity will be preferred.

9. Prior Related Experience (Selection Criteria Of Rocket Propulsion System):

There is a favorable history and valid, available, relevant data of similar propulsion systems supporting the practicality of the technologies, manufacturability, performance, and reliability. Experience and data validating computer simulation programs are available. Experienced, skilled personnel are available.

10. Operability (Selection Criteria Of Rocket Propulsion System):

Simple to operate. Validated operating manuals exist. Procedures for loading propellants, arming the power supply, launching, igniter checkout, and so on, must be simple. If applicable, a reliable automatic status monitoring and check-out system should be available. Crew training needs to be minimal. Should be able to ship the loaded vehicle on public roads or railroads without need for environmental permits and without the need for a decontamination unit and crew to accompany the shipment. Supply of spare parts must be assured. Should be able to operate under certain emergency and overload conditions.

11. Producibility (Selection Criteria Of Rocket Propulsion System):

Easy to manufacture, inspect, and assemble. All key manufacturing processes are well understood. All materials are well characterized, critical material properties are well known, and the system can be readily inspected. Proven vendors for key components have been qualified. Uses standard manufacturing machinery and relatively simple tooling. Hardware quality and propellant properties must be repeatable. Scrap should be minimal. Designs must make good use of standard materials, parts, common fasteners, and off-the-shelf components. There should be maximum use of existing manufacturing facilities and equipment. Excellent reproducibility, i.e., minimal operational variation between identical propulsion units. Validated specifications should be available for major manufacturing processes, inspection, parts fabrication, and assembly.

12. Schedule (Selection Criteria Of Rocket Propulsion System):

The overall mission can be accomplished on a time schedule that allows the system benefits to be realized. R&D, qualification, flight testing, and/or initial operating capability are completed on a preplanned schedule. No unforeseen delays. Critical materials and qualified suppliers must be readily available.

Liquid Propellant Feed Systems:

The propellant feed system has two principal functions: to raise the pressure of the propellants and to feed them to one or more thrust chambers. The energy for these functions comes either from a high-pressure gas, centrifugal pumps, or a combination of the two.

The selection of a particular feed system and its components is governed primarily by the application of the rocket, requirements, duration, number or type of thrust chambers, past experience, mission, and by general requirements of simplicity of design, ease of manufacture, low cost, and minimum inert mass. All feed systems have piping, a series of valves, provisions for filling and removing (draining and flushing) the liquid propellants, and control devices to initiate, stop, and regulate their flow and operation.

Generally, there are two different types of propellant feed systems of liquid propellant rockets and are as follows,

1. Gas Pressure Feed Systems
2. Turbopump Feed Systems

1. Gas Pressure Feed Systems:

One of the simplest and most common means of pressurizing the propellants is to force them out of their respective tanks by displacing them with high-pressure gas. This gas is fed into the propellant tanks at a controlled pressure, thereby giving a controlled propellant discharge. Because of their relative simplicity, the rocket engines with pressurized feed systems can be very reliable.

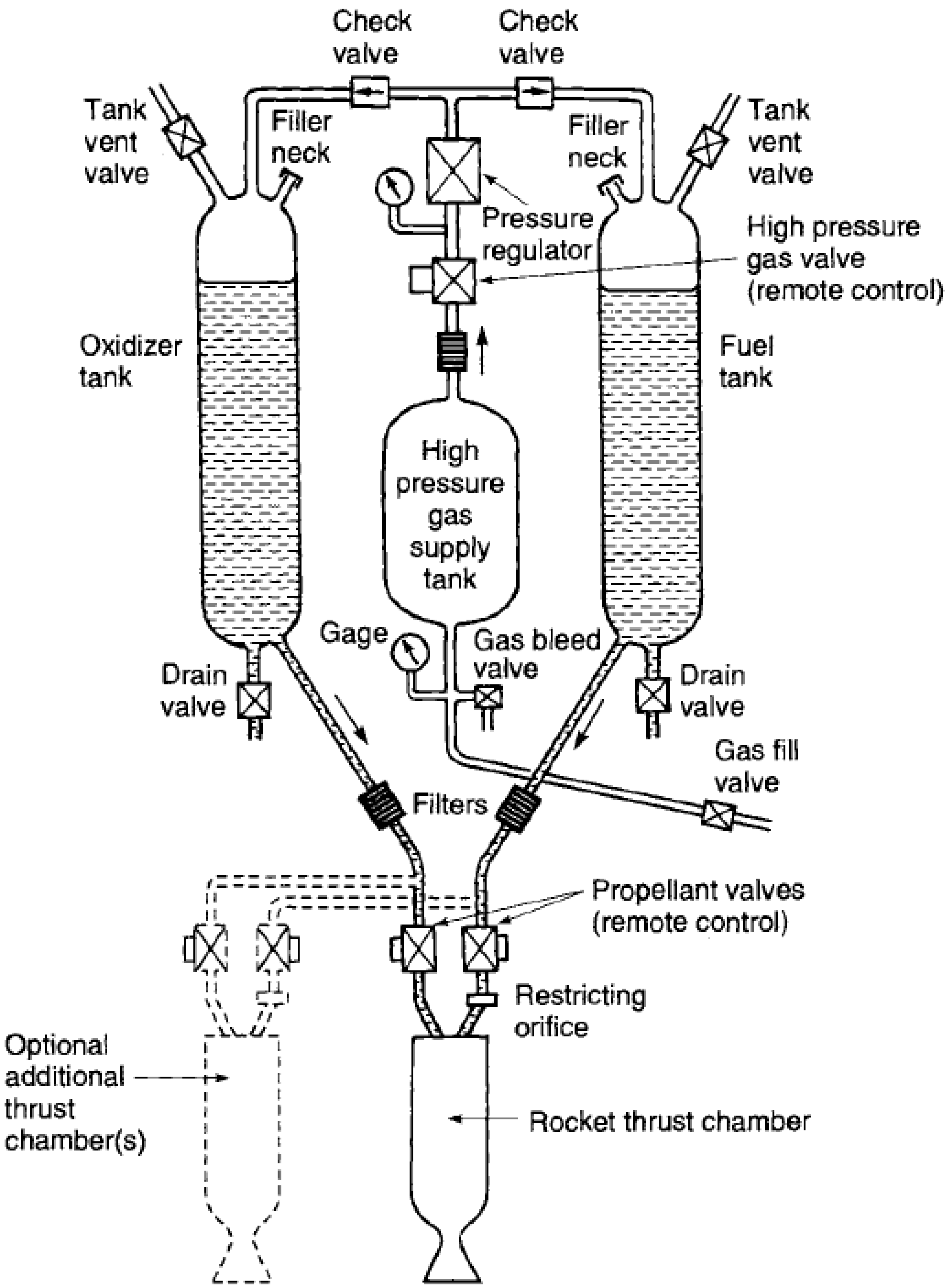
A simple pressurized feed system is shown schematically below. It consists of a high-pressure gas tank, a gas starting valve, a pressure regulator, propellant tanks, propellant valves, and feed lines. Additional components, such as filling and draining provisions, check valves, filters, flexible elastic bladders for separating the liquid from the pressurizing gas, and pressure sensors or gauges, are also often incorporated.

After all tanks are filled, the high-pressure gas valve in the below figure is remotely actuated and admits gas through the pressure regulator at a constant pressure to the propellant tanks. The check valves prevent mixing of the oxidizer with the fuel when the unit is not in an upright position. The propellants are fed to the thrust chamber by opening valves. When the propellants are completely consumed, the pressurizing gas can also scavenge and clean lines and valves of much of the liquid propellant residue.

The variations in this system, such as the combination of several valves into one or the elimination and addition of certain components, depend to a large extent on the application. If a unit is to be used over and over, such as space-maneuver rocket, it will include several additional features such as, possibly, a thrust-regulating device and a tank level gauge; they will not be found in an expendable, single-shot unit, which may not even have a tank-drainage provision.

Some pressure feed systems can be prefilled with propellant and pressurizing agent at the factory and stored in readiness for operation. Compared to a solid propellant rocket unit, these storable prepackaged liquid propellant pressurized feed systems offer advantages in long-term storability and resistance to transportation vibration or shock.

The thrust level of a rocket propulsion system with a pressurized gas feed system is determined by the magnitude of the propellant flow which, in turn, is determined by the gas pressure regulator setting. The propellant mixture ratio in this type of feed system is controlled by the hydraulic resistance of the liquid propellant lines, cooling jacket, and injector, and can usually be adjusted by means of variable or interchangeable restrictors.

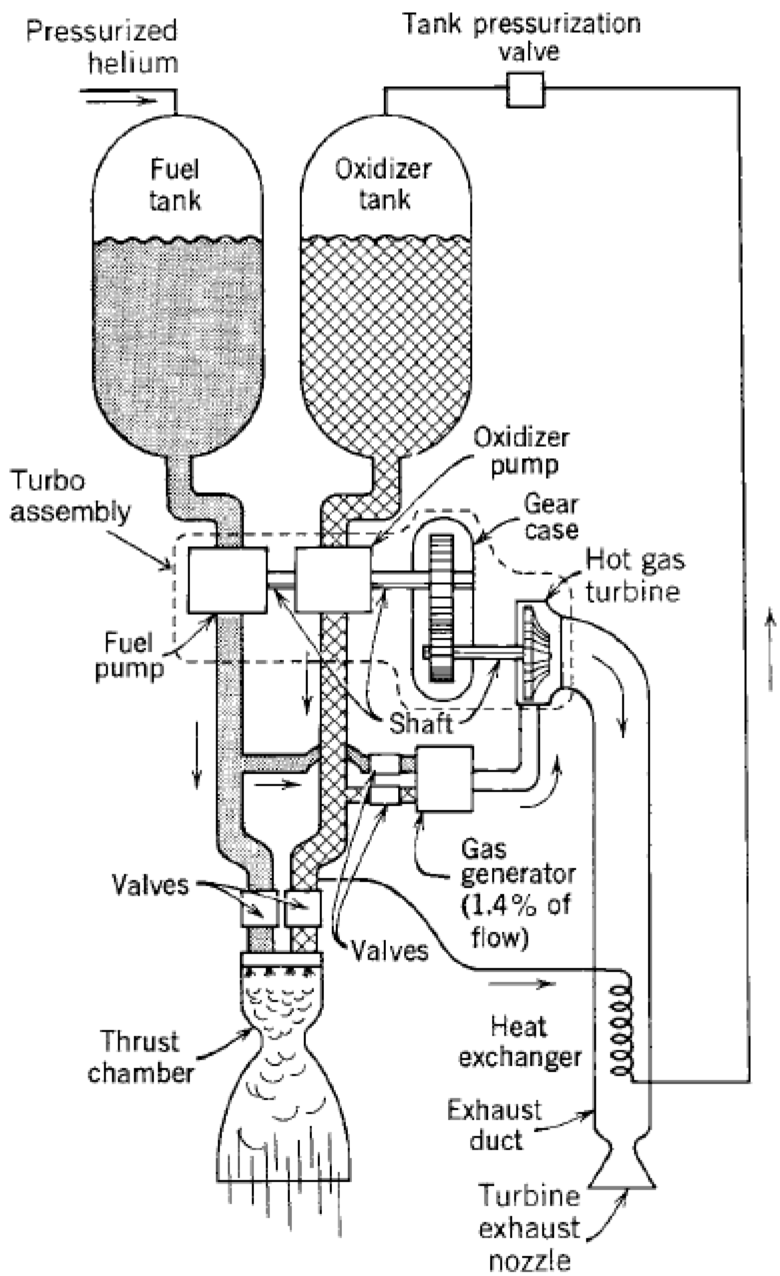


Schematic Flow Diagram Of A Liquid Propellant Rocket Engine With A Gas Pressure Feed System

2. Turbopump Feed Systems:

The principal components of a rocket engine with one type of turbopump system are shown in the below simplified diagram. Here, the propellants are pressurized by means of pumps, which in turn are driven by turbines. These turbines derive their power from the expansion of hot gases.

Engines with turbopumps are preferred for booster and sustainer stages of space launch vehicles, long-range missiles, and in the past also for aircraft performance augmentation. They are usually lighter than other types for these high thrust, long duration applications. The inert hardware mass of the rocket engine (without tanks) is essentially independent of duration.



Schematic Flow Diagram Of A Liquid Propellant Rocket Engine With A Turbopump Pressure Feed System

An **engine cycle** for turbopump-fed engines describes the specific propellant flow paths through the major engine components, the method of providing the hot gas to one or more turbines, and the method of handling the turbine exhaust gases. There are **open cycles** and **closed cycles**.

Open denotes that the working fluid exhausting from the turbine is discharged overboard, after it gets expanded in a nozzle of its own, or discharged into the nozzle of the thrust chamber at a point in the expanding section far downstream of the nozzle throat. In closed cycles or topping cycles, all the working fluid from the turbine is injected into the engine combustion chamber to make the most efficient use of its remaining energy.

In closed cycles, the turbine exhaust gas is expanded through the full pressure ratio of the main thrust chamber nozzle, thus giving a little more performance than the open cycles, where these exhaust gases expand only through a relatively small pressure ratio. The overall engine performance difference s typically between 1 and 8% of specific impulse and this is reflected in even larger differences in vehicle performance.

Most Common Cycles Of Turbopump Feed Systems:

There are three most common cycles of turbopump feed systems and as follows,

- Gas Generator Cycle
- Expander Cycle
- Staged Combustion Cycle

The gas generator cycle and the staged combustion cycle can use most of the common liquid propellants. The expander cycle works best with vaporized cryogenic hydrogen as the coolant for the thrust chamber, because it is an excellent heat absorber and does not decompose. Here, we are going to see each cycle with a separate turbopump for fuel and for oxidizer. However, an arrangement with the fuel and oxidizer pump driven by the same turbine is also feasible and sometimes reduces the hardware mass, volume, and cost.

The “best” cycle has to be selected on the basis of the mission, the suitability of existing engines, and the criteria established for the particular vehicle. There is an optimum chamber pressure and an optimum mixture ratio for each application, engine cycle, or optimization criterion, such as maximum range, lowest cost, or highest payload.

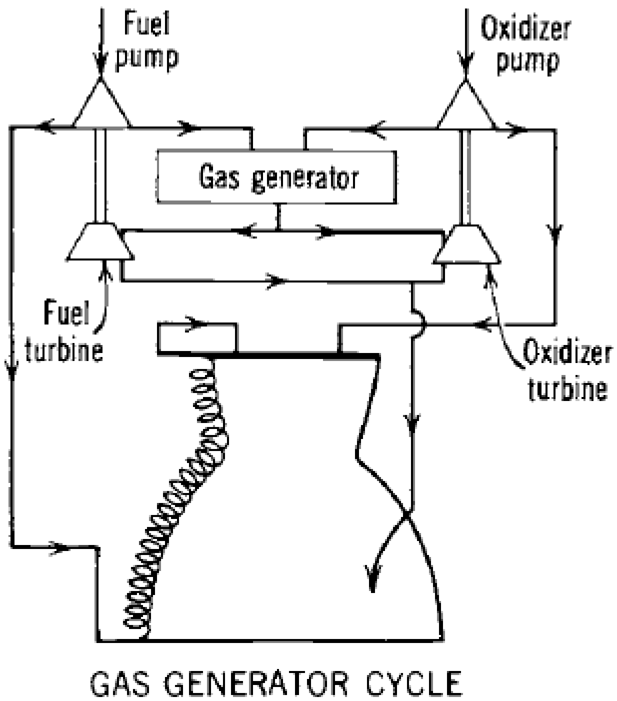
A. Gas Generator Cycle – Turbopump Feed Systems:

In the gas generator cycle, the turbine inlet gas comes from a separate gas generator. It’s propellants can be supplied from separate propellant tanks or can be bled off the main propellant feed system. This cycle is relatively simple; the pressures in the liquid pipes and pumps are relatively low (which reduces inert engine mass). It has less engine-specific impulse than an expander cycle or a staged combustion cycle.

The pressure ratio across the turbine is relatively high, but the turbine or gas generator flow is small (1 to 4% of total propellant flow) if compared to closed cycles. Some early engines used a separate mono-propellant for creating the generator gas. The German V-2 missile engine used hydrogen peroxide, which was decomposed by a catalyst. Typically, the turbine exhaust gas is discharged overboard through one or two separate small lowarea-ratio nozzles (at relatively low specific impulse).

Alternatively, this turbine exhaust can be aspirated into the main flow through openings in the diverging nozzle section, as shown schematically in the below figure. This gas then protects the walls near the nozzle exit from high temperatures. Both methods can provide a small amount of additional thrust.

The gas generator mixture ratio is usually fuel rich (in some engine it is oxidizer rich) so that the gas temperatures are low enough (typically 900 to 1350 K) to allow the use of uncooled turbine blades and uncooled nozzle exit segments. With a gas generator cycle, the specific impulse of the thrust chamber by itself is always a little higher than that of the engine and the thrust of the thrust chamber is always slightly lower than that of the engine.



Gas Generator Cycle – Turbopump Feed System

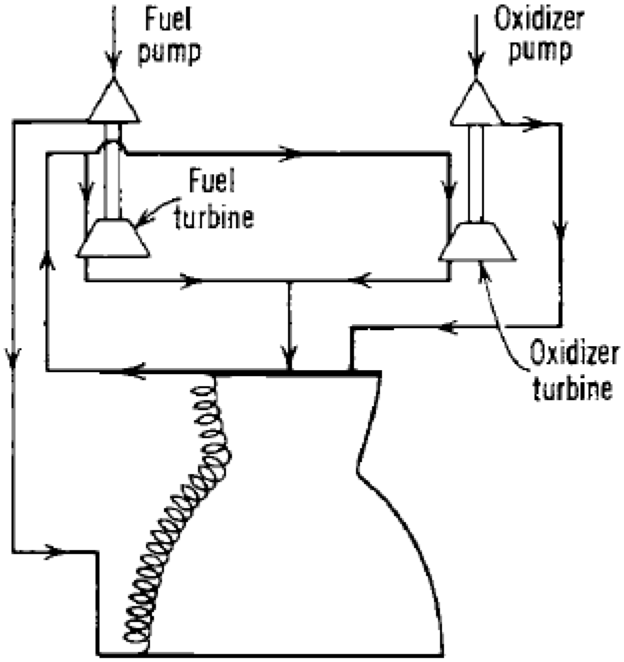
B. Expander Cycle – Turbopump Feed Systems:

In the expander cycle, most of the engine coolant (usually hydrogen fuel) is fed to low-pressure-ratio turbines after having passed through the cooling jacket where it picked up energy. Part of the coolant, perhaps 5 to 15%, bypasses the turbine (not shown in the below figure) and rejoins the turbine exhaust flow before the entire coolant flow is injected into the engine combustion chamber where it mixes and burns with the oxidizer.

The primary advantages of the expander cycle are good specific impulse, engine simplicity, and relatively low engine mass. In the expander cycle, all the propellants are fully burned in the engine combustion chamber and expanded efficiently in the engine exhaust nozzle. This cycle is used in the RL10 hydrogen/oxygen rocket engine, and different versions of this engine have flown successfully in the upper stages of several space launch vehicles.

Heat absorbed by the thrust chamber cooling jacket gasifies and raises the gas temperature of the hydrogen so that it can be used to drive the turbine, which in turn drives a single-stage liquid oxygen pump (through a gear case) and a two-stage liquid hydrogen pump. The cooling down of the hardware to cryogenic temperatures is accomplished by flowing (prior to engine start) cold propellant through cooldown valves.

The pipes for discharging the cooling propellants overboard are not shown here. Thrust is regulated by controlling the flow of hydrogen gas to the turbine, using a bypass to maintain constant chamber pressure. Helium is used as a means of power boost by actuating several of the larger valves through solenoid-operated pilot valves.



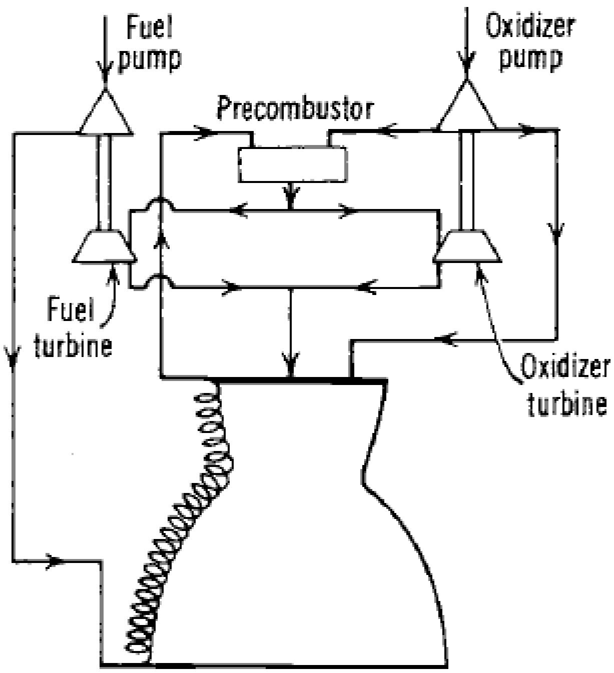
Expander Cycle – Turbopump Feed System

C. Staged Combustion Cycle – Turbopump Feed Systems:

In the staged combustion cycle, the coolant flow path through the cooling jacket is the same as that of the expander cycle. Here a high-pressure precombustor (gas generator) burns all the fuel with part of the oxidizer to provide high-energy gas to the turbines. The total turbine exhaust gas flow is injected into the main combustion chamber where it burns with the remaining oxidizer. This cycle lends itself to high-chamber-pressure operation, which allows a small thrust chamber size.

The extra pressure drop in the precombustor and turbines causes the pump discharge pressures of both the fuel and the oxidizer to be higher than with open cycles, requiring heavier and more complex pumps, turbines, and piping. The turbine flow is relatively high and the turbine pressure drop is low, when compared to an open cycle. The staged combustion cycle gives the highest specific impulse, but it is more complex and heavy.

In contrast, an open cycle can allow a relatively simple engine, lower pressures, and can have a lower production cost. A variation of the staged combustion cycle is used in the Space Shuttle main engine. This engine actually uses two separate precombustion chambers, each mounted directly on a separate main turbopump. In addition, there are two more turbopumps for providing a boost pressure to the main pumps, but their turbines are not driven by combustion gases; instead, high-pressure liquid oxygen drives one booster pump and evaporated hydrogen drives the other.

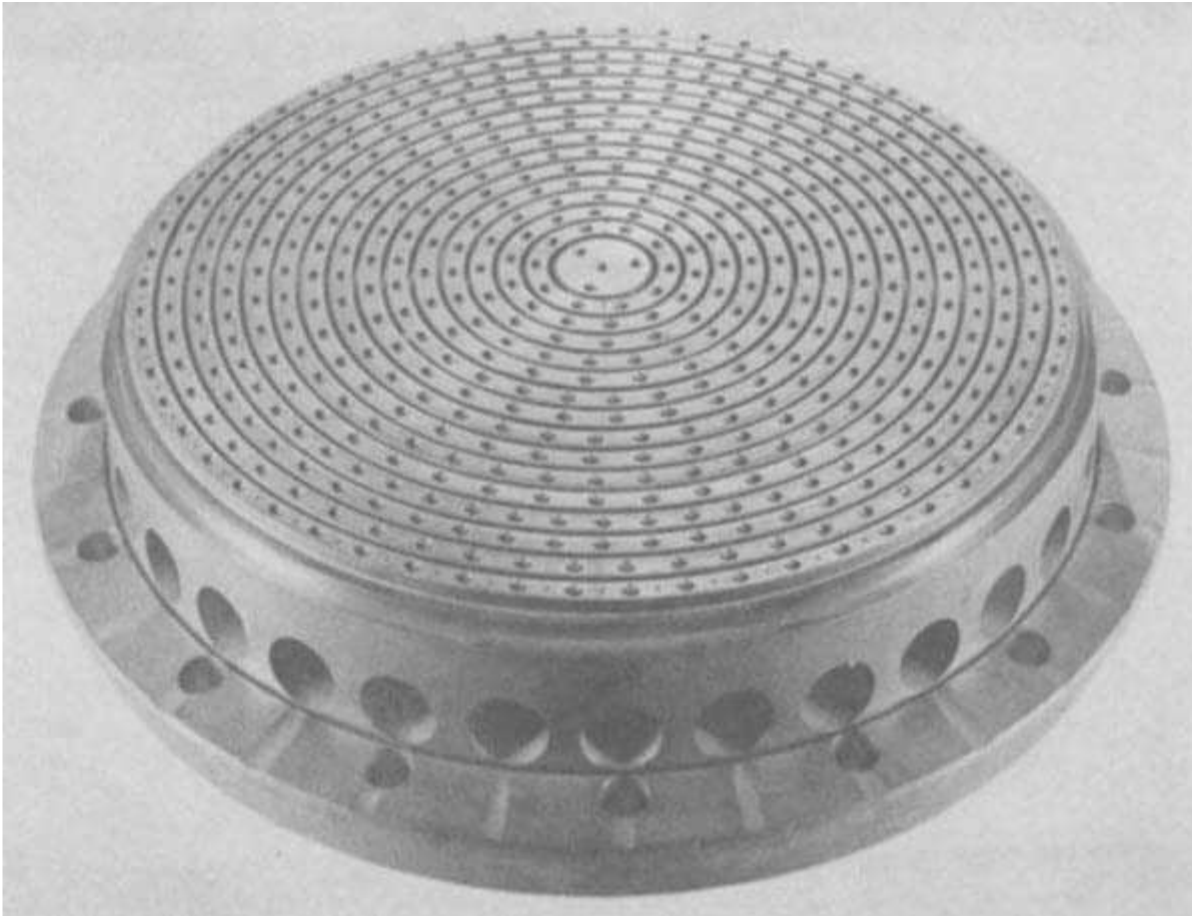


STAGED-COMBUSTION CYCLE

Staged Combustion Cycle – Turbopump Feed System

What Is Injector?

The functions of the injector are similar to those of a carburetor of an internal combustion engine. The injector has to introduce and meter the flow of liquid propellants to the combustion chamber, cause the liquids to be broken up into small droplets (a process called atomization), and distribute and mix the propellants in such a manner that a correctly proportioned mixture of fuel and oxidizer will result, with uniform propellant mass flow and composition over the chamber cross section. This has been accomplished with different types of injector designs and elements.



Injector With 90° Self-Impinging-Type Countersunk Doublet Injector Pattern

Types of Injectors In Liquid Propellant Rockets:

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

The injection hole pattern on the face of the injector is closely related to the internal manifolds or feed passages within the injector. These provide for the distribution of the propellant from the injector inlet to all the injection holes. A large complex manifold volume allows low passage velocities and good distribution of flow over the cross section of the chamber. A small manifold volume allows for a lighter weight injector and reduces the amount of "dribble" flow after the main valves are shut.

The higher passage velocities cause a more uneven flow through different identical injection holes and thus a poorer distribution and wider local gas composition variation. Dribbling results in afterburning, which is an inefficient irregular combustion that gives a little "cutoff" thrust after valve closing. For applications with very accurate terminal vehicle velocity requirements, the cutoff impulse has to be very small and reproducible and often valves are built into the injector to minimize passage volume.

Some of the injectors used in liquid propellant rockets are as follows,

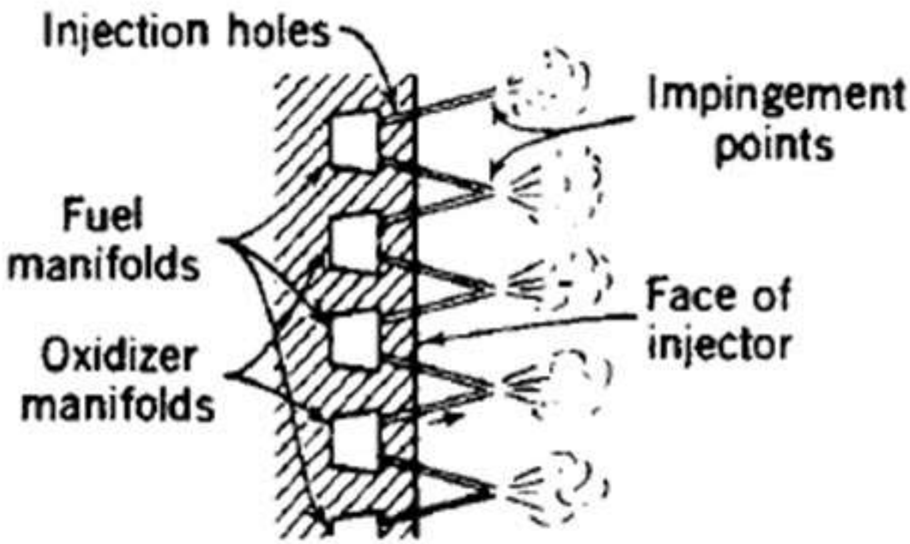
1. Impinging-Stream-Type, Multi-Hole Injector
2. Non-Impinging or Shower Head Injector

3. Coaxial Hollow Post Injector

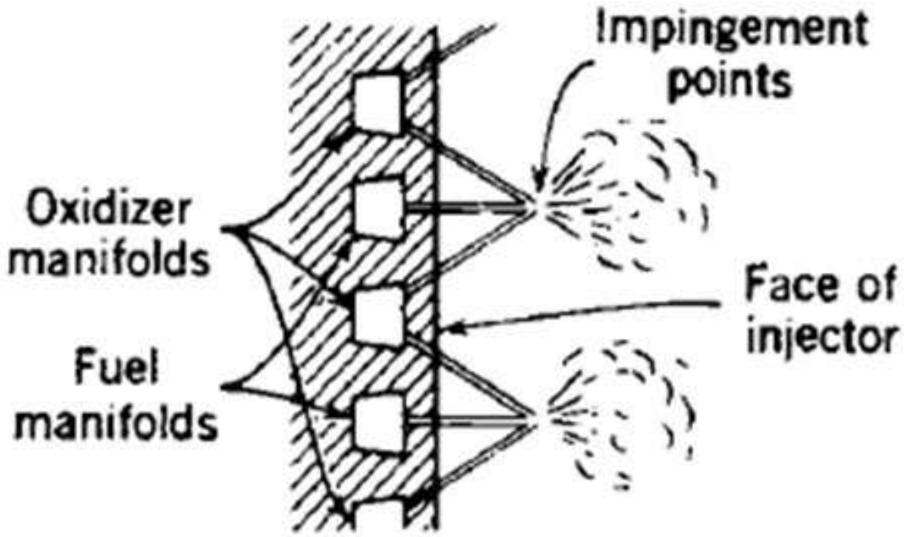
1. <u>Impinging-Stream-Type, Multi-Hole Injector</u>:

Impinging-stream-type, multiple-hole injectors are commonly used with oxygen-hydrocarbon and storable propellants. For unlike doublet patterns, the propellants are injected through a number of separate small holes in such a manner that the fuel and oxidizer streams impinge upon each other. Impingement forms thin liquid fans and aids atomization of the liquids into droplets, also aiding distribution. Impinging hole injectors are also used for like-on-like or self-impinging patterns (fuel-on-fuel and oxidizer-on-oxidizer).

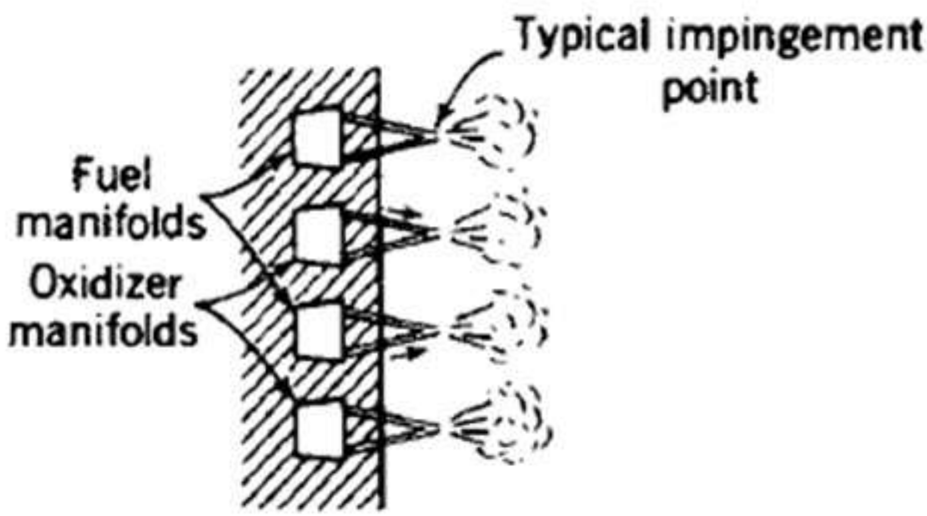
The two liquid streams then form a fan which breaks up into droplets. Unlike doublets work best when the hole size (more exactly, the volume flow) of the fuel is about equal to that of the oxidizer and the ignition delay is long enough to allow the formation of fans. For uneven volume flow the triplet pattern seems to be more effective.



Doublet Impinging Stream Pattern



Triplet Impinging Stream Pattern



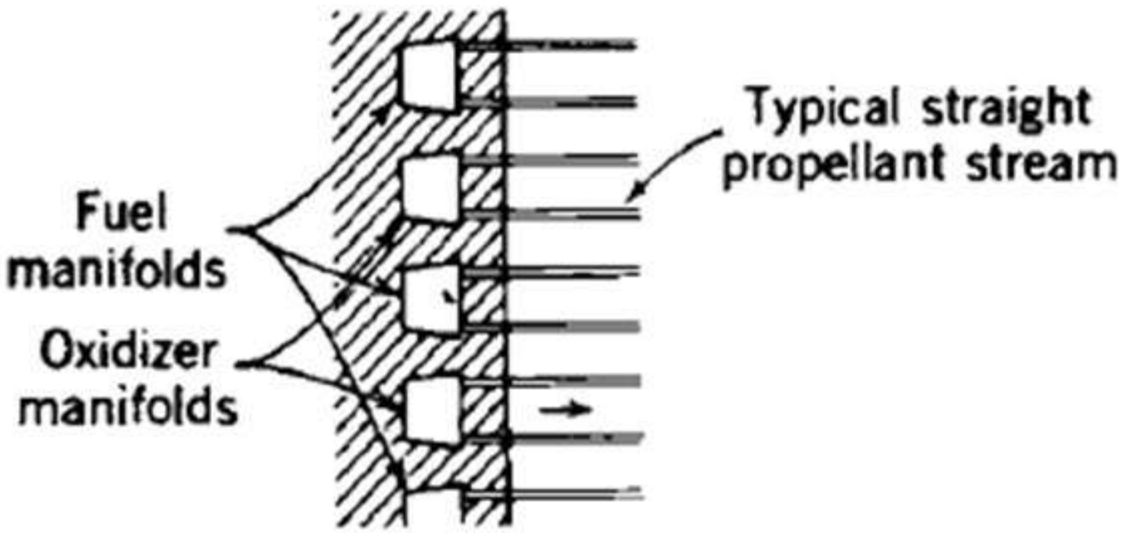
Self-Impinging Stream Pattern

2. <u>Non-Impinging or Shower Head Injector</u>:

The non-impinging or shower head injector employs non-impinging streams of propellant usually emerging normal to the face of the injector. It relies on turbulence and diffusion to achieve mixing. The German World War II V-2 rocket used this type of injector. This type is now not used, because it requires a large chamber volume for good combustion. Sheet or spray-type injectors give cylindrical, conical, or other types of spray sheets; these sprays generally intersect and thereby promote mixing and atomization.

By varying the width of the sheet (through an axially moveable sleeve) it is possible to throttle the propellant flow over a wide range without excessive reduction in injector pressure drop. This type of variable area concentric tube injector was used on the descent engine of the Lunar

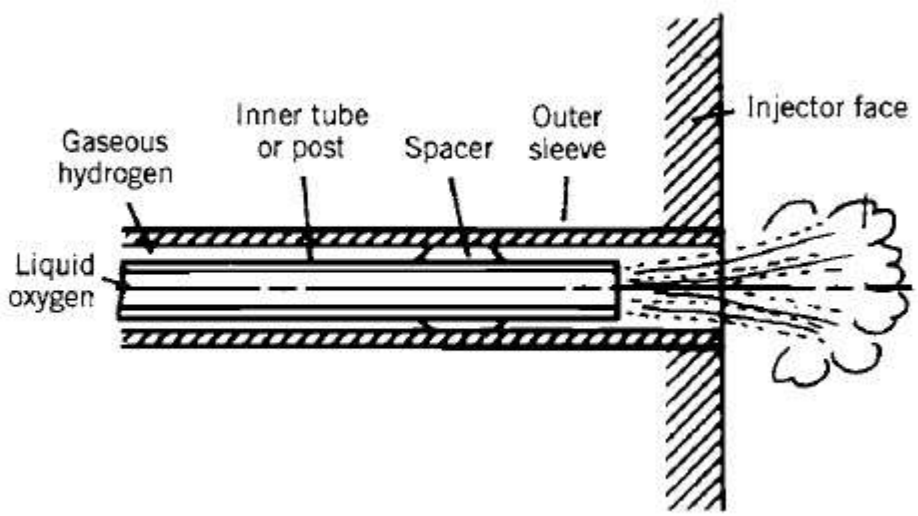
Excursion Module and throttled over a 10:1 range of flow with only a very small change in mixture ratio.



Shower Head Stream Pattern

3. Coaxial Hollow Post Injector:

The coaxial hollow post injector has been used for liquid oxygen and gaseous hydrogen injectors by most domestic and foreign rocket designers. It works well when the liquid hydrogen has absorbed heat from cooling jackets and has been gasified. This gasified hydrogen flows at high speed (typically 330 m/sec or 1000 ft/sec); the liquid oxygen flows far more slowly (usually at less than 33 m/sec or 100 ft/sec) and the differential velocity causes a shear action, which helps to break up the oxygen stream into small droplets. The injector has a multiplicity of these coaxial posts on its face. This type of injector is not used with liquid storable bipropellants, in part because the pressure drop to achieve high velocity would become too high.

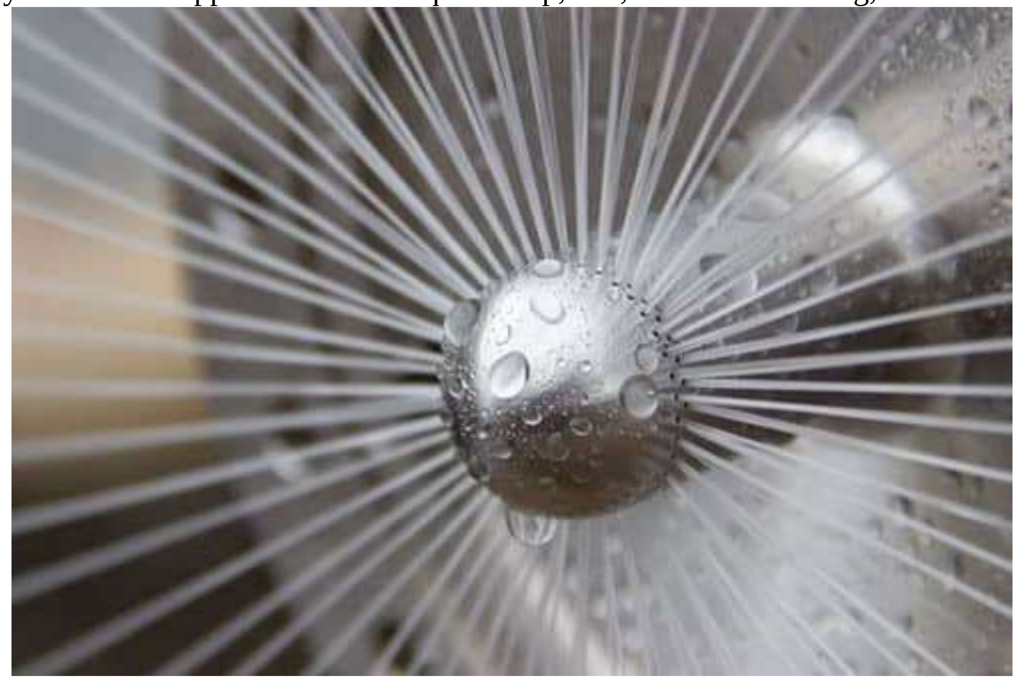


Coaxial Hollow Post Injector

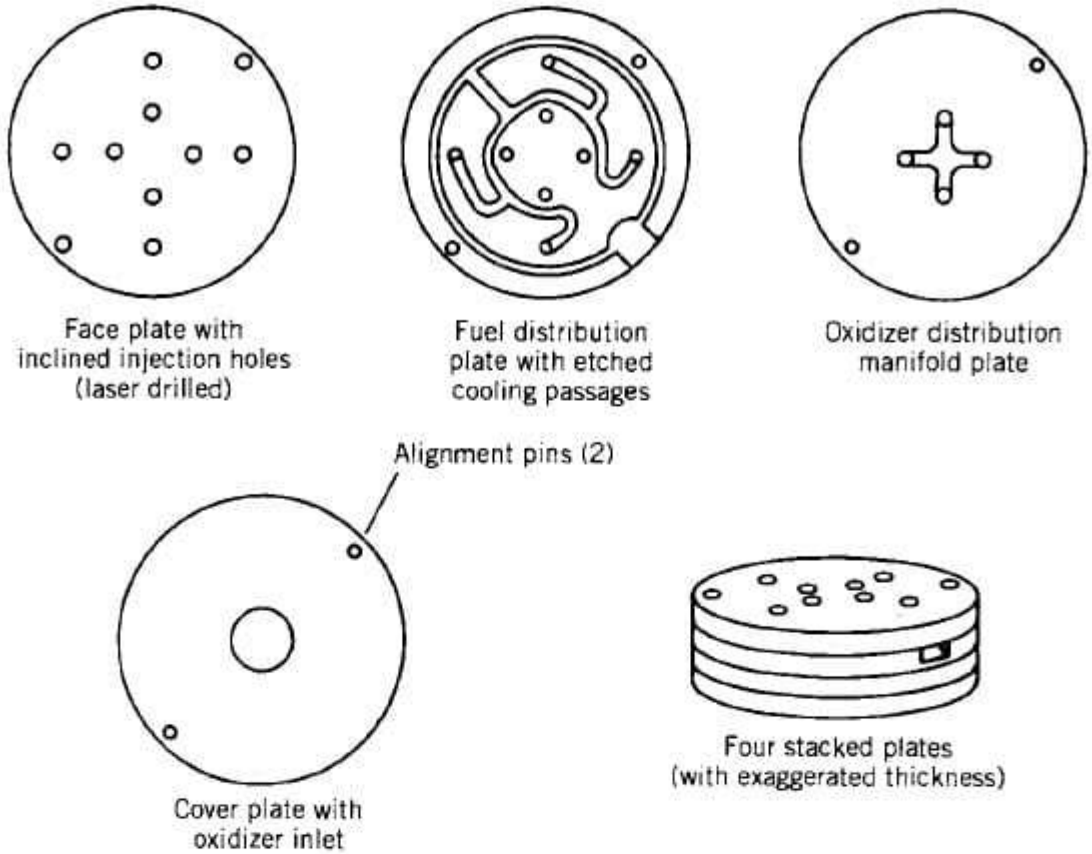
4. Pintle Injector:

The pintle injector is a type of propellant injector for a bipropellant rocket engine. Like any other injector, its purpose is to ensure appropriate flow rate and intermixing of the propellants as they are forcibly injected under high pressure into the combustion chamber, so that an efficient and controlled combustion process can happen.

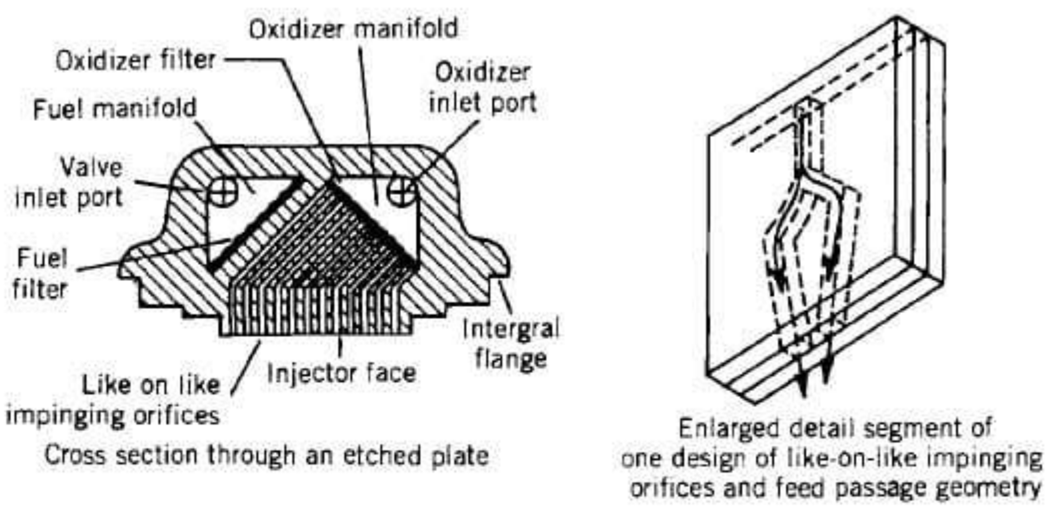
A pintle-based rocket engine can have a greater throttling range than one based on regular injectors, and will very rarely present acoustic combustion instabilities, because a pintle injector tends to create a self-stabilizing flow pattern. Therefore, pintle-based engines are specially suitable for applications that require deep, fast, and safe throttling, such as landers.



The original method of making injection holes was to carefully drill them and round out or chamfer their inlets. This is still being done today. It is difficult to align these holes accurately (for good impingement) and to avoid burrs and surface irregularities. One method that avoids these problems and allows a large number of small accurate injection orifices is to use multiple etched, very thin plates (often called platelets) that are then stacked and diffusion bonded together to form a monolithic structure as shown in the below figure.



Injector For Low Thrust With Four Impinging Unlike Doublet Liquid Streams | Plates Are Parallel To The Injector Face



Like-on-like Impinging Stream Injector With 144 Orifices | Plates Are perpendicular To The Injector Face

The photo-etched pattern on each of the individual plates or metal sheets then provides not only for many small injection orifices at the injector face, but also for internal distribution or flow passages in the injector and sometimes also for a fine-mesh filter inside the injector body. The platelets can be stacked parallel to or normal to the injector face. The finished injector has been called the platelet injector and has been patented by the Aerojet Propulsion Company.

Thrust Vector Control?

In addition to providing a propulsive force to a flying vehicle, a rocket propulsion system can provide moments to rotate the flying vehicle and thus provide control of the vehicle's attitude and flight path. By controlling the direction of the thrust vectors through the mechanisms, it is possible to control a vehicle's pitch, yaw, and roll motions.

Thrust vector control is effective only while the propulsion system is operating and creating an exhaust jet. For the flight period, when a rocket propulsion system is not firing and therefore its Thrust Vector Control is inoperative, a separate mechanism needs to be provided to the flying vehicle for achieving control over its attitude or flight path.

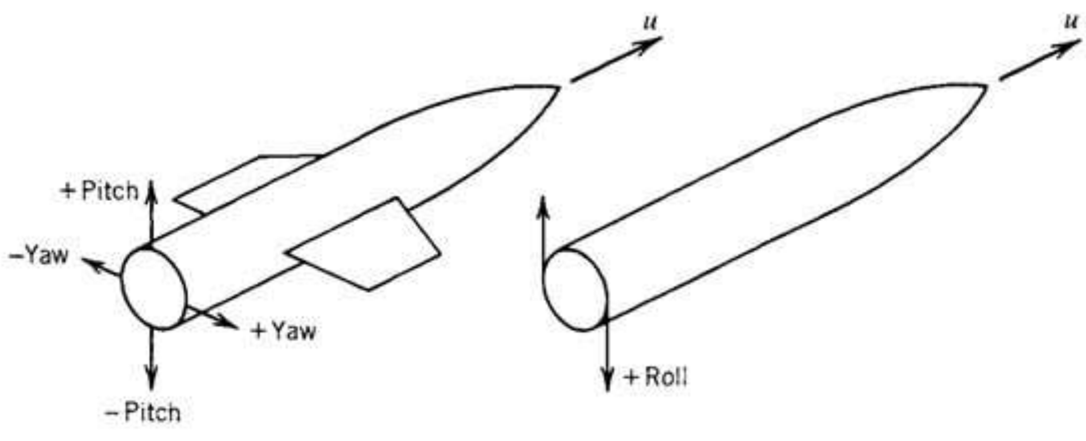
Aerodynamic fins (fixed and movable) continue to be very effective for controlling vehicle flight within the earth's atmosphere, and almost all weather rockets, antiaircraft missiles, and air-to-surface missiles use them. Even though aerodynamic control surfaces provide some additional drag, their effectiveness in terms of vehicle weight, turning moment, and actuating power consumption is difficult to surpass with any other flight control method.

Reasons For Thrust Vector Control:

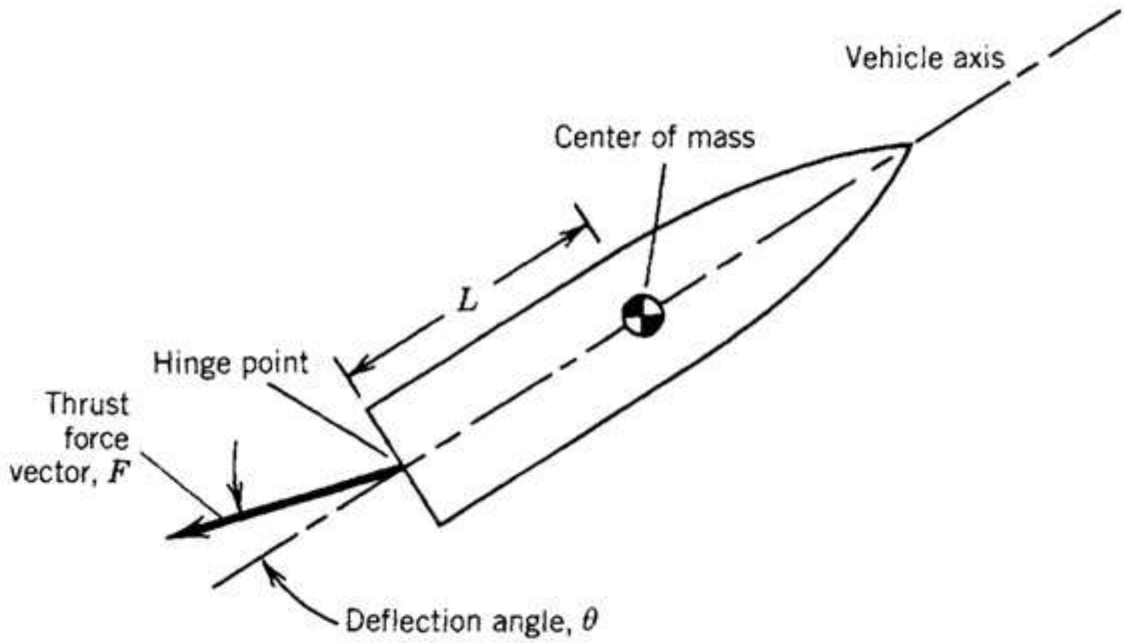
1. To willfully change a flight path or trajectory (e.g., changing the direction of the flight path of a target-seeking missile).
2. To rotate the vehicle or change its attitude during powered flight.
3. To correct for deviation from the intended trajectory or the attitude during powered flight.
4. To correct for thrust misalignment of a fixed nozzle in the main propulsion system during its operation, when the main thrust vector misses the vehicle's center of gravity.

Thrust Vector Control – Pitch, Yaw And Roll:

Pitch moments are those that raise or lower the nose of a vehicle; yaw moments turn the nose sideways; and roll moments are applied about the main axis of the flying vehicle.



Usually, the thrust vector of the main rocket nozzle is in the direction of the vehicle axis and goes through the vehicle's center of gravity. Thus it is possible to obtain pitch and yaw control moments by the simple deflection of the main rocket thrust vector; however, roll control usually requires the use of two or more rotary vanes or two or more separately hinged propulsion system nozzles. The below image explains the pitch moment obtained by a hinged thrust chamber or nozzle. The side force and the pitch moment vary as the sine of the effective angle of thrust vector deflection.



<u>Thrust Vector Control With A Single Nozzle</u>:

Many different mechanisms have been used successfully. They can be classified into four categories:

1. Mechanical deflection of the nozzle or thrust chamber.
2. Insertion of heat-resistant movable bodies into the exhaust jet; these experience aerodynamic forces and cause a deflection of a part of the exhaust gas flow.
3. Injection of fluid into the side of the diverging nozzle section, causing an asymmetrical distortion of the supersonic exhaust flow.
4. Separate thrust-producing devices that are not part of the main flow through the nozzle.

1. <u>Mechanical Deflection Of The Nozzle</u>:

In the **hinge or gimbal scheme** (a hinge permits rotation about one axis only, whereas a gimbal is essentially a universal joint), the whole engine is pivoted on a bearing and thus the thrust vector is rotated. For small angles this scheme has negligible losses in specific impulse and is used in many vehicles. It requires a flexible set of propellant piping (bellows) to allow the propellant to flow from the tanks of the vehicle to the movable engine.

Type	L/S[a]	Advantages	Disadvantages
Gimbal or hinge	L	Simple, proven technology; low torques, low power; ±12° duration limited only by propellant supply; very small thrust loss	Requires flexible piping; high inertia; large actuators for high slew rate
Movable nozzle (flexible bearing)	S	Proven technology; no sliding, moving seals; predictable actuation power; up to ±12°	High actuation forces; high torque at low temperatures; variable actuation force
Movable nozzle (rotary ball with gas seal)	S	Proven technology; no thrust loss if entire nozzle is moved; ±20° possible	Sliding, moving hot gas spherical seal; highly variable actuation power; limited duration; needs continuous load to maintain seal

2. Insertion Of Heat-Resistant Movable Bodies Into The Exhaust Jet:

Jet vanes are pairs of heat-resistant, aerodynamic wing-shaped surfaces submerged in the exhaust jet of a fixed rocket nozzle. They were first used about 55 years ago. They cause extra drag (2 to 5% less Is; drag increases with larger vane deflections) and erosion of the vane material. Graphite jet vanes were used in the German V-2 missile in World War II and in the Scud missiles fired by Iraq in 1991. The advantage of having roll control with a single nozzle often outweighs the performance penalties.

Type	L/S[a]	Advantages	Disadvantages
Jet vanes	L/S	Proven technology; low actuation power; high slew rate; roll control with single nozzle; ±9°	Thrust loss of 0.5 to 3%; erosion of jet vanes; limited duration; extends missile length
Jet tabs	S	Proven technology; high slew rate; low actuation power; compact package	Erosion of tabs; thrust loss, but only when tab is in the jet; limited duration
Jetavator	S	Proven on Polaris missile; low actuation power; can be lightweight	Erosion and thrust loss; induces vehicle base hot gas recirculation; limited duration

3. Injection Of Fluid Into The Side Of The Diverging Nozzle Section:

The **injection of secondary fluid** through the wall of the nozzle into the main gas stream has the effect of forming oblique shocks in the nozzle diverging section, thus causing an unsymmetrical distribution of the main gas flow, which produces a side force. The secondary fluid can be stored liquid or gas from a separate hot gas generator (the gas would then still be sufficiently cool to be piped), a direct bleed from the chamber, or the injection of a catalyzed monopropellant. When the deflections are small, this is a low-loss scheme, but for large moments (large side forces) the amount of secondary fluid becomes excessive.

Type	L/S[a]	Advantages	Disadvantages
Liquid-side injection	S/L	Proven technology; specific impulse of injectant nearly offsets weight penalty; high slew rate; easy to adapt to various motors; can check out before flight; components are reusable; duration limited by liquid supply; ±6°	Toxic liquids are needed for high performance; often difficult packaging for tanks and feed system; sometimes requires excessive maintenance; potential spills and toxic fumes with some propellants; limited to low vector angle applications
Hot-gas-side injection	S/L	Lightweight; low actuation power; high slew rate; low volume/ compact; low performance loss	Multiple hot sliding contacts and seals in hot gas valve; hot piping expansion; limited duration; requires special hot gas valves; technology is not yet proven

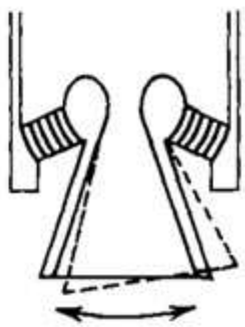
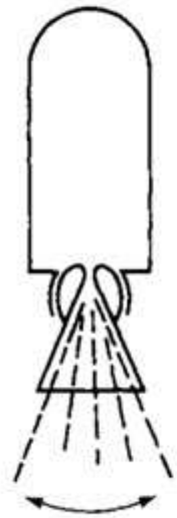
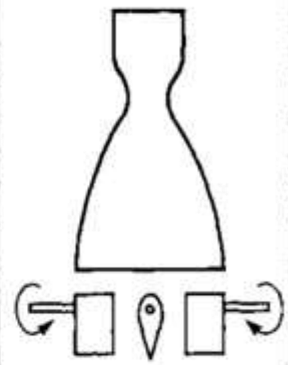
4. Separate Thrust-Producing Devices:

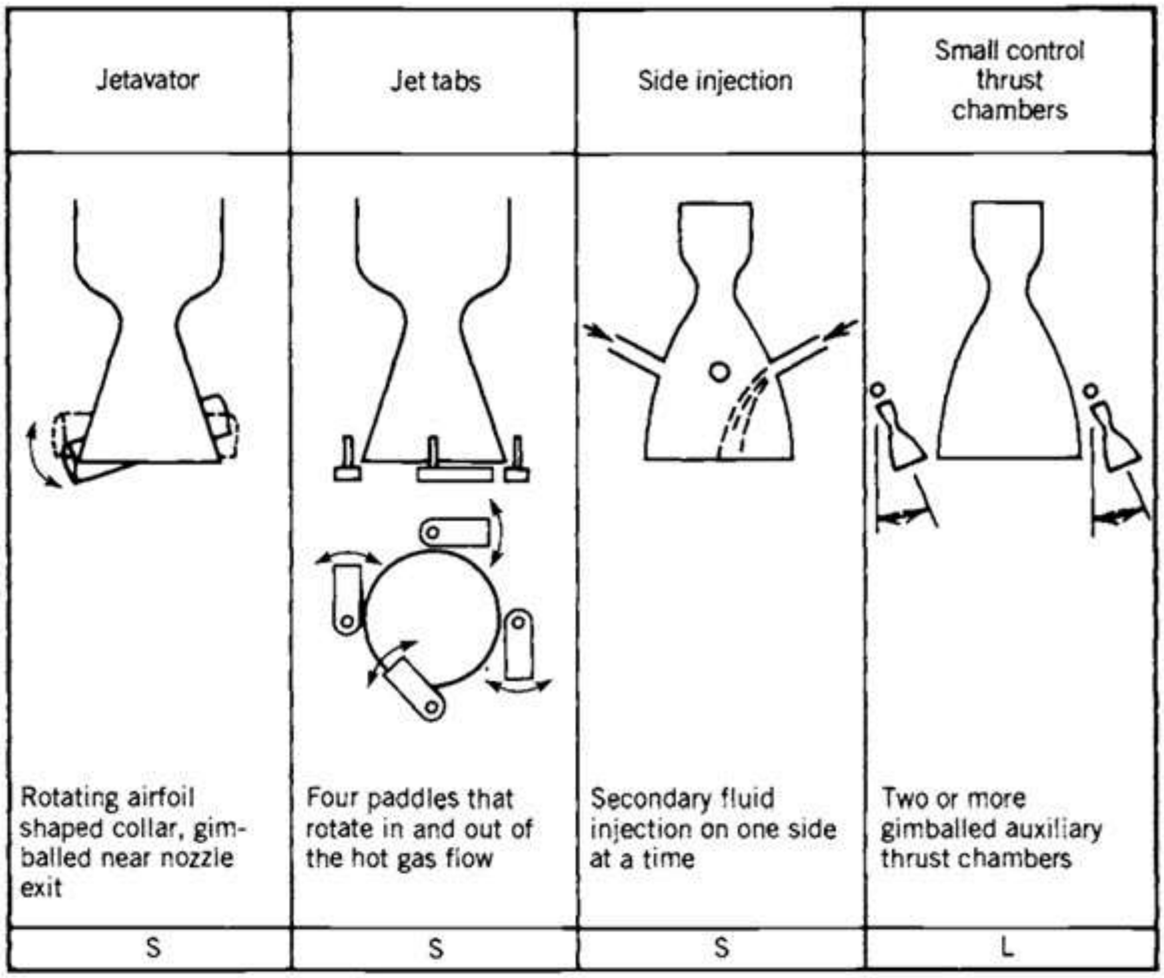
Small auxiliary thrust chambers were used in the Thor and early version of Atlas missiles. They provide roll control while the principal rocket engine operates. They are fed from the same feed system as the main rocket engine. This scheme is still used on some Russian booster rocket vehicles.

Type	L/S[a]	Advantages	Disadvantages
Hinged auxiliary thrust chambers for high thrust engine	L	Proven technology; feed from main turbopump; low performance loss; compact; low actuation power; no hot moving surfaces; unlimited duration	Additional components and complexity; moments applied to vehicle are small; not used for 15 years in USA
Turbine exhaust gas swivel for large engine	L	Swivel joint is at low pressure; low performance loss; lightweight; proven technology	Limited side forces; moderately hot swivel joint; used for roll control only

Schematic Diagrams of Eight Different Thrust Vector Control Mechanism:

Actuators and structural details are not shown. The letter L means it is used with liquid propellant rocket engines and S means it is used with solid propellant motors.

Gimbal or hinge	Flexible laminated bearing	Flexible nozzle joint	Jet vanes
			
Universal joint suspension for thrust chamber	Nozzle is held by ring of alternate layers of molded elastomer and spherically formed sheet metal	Sealed rotary ball joint	Four rotating heat resistant aerodynamic vanes in jet
L	S	S	L/S



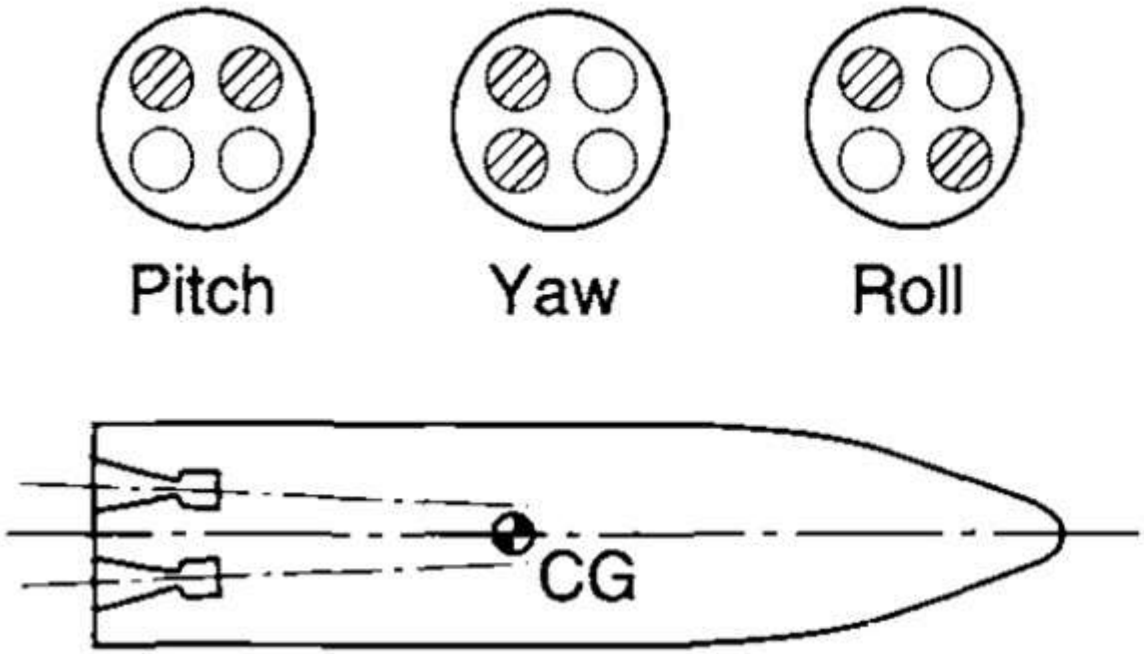
Of all the mechanical deflection types, the **movable nozzles** are the most efficient. They do not significantly reduce the thrust or the specific impulse and are weight-competitive with the other mechanical types. The flexible nozzle is a common type of Thrust Vector Control used with solid propellant motors. The molded, multilayer bearing pack acts as a seal, a load transfer bearing, and a viscoelastic flexure. It uses the deformation of a stacked set of doubly curved elastomeric (rubbery) layers between spherical metal sheets to carry the loads and allow an angular deflection of the nozzle axis.

The flexible seal nozzle has been used in launch vehicles and large strategic missiles, where the environmental temperature extremes are modest. At low temperature the elastomer becomes stiff and the actuation torques increase substantially, requiring a much larger actuation system. There are double seals to prevent leaks of hot gas and various insulators to keep the structure below 200°F or 93°C.

Thrust Vector Control With Multiple Thrust Chambers Or Nozzles:
Roll control can be obtained only if there are at least two separate vectorable nozzles, four fixed pulsing or throttled flow nozzles, or two jet vanes submerged in the exhaust gas from a single nozzle. Several concepts have been developed and flown that use two or more rocket engines or a single engine or motor with two or more actuated nozzles. Two fully gimballed thrust chambers or motor nozzles can provide roll control with very slight differential angular deflections.

For pitch and yaw control, the deflection would be larger, be of the same angle and direction for both nozzles, and the deflection magnitude would be the same for both nozzles. This can also be achieved with four hinged or gimballed nozzles.

The differential throttling concept shown in the below diagram has no gimbal and does not use any of the methods used with single nozzles. It has four fixed thrust chambers and their axes are almost parallel to and set off from the vehicle's centerline. Two of the four thrust chambers are selectively throttled (typically the thrust is reduced by only 2 to 15 %). The four nozzles may be supplied from the same feed system or they may belong to four separate but identical rocket engines.



Differential throttling with four fixed-position thrust chambers can provide flight maneuvers. In this simple diagram the shaded nozzle exits indicate a throttled condition or reduced thrust. The larger forces from the unthrottled engines impose turning moments on the vehicle. For roll control the nozzles are slightly inclined and their individual thrust vectors do not go through the center of gravity of the vehicle.

<u>Methods Of Cooling In Liquid Rocket</u>:

1. Regenerative Cooling
2. Dump Cooling
3. Film Cooling
4. Transpiration Cooling
5. Ablative Cooling
6. Radiation Cooling

1. <u>Regenerative Cooling In Liquid Rocket</u>:

Regenerative cooling is the most widely used method of cooling a thrust chamber and is accomplished by flowing high-velocity coolant over the back side of the chamber hot gas wall to convectively cool the hot gas liner. The coolant with the heat input from cooling the liner is then discharged into the injector and utilized as a propellant.

Earlier thrust chamber designs, had low chamber pressure, low heat flux and low coolant pressure requirements, which could be satisfied by a simplified "double wall chamber" design with regenerative and film cooling.

For subsequent rocket engine applications, however, chamber pressures were increased and the cooling requirements became more difficult to satisfy. It became necessary to design new coolant configurations that were more efficient structurally and had improved heat transfer characteristics.

This led to the design of "tubular wall" thrust chambers, by far the most widely used design approach for the vast majority of large rocket engine applications. These chamber designs have been successfully used and several other Air Force and NASA rocket engine applications. The primary advantage of the design is its light weight and the large experience base that has accrued. But as chamber pressures and hot gas wall heat fluxes have continued to increase (>100 atm), still more effective methods have been needed.

One solution has been "channel wall" thrust chambers, so named because the hot gas wall cooling is accomplished by flowing coolant through rectangular channels, which are machined or formed into a hot gas liner fabricated from a high-conductivity material, such as copper or a copper alloy.. Heat transfer and structural characteristics are excellent.

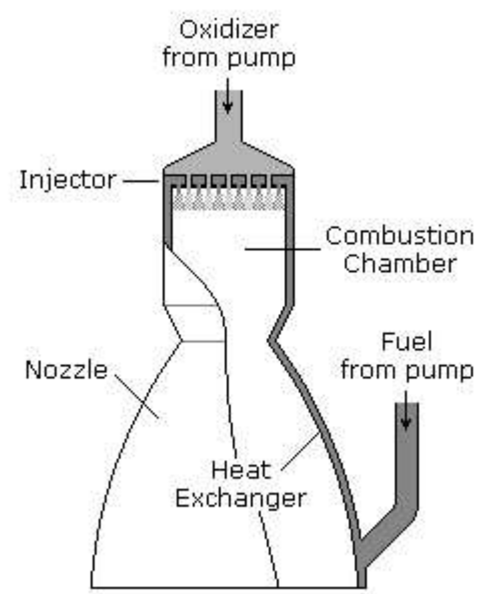
Basically, there are three domains in a regenerative cooled rocket engine.

1. Gas Domain (Combusted Gases) – Convection and Radiation heat transfer
2. Liquid Domain (Coolant) – Convection heat transfer
3. Solid Domain (Thrust chamber wall) – Conduction heat transfer

Heat transfer from the outer surface of thrust chamber to the environment can be neglected and the outer surface wall can be assumed as adiabatic.

In addition to the regenerative cooled designs mentioned above, other thrust chamber designs have been fabricated for rocket engines using dump cooling, film cooling, transpiration cooling, ablative liners and radiation cooling. Although regeneratively cooled combustion chambers have proven to be the best approach for cooling large liquid rocket engines, other methods of cooling have also been successfully used for cooling thrust chamber assemblies.

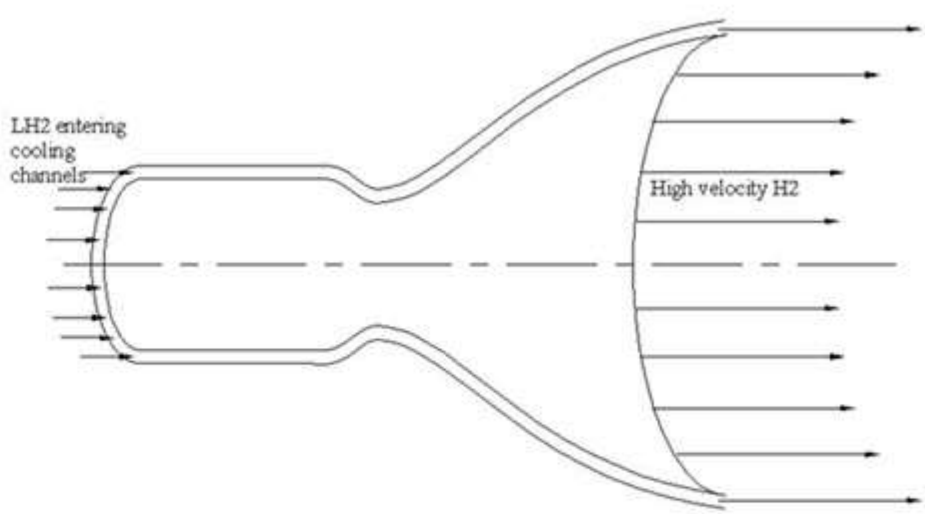
Regenerative Cooling In Liquid Propellant Rocket Engines



2. <u>Dump Cooling In Liquid Rocket</u>:

Dump cooling, which is similar to regenerative cooling because the coolant flows through small passages over the back side of the thrust chamber wall. The difference, however, is that after cooling the thrust chamber, the coolant is discharged overboard through openings at the aft end of the divergent nozzle. This method has limited application because of the performance loss resulting from dumping the coolant overboard. To date, dump cooling has not been used in an actual application.

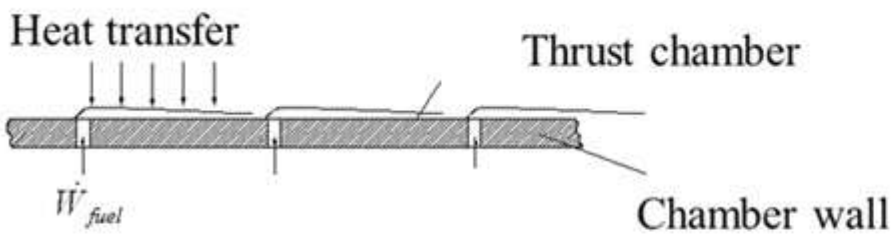
Dump Cooling In Liquid Propellant Rocket Engines



3. Film Cooling In Liquid Rocket:

Film cooling provides protection from excessive heat by introducing a thin film of coolant or propellant through orifices around the injector periphery or through manifolded orifices in the chamber wall near the injector or chamber throat region. This method is typically used in high heat flux regions and in combination with regenerative cooling.

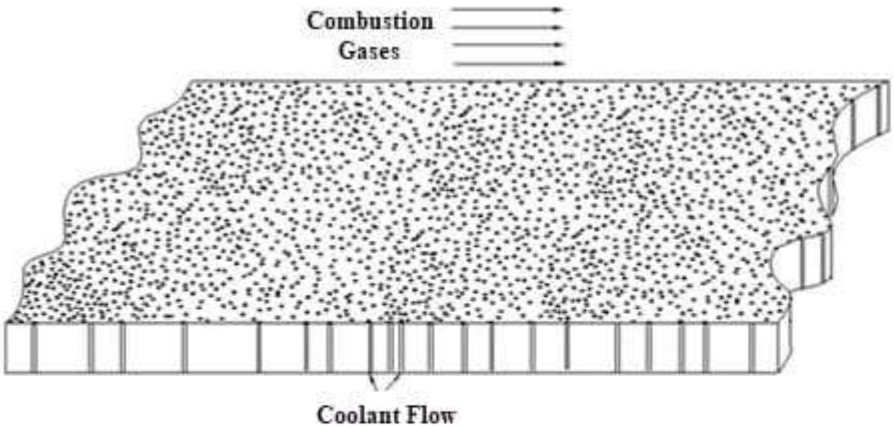
Film Cooling In Liquid Propellant Rocket Engines



4. Transpiration Cooling In Liquid Rocket:

Transpiration cooling provides coolant (either gaseous or liquid propellant) through a porous chamber wall at a rate sufficient to maintain the chamber hot gas wall to the desired temperature. The technique is really a special case of film cooling.

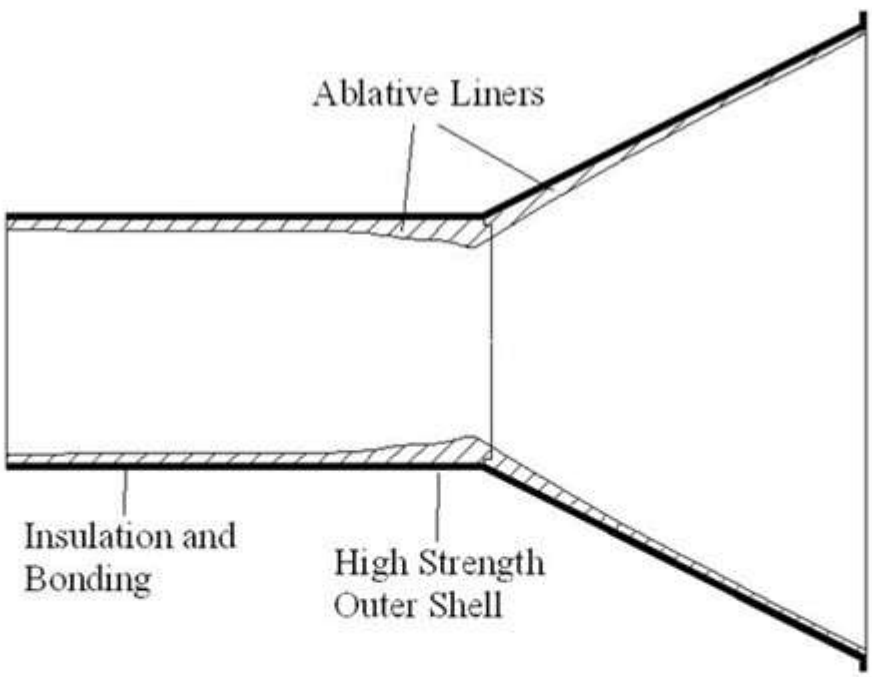
Transpiration Cooling In Liquid Propellant Rocket Engines



5. Ablative Cooling In Liquid Rocket:

With ablative cooling, combustion gas-side wall material is sacrificed by melting, vaporization and chemical changes to dissipate heat. As a result, relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process.

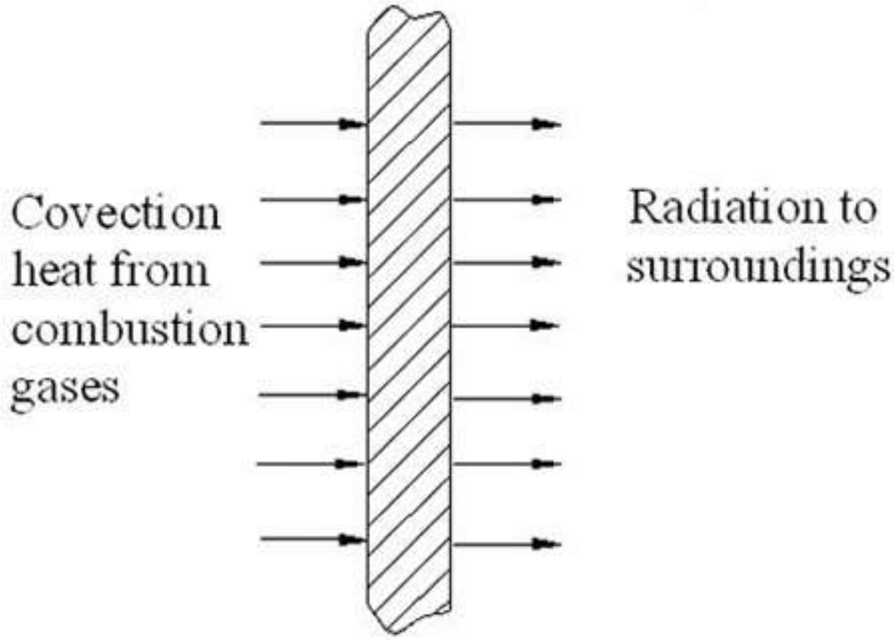
Ablative Cooling In Liquid Propellant Rocket Engines



6. Radiation Cooling In Liquid Rocket:

With radiation cooling, heat is radiated from the outer surface of the combustion chamber or nozzle extension wall. Radiation cooling is typically used for small thrust chambers with a high-temperature wall material (refractory) and in low-heat flux regions, such as a nozzle extension.

Radiation Cooling In Liquid Propellant Rocket Engines



Combustion Instability In Liquid Propellant Rockets?

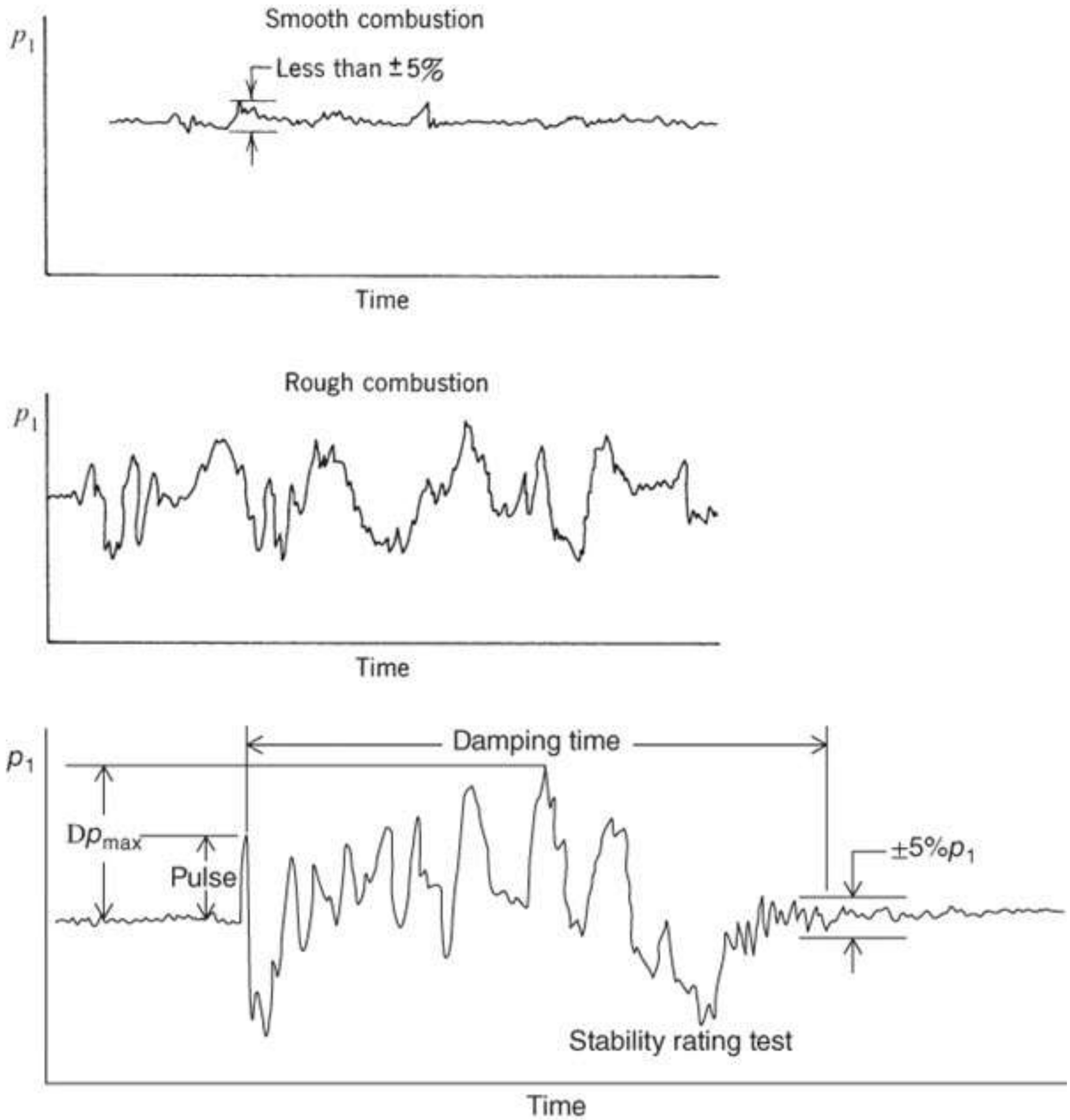
Combustion instabilities are physical phenomena occurring in a reacting flow (e.g., a flame) in which some perturbations, even very small ones, grow and then become large enough to alter the features of the flow in some particular way. In many practical cases, the appearance of combustion instabilities is undesirable. When rocket combustion processes are not well controlled, combustion instabilities may grow and very quickly cause excessive pressure-induced vibrational forces (that may break engine parts) or excessive heat transfer (that may melt thrust chamber parts).

The below table lists the principal types of combustion generated vibrations encountered in liquid rocket thrust chambers. Admittedly, liquid propellant rocket thrust chamber combustion is never perfectly smooth because minor fluctuations of pressure, temperature, and velocity are always present. When these fluctuations interact with the natural frequencies of the propellant feed system (with or without vehicle structure) or with the chamber acoustics, large periodic superimposed oscillations develop.

In normal rocket propulsion practice, ***smooth combustion*** is said to occur when pressure fluctuations during operation do not exceed ± 5% of the mean chamber pressure. Combustion that generates larger pressure fluctuations, as measured at chamber wall locations and occurring at completely random intervals is called ***rough combustion***. Unstable combustion, or ***combustion instability***, displays organized oscillations occurring at well-defined intervals with pressure peaks that may be maintained, may increase, or may die out. These periodic peaks represent fairly large concentrations of vibratory energy and can be easily recognized against the random-noise background using high-frequency pressure measurements.

Type and Word Description	Frequency Range (Hz)	Cause Relationship
Low frequency, called chugging or feed system instability	10–400	Linked with pressure interactions b feed system, if not the entire vehicl chamber
Intermediate frequency, called acoustic [a] instability, buzzing, or entropy waves	400–1000	Linked with mechanical vibrations structure, injector manifold, flow e ratio fluctuations, and propellant fe
High frequency, called screaming, screeching, or squealing	Above 1000	Linked with combustion process fo and chamber acoustical resonance

[a] Use of the word *acoustic* stems from the fact the frequency of the oscillations is related to combustion velocity of sound in the combustion gas.



Typical Simplified Oscillograph Traces Of Chamber Pressure p1 With Time For Different Combustion Events

Types Of Combustion Instability In Liquid Propellant Rockets:

There are three basic types of combustion instability in liquid propellant rockets and are,

1. Low Frequency – Chugging / Feed System Instability
2. Intermediate Frequency – Acoustic Instability / Buzzing / Entropy Waves
3. High Frequency – Screaming / Screeching / Squealing

1. Combustion Instability In Liquid Propellant Rockets – Chugging / Feed System Instability:

Chugging, the first type of combustion instability, stems mostly from the elastic nature of the feed systems and vehicle's structures and/or from the imposition of propulsive forces upon the vehicle. Chugging in an engine or thrust chamber assembly may occur at a test facility or during flight, especially with low-chamber-pressure engines (100 to 500 psia). It may originate from propellant pump cavitation, gas entrapment in propellant flows, tank pressurization control fluctuations, and/or vibration of engine supports and propellant lines. It may also be caused by resonances in the engine feed system (such as oscillating bellows inducing periodic flow fluctuations) or by the coupling of structural and feed system frequencies.

When both the vehicle structure and the propellant liquid in the feed system have similar natural frequencies, then a coupling of forces occurs that may not only maintain but also strongly amplify existing oscillations. Propellant flow rate disturbances, usually at 10 to 50 Hz, give rise to such low-frequency longitudinal combustion instabilities, producing longitudinal vibration modes in the vehicle. ***Pogo instability*** refers to a vehicle's long-feed pipe instability since it is similar to a pogo jumping stick motion. Pogo instabilities can occur in the large, long propellant feed lines of large vehicles such as space launch vehicles or in ballistic missiles.

Avoiding objectionable engine–vehicle coupled oscillations is best accomplished during initial vehicle design, in contrast to applying "fixes" later, as had been the case with rocket engines of older large flight vehicles. Analytical methods exist for understanding most vibration modes and damping tendencies of major vehicle components, including propellant tanks, tank pressurization systems, propellant flow lines, engines, and basic vehicle structures.

The below figure is a simplified spring–mass model of a typical two-stage vehicle, illustrates the complexity of the analytical problem. Fortunately, most vibrational characteristics of the rocket assembly can be substantially controlled by introducing damping into major components or subassemblies. Techniques for damping pogo instabilities include the use of energy absorption devices in fluid flow lines, perforated tank liners, special tank supports, and properly designed engine, interstage, and payload support structures.

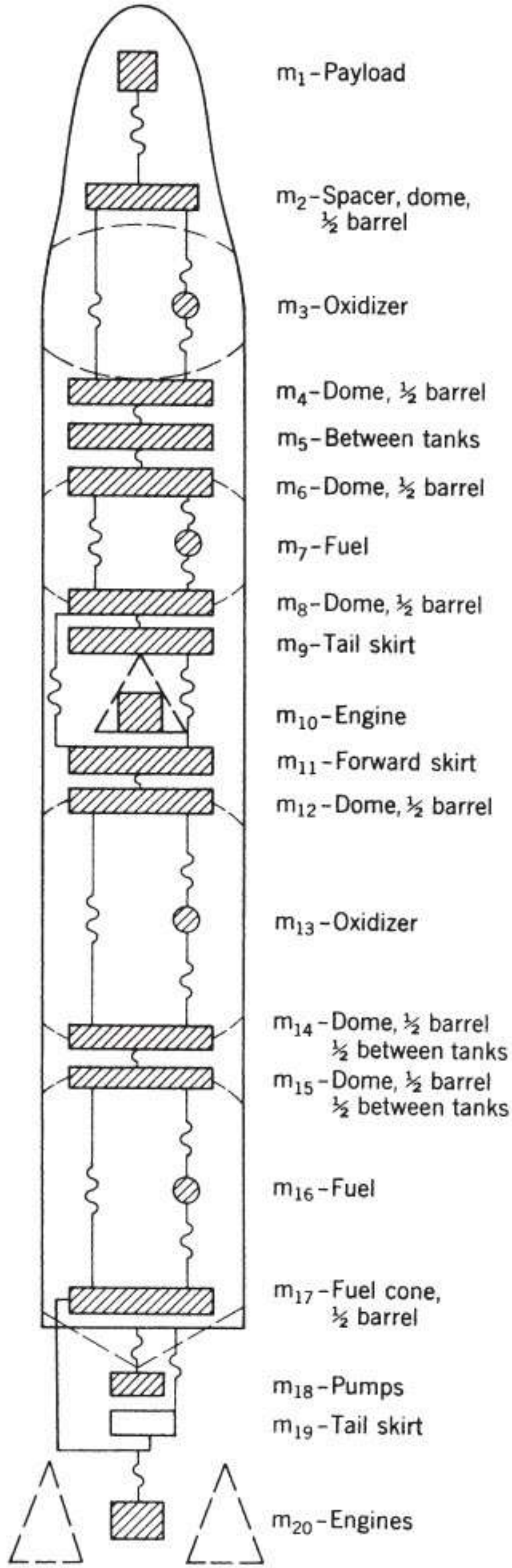
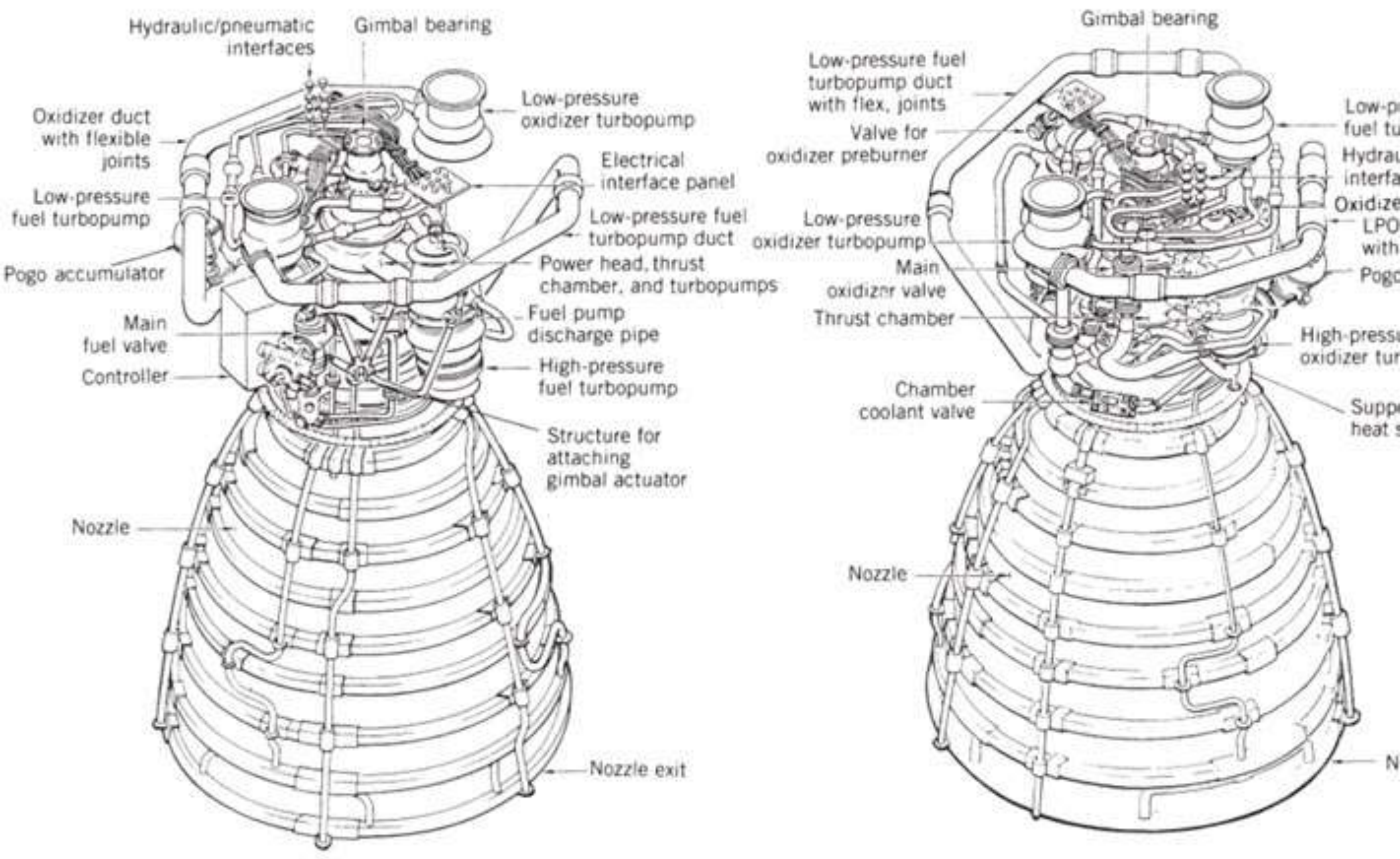

m_1 - Payload
m_2 - Spacer, dome, ½ barrel
m_3 - Oxidizer
m_4 - Dome, ½ barrel
m_5 - Between tanks
m_6 - Dome, ½ barrel
m_7 - Fuel
m_8 - Dome, ½ barrel
m_9 - Tail skirt
m_{10} - Engine
m_{11} - Forward skirt
m_{12} - Dome, ½ barrel
m_{13} - Oxidizer
m_{14} - Dome, ½ barrel ½ between tanks
m_{15} - Dome, ½ barrel ½ between tanks
m_{16} - Fuel
m_{17} - Fuel cone, ½ barrel
m_{18} - Pumps
m_{19} - Tail skirt
m_{20} - Engines

Diagram of a typical two-stage vehicle spring–mass model used in analysis of pogo vibration in the vertical direction. The cross-hatched areas represent masses and the wiggly lines indicate springs.
A partially gas-filled, spherical, small ***pogo accumulator*** attached to the main oxidizer feed line has been used as an effective damping device. It was used in the oxidizer feed line of the Space Shuttle main engine (SSME) between the oxidizer booster pump and the main oxidizer pump.



2. <u>Combustion Instability In Liquid Propellant Rockets – Acoustic Instability / Buzzing / Entropy Waves</u>:

Buzzing, an intermediate frequency type of instability, seldom derives from pressure perturbations greater than 5% of the mean in the combustion chamber and is usually not accompanied by large vibratory energies. It is more often an annoying noise than a damaging effect, although the occurrence of buzzing may sometimes initiate high-frequency instabilities. It is characteristic of a coupling between the combustion process and the flow in a portion of the propellant feed system.
Buzzing initiation is thought to originate from the combustion process itself. Acoustic resonances of the combustion chamber with some critical portion of the propellant flow system, sometimes originating in a pump, promote continuation of these buzzing effects. This type of instability seems to be more prevalent in medium-size engines (2000 to 250,000 N thrust or about 500 to 60,000 lbf) than in larger engines.

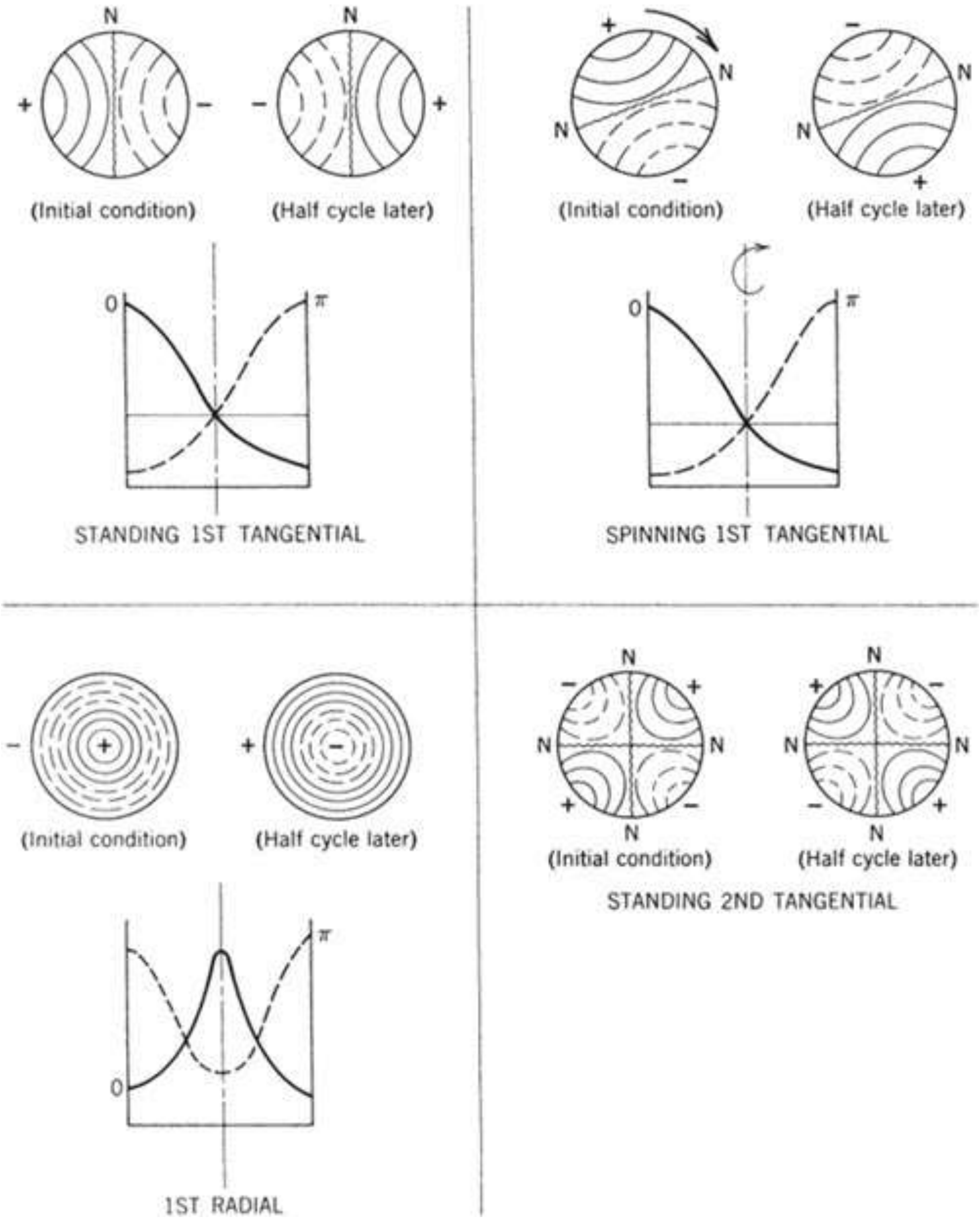
3. <u>Combustion Instability In Liquid Propellant Rockets – Screaming / Screeching / Squealing</u>:

The third type of instability, ***screeching*** or *screaming*, produces high frequencies (4 to 20 kHz) and is the most perplexing and common feature in new engine development. Many liquid rocket engines and solid propellant motors experience some high-frequency instability during their developmental phase. Since energy content increases with frequency, this type can be the most damaging, capable of destroying an engine in less than 1 sec.

Once encountered, it is often difficult to prove that any newly incorporated “fixes” or improvements will render the engine “stable” under all launch and flight conditions. Screeching may be treated as a phenomenon solely related to the combustion chamber and not generally influenced by feed systems or structures.

High-frequency instabilities occur in two basic modes, *longitudinal* and *transverse*. The ***longitudinal mode*** (sometimes called *organ pipe mode*) propagates along axial planes of the combustion chamber, and its pressure waves are reflected at the injector face and at the converging nozzle entrance section. The ***transverse mode*** propagates along planes perpendicular to the chamber axis and can be broken down into *tangential* and *radial* modes. Transverse mode instabilities predominate in large liquid rockets, particularly in the vicinity of the injector.

The below figure shows pressure distributions at various time intervals in a combustion chamber of cylindrical cross section that is encountering transverse mode instabilities. Two kinds of wave form have been observed in tangential vibrations; one can be considered a ***standing*** wave because it remains in a fixed position while its pressure amplitude fluctuates; the other is a ***spinning*** or ***traveling*** tangential wave with an associated rotation of the whole vibratory system—this waveform can be visualized as maintaining constant amplitude while the wave rotates. Combinations of transverse and longitudinal modes may also occur and their frequency can also be estimated.



Simplified representation of transverse pressure oscillation modes at two time intervals in a cylindrical combustion chamber. The solid-line curves indicate pressures greater than the normal or mean operating pressure and the dashed lines indicate lower pressures. The N–N lines show node locations for these wave modes.

Screeching is believed to be predominantly driven by acoustical-energy-stimulated variations in droplet vaporization and/or mixing, by local detonations, and by acoustic changes in combustion rates. Thus, with any assisting acoustic properties, once triggered, high-frequency combustion instabilities can rapidly induce destructive modes. Invariably, any distinct boundary layer may disappear and heat transfer rates may increase by an order of magnitude causing metal melting and wall burn-throughs, sometimes within less than 1 sec. Here, tangential modes appear to be the most damaging, with heat transfer rates during an instability often increasing 2 to 10 times and instantaneous pressure peaks about twice as high as with stable operation.

One source that possibly triggers high-frequency pressure-wave instabilities is a rocket combustion phenomenon called ***popping***. Popping is an undesirable random high-amplitude pressure disturbance that arises during the steady-state operation of rocket engines that use hypergolic propellants. These “pops” exhibit some of the characteristics of a detonation wave. The pressure rise times are a few microseconds and the pressure ratios across the wave can be as high as 7:1.

Some combustion instabilities are induced by *pulsations in the liquid flow* originating at the turbopumps. Unsteady liquid flows may result from irregular cavitation at the leading edge of inducer impellers or main pump impellers. Also, when an impeller's trailing edge passes a rib or stationary vane in the volute, a small pressure perturbation always results in the liquid as it travels downstream to the injector. These two types of pressure fluctuation can be greatly amplified if they coincide with the natural frequencies of combustion induced vibrations in the chamber. Also, liquid propellant feed systems may generate oscillations under certain conditions.

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ESTD : 2001

Department of Aeronautical Engineering

(R18)

AIR BREATHING PROPULSION

Lecture Notes

B. Tech III YEAR – II SEM

Prepared by

Dr.S.R.Dhineshkumar
(Professor)Dept.Aero

SPACE PROPULSION

B.Tech. III Year AE II Sem.LT/P/DC

3 0/0/0 3

Pre-Requisites: Nil

Course Objectives:

- To know about the propulsion system used in rockets and missiles
- Discuss the working principle of solid and liquid propellant rockets and gain basic knowledgeof hybrid rocket propulsion.

Course Outcomes: To be able to

- Understand about trajectory and orbits
- Illustrate electric propulsion techniques, ion and nuclear rocket and the performances ofdifferent advanced propulsion systems.
- Evaluate various space missions, parameters to be considered for designing trajectories androcket mission profiles
- Understand the fundamentals of chemical rocket propulsion, types of igniters andperformance considerations of rockets.

UNIT - I

Principles of Rocket Propulsion: History of rockets, Newtons third law, orbits and space flight, types of orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO,launch assists.

UNIT - II

Fundamentals of Rocket Propulsion: Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Rocket performance considerations of rockets, types of igniters, preliminary concepts in nozzle less propulsion, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.

UNIT - III

Solid Rocket Propulsion: Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Erosive burning in solid propellant rockets, combustion instability, strand burner and T- burner, applications and advantages of solid propellant rockets.

UNIT - IV

Liquid and Hybrid Rocket Propulsion: Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.

UNIT-V

Advanced Propulsion Techniques: Electric rocket propulsion, types of electric propulsion techniques, Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemical rocket propulsion systems, future applications of electric propulsion systems, Solar sail.

TEXT BOOKS:

1. Hill, P.G. and Peterson, C.R., ―Mechanics and Thermodynamics of Propulsion‖, 2nd Edition, Addison Wesley, 1992.
2. Turner, M. J. L., ―Rocket and Spacecraft Propulsion‖, 2nd Edition, MIT Press, 1922.
3. Hieter and Pratt, ―Hypersonic Air breathing propulsion‖ 5th Edition, 1993.

REFERENCE BOOKS:

1. Sutton, G.P., ―Rocket Propulsion Elements‖ John Wiley & Sons Inc., New York, 5th Edition, 1993.
2. Mathur, M. L., and Sharma, R.P., ―Gas Turbine, Jet and Rocket Propulsion‖, Standard Publishers and Distributors, Delhi, 1988, Tajmar, M., Advanced Space Propulsion Systems, Springer 2003

Advanced Propulsion Techniques

UNIT-V

Advanced Propulsion Techniques

Nuclear Rocket

Power-Thrust-Energy: The high specific energy of nuclear fuel is the reason which makes nuclear propulsion ideal for deep space missions including manned missionsto other planets.

For voyages to planets, a spacecraft needs to be given a very high velocity of above11 km/s. The power in the exhaust stream will be

$$P = \frac{1}{2} m v_e^2$$

The thrust and power can be related as

$$F = m v_e$$

$$F = 2\frac{P}{v_e}$$

m is the mass flow rate.

Considering interplanatory mission with a departure velocity of 11km/s, the specific energy/power (per unit mass) required works out to 60.5 MJ/kg.

The maximum exhaust velocity of a **LOX/LH** engine is about 4.5 km/s and it worksout to **energy per kg as 10.4 MJ/kg**. So about 6 kg of propellant is needed to be burnt for every 1 kg of vehicle mass, in order to provide enough energy to set a vehicle off on interplanetary mission.

In comparison, the **energy contained in a kg of pure uranium 235 is 79.3 x 10^6 MJ**.A single kg of uranium 235 can provide energy to place a 1000 t vehicle for interplanetary mission.

The high specific energy of nuclear fuel is a major advantage for high energyinterplanetary missions.

The energy stored in nuclear propellants is 10^7 - 10^9 times higher than chemicalpropellants. A propulsion system using nuclear energy can achieve any specific impulse comparable to the speed of light.

Nuclear Fuel Basics:

Nuclear processes (Fission or Fusion) use very small quantities of matter. A workingfluid, usually is coupled with nuclear reaction products.

After fusion or fission of atoms, the end product will have smaller mass than theinitial atoms**. This mass defect is directly transferred in to energy based on**

Einstein’s equation E = mc^2

Fission Propulsion: Nuclear Fission is a process in which a large nucleus of an atomsplits into two smaller nuclei (lighter nuclei) with release of energy. The fission process often produces free neutrons and releases a very large amount of energy. The splitting of nucleus is as a result of neutron bombardment.

The mass changes and associated energy changes in nuclear reactions are significant. For example, the **energy released from the nuclear reaction of 1 kg of uranium is equivalent to** the energy released during the combustion of **about fourbillion kilograms of coal**.

Nuclear Fission:

Nuclear Fission is used in high thrust applications. Fission is a process where a neutron is absorbed a uranium nucleus, which causes the nucleus to split into twonuclei (of mass about half that of uranium). The mass defect causes release of energy, in the form of kinetic energy of the two fission fragments. The splitting process is also associated with release of two or more neutrons are emitted at thesame time as the fission of the nucleus occurs. These neutrons go to interact with another nucleus and cause to split, thereby, setting up a chain reaction. Since rateat which energy is released depends only on the neutron flux, the power output ofa fission system is controlled by inserting materials that absorb neutrons.

In a controlled nuclear fission, the uranium becomes very hot, leading to melting of

Uranium. Hence, to continue with the energy release, it is essential to cool the uranium

extracting heat. The cooling of Uranium is accomplished using a propellant, which passes through the reactor and then expelled out of the nozzle.

Two isotopes of Uranium, U^{238} and U^{235} are available, of which U^{235} has highprobability of initiating fission process.

Nuclear Fusion:

- If two light nuclear cores are fused together (Eg. hydrogen), the resulting heavier nuclear element has less binding energy than the sum of the two original ones.
- The energy difference is released as heat.
- Fusion is more complex than fission, since in fusion, in order to bring the two positively charged nuclear cores close together, the energy of electrostatic repulsion has to be overcome and maintained
- The energy released in nuclear propulsion is governed by Einstein's equation$E=mc^2$
- Nearly all gained energy through the mass defect is released as heat.
- While Fission and Fusion transfer only part of the nuclear binding energy into heat, the Matter-Antimatter annihilation (eg. Proton-antiproton or hydrogen-antihydrogen etc) can release all the nuclear energy.

Sizing of the Reactor/Ensuring Sustainable Chain Reaction:

There are two approaches that will improve the chances of sustainable chainreaction. They are

- **Enrichment of U^{235}**: It involves increasing the percentage of U^{235} in the natural uranium to a level that highly increases the probability. The process of enrichment is complicated and costly. Natural Uranium contains very low % of U^{235} (0.72%). Although U^{238} also participates in fission process, the probability of fission initiation is very low since U^{238} requires collisions withneutrons with high energy levels, thus reducing the probability very low.

- **Use of moderator**: The second approach is to slow the neutrons quickly andreduce absorption of neutrons by U^{238} nuclei by using a moderator, usually carbon or water. The moderator is mixed with the uranium atoms in a **homogeneous reactor**, or the moderator and uranium can be in separate blocks, as **heterogeneous reactor**.

The heterogeneous reactor which uses cylindrical rods of Uranium separated byblocks of moderator, improves the probability of sustained reaction high and permits use of more natural Uranium. However, this increases the size of the reactor, as more moderator is required.

For space applications, the need to keep size low, requires use of enriched Uranium. Plutonium can also be used in the same way as enriched Uranium, butthe material is poisonous and highly radioactive. Safety issues are complex to handle.

Calculating Criticality: Criticality factor relates to calculating the space fission reactor that can attain sustainable chain reaction with minimum size. The followingkey issues are considered while deciding the size of space reactor:

- In a fission reactor using moderator, sufficient travel distance must be provided for neutrons to slow down adequately and avoid being absorptionby the U^{238} nuclei.
- The slowing down must occur in the moderator.
- When Uranium with low enrichment is used, the Uranium is concentrated inthe fuel rods, separated by blocks of moderator.
- Therefore, the size of the reactor is mainly decided by the dimensions of the moderator.
- Leakage of neutrons from the reactor reduces the neutron flux and leads tolow probability of sustained fission. Neutron leakage must be low.
- Larger reactors will have lesser leakage than the smaller ones.
- Heat generated by fission must be efficiently removed preventing reactorcore from overheating.
- Propellant flow through channels passing through the reactor must be carefully designed for efficient cooling.
- The best shape for the reactor to minimise neutron leakage and provide for propellant channels is cylindrical, with height approximately equal to diameter.

The criticality factor is defined by the “four-factor formula”, as given below:

$$K_\infty = \infty \varepsilon p f$$

$K_\infty$is called “multiplication factor” or “reproduction constant”

K_∞indicates the effective number of neutrons per fission that survive all the lossmechanisms and cause fission in another nucleus.

For $K_{\infty} < 1$, no chain reaction is

possible For $K_{\infty} > 1$, the chain reaction is

possible

$K_{\infty} = 1$ is the critical level and K_{∞} will needs to be controlled at 1 for steadyproduction of heat in the reactor.

The subscript$_\infty$ refers to a reactor size corresponding to infinite, where neutronscannot leak out throgh sides.

The four parameters that influence value of K_{∞} are:

∞ is the number of neutrons that emerge from fission of the nucleus, per incidentneutron. U^{235} nucleus produces 2.44 neutrons on an average per incident. The value of Ƞ for U^{235} is 2.07, available for furthur fission process.

The value of Ƞ must be far higher than unity for catering for loss mechanisms.

ε is the fast fission factor, indicates the probability that a neutron is available forfurthur fission process. Value of ε should be 1.

P is the "resonance escape probability", which indicates chances of absorption by U^{238} nuclei before causing furtur fission process. Value of p depends on fraction ofU^{238} in the fuel and its distribution. If the moderator slow down the neutrons

quickly, their chances of capture are reduced, with value of p high. Value of pranges from 0.6 to 0.8.

The fourth parameter f is the "thermal utilization factor", indicating probability ofcapture of low energy neutrons after slowing down by moderator.

Reactor Dimensions/Neutron Leakage:

As the size of the reactor decreases, the neutron leakage increases, less space isavailable for moderator. Therefore, more neutrons need to be provided which requires enrichment of natural Uranium. For very small reactors, almost 90% enrichment of fuel is needed.

The key factors that determine reactor size are neutron leakage from the core, andthe ability of moderator to prevent neutron absorption. Two properties of neutrons, diffusion length and slowing-down length are critical.

Diffusion length represents the way scattering in the moderator reduces the neutron flux, as the distance from source of neutrons increase. It is about 52 cm ingraphite.

The slowing-down length expresses the mean distance travelled by neutrons,through moderator before reaching thermal energies (escaping absorption).

It is about 19 cm for graphite.

For any reactor of finite dimensions, neutron leakage will occur.

Relation between neutron leakage and reactor size is given by the formula

$$N = N_0 e^{\frac{-r}{L_r}}$$

Where N & N_0 are the number of neutrons crossing a unit volume of materialin atthe source and as the distance increases, situated at a distance r from the is the diffusion length.

The neutron flux also varies with time, depending whether the reactor is sub-critical or super-critical.

The critical link between geometry of the reactor and the criticality is given by the "**buckling factor**".

The buckling factor is calculated based on neutron diffusion in a reactor of differentshapes. It is found to be inversely proporional to the the length L and radius R of the reactor.

Control:

Control of neutron flux and hence the power output is essential for the reactor. Control is maintained by using number of control rods with high absorption in thecore. The control rods move in a channel and be inserted or withdrawn from the core.

When fully inserted, they absorb the neutrons so that the reactor goes sub-criticaland the fission stops. At an intermediate position, the neutrons are absorbed just enough to retain the criticality.

The control rods are connected to a neutron flux sensor with a feedbackmechanism, to hold the reactor at any desired condition.

At the start up, the rods are withdrawn so that k is greater than one and netron fluxand power output increases. Once desired critical level is reached, the rods are inserted in to the intermediate position. Shut down is achieved by fully inserting the rods in to the core.

Reflection:

In normal operation, the neutrons diffusing out of the nuclear core will be lost in fission process or get absorbed. Smaller reactors can be designed to cause the neutrons to diffuse back again in to the reactor, after leaving the core, spending some time scattering off the nuclei in the external moderator. Some neutrons diffusing out of reactor core will participate in the fission process and the remainingcould be made to diffuse back. A core fitted with an external moderator, called “reflector” can be advantageous, in that smaller quantities of U^{235} is needed toachieve criticality.

For space based reactors, ability to control neutron reflection provides a contro element. This reduces the need for internal control rods which are inconvenient ina space reactor.

Reflector will help in

- Reducing the cost of material
- Reduce the neutron leak out of the reactor
- Better neutron density distribution in the core
- More even power distribution in the core
- Can avoid use of internal control rods for regulating neutron flux in the space reactors.

Prompt and Delayed Neutrons:

The fission process inside the nuclear core involves neutrons being released and travelling to the next nuclei/moderator along path. With-in the nuclear dimensions,the travel time is almost instantaneous, with in a few milliseconds. This would make the control mechanism of moving control rods in/out of the core to regulate neutron flux very difficult.

However, the control process is helped/made effective due to presence of “delayed neutrons”. The movement of about 1% of the neutrons is delayed because formation of unstable intermediate nuclei of isotopes like iodine and bromine which undergo decay

during the nuclear process, but will cause induce time lag between prompt neutrons and delayed neutrons.

The delayed neutrons makes the control process though movement of control rodsmore effective.

Thermal Stability:

The multiplication factor k is sensitive to temperature. K decreases when the temperature raises. This is due to the fact that density of core materials increasescausing them to expand, increasing the mean distance between collision and increases the probability of fission.

Thermal stability is a factor that makes the controlled release of fission energyeasier.

As k gets more than 1, the increased release of energy due to neutron flux beingmore, increases the temperature, which in turn, reduces the value of k. Thus thermal stability is established.

There are two factors at work, which govern the power output. For a stable state ofthe core, value of k is one. The power level depends on the neutron flux, which is stable only when k equals one.

To increase the power level, value of k is allowed to become greater than 1. Oncethe desired power level is reached, k is returned to value of 1, and the reactor contimues to produce power at the new level. A decrease of power is also established in a similar way.

Nuclear Thermal Propulsion-Principle:

The engine consists of a nuclear reactor, with the propellant used as a coolant forthe core. The heat generated by fission is carried away by the propellant, and thehot propellant is expanded in the nozzle.

The core contains highly enriched Uranium, mixed with a quantity of moderator. Higher the level of enrichment, difficlt is to control the engine and cost is also high.However, lowering the enrichment increases the size of the reactor.

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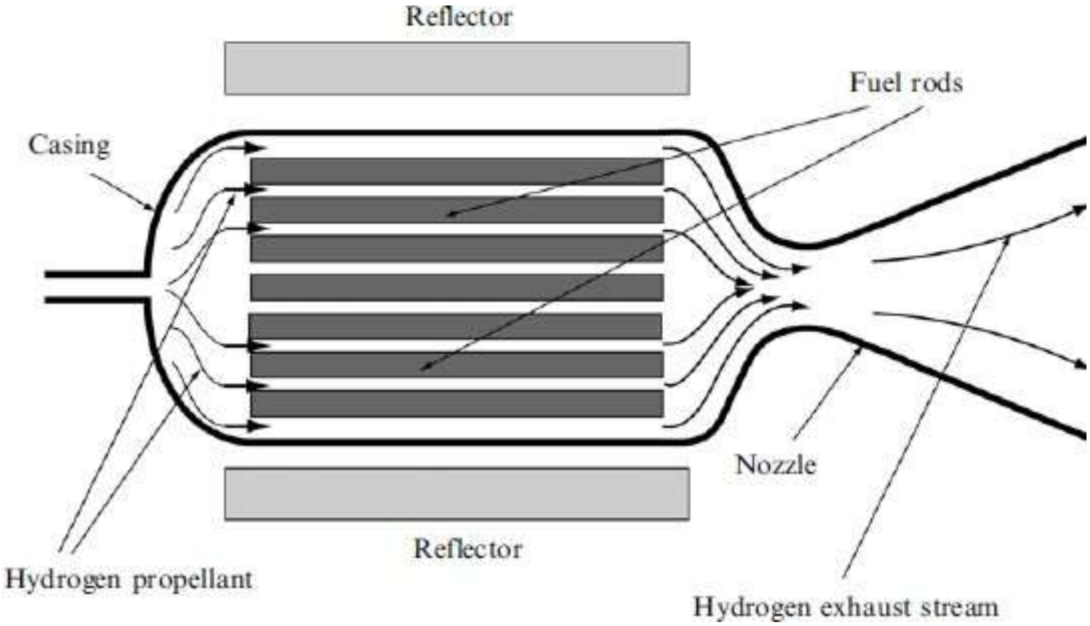
Nuclear Thermal Rocket Engine:



Hydrogen is used as propellant, which gets heated in the core, and expands in theCD nozzle.

The rate of fission and the heat production is controlled by the reflector.

Although, the nuclear thermal engine is smilar to a chemical engine as far as the principle is concerned, there are issues specific to nuclear energy/materials thatneed to be addressed.

There are several very specific engineering details are unique to the fission engine.

Radiation and its management: Nuclear fission produces the radiation effects bothduring the operation and after use. Pure uranium by itself is safe to handle, since itshalf life is very high, The fission rocket engine is safe and non radioactive as long as it has not been fired. The nuclear thermal rocket engine must be launched in space.

Radiation created during operation of the engine is through neutrons, alpha/beta particles and gamma rays. During operation, the entire core is heavy with radiation

flux. Beyond the casing, there is a high flux of both neutrons and gamma rays, which is dangerous to humans and also to electronics, both need protection duringfiring.

A radiation shield made up of one or more discs high-density material is mountedon the forward end of the engine. Any humans can be safely in the cabin well forward the engine.

An additional external shield is also provided to reduce the effect of gamma-rayflux produced by the neutron capture by the internal shield.

Other than the forward side, the radiation shield is not provided any where else onthe spacecraft.

Propellant Flow & Cooling:

The propellant flow is similar to chemical liquid engines except that there are no injectors and need for mixing. There is a need to cool several components of the engine. The power output of the reactor must be matched by the rate at which theheat is extracted by the propellant and exhausted down the nozzle.

The reflector and the casing needs to be cooled. This is done by passing the hydrogen propellant through channels in the reflector, Pumps are provided toensure flow of propellant through the channels at desired rate.

Start-up and Shut-down:

The start up of the nuclear thermal rocket is similar to a cryogenic chemical engine.The whole distribution system has to be cooled down so that the cold hydrogen does not cause thermal shock in the components. Once started, the power output of the reacor will raise very quicky, in matter of seconds. The cooling of the core casing by the propellant must keep pace with rapid heating.

Initially the pressure in the chamber is not adequate to drive the propellant turbo-pumps. Initially, during starting phase, electrical power must drive the turbo- pumps.

Once, the engine is in stable operating mode, the thrust can be varied by positioning the control rods.The power ouput is a function of neutron flux.

About 1% of theneutrons produced by fission are delayed.. When the reactor is shut down, the fission process and hence the power output continues to be produced. So fission heating will go on for several moments measured by the halflife period of the fission material.

Thus the shut down is a complicated process in nuclear fission rocket.

Potential applications of Nuclear Engines:

1. The specific energy of nuclear propellant is far greater than chemical propellant.
2. High Δv values can be obtained by nuclear propulsion.
3. Large increments of Δv are possible with low usage of propellant in nuclear

propulsion.

4. The advantage of nuclear rocket is intermediate between chemical and electrical propulsion when onle exhaust velocity is considered.
5. An ion engine can only generate thrust of fraction of a Newton, but nuclearengine can produce thrust in hundreads of Newtons.
6. Nuclear Engines can provide the high delta velocity required for interplanetary missions to Mars, Venus and beyond.
7. Use of nuclear engines for space journeys can shorten the time of journey toa great extent.

Development Status of Nuclear Thermal Rocket:

Both US and Russia are undertaking development of nuclear thermal rocket.

The ground testing of nuclear thermal rocket has been stopped since 1970 due torestrictions placed on release of nuclear contaminated exhaust from the rocket.

There is renewed interest in the need for a nuclear thermal rocket engine as themain booster for the manned mission to Mars.

One proposal that is feasible, but costly is to test nuclear core in space. And activation and safe disposal of the core needs to be sorted out. The safety issues

also need to be addressed since nuclear core for space applications need to useenriched Uranium.

It is likely that a nuclear propelled mission will be mounted in the next decade. Theproposal under consideration is that a fission reactor will provide the electricity necessary for an electric propulsion.

If the safety aspects and political acceptance can be obtained, then the nuclear thermal engine will take its place in the propulsion systems for space exploration.

Electrical Rocket:

Limitations of Chemical Rocket Engines:

1. **Explosion & Fire Potential**: Explosion and fire potential is larger, failure canbe catastrophic.
2. **Storage Difficulty**: Some propellants deteriorate (self-decompose) in storage. Cryogenic propellants cannot be stored for long periods except when tanks are well insulated. A few propellants like Red Fuming Nitric Acid (RFNA) give toxic vapors and fumes. Under certain conditions, some propellants and grains can detonate.

3. **Loading/Transportation Difficulty**: Liquid Propellant loading occurs at the launch stand and storage facility is needed. Many propellants require environmental permit and safety features for transport on public conveyance.
4. **Separate Ignition System:** All propellants , except liquid hypergolic propellants, need ignition system. Each restart requires separate ignitionsystem.
5. **Smoky Exhaust Plume:** Smoky exhaust plumecan occur with some hydrocarbon fuels. If the propellant contains more than a few percent particulate carbon, aluminum or other metal, then the exhaust will be smokyand plume radiation will be intense.
6. **Need Thermal Insulation:** Thermal insulation is required in all rocket motors.

ELECTRIC THRUSTERS-MISSION APPLICATIONS TO SPACE FLIGHT)

1.1 Limitations of Chemical Rocket Engines:

- **Explosion & Fire Potential (SPR &LPR)**: Explosion and fire potential is larger, failure can becatastrophic.

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- **Need For Thermal Insulation (SPR & LPR):** Thermal insulation is required in almost all motors.

- **Difficult to detect grain integrity(SPR):** Cracks in the grain and unbounded areas are difficult todetect.

- **Toxic Exhaust Gases (SPR)**: Exhaust gases are usually toxic for composite propellants containingAmmonium Perchlorate.

- **Difficult to Re-use (SPR):** If designed for reuse, the motor requires extensive rework and newpropellants.

- **Difficult to change thrust ratings(SPR):** Once ignited, the predetermined thrust and durationcannot be changed

- **Complex Design (LPR) :** Relatively complex design, more parts and hence more probability formalfunction.

- **Sloshing in Tanks (LPR):** Sloshing in tanks can cause flight stability problem**.** Baffles are neededto reduce the sloshing problem.

- **Combustion Instability(LPR):** Difficult to More difficult to control combustion instability.

- **Zero-Gravity Start(LPR)**: Needs special design provisions for start in zero-gravity .

- **Spills & Leaks (LPR):** Spills and leaks can be hazardous, corrosive and toxic. They can cause fires.**More Overall Weight (LPR):** More overall weight for short duration, low-total-impulse applications.

- **Tank Pressurisation (LPR):** Tanks need to be pressurized by separate system. This needs highpressure inert gas storage for long periods of time.

1.2 **Electric Propulsion Systems**:

1.2.1 Structure: The basic subsystems of a electric propulsion thruster are

1. **Energy Source**: Energy source that can be solar or nuclear energy with auxiliary components like pumps, heat conductors, radiators and controls. The energy source isdifferent from the propellant;
2. **Conversion Devices**: The conversion devices transform the energy from above sourcein to electrical form at proper voltage, frequency and current suitable for electric propulsion system;
3. **Propellant System**: The propellant system stores, meters and delivers the propellant tothe thruster;
4. **Thruster**: One or more thruster to convert the electric energy in to kinetic energyexhaust. The term thruster is commonly used to mean the thrust chamber.

1.2.2 Types of ElectricThrusters: Three fundamental types of electric thrusters are available;

1. **Electrothermal**: In this type, the propellant is heated electrically and expanded thermodynamically where the gas is accelerated to supersonic speeds through a nozzle,as in chemical rockets, to produce thrust.
2. **Electrostatic or Ion propulsion engine:** In this type, **a**cceleration is

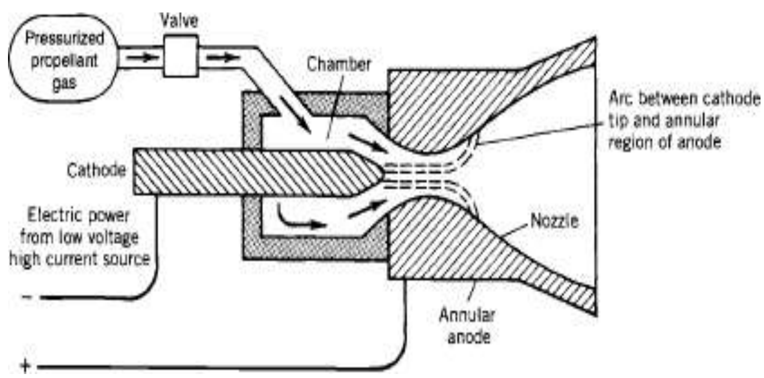
achieved by the interaction of electrostatic fields on non-neutral or charged propellant particles such asatomic ions, droplets or colloids.

3 **Electromagnetic or Magnetoplasma engine:** In this type, the acceleration is achieved bythe interaction of electric and magnetic within a plasma. The plasmas are moderately dense, high temperature gases which are electrically neutral but good conductors of electricity.

Electrothermal Thruster: Electrothermal thrusters use the simplest way to heat the propellant with a hot wire coil, through which an electric current passes. More energycan be delivered from electric current if an arc is struck through the propellant , whichgenerates higher temperature than the resistive approach and therefore produces a higher exhaust velocity.

Thepropellant is heated electrically by heated resistors or electric arcs and the hot gas is thermodynamically expanded in a nozzle and accelerated to supersonic speeds. The electrothermal units have thrust ranges of 0.01 to 0.5 N, with exhaust velocities of 1000to 5000 m/sec. Ammonium, hydrogen, nitrogen or hydrazine are used as propellants.

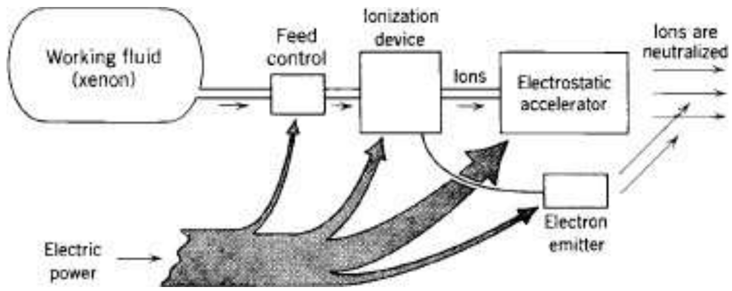
A schematic diagram ofarc-heatingelectric propulsion system is shown below. The arc plasma temperature is around 15,000 K.



Electrostatic and Electromagnetic thrusters accomplish propulsion through differentmeans. They do not use thermodynamic expansion of gas in the nozzle.

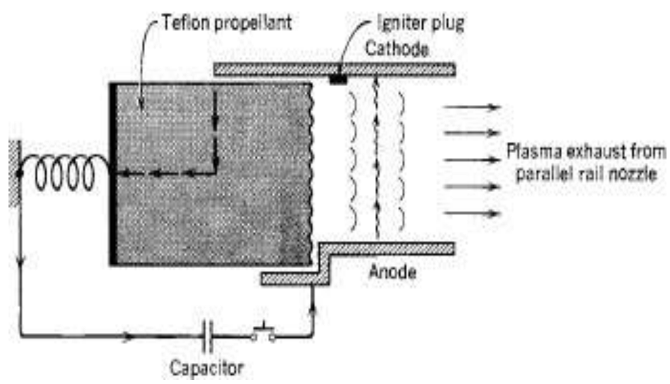
Both Electrostatic and electromagnetic thrusters work only in vacuum.

Ion Rocket Engine (Electrostatic Thruster): In an ion rocket engine, a working fluid, like xenon, is ionized by stripping off electrons. The electrically charged heavy ions are then accelerated to very high velocities (2000 to 60,000 m/sec) by means of electrostatic fields. The ions are subsequently electrically neutralized by combining them with electrons to prevent building up of a space charge on the vehicle. A simplified schematicdiagram of an Ion Rocket is shown below:



Magnetoplasma Rocket (Electromagnetic Thruster): Electrical plasma is an energized hot gascontaining ions, electrons and neutral particles. In the magnetoplasma rocket, an electrical plasma is accelerated by the interaction between electric currents and magnetic fields and ejected at high velocity (1000 to 50,000 m/sec).

A simple pulsed (not continuously operating) unit with a solid propellant is shown below:



The thruster uses a parallel rail accelerator for self-induced magnetic acceleration of a currentcarrying plasma. When the capacitor is discharged, an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of current and the magnetic field causes the plasma to be accelerated at right angles to both the current and themagnetic field, ie in the direction of the rails.

Each time an arc is created, a small amount of propellant (Teflon), is vaporized and converted into a small plasma cloud. The plasma is then ejected giving a small pulse of thrust. The thruster can operate with many pulses per second.

The magnetoplasma rocket is used as spacecraft attitude control engine.

Performance of ElectricThrusters:

- The thrust levels of Electric thrusters are small relative to chemical and nuclear rockets. They have substantially higher specific impulse which results in longer operational life forsatellites whose life is limited by quantity of propellant they carry.

- Electric thrusters give accelerations too low to overcome the high gravity field earth launches.

  They operate best in low vacuum, in space.

- All flight missions envisioned with electric propulsion operate in gravity-free space andtherefore, they must be launched from earth by chemical rockets.

- For electrical thrusters, the key performance parameter is the power-to-mass ratio ie W/kg. Thepower does not diminish with progress through the flight, while the mass of propellant in a chemical rocket decreases as the vehicle accelerates. This is the key difference between Electrical and chemical rockets.

Current Technology: The electrical thrusters need substantial quantities of power on board. All types ofpresent day electrical thrusters depend on vehicle-borne power source-based on solar, chemical or nuclear energy.

- The mass of electric generating equipment, power conversion and conditioning equipment canbecome much higher increasing the mass of thrusters.

- This causes high increase of inert vehicle mass.

1. Overcoming translational and rotational perturbations: These
   - would includeStation keeping for satellites in
   - geosynchronous orbits (GEO),

     Aligning telescopes or antennas in Low Earth Orbits(LEO) and Medium Earth Orbits(MEO)

   - Drag compensation for satellites inLEO and MEOs
2. Increasing satellite speed to overcome weak gravitational field, for Orbit raising from LEO to ahigher orbit even up to GEO. Circularizing an elliptical orbit; This would require velocity increments of 2000m/sec to 6000 m/sec
3. Potential missions as Inter-planetary travel or Deep spaceprobes.

Electric Vehicle Performance: Thepropulsive forcedeveloped by an electric thruster is the momentumtransferred to the propellant. The Rocket equation applies to electric thrusters;

$$V = v_e \log_e R$$, where R is the mass ratio,

$R = \frac{M_0}{M}$; M_0 is the mass of rocket at ignition(initial mass) and M is mass of vehicle (final mass)R can also be expressed as R = $\frac{M_S + M_P + M_E}{M_S + M_E}$.

where M_s is mass of structure including payload, propellant tanks and thrusters, M_p is mass ofpropellant, and $M_E$is mass of power supply equipment on board.

We define the power-to-mass ratio, ξ as

$$\xi = \frac{P_E}{M_E}$$ (W/kg); where P_E is the electric power, M_E is the mass of electric power equipmentThe thrusters have an ɳ, in converting electric power to thrust, which is expressed as

$$\eta = \frac{m v_e^2}{2P_E}$$, where m is the mass flow rate, t is the burn time;

$$m = \frac{M_P}{t}$$

The exhaust velocity v_e can be expressed as

$$v_e = \sqrt{\frac{2P_E}{m}} = \sqrt{\frac{2\xi M_E}{m}} = \sqrt{\frac{2\xi t M_E}{M_P}}$$

or the exhaust velocity can be expressed as, $\frac{M_E}{M_P} = \frac{v_e^2}{2\xi t}$

$$F = m v_e = \sqrt{2m\xi M_E} = \sqrt{\frac{2\xi M_E M_P}{t}}$$

The exhaust velocity v_e is not a free parameter. It is decided by the power P_E and the mass flow rate m.The mass flow rate m, in turn depends on burn time t and mass of propellant M_P

The energy carried away per second by the exhaust is $\frac{1}{2}\dot{m}v_E^2$, this is governed by the power converted in the thruster.

Increasing the exhaust velocity or the mass flow rate, therefore, require an increase in the power supplied to the thruster.

Higher mass flow rate also implies shorter

burn time t. The rocket equation can be

expressed as,

$$V = \sqrt{\frac{2\xi M_E}{\dot{m}}} \log\left(1 + \frac{M_P}{M_S + M_E}\right)$$

The power output P_{jet} is equal to $\frac{1}{2}\dot{m}v_E^2$. The power-to-thrust ratio $\frac{P}{F}$, can be written as

$$\frac{P}{F} = \frac{1}{2}\dot{m}v_E^2 / \dot{m}v_E = \frac{1}{2}v_E = \frac{1}{2}g_0 I_s$$

Example: Determine the flight characteristics of an electrical propulsion thruster for raising a low earth satellite orbit. Data given is:

I_s = 2000 sec; F = 0.20 N; burn time (duration) = 4 weeks = 2.42×10^6 sec ; Pay load mass = 100 kg; ξ = 100 W/kg; η = 0.5

The flight characteristic parameters are $\dot{m}$, M_P, P_E, M_E and Velocity increment ΔV

$\dot{m}$ = F/($g_0 I_s$), since I_s = F/$\dot{m}g_0$

$\dot{m}$ = 0.2/(2000 $\times$ 9.81) = 1.02×10^{-5}

The mass of propellant $M_P = \dot{m}t = 1.02 \times 10^{-5} \times 2.42 \times 10^6$ = 24.69 kg

The electrical power required is $P_E = (\frac{1}{2}\dot{m}v_E^2)/\eta = (\frac{1}{2} \times 1.02 \times 10^{-5} \times 2000^2 \times 9.81^2)/0.5$ =

3.92 kW The mass of electrical power system, M_E will be

M_E = P_E/ξ = 3.92/0.1 = 39.2 kg ;

The initial vehicle mass M_0 = final vehicle mass + propellant mass = 39.2 + 100 + 24.69 = 163.9 kgThe velocity increment Δ V = $v_e \log_e R$ = 2000 9.8 ln (163.9/139.2) = 3200 m/sec

2.1 System Parameters-Interrelations:

1. **Vehicle Velocity V as a function of Exhaust velocity** v_e : The relation between V and v_e is givenby the equation

$$V = \sqrt{\frac{2\xi M_E}{m}} \log\left(1 + \frac{M_P}{M_S + M_E}\right)$$

The exhaust velocity is given by the relation

$$\frac{M_E}{M_P} = \frac{v_e^2}{2\xi t}$$

We can write the mass ratio as, $R = \frac{M_s + M_p + M_E}{M_S + M_E}$ which can be written as

Above equations indicate that

- The mass ratio R (dry vehicle weight divided by propellant weight) for a given dry vehicle weight, decreases as the exhaust velocity increases. This is because higher exhaust velocity needs higherpower supply mass.
- This means that for the electrical thrusters, an increase in v_e requires an increase mass of powersource, or dry vehicle mass, thereby resulting in no improvement of vehicle velocity.
- Figure below shows vehicle velocity as a function of exhaust velocity and burn time t

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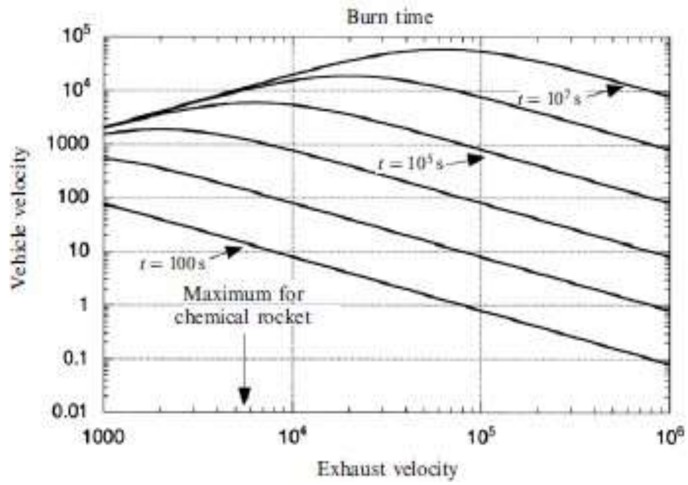
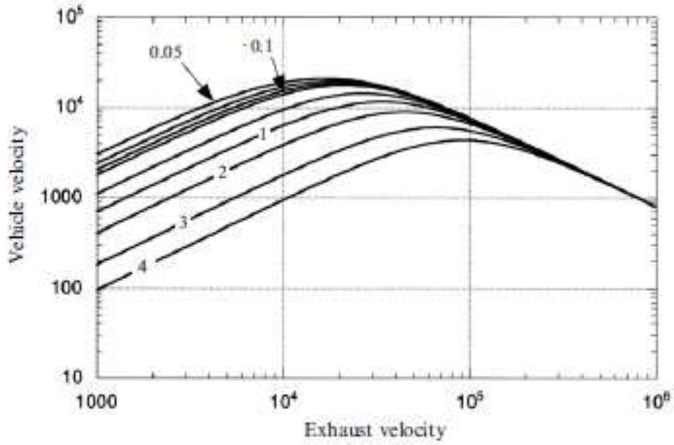


with burn time as a parameter.

- The ratio of structural mass to propellant mass is also fixed at 0.15, equivalent to a mass ratio of6.6.
- It is evident that vehicle velocity does not always increase with exhaust velocity., and peaks for acertain value.
- Increasing the burn time, increases the peak value, both of the vehicle velocity and optimalexhaust velocity.
- The decrease of vehicle velocity beyond a certain point is due to increasing mass of powersupply, and hence reduction in mass ratio.
- With the mass ratio fixed for the rocket, changes in burn time indicate changes in mass flow rate. The exhaust velocity for a given power depends inversely on the mass flow rate. So low mass flow rates or long burn times are beneficial. Also, thrust is inversely proportional to the burn time, and so long burn times and high exhaust velocities imply low thrust.
- In general, electric thrusters have low thrust values, but this is offset by their high exhaustvelocities.

2. **Vehicle Velocity and Structural/Propellant mass:** Electrical thrusters aremeant for bringingsaving of propellant mass. Relation between vehicle velocity as a function of the ratio of payload(structural) mass topropellant mass is indicated below:

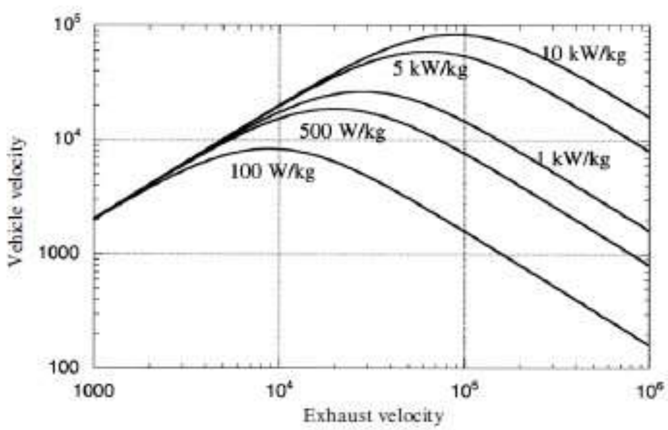
- In the interrelation below, the burn time is fixed at 1 million seconds, and the power-to-massratio, ξ is fixed at 500 W/kg.
- The ratio $\frac{M_s}{M_p}$ is shown as a parameter.



It is evident from the interrelation that

- The vehicle velocity increases as the propellant mass increases
- The peak vehicle velocity shifts to the right ie peak vehicle velocity occurs at higher exhaustvelocities as the payload mass increases

3. **Vehicle Velocity and power-to-mass ratio**: Vehicle velocity is plotted against exhaust velocityfor varying power-to-weight ratios, in the plot below:



The above interrelation shows that

- As the power-to-mass ratio increases, the vehicle velocity increases.
- The peak vehicle velocity also shifts to the right, ie the peak occurs at higher exhaust velocity asthe power-to-mass ratio increases.

Importance of high Exhaust Velocity/high power-to-mass ratio:

- High exhaust velocities allows much higher payload-to-
- propellant mass ratiosHigh power-to-mass ratio allows crucial in
- obtaining the best performance.

The basic characteristics of electric thrusters are **high exhaust velocity, low thrust levels andlong burn times**

2.2 Electric Thrusters : Operation:

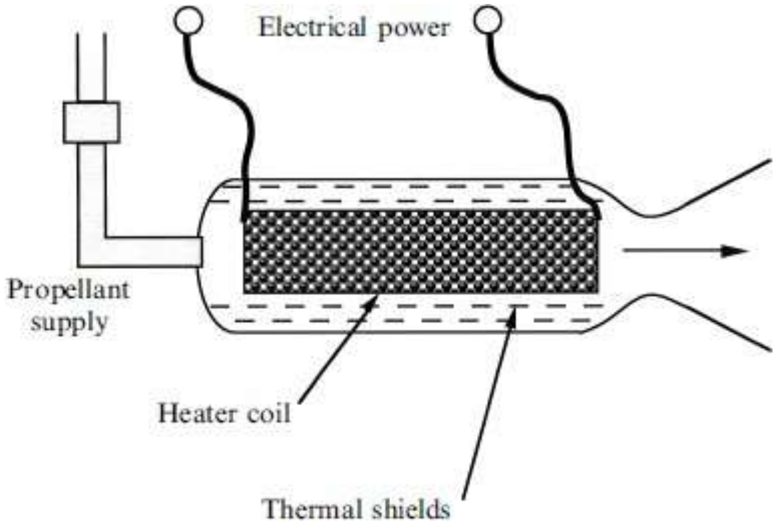
Electric thrusters can be divided into two broad categories: those that use electricity to heat the propellant, which emerges as a neutral gas, and those which use electric or magnetic fields to accelerate ions. The functional form and analysis of these two classes differ.

2.2.1 Resiso-jet:

Operating Principle & Components: The basic electrothermal thruster, resisto-jet, consist ofa nozzle with a high expansion ratio, connected to a chamber in which the propellant is heated by a hot wire through which an electric current passes. The hot gases generated by the heated propellant passes through a nozzle and are expanded thermodynamically. The expansion in the nozzle results in a high velocity exhaust at the end of nozzle. For high exhaust velocity, the temperature and pressure of gasesentering the nozzle should be high. This needs efficient heating of propellant.

To maximize heat transfer to the gas, a multichannel heat exchanger is used to bring as much of gasvolume as possible in contact with the heater.

The resisto-jet thruster is illustrated below:



System Parameters & Performance: The exhaust velocity is calculated using the thrust coefficientand characteristic velocity

$$v_e = C_F C^*$$

Where $C^* = [\gamma(\frac{2}{\gamma-1})^{\frac{\gamma+1}{\gamma-1}} \frac{\mathfrak{M}}{RT_c}]$

- The thrust is a function nozzle exit and chamber pressure (p_e & p_c)
- Since these thrusters are used in vacuum, high nozzle expansion ratios are used, (around 2.25for γ = 1.2)
- While in the chemical rockets C^* depends on T_c & $\mathfrak{M}$, for the electric thrusters, C^* mainly depends on $\mathfrak{M}$. (since there is no combustion and the nozzle exit temperature depends onpower input and mass flow rate)
- The nozzle exit temperature in chemical rockets depends on type of propellant, where as inelectric thrusters, the nozzle exit temperature is an inverse of mass flow rate.
- The melting temperature of heating element limits the maximum temperature levels in thethruster.

Example: Consider following data:

P_E= 1 kW; C_F = 2.25; T_c = 2200 k; Propellant is hydrogen with $\mathfrak{M}$ = 2.

$C^* = \frac{2\gamma}{\gamma-1}\left(\right)^{\frac{\gamma+1}{\gamma-1}} \frac{\mathfrak{M}}{RT_c}$]=4659 m/sec

$v_e = C_F C^*$ = 2.25 × 4659 = 10,483 m/sec

- Electric thrusters can attain very high exhaust velocities

The mass flow rate m is calculated from

$\frac{1}{2} v_e^2 \dot{m} = \eta P_E$ or $\dot{m} = \frac{2\eta P_E}{v_e^2} = \frac{2\times\eta\times1000}{10{,}483^2} = 1.8 \times 10^{-5}$ kg/s

- The mass flow rate of an electric thruster is very small compared to a
- chemical rocketThe thrust for above thruster works out to 0.2 N, which is
- very small

  This means that the vehicle can achieve very high exhaust velocities, but at low thrust values,the time taken to accelerate to such high velocities is very long

-
- This is the fundamental difference between chemical rockets and electric thrusters.The electrical efficiency can be very high at 90%

Propellants used could be hydrogen, helium, water (even waste water can be used) or hydrazene.

Disadvantages: Higher exhaust velocities and power are difficult to achieve since transfer of heat fromfilament to gas is difficult.

2.2.2 Arc-Jet Thruster:

Operating Principle: In the Arc-Jet thruster, the propellant gas is heated by passing an electric arc through the flow. Temperatures in the order 30,000-50,000 k are achieved at the centerline which fullyionizes the propellant.

The anode and cathode are made of tungsten, which has high melting point. The cathode rod is pointed and is supported in an insulator. The insulator also holds the anode. The anode is shaped to create a gapwith the pointed cathode, across which the arc is struck. The propellant flows through this gap and gets ionized. Downstream of this arc, the anode is shaped to form a nozzle, for the expansion of the exhaust.

The propellant gas is introduced annular chamber around the cathode and swirls around it.

The power that can be applied across an arc-jet is up to 100 times higher than the filament of aelectrothermal thruster. The temperature limit can be much higher.

While the propellant is ionized, the electrons and positive ions move towards anode and cathode. Thecathode is struck at high speeds, causing vaporization of the cathode material, thereby limiting it's life.

The arcs cause concentration of energy and cause hot spots leading to erosion of the electrodes.Heat losses due to ionization and dissociation are higher than electrothermal thrusters.

Maximum exhaust velocities are around 20 km/s. Hydrogen, ammonia and hydrazine are used aspropellants.

Power levels can reach up to 200 kW. However, heavier power source is required than theelectrothermal thrusters.

Arc-jets are best suited as station-keeping thrusters.

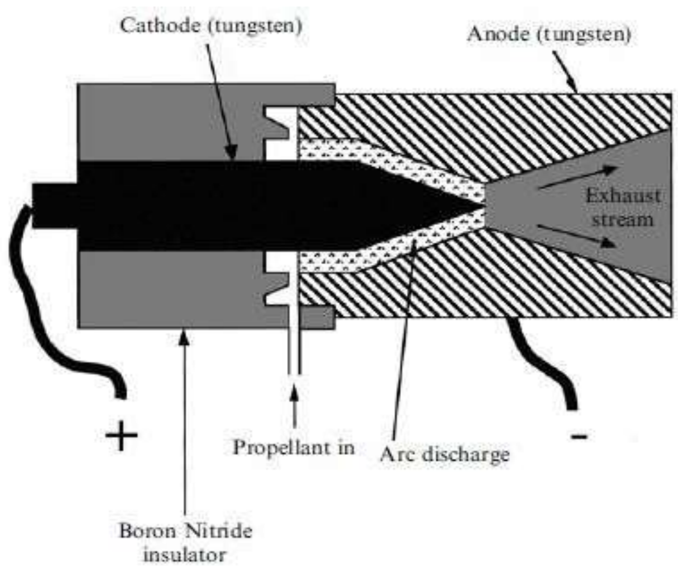


Figure 6.6. Schematic of an arc-jet thruster.

2.2.3 Solar/Laser/Microwave Thermal Propulsion: Beamed energy, for example, a laser can be used for heating instead of on board energy source. Solar/laser/microwave energy source, external to the vehicleis used to heat up the propellant. The external beamed energy may be from an earth or space based infrastructure. The energy is then concentrated on a heat exchanger or directly on the propellant, whichis then heated up and expelled through a conventional nozzle. Specific impulses of 800-1200 sec and thrust levels of several hundred mN are possible using sunlight and hydrogen as propellant.

A reflector is used to collect and concentrate sunlight/laser/microwave energy on to the propellant heldin the chamber of the thruster.

Laser thermal propulsion offers higher specific impulse, but requires very high pointing accuracy.

This concept is under development for using solar thermal propulsion to raise the communications satellite from LEO to GEO in about 20 days. This concept uses very little propellant, saving launch costssignificantly.

3.1 Electrostatic Thrusters:

Performance Parameters:

- If the propellant is ionized, it can be accelerated very effectively by electrostatic fields. The velocity gained for an **ion mass m** and **charge q due to the electric potential difference U isgiven by**

$$\sqrt{\frac{2qU}{m}} ,$$

The mass flow rate is related to the current I, as

$$\dot{m} = \frac{m}{q} I \times$$

And the force generated F, can be

expressed as $F = I \times \sqrt{\frac{2mU}{q}}$

For obtaining very high specific impulse, a multi-ionised, light ion would be ideal. However, since the

thruster should produce high thrust, propellant with heavy ions is preferred.

3.1.2 Ion Thruster:

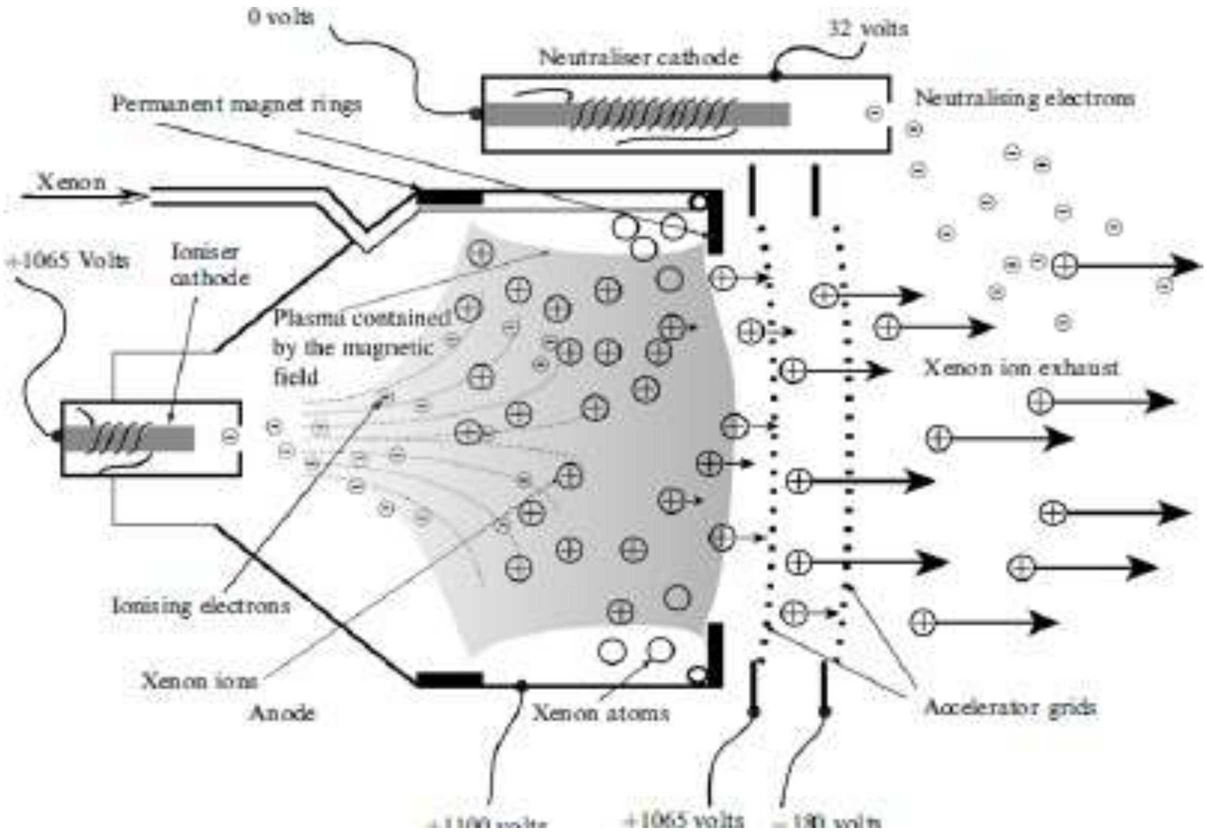
Working Principle: The propellant is ionized, and then enters a region of strong electric field, where thepositive ions are accelerated. The ions are accelerated passing through the grid and leave the engine asa high velocity exhaust stream. Highest exhaust velocities (more than 32,000 m/s) are achieved by accelerating positive ions in an electric field created by two grids having large potential difference.

The electrons do not leave, therefore the electron current is discharged through a neutralising cathode,in to the exhaust.This would neutralise the spacecraft. The electrons discharged carry little momentum,therefore do not affect the thrust.

The thruster is divided into two chambers. Propellant, (usually Xenon gas) enters ionisation chamber in the form of neutral gas molecules.

The cathode at the center, emits electrons, which are accelerated by the electric field. These electrons ionise propellant through electron collision. The ionised propellant drift through the grids with high potential difference and accelerate. The ions gain energy and form the ion beam with high velocities ofaround 32,000 m/sec.

Thrust is exerted by the departing ion stream on the accelerating grids and is transferred through thebody of the thruster to the spacecraft. The exhaust velocity is governed by the potential difference between the grids and the mass flow rate is directly related to the current flowing between the grids.



There is no need for a nozzle to generate thrust .

Applications of Ion Engines:

Ion engines are best used for very high velocity increment missionslike inter-planetary missions andstation keeping.

Ion engines are not used for attitude control due to their low thrust.

Limitation of Ion Thrusters- The space-charge limit**:**

The accelerating grids have an electric field between them, which gets partially blocked as the ions startaccelerating along the grids. As the density of flow of ions increases, a point will reach when the accelerating field at the first grid drops to zero, because the positive charge of the ions passing through cancels the field.

This is the space-charge limit, which limits further ingress of ions and limits thrust levels.

Electromagnetic Thrusters: The low thrust-high exhaust velocity ion thrusters are limited by space-charge limit. Plasma thrusters (electromagnetic thrusters) offer higher thrust values.

In plasma thrusters, an ionised gas passes through a channel across which orthogonal electric and magnetic fields are maintained. The current carried by the plasma (electrons and ions) along the electricfield vector interacts with the magnetic vector, generating a high propulsive force. The plasma accelerates without the need for area change

Magnetoplasma Dynamic (MPD) thrusters and Pulsed Plasma thrusters (PPT) are conventional type ofelectromagnetic thrusters. The Hall Effect thruster is another variant of the electromagnetic thruster.

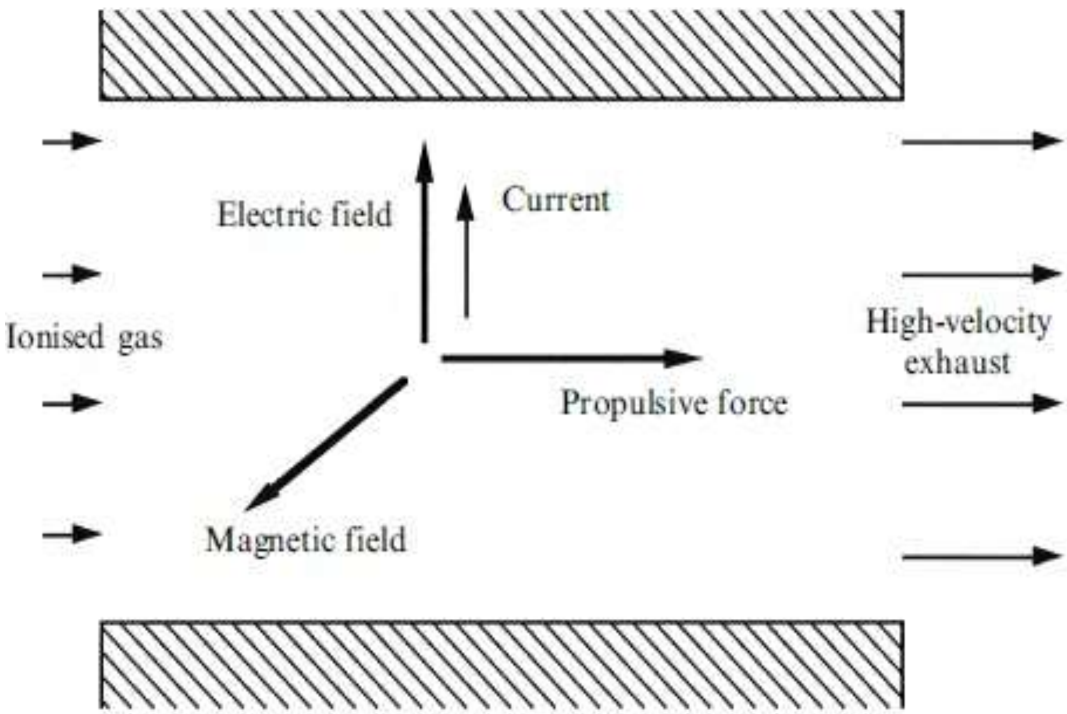
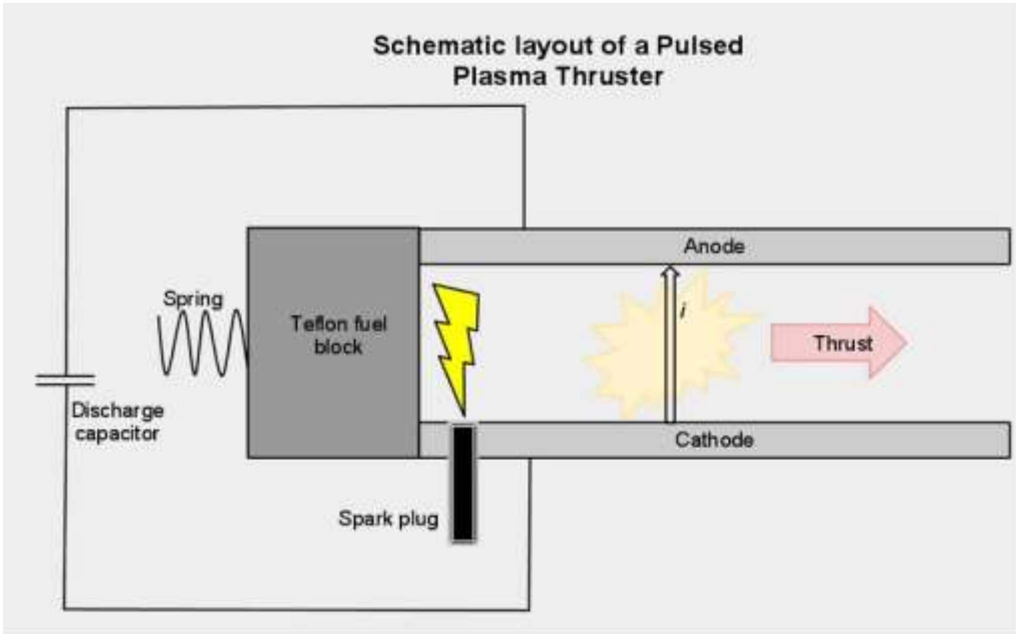


Figure 6.14. Principle of the plasma thruster.

Pulsed plasma thruster (PPT): Plasma thrusters do not use high voltage grids or anodes/ cathodes toaccelerate the charged particles in the plasma, but rather uses currents and potentials which are generated internally in the plasma to accelerate the plasma ions.

While this results in lower exhaust velocities by virtue of the lack of high accelerating voltages, this typeof thruster has a number of advantages.

In the PPT operation, an electric arc is passed through the fuel, causing ablation and sublimation of the fuel. The heat generated by this arc causes the resultant gas to turn into plasma, thereby creating a charged gas cloud. Due to the force of the ablation, the plasma is propelled at low speed between two charged plates (anode and cathode).



Since the plasma is charged, the fuel effectively completes the circuit between the two plates, allowing a current to flow through the plasma. This flow of electrons generates a strong electromagnetic field which then exerts a Lorentz force on the plasma, accelerating the plasma out of the PPT exhaust at high velocity.

The time needed to recharge the plates following each burst of fuel, and the time between each arc causes pulsing. The frequency of pulsing is normally very high and so it generates an almost continuous and smooth thrust.

While the thrust generated by PPT is very low , it can operate continuously for extended periods of time, yielding a large final speed.

A solid material, teflon (PTFE) is commonly used propellant. Few PPTs use liquid or gaseous propellants also.

Magnetoplasmadynamic (MPD) thrusters: MPD thrusters, also referred as Lorentz Accelerators, usethe Lorentz force (a force resulting from the interaction between a magnetic field and an electric current) to generate thrust

The electric charge flowing through the plasma in the presence of a magnetic field causing the plasmato accelerate due to the generated magnetic force.

The operation of MPD thrusters is similar to pulsed thrusters.

Hall Thrusters: Hall Effect Thrusters combine a strong magnetic field perpendicular to the electric field created between an upstream anode and a downstream cathode called neutralizer, to create an area ofhigh density of electrons. The electrons are trapped in a magnetic field and these electrons confined to the field are used to ionise the propellant.

The cathode then attracts the ions formed inside the thruster, causing the ions to accelerate andproduce thrust .

Operation of Hall Thruster: An electric potential between 150 and 800 volts is applied between theanode and cathode. Electrons from a hollow cathode enter a ring shaped anode with a potential difference of around 300 V.

The central spike forms one pole of the magnet, and around the inner pole, an outer circular pole formsan annular radial magnetic field in between. The propellant, usually xenon gas is fed through anode where the neutral xenon atoms diffuse in to the channel, and ionised by colliding with the circulating high energy electrons.

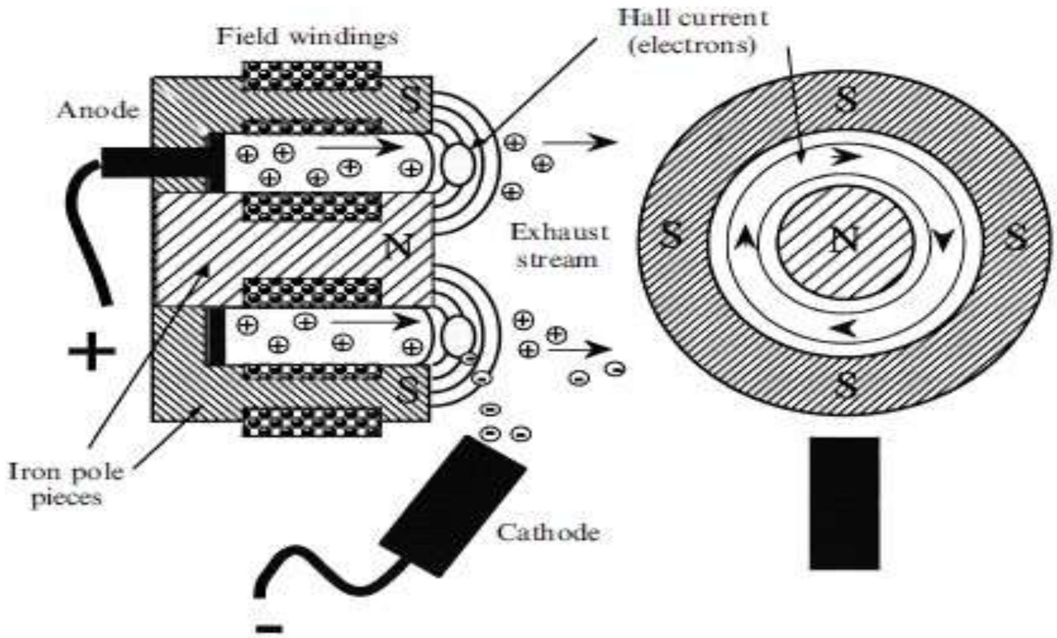


Figure 6.16. Schematic of the Hall thruster.

The xenon ions are then accelerated by the electric field between anode and cathode. Ions reachspeeds of around 15 km/sec with specific impulse of 1500 sec.

Thrust levels are very small, around 80 mN for a 300 V, 1.5 W thruster.

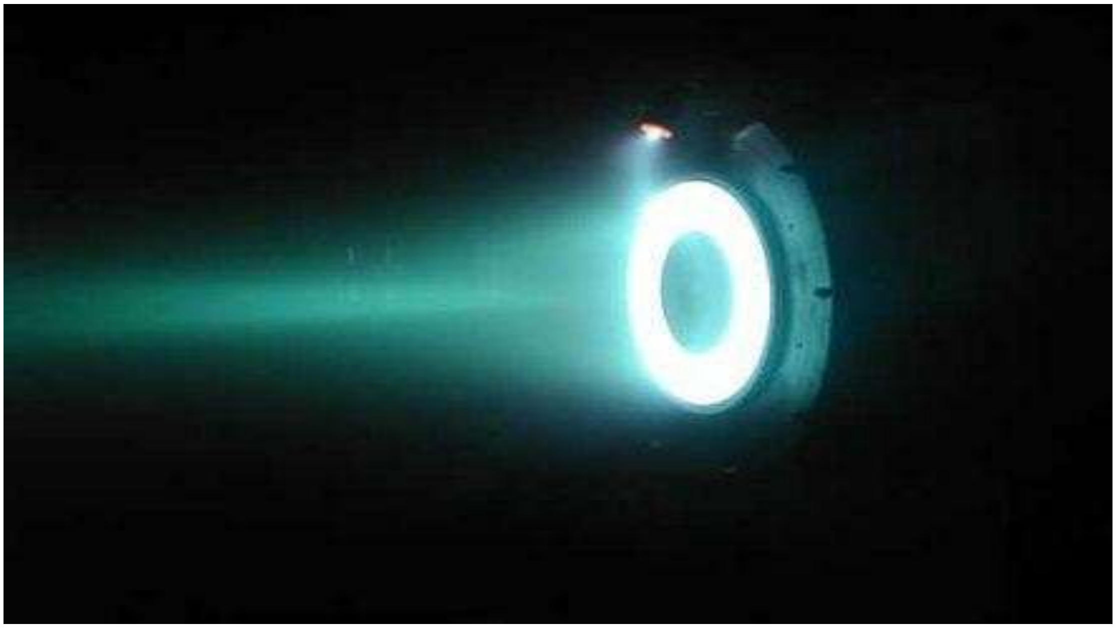
The accelerating ions also pull some electrons forming a plume. The remaining electrons are stuck orbiting the region, forming a circulating hall current. This circulating electrons of hall current ionisealmost all the propellant.

all thrusters can provide exhaust velocities of 10-80 km/s and specific impulse of 1500-3000 sec. Mostcommonly used propellants are xenon, argon and krypton

The applications of Hall-effect thrusters include control of orientation & position of

orbiting satellites and to power the main propulsion engine for medium-size robotic space vehicles.

2 KW Hall Thruster in Operation



Applications of Electric Thrusters: The applications for electrical propulsion fall into broad categories asbelow:

1. **Attitude Correction (Space Station/Spacecraft)**: Overcoming translational and rotational perturbations in orbits; **Drag compensation** for satellites in Low Earth Orbits; **Aligning** telescopes or antennas.**Electro-thermal (resisto-jets)** are preferred using low cost propellantlike cold gas or waste water. MPD thrusters are also being considered for attitude control of space vehicles.
2. **Station Keeping**: For station keeping purpose, savings in propellant mass is very significant.Synchronous and GEO satellites have long life periods need extensive station keeping requirement. **Electro-thermal (Arc-jets) thrusters** have been widely used for this task. Hallthrusters and Ion engines are most suitable.
3. **Raising Orbits**: From low earth to higher orbits (up to Geostationary orbits), circularizing an elliptical orbit Inter-planetary travel and deep space probes. They all require relatively high thrust and power in the range of around 100 kW, much higher velocity increments than thoseneeded for station keeping. Also these corrections need to carried out in reasonable length oftime. **Hall thrusters and Ion engines** are again preferred here.
4. **Inter-planetary missions** : These are deep space long duration applications. **Ion engines** withhigher exhaust velocities are preferred.