POLITECNICO MILANO 1863

Liquid propellant engine

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Introduction to Liquid Rocket Engines and Basic Elements

1.1 Liquid Rocket Engines: Historical Notes

The expression “rocket engine” refers to a reaction engine that contains all the required substances for its operations and is thus capable of generating thrust in space. A rocket engine uses a reactionary propellant mass to generate a thrust force usually in the form of a high temperature gas in accordance with Isaac Newton’s (1643-1727) laws of motion. A liquid propellant rocket engine is a rocket engine that utilizes liquid propellants. The reason for the use of liquid propellant rocket engines was due to their ability for propelling military and space vehicles at a greater capacity than most other forms of chemical propulsion. They also provided certain characteristics required for operation which could not be replicated by other types of propulsion at the time. It took a lot of time, experience and theoretical conjectures for humanity to master the art of rocket design. Historically, the reaction thrust principle was exploited long before science managed to explain it. Around 850 A.D., Chinese alchemists mixed sulphur (S), charcoal (C), potassium nitrate (KNO3) as well as other substances for medical purposes, unknowingly creating an early form of gunpowder. In its first use, gunpowder was inserted into bamboo shoots that would be thrown into fire to produce a loud blast. This is commonly recognised as the invention of fireworks. Between the 10th and 12th century, gunpowder was first used for warfare. It was stored in cylinders and ignited in order to produce the ejection of arrows or bits of metal. By the end of the 13th century, the Chinese fastened such devices to sticks and managed to make them fly, creating the first rockets as seen in Figure 1.1. In the following centuries, the knowledge about gunpowder spread among populations, assuring a trial and error development of rockets for military use.

Figure 1.1: Early example of rocket used for warfare

It is worth noting that only first in 1686 did Sir Isaac Newton present his three laws of motion contained in the ”Principia Mathematica Philosophiae Naturalis”. In particular,

the third law explains how a thrust force originates in a rocket, but it cannot be directly used for its quantitative determination. In the following centuries, the mathematical and speculative bases for the scientific theory of rocket propulsion were developing. In 1812, rockets with a range of about 3000 meters were used in the war between England and the United States, as well as in other conflicts.
The modern idea of a liquid rocket first appears when a school teacher named Konstantin Tsiolkovsky (1857-1935), published a book called "The Exploration of Cosmic Space by Means of Reaction Devices" in 1903, where he presents the famous Tsiolkovsky equation. In the book he demonstrates the possibility of reaching space through the use of rockets as well as the idea of multistaging where a part of the rocket is discarded during flight to reduce the overall mass. His ideas of reaching space involved the use of liquid oxygen and liquid hydrogen as propellants. It was not until 1926 that the first liquid rocket took flight, developed and tested by a professor named Robert H. Goddard (1882-1945), which used gasoline and liquid oxygen as the propellants. Although it only flew for 2.5 seconds and reached a height of 12.5 meters, it proved that liquid propellant rockets were possible. The rocket schematic is represented in Figure 1.2.

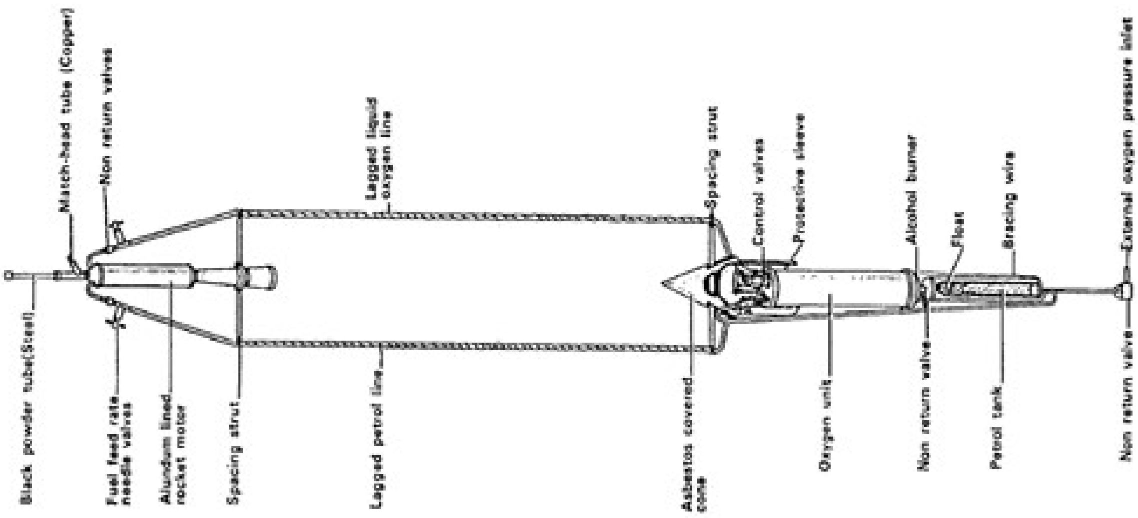

Figure 1.2: Schematic of the first liquid rocket

Interest in liquid rocket engines rose in Europe and the United Stated, in particular for military applications during WWII. In 1939 the first manned flight utilizing a liquid propellant rocket engine came in the form of the Heinkel He 176 which was developed by an aeronautical engineer named Hellmuth Walter (1900-1980). The German rocket V-2 as seen in Figure 1.3, whose technical name was Aggregat 4, reached an altitude of 80km in October 1942. It was developed by Wernher von Braun (1912-1977) and the German Army Research Station and was the first large liquid propellant rocket engine with a thrust greater than any prior such rocket by a factor of between 5 to 10. In June 1944, it crossed the Kàrmàn line reaching an apogee of 176 km and becoming the first man-made object to reach space.

Figure 1.3: The V-2 Rocket

Figure 1.4: Saturn V rocket during launch

During the Cold War, the United States and the Soviet Union engaged in the Space Race. In October 1957, the Soviets succeeded in launching and orbiting the first artificial satellite around the Earth which was named Sputnik-1. In 1961 they managed to put the first man in space by the name of Yuri Gagarin (1934-1968), which prompted the American President John Kennedy (1917-1963) to announce a program to place a man on the moon within 10 years. On 20th July 1969, during the NASA Apollo 11 mission, Neil Armstrong (1930-2012) and Edwin Aldrin (1930-) became the first humans to land on the moon. This was achieved by the Saturn V as seen in Figure 1.4, a three stage liquid-propellant expendable rocket, where Wernher von Braun also contributed to its development after being recruited to NASA after WWII. As of 2020, the Saturn V remains the tallest, heaviest, and most powerful (highest total impulse) rocket ever brought to operational status, and holds the record for the heaviest payload launched.
Nowadays, liquid propellant engines find employment in many stages of rockets, in missions requiring regulatable thrust or re-ignitable engines, in small-size applications for attitude control and adjustments in re-utilizable engines. Recent uses of liquid rocket engines have come in the form of the SpaceX Raptor, as seen in Figure 1.5, and Merlin rocket engines or the European Vulcain rocket engine.

1.2 Fundamentals of Rocket Propulsion

A simplified and generic version of a liquid propellant rocket engine can be described as seen in Figure 1.6: The encapsulating rocket would contain a tank filled with liquid fuel (1), a tank filled with a liquid oxidizer (2), pumps to displace the fuel and oxidizer (3) into a combustion chamber where the fuel and oxidizer are mixed and ignited to form hot expendable gasses (4), which would then need to pass through a throat where the gasses are choked to regulate the amount of thrust produced (5), and finally the hot gasses are exhausted from the exit (6).

From Isaac Newton's laws of motion, we can have an understanding of why a rocket

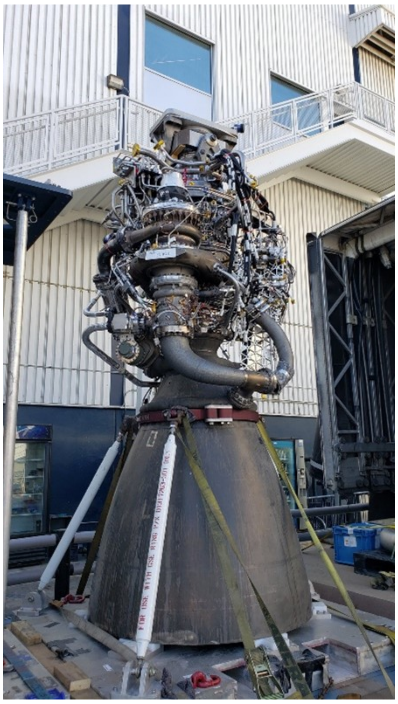

Figure 1.5: SpaceX raptor rocket engine

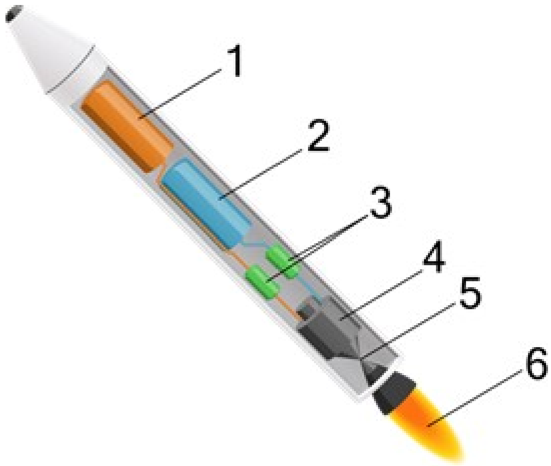

Figure 1.6: Basic schematic of a liquid propellant rocket

is propelled forward. Newton's second law of motion defines a force as the mass times its acceleration. To define the acceleration by the combustion process of the propellant mass m from an initial velocity V_o to an exit velocity V_e, we can state the following formula:

$$a = \frac{V_e - V_o}{\Delta t} \tag{1.1}$$

Where a is acceleration and Δt is the change in time. This can be substituted into Newton's second law by:

$$F = \frac{m\ (V_e - V_o)}{t} \tag{1.2}$$

Where F is the force and t is the time. We also know that m/t can be rewritten as $\dot{m}$ which is the mass flow rate and yielding:

$$F = \dot{m}\ (V_e - V_o) \tag{1.3}$$

Newton's third law of motion states that: "When one body exerts a force on a second body, the second body simultaneously exerts a force equal in magnitude and opposite in direction on the first body". The two forces are referred to as the action and reaction force. Liquid rocket engines are reaction engines as the thrust they generate is the reaction force experienced by their structure due to the ejection of high velocity matter. The origin of the thrust force and the reason why it exists is then to be found in the mechanical interaction between the engine and the working fluid. In this sense, the knowledge of pressure p and shear stress τ distribution over the inner surface S_i of the rocket engine would allow to compute the thrust force through the relation:

$$F = \int_{S_i} (p - p_a)(\mathbf{n} \cdot \mathbf{i}) dS + \int_{S_i} \tau(\mathbf{t} \cdot \mathbf{i}) dS \tag{1.4}$$

Where $\mathbf{n}$ is the unit normal vector to the surface and points outwards from the control volume, $\mathbf{t}$ is the unit tangent vector to the surface and subscript "a" refers to ambient

conditions. Typically, in practical applications there is not sufficient information available to evaluate these integrals. Nevertheless, formula (1.4) gives an interesting insight on the nature of thrust force in rocket engines.
An equation of simpler use can be derived by considering a proper control volume enclosing the rocket and by applying the principle of conservation of momentum, projecting the vector equation along the rocket axis. After a few mathematical manipulations, the following relation can be obtained:

$$F = \dot{m}_p \; u_e + (p_e - p_a) \; A_e \tag{1.5}$$

Where $\dot{m}_p$ is the propellant mass flow rate and subscript "e" refers to the exit section of the nozzle, thus u_e indicates the exit velocity and A_e the exit area. The product $\dot{m}_p \; u_e$ is named "dynamic thrust", while the quantity $(p_e - p_a) \; A_e$ is referred to as "static thrust". Equation (1.5) is effectively represented by Figure 1.7.

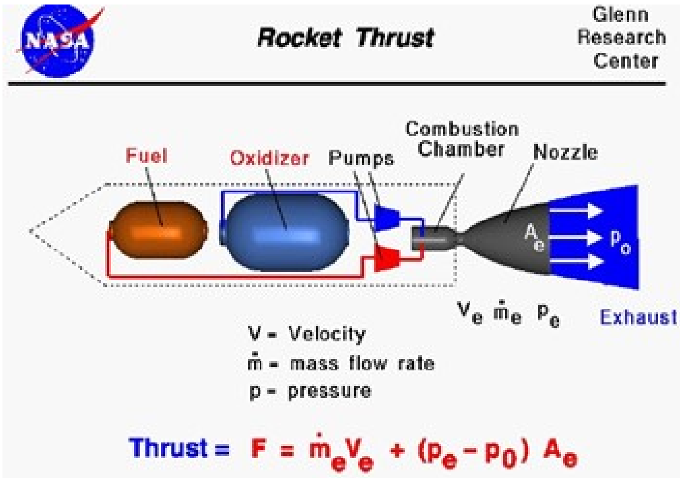

Figure 1.7: Representation of rocket thrust in a liquid propellant rocket

It is convenient to rewrite formula (1.5) by defining a fictitius velocity u_{eq} *effective exhaust velocity* - which takes into account the static thrust whenever different from zero:

$$F = \dot{m}_p \; u_{eq} \tag{1.6}$$

Evidently, the effective exhaust velocity is the ratio between the thrust force and the propellant mass flow rate $u_{eq} = \left(\frac{F}{\dot{m}_p}\right)$ therefore it represents a meaningful parameter to evaluate the performance of a rocket engine. To this end, an analogous quantity – the gravimetric specific impulse $I_{sp,g}$ – can be defined as:

$$I_{sp,g} = \frac{F}{\dot{m}_p \; g_0} = \frac{u_{eq}}{g_0} \tag{1.7}$$

Where $g_0 = 9,80665 \; \frac{m}{s^2}$. The specific impulse is measured in seconds, where a higher specific impulse means a more efficient engine.
It is sufficient to add a few simplifying assumptions and considerations in order to derive a simple relation regarding the rocket motion, that is, the Tsiolkovsky equation. To do this we introduce the following assumptions:

(i) *Single-stage rocket.* Further extension to multi-stage rocket is not complicated.

(ii) *Thrust is the only force acting on the vehicle*: This assumption will not lead to significant error if the resultant of other forces is either zero or negligible with respect to thrust.

(iii) *Thrust and velocity have the same direction.*

(iv) *Effective exhaust velocity is constant* (or in any case its variations can be taken into account through its average value $\bar{u}_{eq}$).

(v) *Rocket is only ignited once*: This assumption simplifies some definitions.

Now consider the total mass m of the rocket: during the mission, the mass decreases in time as propellant is discharged though the nozzle. If m_p denotes the mass of propellant stored in the rocket and $\dot{m}_n$ the mass flow rate through the nozzle, then the following relation is implied:

$$\frac{dm}{dt} = \frac{dm_p}{dt} = -\dot{m}_n \tag{1.8}$$

Under assumptions (ii) and (iii) and recalling equation (1.6), the application of the second Newton's law of motion along the velocity axis yields:

$$F = \dot{m}_n \, u_{eq} = m\frac{dV}{dt} \tag{1.9}$$

That, considering equation (1.8), can be written as:

$$-\,u_{eq}\,\frac{dm}{dt} = m\,\frac{dV}{dt} \tag{1.10}$$

Rearranging:

$$dV = -u_{eq}\frac{dm}{m} \tag{1.11}$$

Integrating:

$$\Delta V = \int_i^f u_{eq}\frac{dm}{m} \tag{1.12}$$

Equation (1.12), under assumption (iv), finally yields Tsiolkowski equation:

$$\Delta V = \bar{u}_{eq}\ \log\left(\frac{m_i}{m_f}\right) \tag{1.13}$$

Where ΔV is the variation of the velocity of the rocket, m_i is its mass before ignition and m_f its mass after shut-off. Equation (1.13) can be used to compute the necessary mass ratio $\left(\frac{m_i}{m_f}\right)$ to obtain a given velocity variation. Moreover, it shows that a rocket which works with a higher $\bar{u}_{eq}$ achieve a fixed ΔV with a lower mass ratio. Figure 1.8 shows a plot of Tsiolkowski equation for different values of $\bar{u}_{eq}$. It is remarkable that with a final mass of 0.15 and a gravimetric specific impulse of 450s (high values for single-stage rockets), the ΔV results to be 9.5 $\frac{km}{s}$, which is lower than the Earth escape velocity. Therefore, Tsiolkolwsky equation also shows that multi-stage rockets are needed in order to reach space.

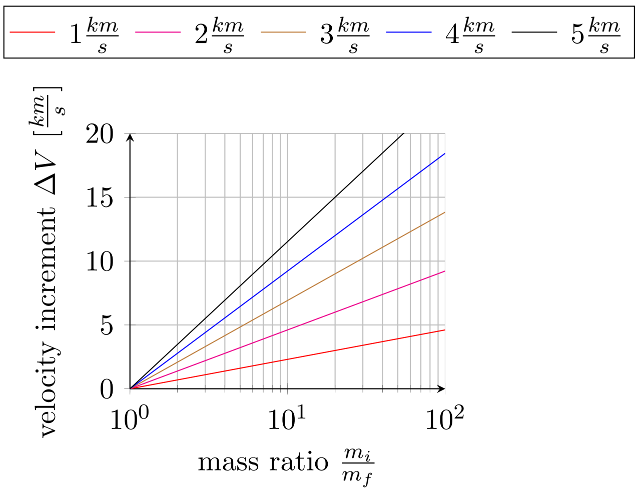

Figure 1.8: Tsiolkovsky equation plotted in a logarithmic chart

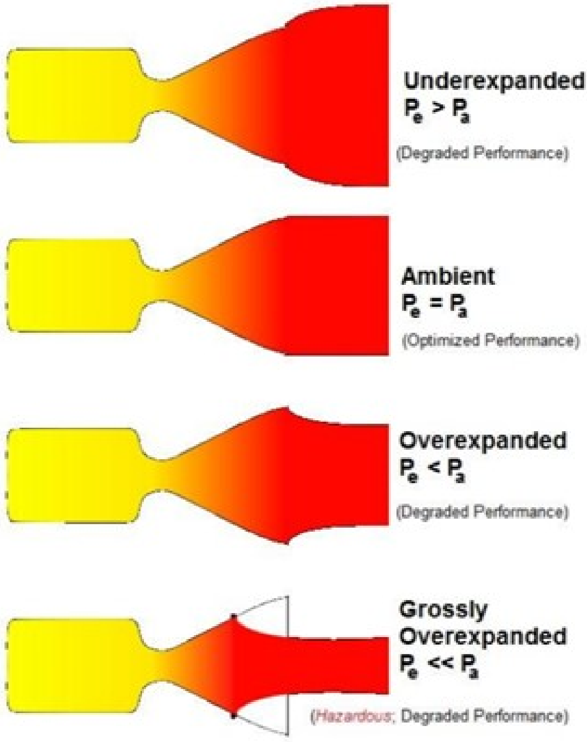

Figure 1.9: Effect of area ratio on rocket plume

Another thing worth looking at is the area ratio or the so called nozzle expansion ratio which determines the exit velocity and the pressure of the exhaust gasses. This is defined as:

$$\varepsilon = \frac{A_e}{A_t} \tag{1.14}$$

Where ε is the area ratio and A_t is the throat area. The exit velocity grows larger and the exit pressure becomes smaller when the area ratio increases and this in turn yields a higher Isp,g. The most optimal thrust operating condition occurs when the area ratio returns an exit pressure which is equal to the external ambient pressure. If the exit pressure is larger than the external ambient pressure, the nozzle is defined as under expanded and does not yield the best performance due to the plume still expanding as it leaves the nozzle. For the reverse when the external ambient pressure is greater than the exit pressure, the nozzle is defined as overexpanded and again yields suboptimal performance due to mitigation of exit shock inside the nozzle and can in turn damage the nozzle. A representation of these configurations can be seen in Figure 1.9.

1.3 Liquid Rocket Engines: Basic Elements

In section 1.2 the generation of a thrust force through the expulsion of high velocity matter was discussed. The purpose of this section is to give a brief overview over liquid rocket engines Figure 1.10, which exploit the presented principles. When working correctly, a liquid rocket engine feeds the combustion chamber with pressurized liquid propellants that ignite, evaporate and expand through the nozzle. Liquid rocket engines are complex devices both in absolute terms and with reference to other propulsion systems. Liquid rocket engines have been some of the most utilised propulsion systems in the past 70 years and even to this day, and this is generally due to the performance advantages that it has over

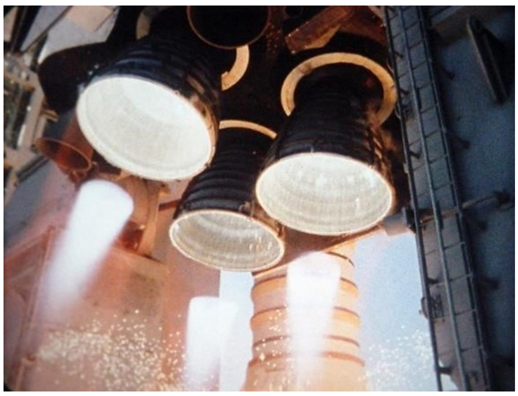

Figure 1.10: Three RS-25 engines during lift-off

other systems. The main advantages could be claimed to be the higher gravimetric specific impulse over solid rocket motors and the flexibility of the liquid rocket engines in allowing for throttling capabilities in order to make thrust changes as a function of time. However, some of its downfalls are having much lower mass fractions as well as being significantly more complex than the rivalling solid systems. This means that the solid systems are more economical and reliable due to their simplicity. Liquid rocket engines also hold the advantage in being able to shut down and re-start on request, they can be re used and also have the ability to use green propellants. It is possible to distinguish three systems with relative sub-systems, as illustrated in Table 1.1.

Table 1.1: Systems and sub-systems of a liquid rocket engine

SYSTEMS	SUB-SYSTEMS
Propellant tanks	
Propellant feed systems	pressurized gas / turbopumps ducts valves
Thrust chamber	injectors combustion chamber nozzle cooling system ignition system

For a liquid propellant rocket engine, the propellant system can be described as either a monopropellant system, where the system uses only one liquid which reacts to form a hot gas when it is in the presence of a catalyst material, or as a bipropellant system where there are two propellant components in the form of fuel as well as an oxidizer. While monopropellant systems usually only have fairly average performance values in the terms of the gravimetric specific impulse which lies somewhere between 200 and 230 seconds, it can be a preferable alternative in uses where simplicity is of importance and the required total impulse is not that high. This means that in today's world, it is used for control rockets for many satellites and spacecraft, and when the engines for these applications are this small they are referred to as thrusters in order to separate them from their much bigger launch vehicle counterparts. As opposed to this, the bipropellant systems are generally used when the required total impulse is high and can be generally divided into even further categories as either storable or cryogenic propellant combinations. The main difference between these types of combinations are that storable propellants are defined by being liquid at ambient conditions as opposed to the cryogenic propellants such as either liquid hydrogen or oxygen, which must be stored at reduced temperatures in order to avoid boiling. The gravimetric specific impulse that each of these two sub categories also varies with the storable propellant values lie between 250 to 350 seconds and the cryogenic propellant values lie in the 425 to 500 second range. While cryogenic propellants seem to be the better option, the additional performance gained can be diminished due to the complexity, such as insulation among other things, required when using these liquids.
Ignition is another thing to consider when using bipropellants, as some mixtures must be ignited while others can spontaneously combust when coming into contact with a suitable catalyst. This is referred to as hypergolic ignition and occurs instantaneously when the oxidizer and the fuel mixes. The feed system, which is the system that moves the propellants from their tanks and into the combustion chamber, can be split into

a pressure-fed (or a blow down system) and a pump-fed configuration. An interesting example about these systems and their assembly is provided by Figure 1.11.

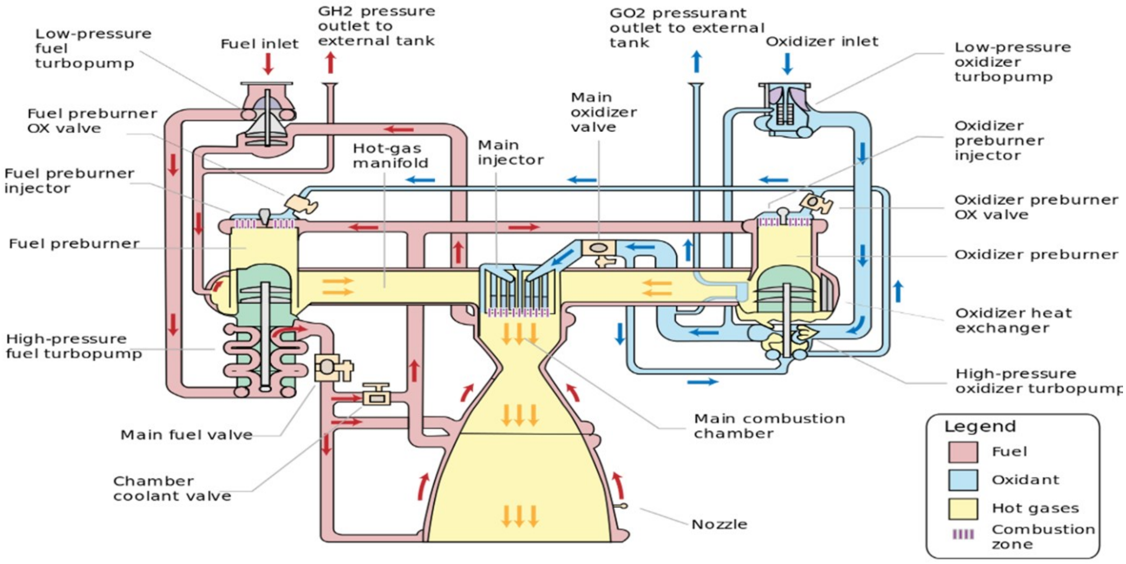

Figure 1.11: Schematic representation of an RS-25 engine

For the pressure-fed system, the propellant is forced into the thrust chamber by the use of a high pressure gas, but is proportional to the required total impulse of the system, and as such can be a very large and heavy system. For such applications, it could be wiser to take advantage of the pump-fed systems where there are turbopumps which force the propellants into the thrust chamber, but the turbine powering the turbopump needs a source of hot gas in order to provide the torque needed to spin the pump.
The combustion or thrust chamber is a large metal component which is designed to withstand the high pressure gases and exhaust them via the nozzle. These components can experience severely high temperatures, especially when considering large liquid propellant rocket engines, where it is required to utilise regenerative cooling chambers. This means that the fuel is moved through a series of tubes adjacent to the chamber wall which subsequently cools the wall and keeps the temperatures at an agreeable level. Several configurations can be seen in Figure 1.12 where the left option is referred to as single pass design, the two middle configurations are referred to as one and a half pass design, and the right option is referred to as a two pass design. As an alternative option to regenerative cooling, a chamber can instead use ablative liners or film cooling techniques to achieve the same effect.

The injector mixes and atomizes the propellants in order to achieve efficient combustion in the thrust chamber. The design procedure for the injector is still conducted in an empirical way because of the severely complicated two phase flow, heat transfer and combustion in this section. It must also ensure quality mixing as well as a thorough combustion while keeping the temperatures in the combustion chamber walls at an agreeable level.

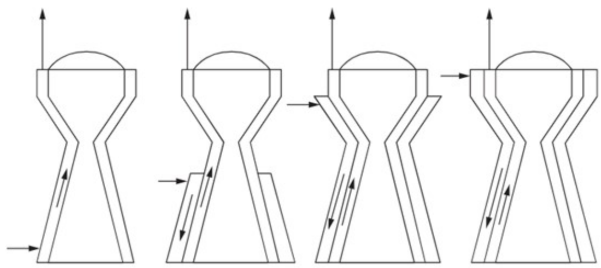

Figure 1.12: Various regenerative cooling configurations

Chapter 2

Monopropellant Rockets

Propellants, the working substances of rocket engines, constitute the fluid that undergoes chemical and thermodynamic changes. The term liquid propellant embraces all the various propellants stored as liquids and may be one of the following.

- Oxidizer (liquid oxygen, nitric acid, nitrogen tetroxide, etc.)
- Fuel (kerosene, alcohol, liquid hydrogen, etc.)
- Chemical compound (or mixtures of oxidizer and fuel ingredients) capable of selfdecomposition, such as hydrazine
- Any of the above, but with a gelling agent (these have yet to be approved for production)

Monopropellants may contain an oxidizing agent and combustible matter in a single liquid substance. It may be a stored mixture of several compounds or it may be a homogeneous material, such as hydrogen peroxide or hydrazine. They are stable at ambient storage conditions but decompose and yield hot combustion gases when heated or catalyzed in a chamber. Furthermore, they enjoy several advantages over the bipropellant systems:

- Only a single tank and propellant feed system is required – so weight is reduced
- Injection is simplified as there is no need to bring two separate fluids into contact
- They are less sensitive to temperature changes. Differences in density changes due to temperature in bipropellant systems can lead to situations in which one propellant changes its density prior to the other due to higher density sensitivity to temperature. This factor also leads to shifts in the mixture ratio.

2.1 Components

A typical monopropellant thruster is shown in Figure 2.1 and 2.2, illustrating its main components.

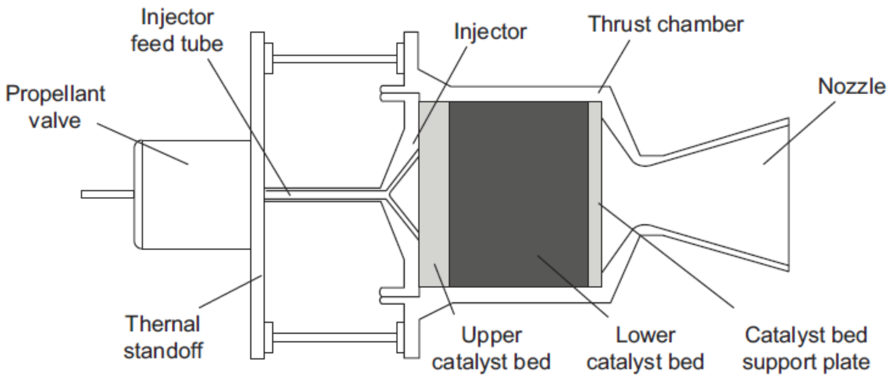

Figure 2.1: A typical monopropellant thruster.

The main components of the thruster are:

- Propellant valve – the only moving part of the system
- Thermal standoff – a spacer used to protect the incoming fuel from the heat released by the combustion chamber, so it won't decompose
- Injector feed tube and injector
- Catalytic bed – or 'catbed'
- Bed heater – some monopropellants require heated beds in order to decompose
- Heating coil – adds additional energy into the decomposition gases to raise temperature and specific impulse
- Nozzle

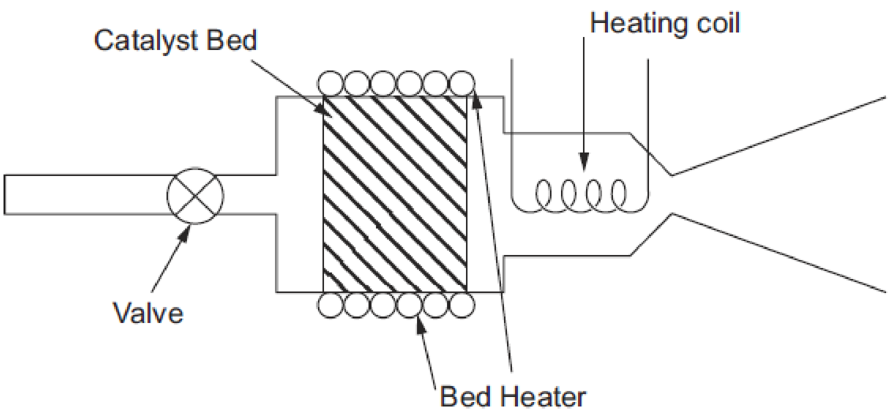

Figure 2.2: Hydrazine thruster with bed heater and electrothermal augmenter.

The bed heater is a simple electrical coil used to preheat the catbed before the thruster operation. Preheating it, minimizes the thermal shock to the catalysts particles during the ignition phase and enhances the decomposition rate.

2.2 Uses of the Monopropellant Systems

The simplicity associated with monopropellant feed control systems makes this kind of propellant very attractive for certain applications.

2.2.1 Gas Generators for Tank Pressurization

The objective of feed systems is to move propellants under pressure from propellant tanks to thrust chamber(s). The tank pressurization system is that part of the feed system that provides such a propellant expellant gas. There are two types:

- In a pressurized gas feed system, a relatively high-pressure gas displaces the propellants from the tanks.
- In a pumped feed system, the main energy for feeding the propellants comes from one or more pumps.

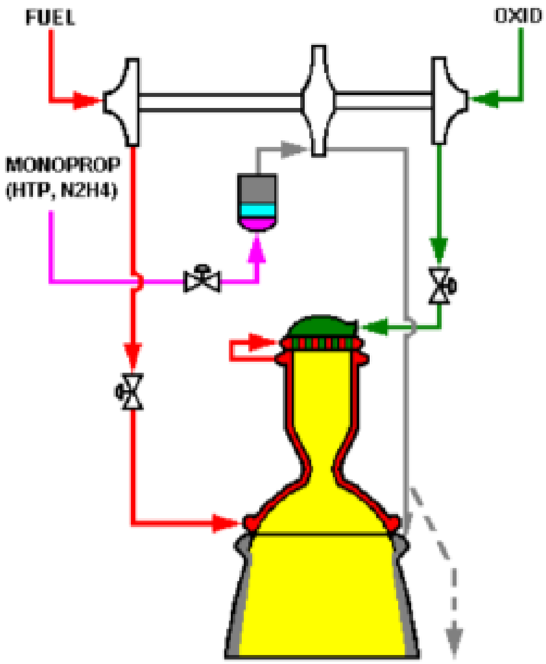

Figure 2.3: Gas generator used to drive the pumps.

There are several sources of pressurizing gas used in tank pressurization systems, one of these is gas created by a chemical reaction using either liquid bipropellants or a monopropellant, or alternatively a solid propellant, all at mixture ratios or compositions that result in "warm gas". The term warm gas (say $400°F$ to $1600°F$ or $204°C$ to $871°C$) distinguishes such gas from the "hot gas" ($4000°F$ to $6000°F$, or $2204°C$ to $3319°C$) in the main combustion chamber.
In the past, monopropellant gas generators were commonly using either 80% or 90% hydrogen peroxide or pure hydrazine to provide the warm gases, usually through catalytic decomposition in a bed of solid catalysts. For example, the V2 rocket's gas generator cycle was using a hydrogen peroxide with sodium permanganate catalyst system to drive the pumps, a schematic with a similar system can be found in Figure 2.3

2.2.2 Rocket Engines for Maneuvering, Orbit Adjustments, or Attitude Control

These engines usually include a set of small thrusters that are installed at various places in a vehicle and a common pressurized feed system. They are called attitude control systems in contrast to higher-thrust primary or boost propulsion systems. Most have multiple small thrusters, produce low thrust, use storable liquid propellants, and require accurate repeatable pulsing, a long life in space and/or long-term storage with loaded propellants in the flight tanks. Such auxiliary rocket engines are commonly used in spacecraft and missiles for accurate control of flight trajectories, orbit adjustments, or attitude control of the vehicle.

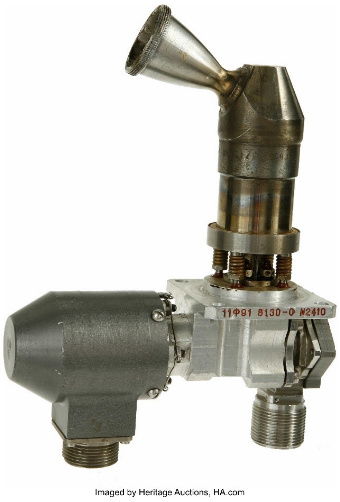

Figure 2.4: The Soyuz capsule's hydrogen peroxide yaw thruster.

Attitude control can be provided during two occasions, while a primary propulsion system (of a vehicle or of a stage) is operating and while its small thruster rocket system operates by itself. For instance, this is done to point satellite's telescope into a specific orientation or to rotate a spacecraft's main thrust chamber into the desired direction for a vehicle turning maneuver. A common method for achieving accurate velocity corrections or precise angular positions is to operate some of the thrusters in a pulsing mode. The guidance system determines the maneuver to be undertaken and the vehicle control system sends command signals to specific thrusters for the number of pulses needed to accomplish such maneuver. Small liquid propellant engine systems are uniquely capable of such pulsing operations.

Propellants for auxiliary rockets fall into three categories: cold gas jets (also called inert gas jets), warm or heated gas jets, and chemical combustion gases, such as fuel-rich liquid bipropellants. Warm gas systems can use inert gases from an electric heater or a monopropellant, which is catalytically and/or thermally decomposed.

Small liquid monopropellant and liquid bipropellant rocket units are commonly used as auxiliary rocket systems for thrust levels typically above 1.0 N and total impulse values above 3000 N-sec. Hydrazine is the most common monopropellant used in auxiliary control rockets. The MESSENGER's probe propulsion system, for example, has monopropellant thrusters.

2.3 Catalysts

A liquid propellant decomposes into hot gases with the aid of a catalyst. The function of the catalyst is to promote rapid decomposition of the liquid by reducing the activation energy required to initiate chemical decomposition reactions. Catalysts need to provide fast and repeatable performance as well as be insensitive to poisoning by the stabilizers and impurities in the propellant. They should not require preheating for efficient operation and should also be capable of sustaining the large number of thermal cycles imposed by typical mission profiles.

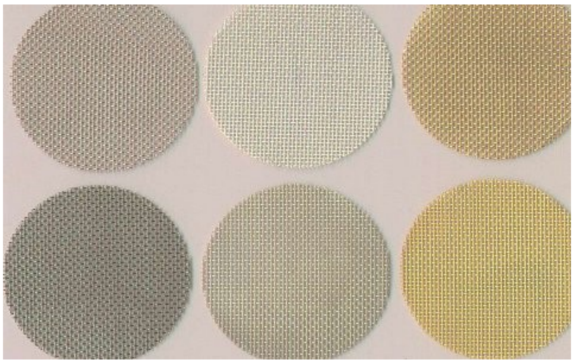

Figure 2.5: Catalysts coated in gold, silver rhodium, palladium, silver and platinum.

The catalyst material is generally applied as a coating and housed within a catalyst bed. The “catbed” is composed of high surface area features (either granular material or a series of finely woven screens) in order to rapidly expose liquid molecules to the bed surface and to initiate the exothermic reaction for the relatively cold propellant entering the chamber.

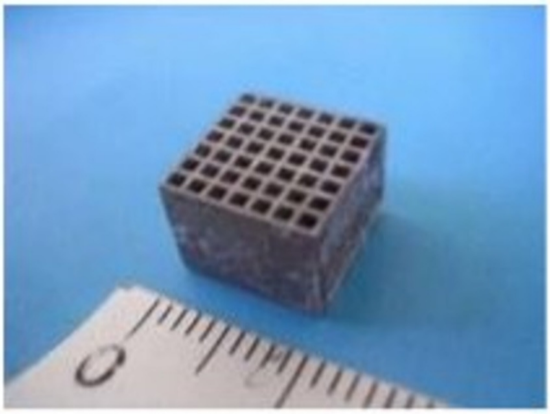

Figure 2.6: Ceramic monolithic catalyst for hydrogen peroxide

Flow processes within the catalyst bed are poorly understood as a variety of two-phase flow conditions (from bubbly flow that is mainly liquid to dispersed flow that is mainly gaseous with just a few drops) and the process of thruster development tends to be highly empirical. The main parameter governing the flow processes is the massflux or bed loading, $G = \dot{m}/A$, where A is the cross-sectional area of the catbed. Here G has units of $lbm/(in^2 * s)$ or $kg/(cm^2 * s)$. Higher G values are desirable as this implies that we can achieve higher propellant flows within the same cross-sectional area.

However, raising G also raises the velocities in the bed and reduces residence time, so there is a threshold beyond which the bed will become flooded and the conversion from liquid to gas will become incomplete. Generally, this limit can only be determined empirically due to the complexity of the two-phase flows within the bed itself. The other aspect of increasing the bed loading is that the pressure drop of the fluid tends to increase due to higher frictional losses within the tortuous path. The pressure drop characteristics are important at a system level as the tankage or fluid delivery system must provide sufficient pressure to overcome this loss and sustain the desired thruster chamber pressure.

2.4 Liquid Monopropellants

To date, only four monopropellants have been used in spacecraft: hydrazine, hydrogen peroxide, propyl nitrate, and LMP-103S. Several others have been considered, however, and at least one, AF-M315E, is scheduled for flight with the Green Propellant Infusion Mission (GPIM) on the second flight of the SpaceX Falcon Heavy rocket.

Out of the four monopropellants used in space flight systems, hydrazine is by far the most common and will therefore be reviewed first. The second most common monopropellant is hydrogen peroxide with significant use up to the 1970s, current use in the Soyuz spacecraft, and a regular interest within the aerospace community for various missions. LMP-103S is a Swedish developed monopropellant based on ammonium dinatramine (ADN). It has similar properties to the hydroxylammonium nitrate (HAN)-based AF-M315E so they will be reviewed together. Finally, propyl nitrate, or more precisely n-propyl nitrate, is a liquid monopropellant over a large temperature range but is significantly more shock sensitive than other flight-proven options. The same less-common monopropellants may be found by the reader in older textbooks or specialized literature in which fluids as reactive as nitromethane and nitroglycerin are mentioned for use in rockets.

One final but important aspect of monopropellants is the three classes used to distinguish between them. The three classes, A, B, and C, have the merit of providing clear distinctions between different "types" of monopropellants and will help the description of monopropellant currently available or in development.

- Class A includes single molecule compounds with both a fuel and an oxidizer included in the molecular structure. As can be expected, these compounds tend to exhibit stability issues under certain conditions or simply, like nitroglycerin as an example, are explosives. Others, like hydrogen peroxide, are purely the definition of a fuel and an oxidizer included in a molecular structure with two hydrogen atoms each connected an oxygen atom (H_2O_2).

- Class B encompasses single molecules which structures make them either fuels or oxidizers. These molecules behave as monopropellants when, upon activation with a catalyst or a heat source, the chemical structure of the atoms in the molecule becomes unstable. Hydrazine is the most common Class B monopropellant.

- Class C monopropellants are those synthetic mixture of two or more compounds of fuel and oxidizer. Unlike Class A monopropellants in which a single molecule is involved, constituents of a Class C monopropellant are not spontaneously reactive with one another but react sustainably once activated via ignition has occurred. AF-M315E and LMP-103S are recently developed Class C monopropellants.

2.4.1 Hydrazine

Hydrazine is a derivative of ammonia with the chemical formula N_2H_4. Liquid hydrazine has been, by far, the most commonly used monopropellant in space propulsion since the 1950s. It is both a monopropellant and a bipropellant fuel used in early rocket system with oxidizers such as hydrogen peroxide (with which it is self-igniting, or hypergolic). The applications for hydrazine include attitude control thrusters, insertion stages, and gas generators.

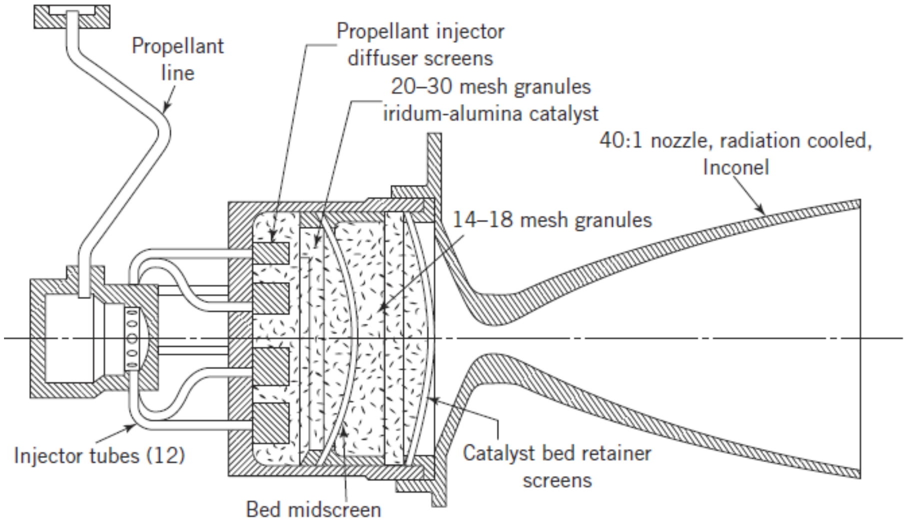

Figure 2.7: Typical hydrazine thruster.

All the monopropellant engines that use hydrazine require a catalyst which decomposes it into ammonia, nitrogen gas, and hydrogen gas. While several catalyst materials have been suggested and tested with hydrazine, a key aspect, valid for any monopropellant system, is its ability to start readily and operate steadily from low temperatures while sustaining the stresses induced by the exothermic decomposition process. The definition of "low temperature" is naturally quite subjective, but the high-altitudes thrusters typical of monopropellant systems certainly mean below Earth ambient temperatures. Most tested materials did not qualify as catalysts for hydrazine until the development of Shell 405, a solid catalyst composed of$30 wt.\%$ iridium deposited on aluminum oxide (Alumina, Al_2O_3). Alumina is a highly crystalline support with interlinked micro and macro-pores, providing a large surface area for the iridium and a highly efficient catalyst material.

The catalytic decomposition of hydrazine can be described ideally as a two-step process; such a simplified scheme ignores other steps and intermediate products. First, hydrazine (N_2H_4) decomposes into gaseous ammonia (NH_3) and nitrogen (N_2); this reaction is highly exothermic. Second, hot ammonia decomposes further into nitrogen and hydrogen gases, but this reaction is endothermic and absorbs heat. These simplified reaction steps may be written as:

$$3\ N_2H_4 \rightarrow 4\ (1-x)\ NH_3 + (1+2x)\ N_2 + 6\ x\ H_2 \tag{2.1}$$

Here, x is the degree of ammonia dissociation; it is a function of catalyst type, size, and geometry; chamber pressure; and dwell time within the catalyst bed, the operating parameters for decomposed hydrazine after leaving the catbed as a function of ammonia decomposition fraction can be found in Figure 2.8.

Both the characteristic velocity (c^*) and the specific impulse of the catalyst bed are affected

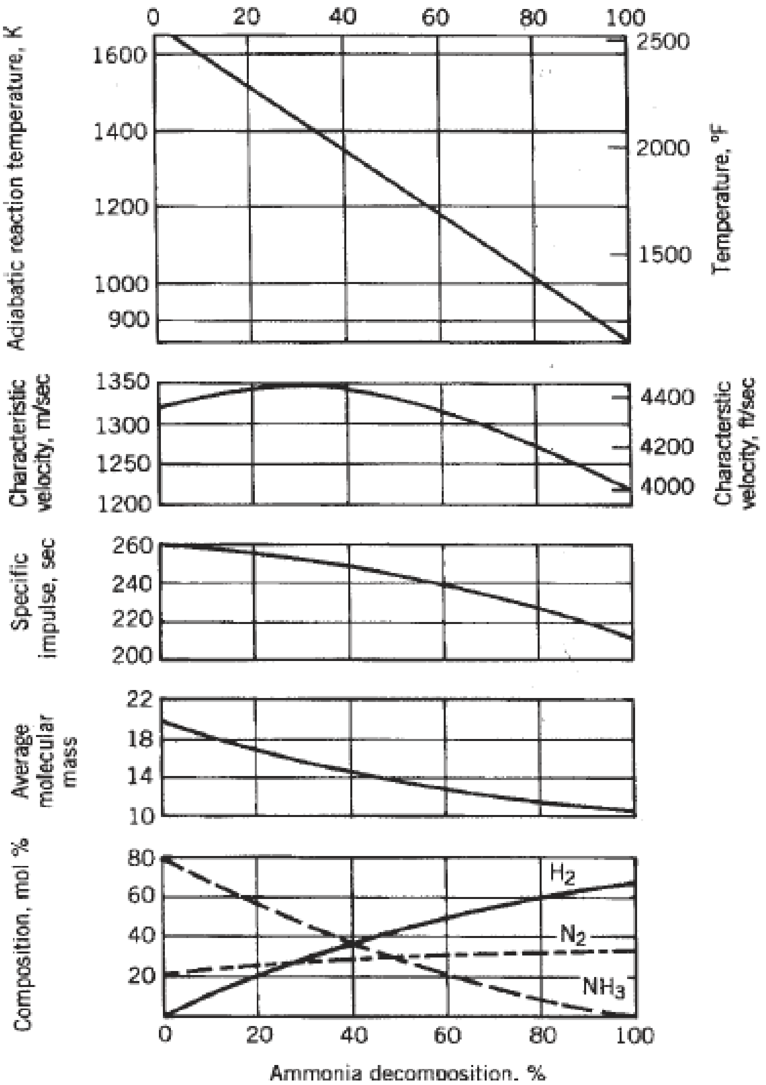

Figure 2.8: Operating parameters of decomposed hydrazine.

by the amount of ammonia decomposition, with a maximum c^* value near $1350m/s$ with about 30% of the ammonia decomposed and a maximum theoretical vacuum Isp near 260s with no ammonia decomposition. In practice, the ammonia dissociation is limited from 30% to 80% by limiting the catalyst bed thickness (with a typical value around 55% depending upon on the design). Overall, the exothermicity of the hydrazine decomposition reaction leads to a demonstrated vacuum specific impulse (Isp) exceeding 220 seconds.

With impurities or at higher temperatures hydrazine decomposes releasing energy. Under pressure shock (blast wave or adiabatic compression) hydrazine vapor or hydrazine mist can decompose at temperatures as low as $367K$. Under some conditions this decomposition can be a violent detonation, and this has caused problems in cooling passages of experimental injectors and thrust chambers. Harmful effects to personnel may result from ingestion, inhalation of its toxic vapors, or prolonged contact with skin. Hydrazine vapors may form explosive mixtures with air. If hydrazine is spilled on a porous surface or a cloth, spontaneous ignition with air may occur. It reacts with many materials, and care must be exercised to avoid contact with storage materials that cause decomposition. Hydrazine has a relatively high freezing point $(275K)$ and tanks, pipes, injectors, catalysts, and valves are usually electrically heated to prevent freezing in cool ground weather or in outer space.

2.4.2 Hydrogen Peroxide

Hydrogen peroxide is the simplest "peroxide," a compound with an oxygen–oxygen single bond, with the chemical formula H_2O_2. It has been extensively used as a rocket propellant, having the longest history of any liquid propellant. It was used in the Henkel He 52 aircraft in 1937, by Messerschmidt Me 163 in 1938 and by the V2 rocket to drive its turbopump in World War 2.

Rocket grade hydrogen peroxide (RGHP) can be decomposed reliably into superheated oxygen gas and water vapor by a variety of metals, oxides, and salts. The most common catalyst in RGHP monopropellant systems is silver, typically arranged in screens (either pure or coated on stainless steel or another alloy). As for any catalyst bed, preheating can accelerate decomposition and provide smoother start transients. However, any preheating is a tax on the power systems of a spacecraft and should be therefore be minimized, no matter for which monopropellant it is. With RGHP, even cold RGHP, catalyst beds can and have been designed that do not require preheating. It decomposes according to the following chemical relation, forming superheated steam and hot gaseous oxygen:

$$H_2O_2 \rightarrow H_2O + \frac{1}{2}O_2 + heat \tag{2.2}$$

This decomposition is brought about by the action of catalysts such as silver screens, various liquid permanganates, solid manganese dioxide, platinum, or iron oxide but, in fact, many materials can catalyze hydrogen peroxide. The propulsive performance of hydrogen peroxide monopropellant rockets is about 20% lower than hydrazine, but the volume specific impulse achievable with $90\% H_2O_2$ is higher than most other propellants due to its high density.

RGHP is characterized by a high mass ratio of hydrogen peroxide to water (usually greater than 85%) and by less than $1.0 mg/l$ of phosphorus, tin, or sodium ions. RGHP is also referred to as high test peroxide (HTP). All hydrogen peroxide solutions, including RGHP, decompose naturally at a very low rate of nominally $0.5 - 2\%$ a year. The decomposition rate of hydrogen peroxide increases by a factor of 2.3 for each $10°C$ rise in temperature, in accordance with Arrhenius' Law. The rate of decomposition of high-purity 90% hydrogen peroxide is listed at various temperatures in the Table 2.1.

Table 2.1: The rate of decomposition of 90% hydrogen peroxide at different temperatures.

Temperature [$°C$]	Approximate rate of decomposition
30	1% per annum
66	1% per week
100	2% per day
140	Rapid decomposition

In the past, concentrated hydrogen peroxide would self-decompose during storage at about 1% per year. This progressively diluted this propellant until no longer useful. As oxygen gas would bubble out of the stored liquid, the decomposition rate would accelerate. Small amounts of impurities in the liquid or in wall materials would also accelerate decomposition and cause a rise in liquid temperature. Before reaching $448K(175°C)$, warm stored H_2O_2 or contaminated H_2O_2 must be diluted and discarded because an explosion may occur.

Recent progress in the manufacture of storage tanks and piping materials (and stricter cleaning methods) has much reduced the amount of impurities and lengthened storage life of HTP from about 3 - 4 years to 12 - 16 years, and there may further improvements. This has much renewed interest in this dense oxidizer and several different investigations and development programs have been underway to date. The most significant technology challenge for the realization of hydrogen peroxide monopropellant thrusters is the development of effective, reliable, long-lived catalytic beds, giving fast and repeatable performance, insensitivity to poisoning by the stabilizers and impurities contained in the propellant, capable of sustaining the large number of thermal cycles imposed by typical mission profiles and not requiring (if possible) pre-heating for efficient operation.

As a direct consequence of the renewed interest in the use of hydrogen peroxide shown by the rocket propulsion community, various companies have developed several prototypes of hydrogen peroxide monopropellant thrusters. Alta S.p.A. (Italy) and DELTACAT Ltd. (United Kingdom) have developed two prototype thrusters (a 5 N and a 25 N one) based on the use of advanced catalytic beds.

It is interesting to see how the specific impulse and other performance parameters change in function of the concentration of hydrogen peroxide. The values for the 25 N thruster are shown in Table 2.2. With regard to operating pressures, a nominal operating pressure of approximately 10 bar has been chosen for the combustion chamber, taking into account the typical values used in similar applications in the past, while the exhaust/throat area ratio has been set on the value of 8. A very important aspect is the relatively low temperatures the system faces. The temperatures are lower than 1300K the annealing temperature of stainless steel, so this relatively cheap aerospace material can safely be used.

Table 2.2: Performance parameters as a function of concentration.

	85%	87.5%	90%	92.5%	95%	98%
Combustion chamber temperature (K)	907.27	968.46	1029.59	1090.71	1151.87	1225.41
Exhaust temperature (K)	318.07	344.21	370.99	398.41	426.43	460.82
Thrust Coefficient (C_T)	1.5604	1.5613	1.5622	1.5631	1.5639	1.5648
Gravimetric specific impulse (m/s)	1381.7	1426.0	1468.6	1509.6	1549.2	1594.9

2.4.3 Ionic Liquid-based Monopropellants

In general terms, ionic liquids (ILs) are salts in which the ions are poorly coordinated. This property makes ionic liquids melt at or below $100°C$ or, even more conveniently for many applications, at room temperature. Along with low to almost non-existent vapor pressure, high thermal stability, and the ability to dissolve a wide range of chemical species, it is said that ILs have brought a green revolution to chemistry and chemical engineering in the past dozen years.

While highly promising, the relative novelty of ILs and the virtually infinite number of building blocks to form new ILs complicates the generalization of their potential application to rocket systems. However, over the past 20 to 25 years, several organizations around the world have made significant progress in developing rocket-grade and low-toxicity monopropellants taking advantage of the properties of ILs. The key objective of this work is the development of candidates to replace hydrazine as a primary propellant for space propulsions systems.

Particularly, the US Department of Defense (DoD) and the Swedish Space Corporation (SSC) subsidiary Ecological Advanced Propulsion Systems (ECAPS) have developed IL propellant options based on ammonium salts, and, respectively, on hydroxylammonium nitrate (HAN) and ammonium dinitramide (ADN). These propellants, called AF-M315E for the HAN-based option and, for the ADN-based options LMP-103S and FLP-106, contain an IL as an ionic oxidizer associated with a fuel as a reducing agent. Water is added as a solvent, a stabilizer, and to tame the combustion temperature in catalyst beds and thrusters. Fuel components are added in the blend to increase propellant performance through higher energy release and higher flame temperature. The common fuel components are methanol, glycine, or ethanol. Overall, AF-M315E, LMP-103S, and FLP-106 can offer up to a 50% improvement in available density-specific impulse over hydrazine. One of the main advantages of these two salts is that they have a very low vapor pressure at room temperature and thus do not give off any toxic vapors, characteristic of hydrazine-based propellants.

The undeniable toxicity benefits of AF-M315E and LMP-103S over hydrazine should be counterbalanced by the potential hazards associated with the crystalline solids of HAN and ADN, two highly energetic salts with both reducing and oxidizing components. This means that in the pure form both are unstable and potentially explosive. In propellant applications, however, both HAN and ADN are used in concentrated aqueous solutions to limit the explosion potential. In addition, the water content in both AF-M315E and LMP-103S complicates their ignition processes and, at least to date, has meant that significant preheating on the catalyst beds is required for those to operate reliably. Cold-starts with AF-M315E and LMP-103S are not yet possible, thus presenting significant challenges to the power systems on-board smaller spacecraft.

Chapter 3

Bipropellant Systems

Liquid bipropellant rocket engines are the most used type of rocket propulsion systems for their special characteristics which will be discussed in this section. Liquid bipropellant rocket engines are highly complex systems which consist of the three main components:

- Oxidizer and fuel tanks.
- The feed system, whose aim is to bring fuel and oxidizer to the combustion chamber.
- The thrust chamber and the nozzle.

3.1 Desireable Properties of Liquid Propellants

Before addressing the most common fuels and oxidizers, it is worth noting to review the determinative properties of liquid propellants. Great knowledge of the physical and chemical properties of each unburned propellant is an important factor for the design and selection of propellants for rocket engines. To do so, the properties categorized into two parts, performance and physical properties. It is desirable to increase these three main performance properties:

- The specific impulse
- The exhaust velocity
- The characteristic velocity

The desirable physical properties of propellants are:

1. **Low freezing point**: Which allow the rocket operation in cold environments

2. **High specific gravity**: For a given mass a denser propellant permits a smaller tank volume and, consequently, a lower structural vehicle mass and lower aerodynamic drag.

3. **Storability**: This is a determinative factor for the selection of propellants for a defined mission. Cryogenic propellants are liquified gases at a less temperature than the environment which required a precise tanking material design, also they have a time limit for the operation while storable propellant can be used for long term mission based on their tanking property.

4. **Heat transfer properties**: High thermal conductivity, high decomposition temperature and high boiling temperature are desirable because usually the propellants are also used to cool down the nozzle's walls or combustion chamber's walls (storing some refrigerant on board means another tank and more weight).

5. **Pumping properties**: Low vapour pressure allows easier handling of the propellants and it reduces the potential for cavitation, while low viscosity is fundamental in order to make the pumping of propellant possible.

6. **High content of chemical energy**: to release the greatest amount of enthalpy during the combustion process.

7. **Spontaneous ignition**: Ignition is an important factor of propellants. Spontaneously ignitable propellants are called hypergolic propellants. This means combustion starts as an oxidizer and fuel meet each other. In this case, we don't need an ignition system which results in saving weight, complexity and cost.

8. **Homogeneity**: Propellant must have the same properties, same composition and same operative characteristics at the different condition and any time.

9. **Stability**: Good chemical stability means no deterioration and no decomposition of propellant in the long term storage and various operating condition.

3.2 Types of Propellants

Various kinds of fuels and oxidizer choose based on the mission requirements and the property of the raw propellants. Among all fuels and oxidizers, the most commonly used ones are listed below by their important specifications.

3.2.1 Oxidizers

- **Liquid oxygen** (O_2): Liquid oxygen is a cryogenic oxidizer which is widely utilized in space missions. As it is Cryogenic, it evaporates quickly, so it is impossible to store it for a long time and, consequently, it is produced near and just before the launch of the space vehicle. So, it requires to insulate tank, pipeline, valves which come in contact with it to reduce evaporation loss. Insulation means inert mass and inert mass means less payload transportable. It boils at $90K$ at atmospheric pressure and has a specific gravity of 1.14. Also, produces white-yellow flames when burns with hydrocarbons. The handling and storage of liquid oxygen could be safe only when all contact materials are clean.

- **Hydrogen peroxide** (H_2O_2) This is not just a powerful oxidizer but also it is clean burning oxidizer. It is used as a monopropellant with a solid catalyst bed in high concentrations, between 70% to 99%, and the remainder is mostly water. Furthermore, if it is well stored, it naturally decomposes (about 1% a year) and some gas will exit out of the liquid. When the contaminated liquid peroxide reaches a temperature of $448K$, an explosion usually occurs. It is dense and it doesn't produce any toxic products.

- **Nitric acid** (HNO_3): Different kinds of Nitric acid were used as an oxidizer, however, they are not in use today. The most common type of HNO_3 the red fuming nitric acid ($RFNA$), which consists of concentrated nitric acid that contains between 5% and 20% of dissolved nitrogen dioxide. The evaporating red-brown fumes are poisonous. Compared to concentrated nitric acid (white fuming nitric acid, $WFNA$), $RFNA$ is more energetic, more stable in storage and less corrosive to many tank materials. The specific gravity is in the range of 1.5 - 1.6, this value depends on the percentages of nitric oxide, water and impurities. This high-density feature results in more compact vehicle construction.

- **Nitrogen tetroxide** (N_2O_4) This propellant is the most commonly used storable oxidizer. It is a high-density yellow-brown liquid (specific gravity of 1.44). Due to this characteristic, it is largely used in the U.S.A., however, its liquid temperature range is narrow (which means it is prone to freezing or evaporation accidentally) and if humified or mixed with water it forms strong acids (and it readily absorbs moisture from the air). For this reason, it must be stored in sealed containers and it can be stored indefinitely without a change in its characteristics. Also, it is hypergolic with many fuels like hydrazine and unsymmetrical dimethylhydrazine. The fumes produced after combustion are red-brown and extremely toxic. It is possible to decrease the freezing temperature of the nitrogen tetroxide by adding a small amount of nitric oxide (NO) but this costs a higher vapour pressure. The mixture between N_2O_4 and NO is called "mixed oxides of nitrogen" (MON) and it has a percentage of NO between 2% and 30%.

3.2.2 Fuels

1. **Hydrocarbon fuels**: They represent a wide class of fuels, all obtained from the petroleum, with a wide range of characteristics that can be used for different missions for different applications. The most common type used in rocket engines is the $RP-1$. It is a kerosene-like mixture of saturated and unsaturated hydrocarbons with a narrow range of densities and vapour pressures. It is low in olefins and aromatics, in order to avoid the formation of crystals or rubber inside of the fuel cooling passages and pipelines. It is usually used with liquid oxygen in missiles such as ATLAS, THOR, DELTA, TITAN I and SATURN. Another interesting hydrocarbon for space propulsion applications is the cryogenic liquid methane, mainly for its high density which permits a compact structure and for its low cost while the characteristic velocity is relatively lower than liquid hydrogen.

2. **Liquid hydrogen**: It is commonly used as a fuel with liquid oxygen and releases a huge amount of energy. When it reacts with oxygen, the flame produced is colourless (maybe only the shockwaves in the plume are visible). Liquid hydrogen is the lightest and coolest fuel among all non-toxic fuels, therefore in one hand they can be used as a coolant in the propulsive system, on the other hand, they are not volume efficient this means it must be stored in large bulky insulated tanks, so its usage needs large vehicle volumes and it adds heavy inert mass. Some attempts to increase its density were made and one possible solution is to create a mixture of liquid hydrogen and suspended frozen small particles of solid hydrogen, which is obviously denser than the liquid form. However, this combination has not yet been used in a flight vehicle. Considering the first stages of a launcher vehicle system, it's better to use denser hydrocarbons such as RP-1 with liquid oxygen, because the gain obtained by reducing the volume of the vehicle in terms of aerodynamics drag overcomes the losses of thrust due to a low-energetic fuel. In Russia, there are experiments about a double-fuel booster stage, which starts using hydrocarbons and then switch to hydrogen during the flight. This brings to smaller volume vehicle than the only-hydrogen-fuel vehicle and to better performances vehicle in confront of only-RP-1-fuel vehicle.

3. **Hydrazine** (N_2H_4): It is not only used as the fuel in bipropellant but widely is utilized as a monopropellant. It is a colourless, toxic liquid with a high freezing point ($274.3K$). It is hypergolic with nitric acid, nitrogen tetroxide, etc. It also benefits from instant readiness which means, it has a short ignition delay and is

spontaneously ignitable with nitric acid and nitrogen tetroxide. Its vapour can form explosive mixtures with air. It can be stored in sealed tanks for 15 years, So is applicable for long term missions, but it is important to avoid contamination with materials that can cause decomposition. There are two derivatives of Hydrazine with better physical property and stability which listed below:

- **Unsymmetrical dimethylhydrazine** ($UDMH$): It is used instead of or in a mixture with hydrazine because it forms a more stable liquid at high temperature. Furthermore, it has a lower freezing point ($215.9K$) and a higher boiling point ($336.5K$) than common hydrazine. When it is burned with an oxidizer, it gives only slightly lower values of specific impulse than pure hydrazine. Compromising between the physical and performance properties, usually, it is mixed with 30% to 50% of pure hydrazine. It is used in Titan missile, in the lunar landing and take-off engines and in most of Russia and China's rocket engines. If the mixture froze, there would be the separation in two layers, and it would be necessary a remixing operation.
- **Monomethyl-hydrazine** (MMH): It is usually used with nitrogen tetroxide as oxidizer. It has better heat transfer properties and a better liquid temperatures range than pure hydrazine. Like hydrazine, its vapour is easily ignited in the air. It produces a 1% lower specific impulse than common hydrazine and it is the most toxic type of hydrazine.

It is hard to find a propellant that shows all of these characteristics, and gives the best performances, so it is important during the design of the bipropellant system to find a trade-off, suitable for the goals of the mission. It is noteworthy that additives can be applied to propellants in order to improve their properties. An important case is represented by gelled propellants, which are obtained through the action of some additives. This type of propellants has the consistency of jelly at rest, but they liquefy and flow through valves, pipes, pumps when adequate shear stress is applied. This structural configuration is useful to add suspended particles of aluminium, for example, which increase the heat released during combustion or suspended fuel/oxidizer particles, in order to increase the density. They also show long-term storability and reduce the possibility of leakage. However, they show a smaller specific impulse, due to the dilution with a gelling agent, and worse atomization into small droplets, due to high viscosity.

3.2.3 Common Bipropellant Combinations

Besides the properties of raw propellants, choosing the type of fuel and oxidizer to be combined is very important for the final performance of the system. The result of `NASA CEA` code reveals a comparison between the three examples of the most common combinations used in bipropellant systems we have seen before:

Table 3.1: Comparison between the most common rocket propulsion propellant combination for nozzle area ratio was set to 40 and combustion chamber pressure of $70bar$

Combinations	Fuel and oxidizer	$I_{sp,vac}$ [a] $[\frac{m}{s}]$	T_C $[K]$	C_f	ρ $[\frac{kg}{m^3}]$
Cryogenic propellants	$O_{2,liq}$; $H_{2,liq}$	4462.9	3133.74	1.8	2.93
LOX and liquid HC [b]	$O_{2,liq}$; $CH_{4,liq}$	3613.2	3559.12	1.86	5.19
Storable propellants	$N_2O_{4,liq}$; $N_2H_{4,liq}$	3232.0	3108.68	1.8	5.7

[a]Vacuum Specific Impulse
[b]Liquid Hydrocarbon

As can be interpreted from Table 3.1 each combination has its own advantages and disadvantages which discussed briefly below.

1. **LOX + Cryogenic liquid hydrogen**: It gives the highest specific impulse among all non-toxic propellant combination, therefore, it's a good choice for high-velocity missions. However, it has a low density, which requires larger storage tanks, following with higher structural weight. This combination is mostly used in the upper stage and sometimes in booster stages of space lunch vehicles

2. **LOX + Liquid hydrocarbon**: This fuel gives a smaller specific impulse, but much more thrust. It also requires smaller tanks than the previous combination, due to its higher density, thus permits more compact boosters. It is used in boosters or first stages.

3. **Hydrazine + Nitrogen tetroxide**: This storable propellant combination has a great feature named instant readiness (means, starting instantly without delay) which makes them a good choice for reaction thrusters and control application. It has a smaller specific impulse and smaller thrust which is sufficient for the kind of application they are used, so it is used for secondary engines or for a ballistic missile.

3.3 Mixture Ratio

The mixture ratio is one of the determinative factors in bipropellant systems to be sure the right amount of fuel and oxidizer get mixed together. For this purpose, it required extra components with respect to monopropellant systems which make them more complex but more efficient, too. Bipropellant systems are more efficient than monopropellant systems, mostly in terms of performance and overall vehicle mass. For high performance, high content of chemical energy per unit of the propellant mixture is desirable because it permits a high chamber temperature.

Also, the low molecular mass of the product gases of the propellant combination is also desirable which follows with a higher specific impulse. It can be accomplished by using fuels rich in combined hydrogen. In general, therefore, the best mixture ratio for many bipropellants is not necessarily the stoichiometric one (which results in complete combustion and yields a high flame temperature) but, a fuel-rich mixture containing a large portion of low-molecular-mass reaction products as shown in Figure 3.1. The other limit for the mixture ratio is the weight of the structure. Since for the fuel-rich mixture the extra hydrogen is needed and as hydrogen has low density, carrying extra excess hydrogen requires a huge volume which costs a higher overall structural weight. For this reason, the O/F ratio is slightly higher than this value for the pick specific impulse that is equal to 5.3 in this case.

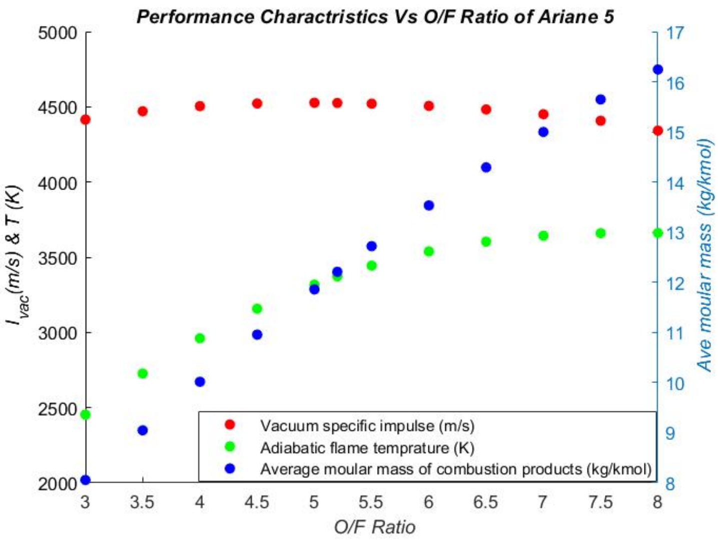

Figure 3.1: `NASA CEA` result of Vulcain (rocket engine of Ariane 5), for ($3 < \frac{O}{F} < 8$), the nozzle area ratio of 40 in 70 bar chamber pressure with cryogenic LOX and LH_2.

Chapter 4

Liquid Propellant Feed Systems

In order to ensure proper combustion in the thrust chamber and thus adequate rocket performance, propellant feed systems have two main tasks: to raise the pressure of the propellants to a suitable value for the combustion process and to feed them at the predicted mass flow rates at one or more thrust chambers. To accomplish this, generally one of two available propellant feed systems is selected: a pressure-fed system or a pump-fed system. Both concepts have their advantages and drawbacks for different mission scenarios, which will be discussed hereinafter.

4.1 Pressure-Fed Systems

A very simple and common approach to transport the propellants from their respective tanks to the combustion chamber is to pressurize them. In pressure-fed systems, the propellant tanks' pressure is raised to a level, which is higher than the thrust chamber pressure, consequently forcing the transport of the propellants into the chamber. This is achieved through compressed gases inside the propellant tanks, which can either be:

- inserted from an external source (e.g. high-pressure gas tank or gas generator) therefore allowing for a *pressure regulated operation* or
- stored inside the propellant tanks throughout the entire mission ("*blow down operation*", as the tank pressure decreases with propellant consumption).

Generally, pressure-fed systems are a simple, reliable, and low-cost solution. As they require heavy, pressurizable tanks, whose weight further increases with more propellant mass, pressure-fed systems are best suitable for small- and medium-sized rockets engines (e.g., for satellite and orbital propulsion).

4.1.1 Blow Down Operation

Blow down feed systems require large pressurizable propellant tanks, in order to additionally store the pressurizing gas, as can be seen in Figure 4.1. When the valves of the tanks are opened, the pressurized gases push the fuel and the oxidizer towards the combustion chamber. The main advantages of this system are its simplicity and its lower weight compared to the pressure regulation operated system. On the other hand, as the liquid fuel and oxidizer are consumed and the "available volume" in the tanks increases, the pressurized gases face an expansion process which leads to a decrease in their pressure. If the pressure of the gases decreases, also the push over the liquid propellant decreases, and, therefore, the mass flow rates. A decrease in mass flow rates leads a decrease of the thrust and of the specific impulse. Even though the rocket performance declines during the mission, blow down operation might still be suitable, as it is a very simple potentially light-weight solution.

4.1.2 Pressure Regulated Operation

This type of pressure-fed systems uses gas pressure regulators in the gas feed line, enabling the engine to operate with constant tank pressures and therefore essentially constant thrust. These systems are usually composed by an external high-pressure gas tank, a gas starting valve and a pressure regulator in addition to the propellant tanks, propellant valves, and feed lines. A schematic diagram of a typical pressure regulated feed system can be seen in Figure 4.1. The pressurized gas starts to flow to the propellant tanks, pushing the fuel and the oxidizer to move towards the combustion chamber. The main advantage of this system is that it allows for a constant pressure feed because the pressure regulator is able to control the tank pressures independently of the change in the volume occupied by the propellants stored in the tanks. Even though pressure regulated systems are a slightly more complex solution, they are still reliable and the advantage outweighs the added complexity: Constant pressure feed means constant propellant flow and, consequently, approximately constant thrust and specific impulse. It also permits a better control on the fuel/oxidizer mixture ratio.

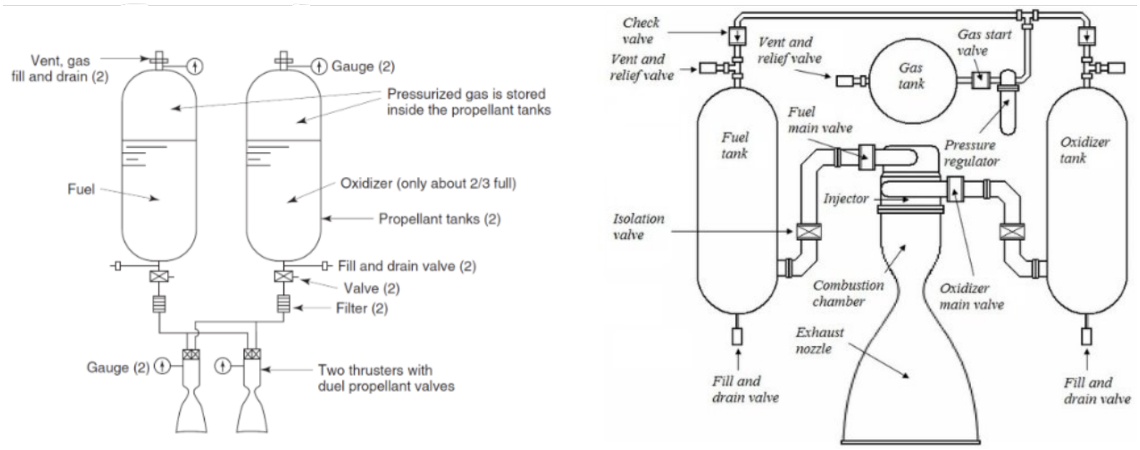

Figure 4.1: Different pressure-fed systems [1, 2]

4.1.3 Selection of Pressurizing Gas

There are several sources of pressurizing gas commonly used in pressure-fed systems:

1. High-pressure inert gas stored at ambient temperature is the most common choice. Common gases in this case are helium, nitrogen or air. This is by far the easiest solution, which is why it is often chosen, even though it theoretically offers a slightly smaller efficiency than the following "heated gas" approaches.

2. Heated high-pressure inert gas ($\approx 100°C \div 500°C$). Heating the pressurizing gas is useful to reduce the required amount of mass of gas and thus the inert mass of the system. Usually, electrical heaters are placed in the tanks.

3. Gas created through a chemical reaction. Liquid bipropellant, liquid monopropellant or solid propellant can be reacted in order to obtain a gaseous products mixture

which can be used as a "pressurized gas" in the pressure-fed system. It is important to use only fuel over oxidizer ratios that allow to obtain "warm gases", so they don't need to be cooled down. "Warm gases" are gases whose temperature is much lower than the temperature reached by the combustion products in the combustion chamber (so, for example, in the order of $500°C$). Commonly, there are two different "pressurizing gas" generators: one that produces pressurizing gas for the fuel tank, another that produces pressurizing gas for the oxidizer tank. The two pressurized gases are different: the one which is used for the fuel feeding system is a fuel-rich gas, while the one which is used for the oxidizer fuel system is oxidizer-rich. Nowadays, some experiments have been made in order to obtain a system of this type but without adding other chambers of reaction in which the gases are generated. One attempt recently made consists in the injection of some hypergolic fuel inside of the oxidizer tank and some oxidizer inside of the fuel tank, in order to obtain little reactions that bring to the formation of gases, but the only successful flight application is reported to be a upper stage engine of a Russian ICBM.

4. Gas obtained through the evaporation of a propellant. The evaporation of the propellant, usually liquid hydrogen, is obtained by applying heat taken from the thrust chamber and nozzle cooling jacket. Therefore, part of this evaporated propellant is used as a pressurizing gas in the pressure-fed system. Pressure regulators are needed to control the pressure inside of the tanks.

5. Self-pressurization of cryogenic propellants obtained through the evaporation of part of the liquid fuel and oxidizer. This method is difficult to control.

4.1.4 Design of Pressure-fed Systems

In the pressure-fed system design, a key factor is the amount of pressurizing gas mass necessary to ensure the operation of the system. In the following, the exemplary calculation of the required gas mass is shown for a pressure regulated system using an external high-pressure gas tank.

First of all, some assumptions must be made:

1. The pressurizing gas goes through an adiabatic expansion, so no heat is exchanged with the system's hardware. This hypothesis can be taken because the tank is insulated and the expansion process is quite fast, so the amount of heat exchanged is minimum and negligible.

2. The pressurizing gas behaves as an ideal gas.

3. The conservation of energy can be applied.

4. The gas is inert.

5. There is no evaporation of the propellant (this assumption is valid only if the vapor pressure of the species is low).

6. The pressurizing gas does not dissolve inside of the propellant.

Considering the initial conditions of the gas in the gas tank expressed with the subscript 0, the conditions at a certain instant of time t in the gas tank expressed with g and the conditions at a certain instant of time t in the propellant tank with p, the conservation of the energy of the gas between the initial instant of time and a following instant of time t is equal to:

$$m_0\ C_v\ T_0 - m_g\ C_v\ T_g - m_p\ C_v\ T_p = p_p\ V_p \tag{4.1}$$

Where:

- $m_0\ C_v\ T_0$ = initial energy of the pressurizing gas.
- $m_g\ C_v\ T_g$ = energy of the gas in the gas tank at t.
- $m_p\ C_v\ T_p$ = energy of the gas in the propellant tank at t.
- $p_p\ V_p$ = energy given to the propellant at t. This is the work done by the gas in displacing the propellants.

Applying the perfect gas state equation:

$$\frac{C_v\ p_g\ V_g}{R} + \frac{C_v\ p_p\ V_p}{R} + p_p\ V_p = m_0\ C_v\ T_0 \tag{4.2}$$

And considering that $V_g = V_0$, the equation becomes:

$$\begin{aligned} m_0 &= \frac{C_v\ p_g\ V_0}{R\ C_v\ T_0} + p_p V_p \frac{C_v + R}{R\ C_v\ T_0} = \frac{p_g\ V_0}{R\ T_0} + p_p\ V_p \frac{\gamma}{R\ T_0} \\ m_0 &= \frac{p_g\ V_0 + \gamma\ p_p\ V_p}{R\ T_0} \end{aligned} \tag{4.3}$$

But:

$$\frac{V_0}{RT_0} = \frac{m_0}{p_0} \tag{4.4}$$

So:

$$m_0 = \frac{p_g}{p_0} m_0 + \frac{\gamma\ p_p\ V_p}{RT_0} \rightarrow m_0 = \frac{p_p\ V_p}{RT_0}\left(\frac{\gamma}{1 - \frac{p_g}{p_0}}\right) \tag{4.5}$$

The first term of the equation expresses the amount of mass of the gas needed to empty a full propellant tank if the temperature of the gas is maintained at the storage temperature T_0 while in the second term $p_g = p_p$ = required propellant tank pressure.

4.2 Pump-fed Systems

The pump-fed system utilizes pumps to pressurize the propellants in order to push them to the combustion chamber. With a few exceptions, these pumps are always turbopumps (pumps connected to and powered by turbines). The turbines take the energy from the expansion of hot gases, which are usually produced from the propellants. The shaft on which the pump and the turbine are mounted is sealed, in order to prevent leakage of propellant and to prevent the fuel and the oxidizer from mixing with each other inside the turbopump.

4.2.1 Open vs. Closed Engine Cycles

In order to describe how a turbopump-fed system works, it is necessary to introduce the concept of engine cycles. An engine cycle describes the flow path followed by a propellant through the main components of the liquid rocket engine, the method to provide to the turbines the hot gas they need and how the exhaust gas exiting from the turbines is treated.

There are two categories of engine cycles:

- Open cycles: in an open engine cycle, the exhaust working fluid from the turbines is discharged overboard, usually through its own nozzle (in order to obtain some thrust by its expansion) or through the "main nozzle" of the combustion chamber (the injection of the exhaust gas happens in a point after the nozzle's throat).

- Closed cycles: in a closed engine cycle, the exhaust working fluid from the turbines is completely injected in the combustion chamber. Consequently it contributes to the combustion process and undergoes full expansion through the "main nozzle".

The difference between the two types of cycles is in the way they exploit the remaining energy of the exhaust gas. Usually, a closed cycle gives a slightly better performance than an open cycle, because the gas (in addition to contributing to the combustion) experiences a full expansion process after the injection in the combustion chamber, using the full pressure ratio of the "main nozzle" to expand. However, closed cycles are more complex to design and have higher requirements for the turbopumps and are therefore more expensive. Open cycles have efficiency losses, because they do not use all of the propellants for the combustion. Furthermore, a shift of the mixing ratio of the rocket combustion chamber away from the impulse maximum as a result of the propellant tapping is possible.

4.2.2 Most Common Engine Cycles

The most common engine cycles used nowadays are the gas generator cycle, the staged-combustion cycle and the expander cycle.

Gas Generator Cycle

In the gas generator cycle, the hot working gases for the turbines are produced in a pre-combustion chamber (*gas generator*), powered by propellants coming from additional propellants' tanks or from the main propellants' tanks. The gas mixture is usually fuel-rich, in order to obtain a low temperature and so to avoid cooling systems for the turbines' blades and for its own nozzle. Typically, a gas generator system is an open cycle. The schematic structure can be seen in Figure 4.2. It is simpler than other cycles, it requires a smaller inert mass and it is usually cheaper. On the other hand, its performance (the specific impulse) is lower by a certain percentage than the performance obtained with the other engine cycles. Despite this aspect, its performance is adequate for a wide range of space flight and military applications. A recent example of rockets using this type of cycle is given by the Vulcain 2, which is the main engine of ESA's Ariane 5.

Expander Cycle

The expander cycle is based on the evaporation of the engine coolant (such as the hydrogen or methane fuel). The coolant passes through the cooling jacket, where it picks up energy and cools down the walls of the combustion chamber and of the nozzle. As a consequence, it evaporates and the resulting gases are fed to the turbines. The schematic structure can be seen in Figure 4.2. Usually, 5% to 15% of the coolant gas bypasses the turbines and rejoins the exhaust gases exiting from the turbines before the entire coolant is injected in the combustion chamber, where it is mixed with the oxidizer and goes to combustion. The expander cycle in most realizations is a closed cycle, so it gives a good specific impulse. However, it can only be operated at low pressures, which is why it is only suitable for upper stage engines, e.g. the Vinci engine.

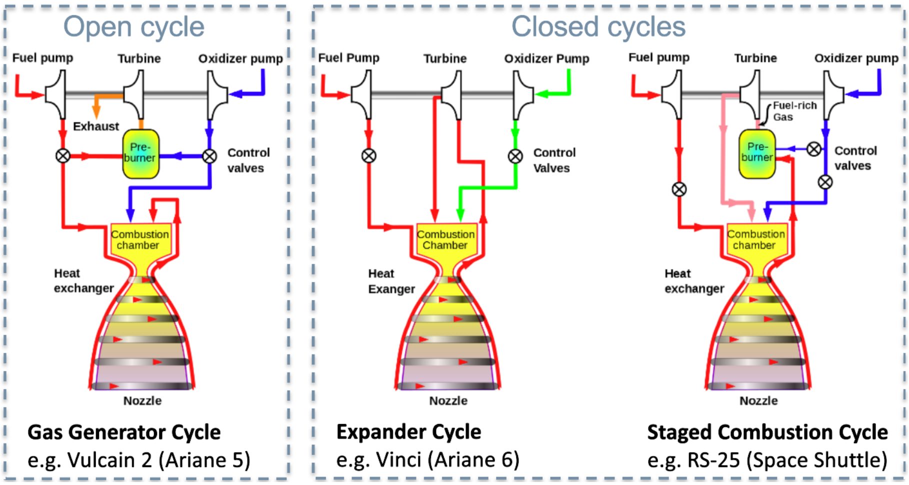

Figure 4.2: Schematic structure of the most common engine cycles [3, 4, 5]

Staged Combustion Cycle

The staged combustion cycle shows a similar coolant path to the one of the expander cycle. However, before the coolant gases enter in the turbines, they are burned in a preburner with a part of the oxidizer. In this way, the system provides to the turbines a high-energy gas, which allows them to extract a lot of work. Therefore, the pumps linked to the turbines have available more energy to compress the propellants. This leads to a high-pressure combustion in the combustion chamber, which allows to have a smaller combustion chamber. The exhaust gases of the turbines are all injected in the combustion chamber, where they go to combustion with the remaining part of the oxidizer. The schematic structure can be seen in Figure 4.2. The high-pressure operations require heavier and more complex pumps, turbines and pipelines. The staged combustion cycle gives the highest specific impulse in comparison with the two other cycle, but it is more complex and heavier. A variant of this system is used in the Space Shuttle Main Engine.

Comparison

A comparison of the most important properties of the three cycles is presented in Table 4.1. As can be seen from the table, the expander cycle and the staged combustion cycle give a better specific impulse than the gas generator cycle, however the gas generator cycle is nowadays the most common feed system used in the turbopump-fed rockets. This is because it is simpler, cheaper and it requires a smaller inert mass than the other two systems, which allows to have a lighter rocket.
The areas of operation (in terms of chamber pressure and thrust) are shown for the different engine cycles in Figure 4.3.

4.2.3 Special Engine Cycles

There are different engine cycles, which are not as commonly used. Two of them are modifications of the aforementioned common cycles: The expander bleed cycle and the full-flow staged combustion cycle. Additionally, the development of the so-called tap-off cycle has been achieved in recent years by Blue Origin. Their schematic structure can be seen in Figure 4.4.

Table 4.1: Comparison between the 3 most used engine cycles in liquid rocket propulsion [2]

Characteristics	Gas generator cycle	Expander cycle	Staged combustion cycle
$\frac{I_{sp,cycle}}{I_{sp,gas\ generator}}$	100%	102% – 106%	102% – 108%
Turbine flow as % of the total propellant flow	1.5% – 7%	12% – 20%	60% – 80%
Propellant type	All types	Cryogenic propellants	All types
Pump discharge pressure at % of combustion chamber pressure	130% – 180%	150% – 200%	170% – 250%
Turbine exhaust gas	Discharge overboard through own nozzle or through the main nozzle	Fed into the main combustion chamber	Fed into the main combustion chamber
Maximum pressure in feed system	Relatively low	Higher	Highest
Relative inert mass of engine	Relatively low	Higher	Highest

Expander Bleed Cycle

The expander bleed cycle is a open cycle variation of the expander cycle. As a consequence, it has a reduced effiency, but in return it is robust regarding turbomachinery efficiencies and facilitates component tuning. Furthermore it can be used in wide combustion chamber pressure ranges. So far it has only been realized in Japanese rocket engines, e.g. the LE-5B or the MB-60.

Full-Flow Staged Combustion Cycle

The full-flow staged combustion cycle is the advancement of the staged combustion. It features two pre-combustion chambers - one fuel-rich, one oxidizer-rich. Their combustion products are used to power the fuel and the oxidizer turbopump, respectively, and are then guided into the main combustion chamber. It allows for very high combustion chamber pressures, but has a very high complexity. It has only been operated in-flight with one engine, so far: SpaceX's Raptor engine.

Tap-off Cycle

The tap-off cycle uses hot gases from the combustion chamber to power the turbopumps. Its advantage is a low inert mass and low cost because of the reduced number of required components. According to Blue Origin, the cycle is particularly suited to human space-flight due to its simplicity, with only one combustion chamber and a less stressful engine shutdown process. However, engine startup is more complicated, and due to its nature of

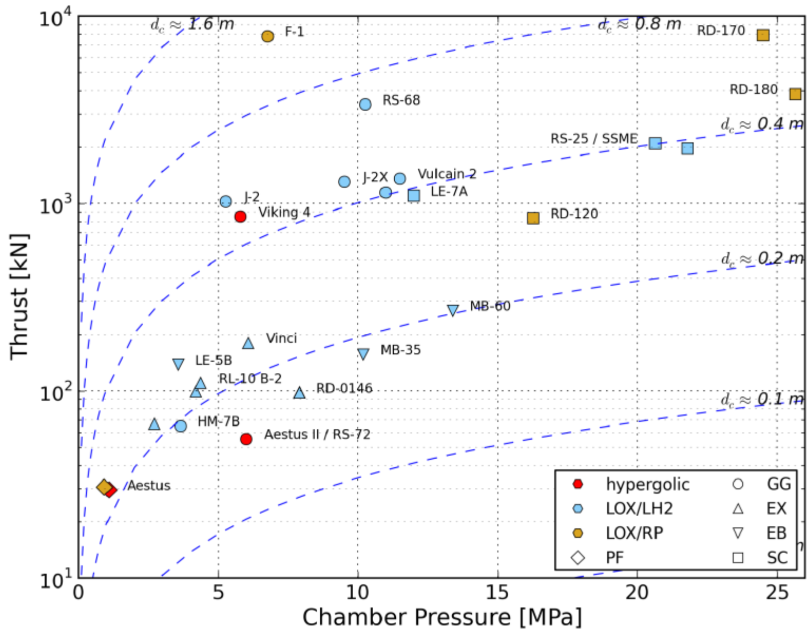

Figure 4.3: Operational thrust and chamber pressures for different rocket engines [6]

feeding gases from the main combustion chamber into the turbopumps, the turbine must be built to withstand higher-than-normal temperatures. Tap-off cycles are used in the BE-3 and BE-3U.

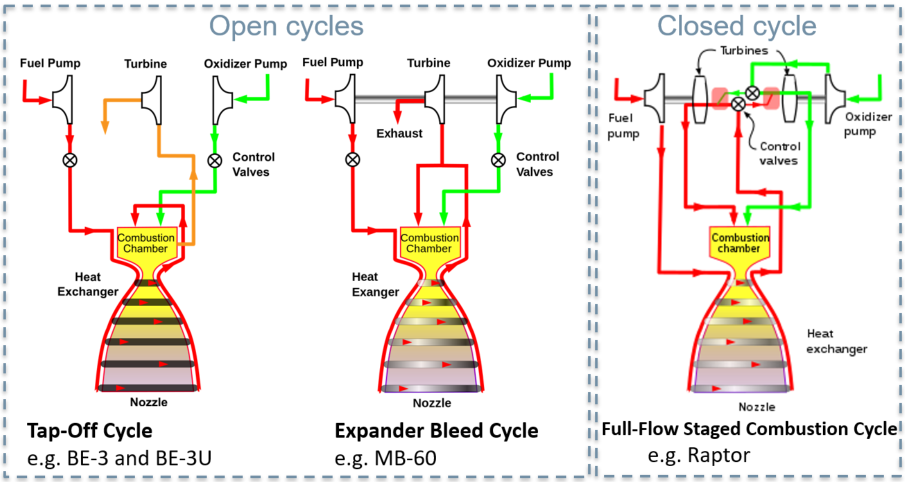

Figure 4.4: Schematic structure of special engine cycles [7, 4, 5]

Chapter 5

LRE Thrust Chambers

5.1 LRE Thrust Chambers: Introduction

To launch a rocket or to move a rocket through space, we must use a propulsion system to generate thrust. Thrust is generated through an application of Newton's third law of motion; a working gas is accelerated to the rear of the rocket engine, and the re-action is a thrust force applied to the engine in the forward direction. In solid and liquid fueled rocket engines, the working gas is produced through the burning of a fuel to produce power. Burning a fuel is called combustion, a chemical process. Because combustion is so important for rocket propulsion. Combustion is a chemical process in which a substance reacts rapidly with oxygen and gives off heat. The original substance is called the fuel, and the source of oxygen is called the oxidizer. The fuel and oxidizer can be a solid, liquid, or gas. For rocket propulsion the fuel and oxidizer are usually stored as either a liquid or a solid.

During combustion, new chemical substances are created from the fuel and the oxidizer. These substances are called exhaust. Most of the exhaust comes from chemical combinations of the fuel and oxygen. When a hydrogen-carbon-based fuel (like gasoline) burns, the exhaust includes water (hydrogen + oxygen) and carbon dioxide (carbon + oxygen). But the exhaust can also include chemical combinations from the oxidizer alone. If gasoline is burned in air, which contains 21% oxygen and 78% nitrogen, the exhaust can also include nitrous oxides (NOX, nitrogen + oxygen). The temperature of the exhaust is high because of the heat that is transferred to the exhaust during combustion. Because of the high temperatures, exhaust usually occurs as a gas, but there can be liquid or solid exhaust products as well. Soot, for example, is a form of solid exhaust that occurs in some combustion processes.

During the combustion process, as the fuel and oxidizer are turned into exhaust products, heat is generated. Interestingly, some source of heat is usually necessary to initiate combustion. Gasoline and air are both present in your automobile fuel tank; but combustion does not occur because there is no source of heat. Since heat is both required to start combustion and is itself a product of combustion, we can see why combustion takes place very rapidly. Also, once combustion gets started, we do not have to provide the heat source because the heat of combustion will keep things going. We do not have to keep lighting a campfire, it just keeps burning.

To summarize, for combustion to occur three things must be present: a fuel to be burned, a source of oxygen, and a source of heat. As a result of combustion, exhausts are created, and heat is released. You can control or stop the combustion process by controlling the amount of the fuel available, the amount of oxygen available, or the source of heat.

There are some unique liquid rocket propellants, called hypergolic propellants, that do

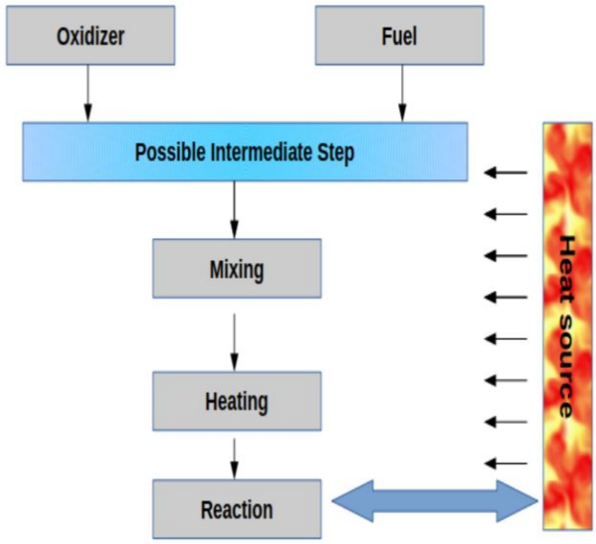

Figure 5.1: Combustion process

not require an external heat source to begin combustion. Hypergolic propellants burn on contact with one another. The advantage of using hypergolic propellants is that the engine design is simplified because no igniter is required, and the engine "fires" every time. The disadvantage is that the propellants are highly toxic and must always be kept separate to prevent unwanted combustion. Hypergolic propellants have been used on missiles like the Titan series and for thrusters on some spacecraft.

5.2 Combustion Chamber

The main combustion chamber contains a physical process that converts chemical reactions into a hot gas which is accelerated at the nozzle of the rocket. The pressure in the combustion chamber is proportional to the propellant flow rate since choked flow occurs in the throat. The geometry of the chamber is the trade-off between large volumes with long residence times, which is much better in terms of thermodynamics and higher combustion efficiency, and small volumes, which are much better in terms of reducing cooling requirements and space weight. The specific shape of the combustion chamber also makes the manufacturing process sometimes more complex. The combustion chamber which uses a pintle injector, the round head end much prefers to obtain a better radial flow pattern. For the low-thrust engine, the combustion chamber usually in one part with a nozzle. Instead, for the large engine, the combustion chamber usually ends downstream of the throat with the expansion ratio of the combustion chamber smaller than the overall expansion ratio of the nozzle.
Before discussing the details of actual thrust chamber design. The following summarizes these parameters and illustrates their use for design calculations by applying them to the engine system.

1. Specific Impulse. I_{sp} (sec):

$$I_{sp,tc} = \frac{F}{\dot{W}_{tc}} = \frac{c^* \, C_f}{g} \tag{5.1}$$

   The specific impulse figure indicates the overall quality of the thrust-chamber design. It shows how much thrust is generated while running propellant expenditures.

2. Characteristic Velocity, c^* ($\frac{ft}{sec}$)

$$c^* = f(\gamma, R, (T_c)_{ns}) \tag{5.2}$$

Assuming that the propellant and mixture ratio selection has been made. It can be expected that the gas properties (γ, R) will fall into a known band. From there on c^* almost entirely depends on the temperature of the gases. Obviously, this temperature has a theoretical maximum for a selected propellant combination. How close to this maximum the chamber will operate depends on influences for mixture ratio. It is seen that c^* peaks at combustion temperatures somewhat lower than maximum. Other considerations, such as bulk densities, which affect vehicle tank sizes, may cause further adjustments of the mixture ratio for optimum overall vehicle performance. Within these boundaries, the quality of the combustion process greatly depends on the design efficiency of a thrust chamber assembly, in particular the injector.

3. Thrust Coefficient, C_f (dimensionless)

$$C_f = f(\gamma, \varepsilon, p_a) \tag{5.3}$$

Let us assume that the performance of energy generation through the combustion process, the effects of which were just summarized for c^*, has been determined. Then, with a given gas property (γ), the performance of the remaining thrust generating functions of the thrust chamber, essentially those of the divergent nozzle. Will depend on the nozzle geometry, mainly ε which determines the pressure ratio $\left(\frac{p_e}{(p_c)_{ns}}\right)$ and the ambient pressure (p_a).

And also the another important parameter of combustion chamber is contraction ratio. The chamber contraction ratio (CR) is the ratio between the cross-sectional area of the chamber and the throat area. The value of CR usually lays in the range of 1.3 up to 10. The reduction of CR will have as an effect the increase of the Mach number in the combustion chamber. Rayleigh loss becomes more pronounced. Rayleight loss are stagnation pressure loss due to affect of adding heat at finite Mah number. The loss of stagnation pressure will directly affect to the reducing thrust. Figure 5.2 shows the relationship between chamber pressure and the contraction ratio. On the other hand, increasing CR also increases the chambers weight, therefore making the entire rocket heavier. The large engine tends to have a small CR, but for mid-size or smaller engines Rayleigh loss becomes the dominant consideration. So this engine tends to have a larger CR value.

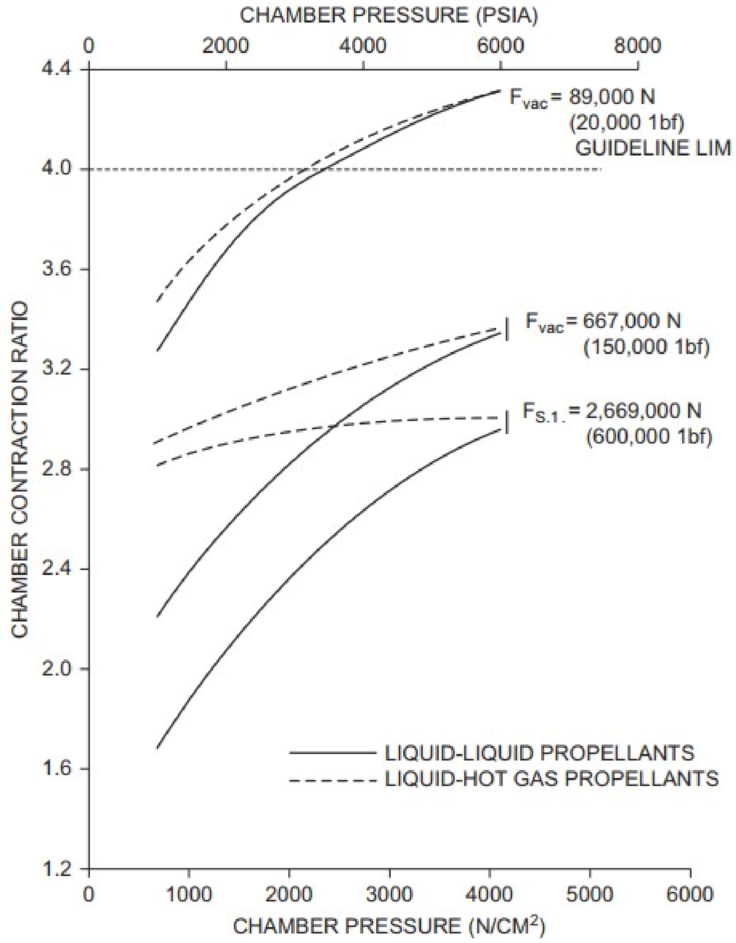

Figure 5.2: Chamber pressure vs contraction ratio

5.3 Thrust Chamber Configuration Layout

After major thrust chamber operating parameters such as type of propellants, thrust level, chamber pressure, C, c^*, and is have been established from engine system requirements and performance calculations, one of the fundamental dimensions of the thrust chamber, the throat area A_t, can be readily derived. The throat area A_t or throat diameter D_t usually is the starting point of a thrust chamber configuration layout. The combustion chamber and nozzle section are commonly designed as an integral thrust chamber body. For light weight and ease of manufacture, thrust chambers will have the general shape of a pressure vessel with wall surfaces of rotation and smooth contours. The combustion chamber serves as an envelope to retain the propellants for a sufficient period (stay time) to assure complete mixing and combustion before entering the nozzle. The volume of the combustion chamber thus has a definite effect on combustion efficiency. The theoretical required chamber volume is a function of the mass flow rate of the propellants and their average density, and of the stay time needed for efficient combustion. The relationship can be expressed by the following equation:

$$V_c = \dot{W}_{tc} \; V \; t_s \tag{5.4}$$

where:

- V_c = chamber volume, ft^3

- $\dot{W}_{tc}$ = propellant mass flow rate, $\frac{lb}{sec}$
- V = average specific volume, $\frac{ft^3}{lb}$
- t_s = propellant stay time, sec

A useful parameter, the characteristic length L^* (commonly pronounced "L-star"), can be used to specify the propellant stay time in the combustion chamber. L^* is defined as the ratio of chamber volume to nozzle throat area, and can be expressed by the following equation:

$$L^* = \frac{V_c}{A_t} = \frac{\dot{W}_{tc}\ V\ t_s}{A_t} \tag{5.5}$$

A_t is throat area. Since the value of At is in nearly direct proportion to the product of $\dot{W}_{tc}$ and V, L^* is essentially a function of t_s. The effect of L^* on c^* in an experimental combustion chamber is shown in Figure 5.3. The c^* value increases with L^* to an asymptotic maximum. Increasing L^* beyond a certain point tends to decrease overall engine system performance because of the following:

1. Larger L^* results in higher thrust chamber volume and weight.
2. Larger L^* creates more surface area in need of cooling.
3. Larger L^* increases friction losses at the chamber walls.

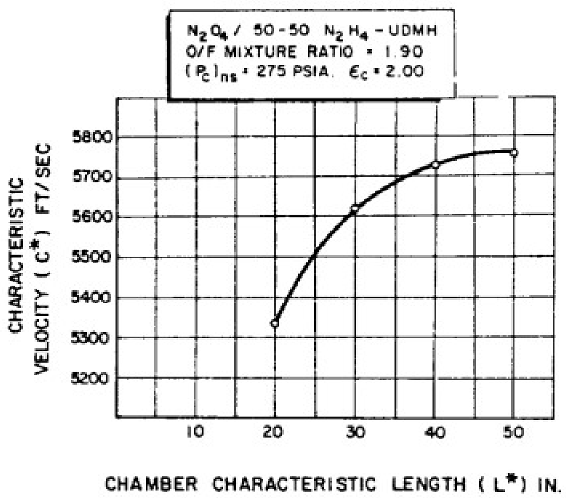

Figure 5.3: The effect of L^* on c^* value of experimental thrust chamber

In actual design practice, optimization analyses will determine the minimum possible combustion chamber L^* consistent with efficient combustion. Under a given set of operating conditions, such as type of propellants, mixture ratio, chamber pressure, injector design, and chamber geometry, the value of the minimum required L^* can only be evaluated by actual firings of experimental thrust chambers. L^* values of 15 to 120 inches for corresponding propellant stay-time values of 0.002-0.040 second have been used in various thrust chamber designs. With A_t and minimum required L^* established, the required combustion chamber volume V_c can be calculated by equation (5.5).

As can be seen from equation (5.4) the stay time t_s is independent of the combustion chamber geometry. for a given required volume, the chamber can be of any shape. In actual design, however, the choice of the combustion chamber configuration is limited. This approach also dictates a longer thrust chamber space envelope and imposes certain space limitation on the injector design to accommodate the necessary number of injector holes. With a short chamber of large cross section, the propellant atomization or vaporization zone occupies a relatively large portion of the chamber volume, while the mixing and combustion zone becomes too short for efficient combustion. Other factors, such as heat transfer, combustion stability, weight, and ease of manufacturing, are to be considered in determining the final combustion chamber configuration.

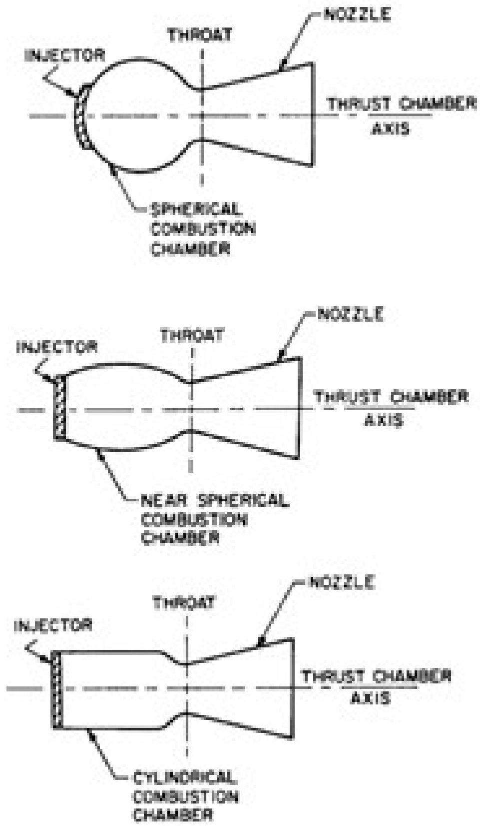

Figure 5.4: Geometrical shape of combustion chamber

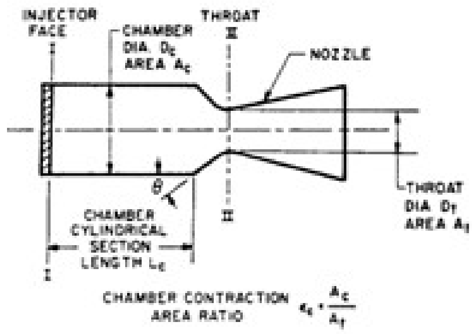

Figure 5.5: Combustion chamber model

Three geometrical shapes which have been used in combustion chamber design are shown in Figure 5.4. While the spherical and the nearspherical chambers were used in earlier European designs, the cylindrical chamber has been used most frequently in the United States. The spherical or nearly-spherical chamber, as compared to the cylindrical one of the same volume, offers the advantage of less cooling surface and weight. A sphere has the smallest surface-to-volume ratio. For equal strength of material and chamber pressure, the structural walls of the spherical chamber are about half the thickness of the walls of a cylindrical chamber. However, the spherical chamber is more difficult to manufacture and has poorer performance under most circumstances. For these practical reasons, the design details of the cylindrical combustion chamber will be treated in this book. Several novel thrust chamber designs will also be discussed. In the design layout of the cylindrical combustion chamber of a given A_t and L^*, the value of the contraction area ratio, can be optimized through careful studies of the following factors:

1. Combustion performance in conjunction with the injector design
2. Chamber gas flow pressure drop
3. Chamber wall cooling requirements
4. Combustion stability
5. Weight
6. Space envelope
7. Ease of manufacturing

For pressurized-gas propellant feed, lowthrust engine systems contraction area ratio values of 2 to 5 have been used. For most turbopump propellant feed, high thrust and high chamber pressure engine systems lower ratio values of 1.3 to 2.5 are employed. The reader is also referred to "The Gas-flow Processes in the Combustion Chamber and the Nozzle." The basic elements of a cylindrical combustion chamber are identified in Figure 5.5. In design practice, it has been arbitrarily defined that the combustion chamber volume includes the space between injector face I-I and the nozzle throat plane II-II. The approximate value of the combustion chamber volume can be expressed by the following equation:

$$V_c = A_t\Big[L_c\ \varepsilon_c + \frac{1}{3}\sqrt{\frac{A_t}{\pi}}\cot\theta\Big(\varepsilon_c^{\frac{1}{3}} - 1\Big)\Big] \tag{5.6}$$

The total surface area of the combustion chamber walls excluding injector face can be approximated by the following expression:

$$\text{Total area} = 2L_c\sqrt{\pi\ \varepsilon_c\ A_t} + \csc\theta(\varepsilon_c - 1)\ A_t \tag{5.7}$$

Table 5.1 is shown some characteristic chamber length (L^*) for some propellants. The heavier hydrocarbon molecule contained in kerosine (RP-1) needs more chamber length than lighter fuels like hydrogen because hydrogen can burn rapidly , therefore the residence time will be shorter. In addition, hydrogen has a high evaporation rate, and also due to the fast-kinetic reaction speed in the combustion process. In general, the cryogenic engine normally has a shorter chamber length than a comparable engine that using a storable propellant, because cryogenic propellant can easily evaporate and therefore have a lower residence time.

Table 5.1: Characteristic chamber length (L^*) values for selected propellants

Propellant combination	Minimum L^*, m
$LOX/RP-1$	$1.0 - 1.3$
LOX/LH_2	$0.7 - 1.0$
LOX/GH_2	$0.5 - 0.7$
Hydrazine family / NTO	$0.7 - 0.9$
$H_2O_2/RP-1$ (including catalyst)	$1.5 - 1.8$

5.4 Liquid Propellant

The combustion chamber of a liquid propellant unit is much more complex since injection, mixing, and reaction must occur within a few tens of milliseconds. The injection head is a complex device which ensures proper oxidizer-to-fuel mixture ratio. Several tens to some hundreds of injectors are in the plate to enable proper atomization and mixing.

5.5 Thermodynamics and Chemistry

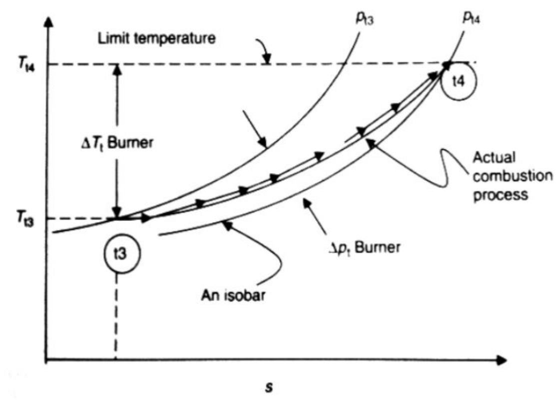

Figure 5.6: Thermodynamic transformation in a combustion chamber

From a thermodynamic point of view the ideal transformation occurring in a combustion chamber consists of a heat addition at constant pressure. The actual transformation is similar to this model, but some inefficiencies must be taken into account. First, total pressure losses are introduced by mixing, injection, and heat addition. Moreover, incomplete fuel burning should be considered (Figure 5.6). The heat addition process can be modelled with different degrees of complexity. We have already addressed the thermochemical approach based on equilibrium (space part of the lectures). The method applies also in this case, using the right fuel (kerosene) and oxidizer (air). If one chooses to maintain a simpler approach, the stoichiometric reaction can be used to have an initial evaluation of final temperature and heat of reactions involved.

5.6 Geometry for Space Combustion Chambers

In space applications, a distinction must be done between heating and combustion chambers. The former group of devices are mostly involved in a heating process of the propellant (or expellant), regardless of the type of energy source that is used. A nuclear propulsion unit or a resistojet represent two examples of heating chambers. Performances are bounded by the energetic level of the source (e.g. nuclear, battery) and by the strength of the materials that transfer the energy to the expellant. When chemical energy source is used, we talk about a combustion chamber, which produces high temperature and high-pressure gas mixture by heating. There are limitations by the nature of the energy source. A chemical propulsion unit makes use of the energy contained in the molecule bonds. A nuclear reactor uses the energy contained in the atoms, which is much higher. In this section, only combustion-kind transformations are considered, leaving the description of other units to next courses.

Chapter 6

Ignition in Liquid Rocket Engine

6.1 Introduction

Ignition phase is one of the most important phases where the pressurised fuel and oxidizer are **ignited** using ignition devices in the liquid rocket engine. LRE is commonly used as upper stage propulsion systems on multiple stage launch vehicles. So, it necessary to ensure that the ignition system highly reliable, less complex, optimal space and weight constraints, safety and capable of multiple engine re-lights.

This process is so critical because it is important to precisely define the operations sequentially. The propellant should be injected, atomised and then combustion initiation process takes place where a reliable ignition device plays a vital role and subsequently followed by flame stabilization under nominal conditions.

6.1.1 Selection of Ignition System

Selection of ignition system varies from one mission to another. So, the selection of ignition system according to the objective and constraints of the mission is vital. Selection of ignition system should be based on following factors [8]:

1. **Nature of propellant**: It is quite standard to describe ignition by first distinguishing two categories of propellant:

    - **Hypergolic propellant** that ignites spontaneously after contact without any source of external energy
    - **Non-hypergolic propellants** that are inert under standard injection conditions and requires an external source of energy to initiate the reaction.

2. **Constraints and considerations**: According to the misson aim, considerations and constraints the ignition system should be selected. Some of the basic constraints and considerations are;

    - Space
    - Weight and Size

3. **Nature of the system**: We should also ensure that the nature of the selected system is favourable for the mission. Some of the factors of favourable system,

    - Reliability
    - Safety

- Less complex

For both types of propellants, the ignition delay τ_i is the fundamental parameter of the process. This delay governs the ignition overpressure. Since, the value of τ_i determines the starting sequence, it is important to have a precise knowledge of this parameter under the operating conditions. For hypergolic propellants, the ignition delay is essentially governed by the kinetics of the propellants during which exothermic liquid/vapor reactions initiate combustion throughout the chamber at the many surfaces of contact of the reactants.

For non-hypergolic propellants, the delay depends on the gas-phase kinetics of the reactants, but it is also governed by the initial heat deposition, the rate of vaporization of the propellant injected as a liquid, and by the rate of turbulent mixing during the starting phase. Because the external source of energy is generally localized, ignition will involve flame propagation from the first flame kernel toward the surrounding reactants. In this case there are two important physical concepts that must be taken into consideration:

- The minimum ignition energy (E_{min}).
- The quenching distance (d_c).

The minimum ignition energy is the quantity of energy that when added to a flammable system, will lead to sustained flame propagation. The minimum ignition energy is then given by:

$$E_{min} = \rho \; \pi \; \frac{d_c^3}{6} \; c_p \; (T_{ad} - T_0) \tag{6.1}$$

This quantity is closely related to the quenching distance, dc, which corresponds to the size at which a combustion element of reactants (kernel) brought at its adiabatic flame temperature will grow unaided. Below the critical size, combustion is extinguished because the heat removed from the kernel exceeds the energy produced by the chemical reaction. This chapter focuses on the ignition in liquid rocket propulsion engine and the structure of this chapter is divided in two main sections,

- Ignition devices
- Ignition detection

6.2 Ignition Devices

Ignition device is an external device or tool to initiate the combustion process to release of chemical energy stored in liquid propellants and there are several methods and devices for ignition [9]. First, selection of an ignition system depends on the nature and phase of the propellants (bipropellants and monopropellants), need for altitude start, need for restart, system safety, compatibility of the ignition method with overall engine design, space and weight considerations.

6.2.1 Pyrotechnic Igniters

Pyrotechnic igniters are electrically initiated slow-burning pyrotechnic torches. They are designed with solid propellant technology and typically incorporated one or more solid propellant charges. The ignition of solid propellant produces a flow of hot gas and incandescent particles into the thrust chamber to ignite the rocket propellant as soon as it arrives.

Pyrotechnic igniters can be mounted to the thrust-chamber injector at the injector face, or through the injector from the manifold side. Particularly injector-mounted igniters with radial flame output provide a sheet of flame immediately downstream of, and parallel to, the injector face. Although pyrotechnic igniters provide excellent ignition sources, they have not been popular in recent engine, for reasons such as these:

- Electro-explosive interfaces and electrical components necessitate redundancy.
- Many large injectors are compartmented, requiring separate ignition source of each compartment.
- Pyrotechnic devices are subject by stringent safety requirements that raise hardware costs.
- Field checkout and installation of components are frequently required. Field sites must provide facilities for receiving, handling, and storing pyrotechnic devices.
- The igniter must be installed for each engine test and for launch.

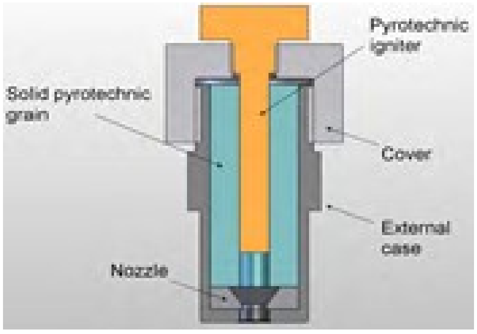

Figure 6.1: Pyrotechnic igniter model

Technology is now available for design of pyrotechnic igniters with laser-pyrotechnic interfaces replacing the electro-explosive interfaces. Elimination of the electro-explosive interfaces enables design of devices insensitive to radio-frequency-induced power and electrostatic discharge [10].

6.2.2 Hypergolic Igniters

The term 'hypergolic' denotes a bipropellant combination that ignites spontaneously when the two components meet. On successful ignition, a signal is sent to actuate remotely controlled valves for propellant feed line for supplying the main propellants to the combustion chamber to get requisite thrust. However, its clumsiness, the frequent clogging of feed lines, and the need to eject a considerable amount of inert solid material made it undesirable.

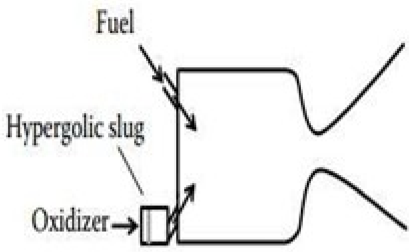

Figure 6.2: Hypergolic igniter model

To overcome this problem, a small amount of fluid that is hypergolic with one of the propellants is stored in a cylindrical cartridge that has burst diaphragms at both ends ('Hypergolic slug') as shown in the above figure. Generally, this loaded cartridge is placed as a bypass to the high-pressure main feed line. The diaphragm of this cartridge gets ruptured when requisite pressure is applied to it for the initiation of ignition. Note that a fuel that is hypergolic to the oxidizer is preferred over the opposite combination in actual practice. When the pump starts and outlet pressure rise, the oxidizer valve is opened. As pressures rise further, burst diaphragms in the hypergolic-fluid cartridge rupture and the fluid meets with the oxidizer in the chamber, igniting spontaneously. Subsequently, the main fuel propellant is fed into the combustion chamber to sustain ignition flame to reach main stage level.

However, the hypergolic slug provides only a single ignition event. The common hypergolic propellant combination includes: Triethylaluminum paired with liquid oxygen and Triethylboron paired with oxygen. The first one offers excellent ignition delay characteristics, but the major combustion product is aluminum oxide instead the second one produces a less tenacious residue but, has a long ignition delay. In general, a mixture of $10\% - 15\%$ by weight of Trienthylaluminum in Triethylboron provide satisfactory ignition-delay characteristics with liquid oxygen, while producing an acceptable residue.

Most importantly, due to their high levels of reactivity all commonly hypergols implemented have the disadvantage of high toxicity and/ or carcinogenicity. Additionally, hypergolic propellants present objective hazards like detonability or corrosiveness and require special material handling considerations that drive up operating costs.

6.2.3 Spark Plug and Spark-torch Igniters

Spark ignition system can be easily used in liquid- propellant rocket engines in which the initial energy required for ignition is provided by producing an electrical discharge(spark) between two electrodes positioned in a propellant stream. Particularly, typical spark igniter exciter can deliver energy in the range of $10mJ$ to $300mJ$ per spark and can produce 50 to 75 spark per second [11].

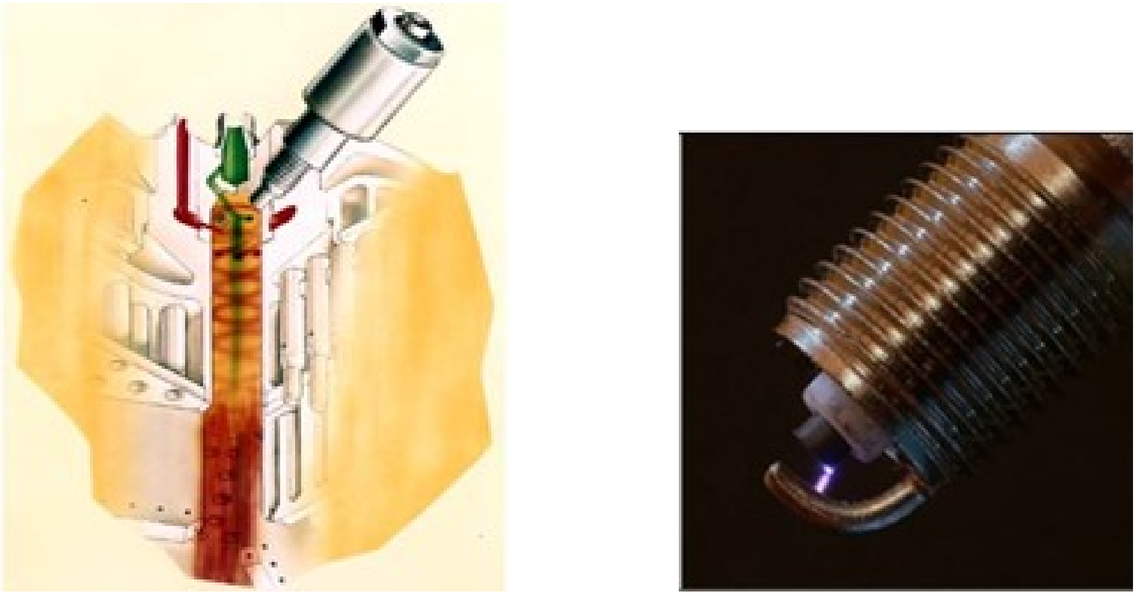

Figure 6.3: Spark plug igniter model.

The spark igniter may be mounted directly in the engine chamber ('direct spark ignition'), but this system has a limited spark energy thus may not be suitable for the larger rocket engine. This limitation has led to development of spark-torch ignition systems ('Augmented spark ignition') [12], in which small amounts of engine propellant are fed into an igniter combustor and ignited by electric spark. The hot flame generated is ejected out to ignite the main propellants in the thrust chamber.

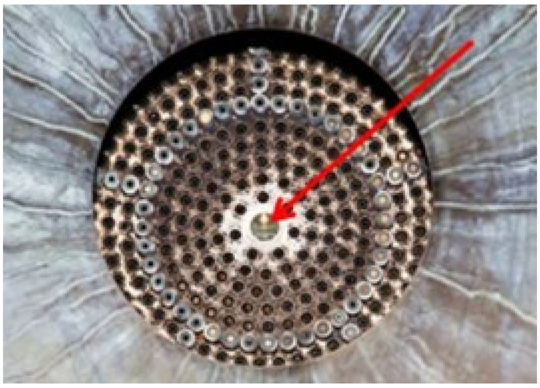

Figure 6.4: Augmented spark igniter model.

This is a picture "up the gullet" of an old SSME showing the main injector. The red arrow is pointing to where the torch ingniter injects into the chamber. This torch system is called Augmented Spark Igniter (ASI).

Generally, it is mounted on the centreline of the injector as it provides better performance. The spark-torch can deliver a very large number of starts, including ignition at high altitude. It can operate over wide ranges of propellant-inlet condition, mixture ratio, flow rates and is always present a redundant spark igniter to avoid system failure.

6.2.4 Laser Igniters

The fiber optically coupled laser ignition approach summarized is under consideration for use in igniting bi-propellant rocket thrust chambers. This laser ignition [13] approach is based on a novel dual pulse format capable of effectively increasing laser generated plasma life times up to 1000% over conventional laser ignition methods. In the dual-pulse format tinder consideration here an initial laser pulse is used to generate a small plasma kernel. A second laser pulse that effectively irradiates the plasma kernel follows this pulse. Energy transfer into the kernel is much more efficient because of its absorption characteristics thereby allowing the kernel to develop into a much more effective ignition source for subsequent combustion processes.

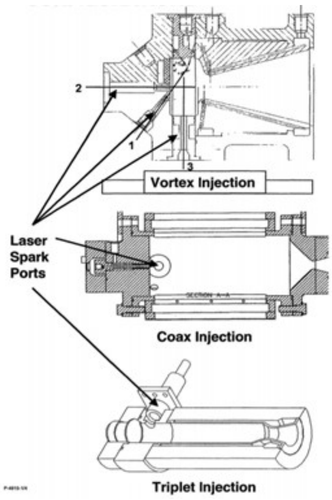

Figure 6.5: Laser spark port in different injections device model

Some of the important variables are spark location, pulse timing, and relative pulse energy. These variables were evaluated in an effort to identify the conditions in which laser ignition of bi-propellants [14] is feasible. Preliminary results and analysis indicates that this laser ignition approach may provide superior ignition performance relative to squib and torch igniters, while simultaneously eliminating some of the logistical issues associated with these systems. Further research focused on enhancing the system robustness, multiplexing, and window durability/cleaning and fiber optic enhancements is in progress.

6.2.5 Plasma Igniters

A plasma igniter (Figure 6.6) has been developed for initiating combustion in liquid-propellant rocket engines. The device propels a hot, dense plasma jet, consisting of elemental fluorine and fluorine compounds, into the combustion chamber to ignite the cold propellant mixture. The igniter consists of two coaxial, cylindrical electrodes with a cylindrical bar of solid Teflon plastic in the region between them. The outer electrode is a metal (stainless steel) tube; the inner electrode is a metal pin (mild steel, stainless steel, tungsten, or thoriated-tungsten).

The Teflon bar fits snugly between the two electrodes and provides electrical insulation between them. The Teflon bar may have either a flat surface, or a concave, conical surface at the open, down-stream end of the igniter (the igniter face). The igniter would be mounted on the combustion chamber of the rocket engine, either on the injector-plate at the upstream side of the engine, or on the sidewalls of the chamber. It also might sit behind a valve that would be opened just prior to ignition, and closed just after, in order to prevent the Teflon from melting due to heating from the combustion chamber.

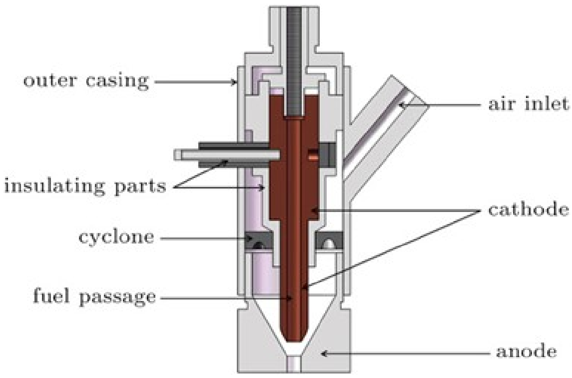

Figure 6.6: Plasma igniter model

The plasma jet [15] deposits the energy required to initiate combustion, while highly reactive fluorine and fluorine compounds create free-radicals in the flow-field to further promote rapid ignition. The plasma jet is created and accelerated electrically, and the feedstock for the plasma is maintained in a solid, inert form, leading to a rugged, reliable and compact design. The device should promote rapid and reliable ignition in LOX/LCH_4 engines, and in liquid propellant engines in general. It could also be used in gas-turbine engines where prompt and reliable restart is critically important, for example, in helicopter and jet aircraft engines.

6.2.6 Combustion-wave Igniters

The combustion-wave ignition system (Figure 6.7) utilizes a weak electric spark to ignite propellants in a pre-combustor. A combustion wave, or flame front, propagates from the pre-combustor to the system element tubes, gaining in intensity until, if the tubes are long enough it transitions into a detonation wave. This fast-moving flame front produces hot combustion products which, upon exiting the ignition system, ignite the propellants in the main combustion chamber.

Particularly, it becomes useful when large-segmented engines and multi-compartmented injectors with numerous individual compartments requires near simultaneous ignition at many locations. Otherwise, it would be necessary to have many single-point ignition sources. However, this system requires a large amount of hardware.

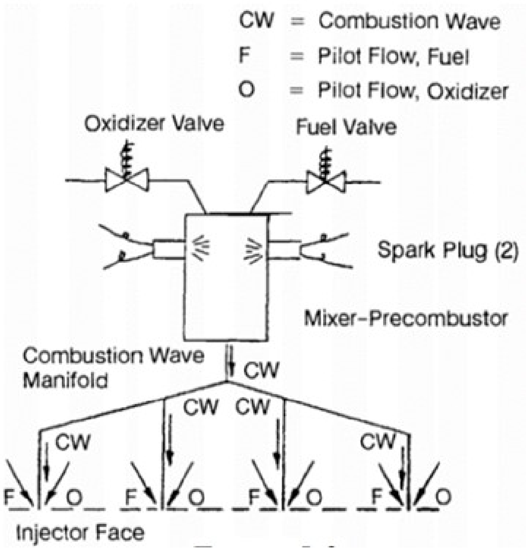

Figure 6.7: Combustion wave igniter

Moreover, the combustion wave ignition [16] concept was experimentally studied in order to verify its suitability for application in baffled sections of a large booster engine combustion chamber. Gaseous oxygen/gaseous methane (GOX/GH_4) and gaseous oxygen/gaseous hydrogen (GOX/GH_2) propellant combinations were evaluated in a subscale combustion wave ignition system. The system included four element tubes capable of carrying ignition energy simultaneously to four locations, simulating four baffled sections. Also, direct ignition of a simulated Main Combustion Chamber (MCC) was performed.

Tests were conducted over a range of mixture ratios and tube geometries. Ignition was consistently attained over a wide range of mixture ratios. And at every ignition, the flame propagated through all four element tubes. For GOX/GH_4, the ignition system ignited the MCC flow at mixture ratios from 2 to 10 and for GOX/GH_2 the ratios is from 2 to 13. The ignition timing was found to be rapid and uniform. The total ignition delay when using the MCC was under 11 ms, with the tube-to-tube, as well as the run-to-run, variation under 1 ms. Tube geometries were found to have negligible effect on the ignition outcome and timing.

6.2.7 Resonance Igniters

Resonance igniters use a flow of pressurized gaseous propellant to create resonance heating that will start the igniter propellant (Figure 6.7). Igniter exhaust, with or without oxidizer augmentation is directed to the injector face, where it is used to ignite the main propellants as they enter the combustion chamber. A typical sketch of resonance igniter is shown in the figure below, where the Gas (hydrogen) introduced from an elevated pressure source upstream of the sonic orifice expands into the mixing chamber and is directed into the resonator cavity, where it is cyclically compressed and expanded. This raises the temperature of the gas at the closed end of the cavity.

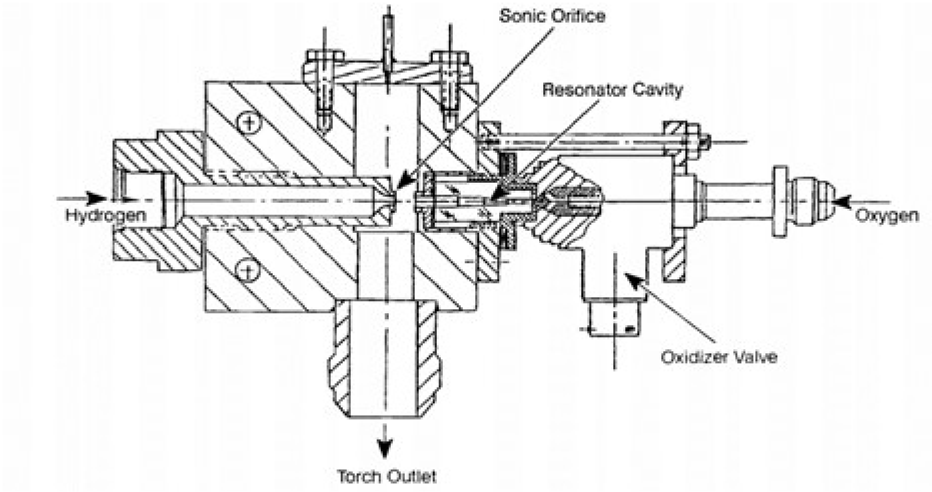

Figure 6.8: Resonance igniter

After the fuel in the cavity has reached a sufficient temperature, oxidizer is introduced at the 'hot' zone to effect ignition. This igniter does not require external electrical power or catalyst moreover in the case of segmented thrust chamber, where each segment must be ignited individually and simultaneously, the resonance igniter promises several advantages over other ignition system in terms of integration, simplicity, ease of operation and light weight packaging.

6.2.8 Catalyst Igniters

In a general sense, catalysts are not igniters but, initiators and sustainers of reaction and remain unchanged during these reactions. In rocketry, catalysts have been used predominantly to initiate and sustain the decomposition of monopropellants such as hydrazine and hydrogen peroxide. Since the monopropellants have a lower specific impulse, they are used along with a solid catalyst for igniting other liquid propellants like RP-1.

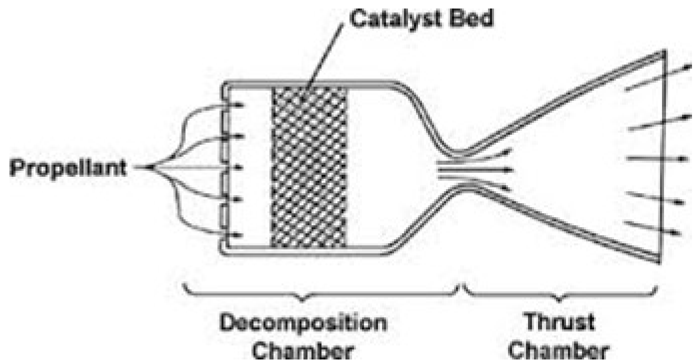

Figure 6.9: Catalyst igniter

In this manner, the solid catalyst indirectly serves an ignition system. Such a system offers versatility, storability, simplicity, and restart-ability. In recent times, catalyst for ignition is emerging as an alternative to the augmented spark ignition system. Efforts are being made to develop better catalysts, which must have a longer life with least cost.

6.3 Ignition Detection

The potentially catastrophic consequences of ignition failure have prompted extensive investigations of means to detect reliably the presence or absence of ignition in liquid propellant rocket engines. The only verification required is that safe, damage-free ignition has been achieved. Ideally, the ignition-detection system should be capable of instantaneous, 100% - reliable ignition detection — detection so rapid that, proper ignition not achieved, the engine can be shut down safely. With this system, the start sequence can proceed only after receipt of an 'ignition OK' signal. The main ignition detection system used are:

- **Image detectors**: Ground-mounted optical devices can be moved close to chamber exit. Several types have been investigated, such as simple light- or infrared sensitive cells. It is possible to mount the optical devices in the chamber wall facing toward the inside near injector face. but these devices become vehicle-mounted and require interfaces to ground support equipment.
- **Fusible-wire links**: For many applications, fusible-wire links prove simple and reliable devices. A wire is strung across the chamber exit, which when fused by the ignition flame, interrupts a circuit and signals 'ignition OK'. Proper selection of wire gage, material, and distance from the chamber exit and/or centre, will give some qualitative judgment. The wire can be ground-mounted or chamber-mounted. The wire links have several shortcomings that can be prevented by providing redundancy with several wires in parallel that must be interrupted before the sequence can proceed.
- **Pressure-sensing devices**: It has been attempted to sense the pressure rise in the combustion chamber resulting from the burning igniter flame. Since the pressure rise is small; however, reliable discrimination is difficult.
- **Resistance wires**: Constructed like a glow plug and connected to a bridge circuit, the resistance wire signals by a distinct change in resistance the presence or absence of ignition. Resistance wire sensors are ideally suited for repeatable-start engines.

Chapter 7

LRE Injection Plate and Injectors

7.1 Introduction

The injection system in a rocket injects the reactives in such a manner that guarantees an optimal and homogeneous mixture of fuel and oxidizer ratio, to ensure a proper atomization and, therefore, a stable combustion.
An optimal design of the rocket thrust chamber is required due to all the various forces and extreme heat generated that each component has to withstand or transmit to the vehicle.
The main design parameter of an injection plate are:

- The total number of elements and their distribution with minimal hot spots due to local flame impingement or gas recirculation
- Geometry of orifices to achieve the best performance
- Upstream flux

Through the injectors, fluids are injected at supercritical conditions with high values of pressure and temperature. It can also be shown that the fluids are always in turbulent regime because is recommended for propulsion systems, therfore, it is challenging to be implement for CFD because molecular mixing is the goal.

A fundamental aspect in a preliminary design phase will be the selection of the optimal type of element, their number and distribution over the plate. In fact, this parameter will affect the mixture of the propellant, the atomization and the mass distribution of the injected mixture in the combustion chamber. This will have a strong effect on wall resistance and on other components inside the combustion chamber, indeed an erroneous selection of the elements can cause impingement or erosive phenomena due to chemical reactions on the wall surface. The injection dynamic will be also responsible of some instability phenomena that will be further presented.

The main parameters that affect the injection processes and influence the selection of the elements are:

- **Type of propellant**: hypergolic, cryogenic or storable
- **Phase of propellant**: liquid, gaseous or gel
- **Combustion chamber**: not cooled, ablative, regenerative cooling methods
- **Length of the combustion chamber**

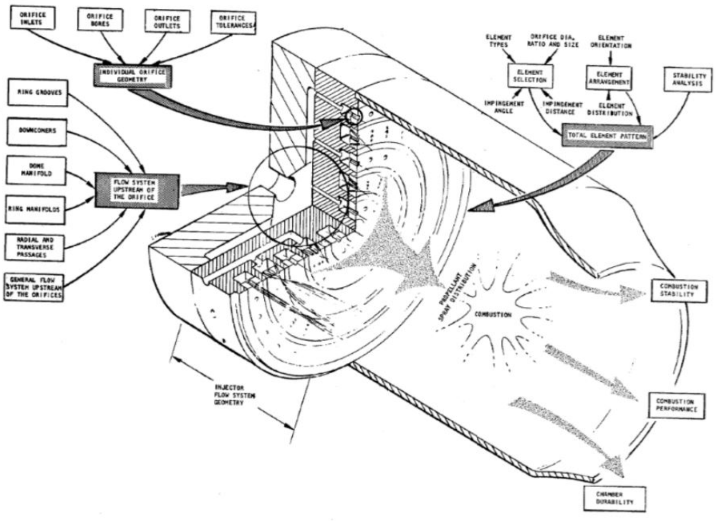

Figure 7.1: Injector scheme of F-1 rocket engine [17].

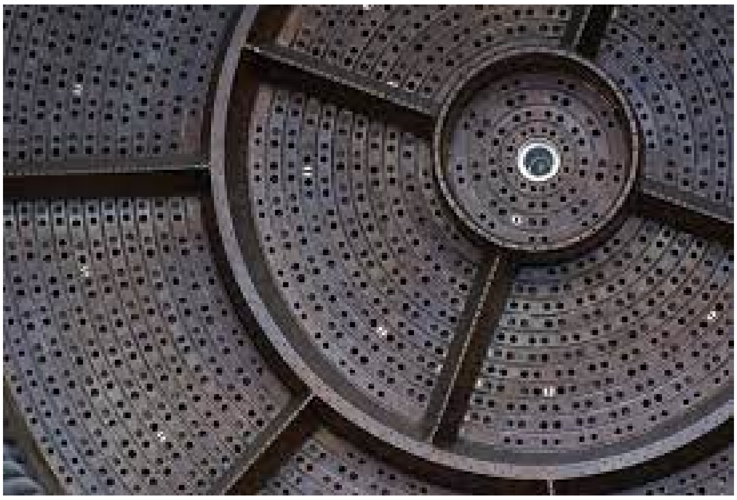

Figure 7.2: Injector plate of F-1 rocket engine [17].

- **Conditions of operation**: thermodynamic parameters in combustion chamber
- **Throttling** required
- **Pressure drop limitations**
- **Engine life**: restart, total duration

7.2 Elements Types

The different type of elements (injectors) can be subdivided in four categories:

- **Unlike-impinging**
- **Like-impinging**
- **Non-impinging**

• **Hybrid**

Each category differs in mixtures and atomization properties, furthermore they do not always guarantee a full compatibility with the chamber's wall; hence, during the design, the different types of elements must be selected accurately to ensure good performance and a suitable compatibility with the walls. The injection plate with multiple holes are usually employed for bipropellant systems with oxygen and hydrocarbons or with storable propellants.

7.2.1 Unlike-impinging Elements

The unlike-impinging elements are characterized by a series of small orifices distributed on the injection plate, they allow the atomization of the mixture through a direct impingement process between the oxidant and fuel jets.
The mixture and the atomization of the jets will occur in the immediate surroundings of the impingement point. Due to this, chemical reactions, ignition or impinging phenomena can occur on the injection plate.
The most employed element is the doublet impinging element. This kind of element is formed by two jets, one for the fuel and the other for the oxidizer. To allow optimal performance the diameter of the duct of oxidant and fuel need to be approximately the same and also must be guaranteed a sufficient injection delay to ensure the formation of fans.
Another largely used element of this kind is the triple-unlike impinging element, especially for nonsymmetric volumetric fluxes. This typology is formed by two external oblique jets, normally of oxidant, and a central one oriented axially. This symmetry of the system, allows the generation of a resultant jet directed accordingly to the axis of the combustion chamber in every operative condition.
In general, unlike configurations are preferred for hypergolic propellants as this promotes a rapid ignition. Additionally, with hypergolic propellants, the mixture will start to evolve

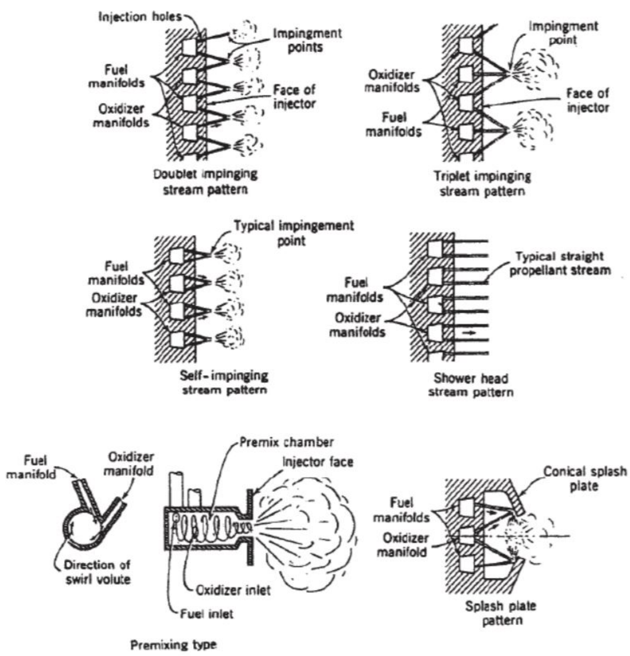

Figure 7.3: Different types of injector [2, Ch. 8]

in the contact region, and it may reduce the effectiveness of the mixture while blowing apart each stream, therefore high velocities are needed.

7.2.2 Like-impinging Elements

In this type of elements the process is analogous to the previous one but it will occur only downstream to the impinging point. In fact, we will have a first impinging between jets of oxidant (or fuel) and subsequently they will mix, in the form of spray, with the fuel (which is also atomized) or oxidant.
The uniformity of the mixture will be influenced by the dimensions of the element, their distance and the characteristics of the spray.
This type of element allows a good uniformity of the mixture and avoids problems of separation of reactive fluid (blowpart) when used with hypergolic propellants. However, when the system is designed to create impingement phenomena on the walls, can still occur blowpart.
As the unlike- element, there are different types of elements, the most used is the double impinging element, which guarantees a good mixture of uniformity and compatibility with the walls of the chamber. In order to cool down the combustion chamber's walls, elements which inject only fuel are arranged on the border of the injection plate, and the oxidizer elements slightly inboard.
These injectors are preferred for non-hypergolic propellants as this permits the mixing region to be further from the injector plate to mitigate overheating.

7.2.3 Non-impinging Elements

This injectors produce a stream which is normal to the injector plate, henceforth, relies on turbulence to achieve mixing.
The main element of this type are the showerhead and concentric tubes element.

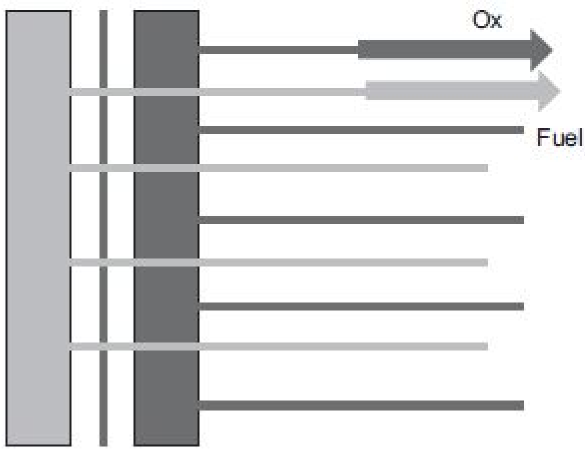

Figure 7.4: Schematic of showerhead injector [18, Ch. 8]

Showerhead elements have the simplest geometry that can be selected. They consist of a series of small duct parallel between themselves, even though, nowadays, they are no longer used. They may be used on the border of the injection plate for the cooling of the chamber's walls through the injection of fuel. Even though it was cheap to manufacture, it is no longer used because a large chamber volume was required to achieve a good combustion.

Figure 7.5: Showerhead injector [18, Ch. 8]

7.2.4 Hybrid Element

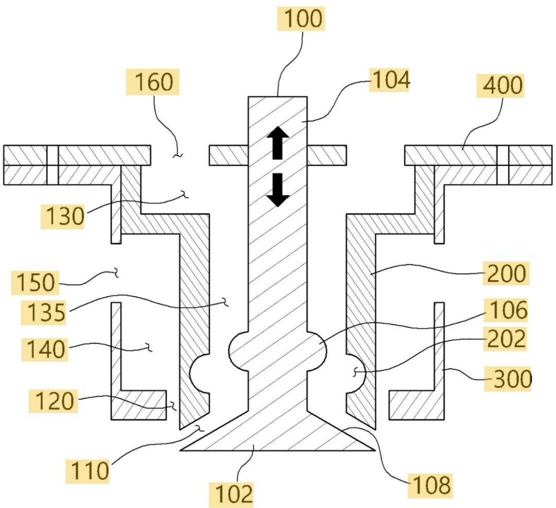

Figure 7.6: Pintle injector: Patent No. KR101710411B1 South Korea.

This kind of elements presents some peculiarity that do not fit with the categories described above, the main elements of this kind are the pintle and the splash plate injectors.

Pintle injectors is a type of coaxial injector. It consists of two concentric tubes and a central protrusion. First, one propellant flows through the inner tube, while the second one passes in the outer tube. Furthermore, the first propellant impinges at the end of the tube on a pintle-shaped protrusion which allow to create a broad cone or a flat sheet that intersect the outcoming stream from the outer tube. This type of elements can guarantee a greater throttling range respect to regular injectors (suitable for landers), high efficiencies (typically 96% – 99%) and will rarely present acoustic combustion instabilities. In fact, motors which presents pintle injection element has only a single central injector, this permits to create two recirculation zones inside the chamber that helps in the decrease of the acoustic instabilities. The principal problems of this injector system are an excessive thermal stress on the chamber walls.

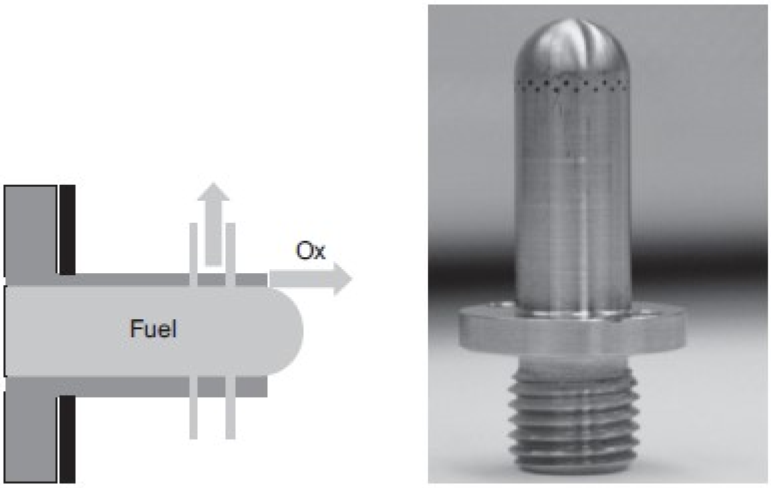

Figure 7.7: Pintle injector [18, Ch. 8]

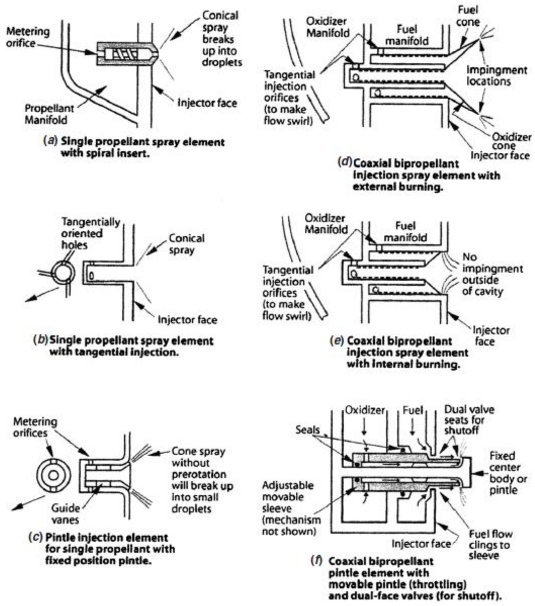

Figure 7.8: Simplified cross-section sketches of several injectors [2, Ch. 8]

7.3 Injector Flow Characteristics

The characteristic of an hydraulic injector can be evaluated accurately allowing us to design orifices with the desired values of injection pressure, injection speed, flows and mixture ratio.
For a given thrust F with an effective exhaust speed u_{eq}, the mass flow rate $\dot{m}$ is given by $\dot{m} = \frac{F}{u_{eq}}$. We can define the mass and volumetric flow rate for a given flux through an orifice:

$$
\begin{aligned}
Q &= C_d \; A \sqrt{\frac{2\Delta p}{\rho}} \\
\dot{m} &= C_d \; A \; \sqrt{2\rho\Delta p}
\end{aligned}
\tag{7.1}
$$

where C_d is the dimensionless discharge coefficient, ρ the propellant density, A the

area of the orifice and Δp is the pressure drop across the injector due to the loss of energy resulting from friction and turbulence.
The greater the pressure difference, the more flow you will get through the opening of a given size. Since we are designing for a specific flow rate, if we increase the pressure difference you want to decrease the orifice size to keep the flow rate the same. A value recommended for the pressure difference is of 4 to 10 atm as shown in Figure 7.9.
In Figure 7.9, a series of pressure drop against the characteristic of the injection orifice are shown. The graph shows that small differences in chamfer, hole entry radius or burrs at the edge of the hole can cause significant variations in the discharged coefficient and the jet flow patterns. This influences the atomization, the quality and distribution of the atomized small droplets, local mixture ratio and the local heat transfer rates. They can also cause burnout phenomena.

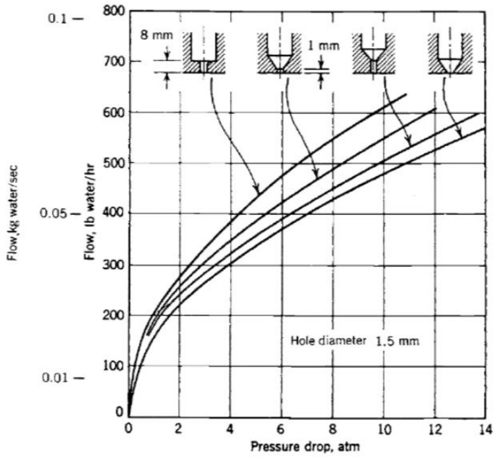

Figure 7.9: C_d for different orifices.

A fundamental aspect will be the determination of the mixture ratio, for a given pressure drop in the orifice:

$$r = \frac{\dot{m}_O}{\dot{m}_f} = \frac{Cd_O}{Cd_f}\ \frac{A_O}{A_f}\sqrt{\frac{\rho_O}{\rho_f}\frac{\Delta p_O}{\Delta p_f}} \tag{7.2}$$

The quantities in the equation have to be chosen so that the correct design mixture ratio is obtained. The actual mixture ratio can be obtained by experimental measurments through cold flow test that allows us to determine the optimal hole areas and discharge coefficient. A suitable fluid for this test is water. It will be feed alternatively both in fuel and oxidizer side, the mixture ratio obtained will be given by the equation:

$$r = \frac{Cd_O}{Cd_f}\ \frac{A_O}{A_f} \tag{7.3}$$

We notice that in order to obtain the equation written above we just need to multiply the square root of the density of the propellant times the pressure drops. Through the parameters yet obtained we can compute the injection velocity:

$$v = \frac{\dot{m}}{A} = C_d\sqrt{2\frac{\Delta p}{\rho}} \tag{7.4}$$

After the impingement of two jets ($f_l{-}f_l$, $f_l{-}O_x$, $O_x{-}O_x$) resultant momentum can be computed by the following equation based on the principle of conservation of momentum. So, if the momentum of the jet is the same before and after the impingement, we have:

$$tan(\delta) = \frac{\dot{m}_O\ v_O\ sin(\gamma_O) - \dot{m}_f\ v_f\ sin(\gamma_f)}{\dot{m}_O\ v_O\ cos(\gamma_O) + \dot{m}_f\ v_f\ cos(\gamma_f)} \tag{7.5}$$

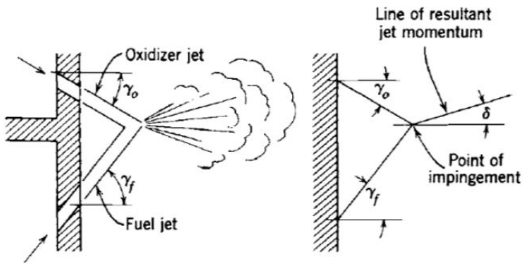

Figure 7.10: Jets impingement [2, Ch. 8]

where γ_O is the angle between the chamber axis and the oxidizer stream, γ_f is the angle between the chamber axis and the fuel stream while δ is the angle between the chamber axis and the average resultant stream. Often the optimal injection direction, is the one for which the flux is aligned with the axis of the chamber, so for $\delta = 0$, we have:

$$\dot{m}_O\ v_O\ sin(\gamma_O) = \dot{m}_f\ v_f\ sin(\gamma_f) \tag{7.6}$$

7.4 Orifice Diameter

The orifice dimension and the ratio between the diameter of the orifice and the fuel duct, strongly influence the quality of the atomization and of the mixture. Even though those characteristics principally depend on the type of element (injector) selected, in general we have that for every kind of element, it is possible to obtain a better performance through the use of small orifices.

Through experimental measurements, it has been shown that a lower limit for the diameter can be set, below $0.5mm$ no more significant advantages are shown. A smaller size of orifices allows the formation of smaller drops that guarantee a faster vaporization and consequently a better engine performance; moreover they allow us to obtain a more uniform mixture which is another desired effect.

Another important aspect that can be obtained by a proper design of the orifice is the minimization of the pressure drop due to the passage of the propellant through the duct. For the like-impinging elements the dimensions of the orifices will be the same for both the fuel and the oxidizer, so the ratio between the two diameters will no longer be a characteristic parameter of our design, but a small diameter of the orifices will still be important to guarantee a correct atomization.

7.5 Orifice Geometry

In the design of an injection system a crucial aspect will be the study of the orifice's geometry, in fact design or production errors can cause problems to the proper behaviour of the engine or even the failure of the mission. The main issues that can occur due to a wrong design are hydraulic vibrations, for example due to roughness, a small bore length-diameter ratio or to an excessive pressure drop.

The structure of the orifice can be divided in three main parts: orifice inlet, orifice bore and orifice outlet.
The different types of orifice inlet are classified as: rounded, sharp-edged or chamfered. The main problems due to the inlet are pressure drop phenomena or the separation of the fluxes. Rounded inlets prevent cavitation and provide a better flow direction but produce an higher C_d, with sharpedged orifices we reduce this problem but they present more production difficulties due to the presence of more burrs.
The study of the orifice outlet is also important, a wrong design can cause unsuitable directions of the injected flows.

7.6 Impingement Angle

The impingement angle influences several factors that characterize the mixture, for example: uniformity of the mixture, backsplash of the propellant or the atomization characteristic. The back splash of the propellant can cause the so called phenomenon of the injection-face burnout. However, the specific importance of this design parameter varies for each type of element.
For example, for an unlike impingement element as the impingement angle increase we have an increase of the back flow phenomena of the mixed fluid. It is possible to observe that we have a direct correlation between this effect and the semi-angle of impingement. For angles higher then 90° it is possible to observe an high heat flux directed to the face of the injection plate.
It has also effects on the mixture ratio and on the uniformity of the obtained mixture, in fact they increase for lower impingement angles. For unlike-impingement double element we obtain an optimal mixture for angles included between 40° and 80°, it is also present a correlation between the impingement angle and the dimension of the produced droplets.

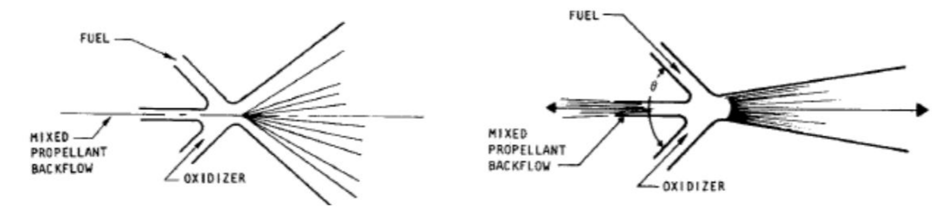

Figure 7.11: impingement angle and backflow

If we consider like-impinging elements we do not have the backflow problem, indeed in this type of element the two propellant aren't mixed until the two sprays collide, so contrary to the previous case on the injection plate face we have the impact of a non reactant fluid. The like-impinging double element has two impingement angles: primary angle (between Ox and Ox or between fuel and fuel) and a cant angle, which is the angle included between the chamber axes and the fluid jets generated by the element. Through experimental measurements, it is possible to notice that the dimension of the produced droplets decreases with the increase of the impingement angle. Instead an increase of the cant angle allows a better mixing.
For non-impingement element we are unable to speak about impingement angle, however it can be defined an equivalent parameter, such as the inner-post chamfer angle for concentric injectors and the deflection angle for pintle-injectors. The latter will be really important, because an excessive value can cause the impingement of the oxidant on the chamber walls, causing elevated heat flux on the injection plate face.

7.7 Impingement Distance

The impingement distance, or free stream jet length, is the distance that must be covered by the jet from the outlet of the orifice to the impingement point, measured along the jet axes.

For all type of elements a wide impingement distance cause the impingement of jets that are yet degraded, this effect added to the problems due to the production tolerances, can cause the misimpingement of the jet. Other problems connected with the impingement distance are for example the corrosion of the injection plate (due to too short impingement distances or for excessive values of the ratio $\frac{L_f}{L_0}$) or a lower quality of the mixture caused by the misimpingement, or by a rotation of the generated spray with respect to the optimal condition. The concentric tube injectors do not present an impingement distance, however an equivalent parameter can be the recess depth of the inner tube. Recess of the inner tube results in an increase of the atomization and mixture quality.

7.8 Element Arrangement

Element arrangement on the injection plate, depending on the type of element selected, must guarantee the uniformity of the mixture and optimal mass distribution. The only type of element not influenced by this parameter will be the symmetric ones.

It will be necessary to make specific studies during the design process to find out the optimal distance for each kind of element. An example could be the opposed impingement doublet, which allows us to obtain a secondary mixture thanks to a specific element arrangement.

It is possible to have some problems with the elements that present a uniform mixture ratio distribution both close and far away from the chamber's walls. Moreover, radial and transverse wind produced by gross mass and mixture ratio maldistributions can cause severe overheating and erosion of the injectors. A possible solution is to alter injection element arrangement to achieve a better distribution of mass flow rate. In some cases non uniform mass distribution are intentionally created, for example in the outer injectors to reduce the heat flux to the chamber's walls.

Another important parameter to take into account is the orientation of the single element, this parameter can influence the heat flux transfer to the baffles. A possible solution can be the use of concentric tube systems, which remove the problem of the orientation of the elements. A uniform spacing usually guarantee a lower heat flux to the injection plate and so is preferable.

7.9 Atomization Process

As explained earlier, the aim of the combustion is that of generating thrust. The different zones (Figure 7.12) the flow goes through can be divided into:

- **Injection-atomization zone**. In this zone, the reactants mix together with the help of injectors. The injectors try to generate favorable conditions for reactants atomization[1] and for reactants mixing. It is favorable to have a fast and reliable mixing in order to reduce the combustion chamber length and weight. At the same time it is strictly necessary to have a stable flow mixture and to assure low temperature and mechanical stresses at the combustion chamber wall.

[1] Note: only liquid reactants need to be atomized.

After atomizing the reactants, droplets of fluid need to be heated in order to evaporate and burn. Because of the high temperature of the combustion chamber, this process is quite fast but it is mainly related to the droplet size. So aiming for good atomization brings to faster evaporation of droplets and a more complete combustion.

- **Rapid combustion zone**. This process is very fast because the mixture is in the ideal conditions to be burnt. It is important to note that there is heat flux towards the injection plate, this is because part of the heat of combustion is necessary to heat up the cool droplets. It is favorable to have a good mixing in order to avoid diffusion phenomena in this zone (from too rich mixture points to too lean mixture points in the combustion chamber), but this is only achievable by a good design of the injection plate and atomization process.

- **Stream tube combustion zone**. The pressure gradient generated by the combustion starts to displace the mixture downward[2]. After a turbulent flow from the previous zone, the flow switches to an almost laminar-like flow. This is achievable only because of the huge pressure gradient due to combustion. Changes still exist in the mixture composition (shifting equilibrium) but these are of the second order with respect to the main combustion process.

- **Supersonic expansion zone**. Due to thermodynamics, pressure and temperature drop down in order to accelerate the fluid. Because of the supersonic velocity of the mixture, it is possible to say that the chemical kinetics are much slower than the fluid permanence inside the diverging part. These two considerations allow to say that it is possible to consider the composition of the mixture in the diverging part of the nozzle as fixed (frozen equilibrium). However, this zone has its problems such as: flow instabilities (overexpanded configuration), boundary layer control, creation of condensate species, etc.

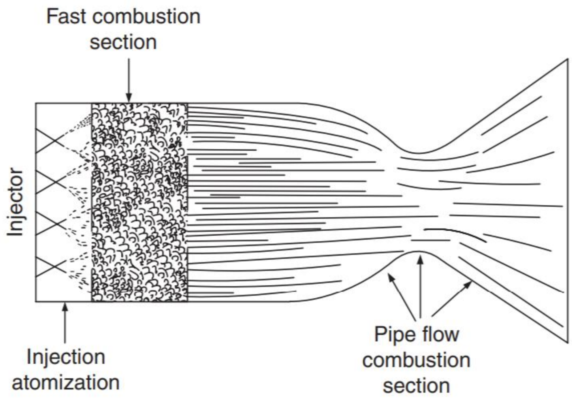

Figure 7.12: Combustion sections [19, Ch. 1].

7.9.1 Droplet Analysis

Droplet behaviour analysis is a really difficult aspect of LRE combustion mechanism. Although it is possible to have good atomization of the reactants inside the combustion

[2]It is important to keep in mind that, in most of the cases, the combustion is incomplete due to not stoichiometric $\frac{O}{F}$, dissociation phenomena, chemical kinetics and atomization limitations.

chamber, it is necessary to study how droplets behave in different system conditions and how they interact with the external aerodynamic forces (essentially the interaction between droplets and combustion products gases). It is also important to understand the forces that dictate droplets behaviour.
It is possible to see the atomization as the process where surface tension action is disrupted by the internal liquid force and/or external forces. Good atomization of reactants results in smaller droplets size; this allows:

- **High surface over volume ratio** $\left(\frac{S}{V}\right)$. This ratio can be computed in different ways (also using probability and statistics) but one of the most used parameters is the Sauter mean diameter (SMD)[3]. A high $\frac{S}{V}$ ratio can be related to a faster evaporation of liquid, a fast combustion of mixture and, if the mixture is well distributed inside the system (due to a good engineering design of the injection plate and injection process), a homogeneous mixture inside the system. A good particle diameter is in the range $25 - 500\ \mu m$.
- **Faster mixing and combustion**. Due to evaporation speed and mixing time inside the system, small droplets allow faster combustion.

Forces

The forces acting on the droplet can be divided into:

- **Aerodynamic forces**. These forces are encountered when the jet flow is modified by pressure gradients and/or interactions with other gaseous fluids (such as hot mixture, due to a partial backflow of the combustion products).
- **Surface tension**. This is the force that leads the liquid to form a spherical shape (minimization of surface energy) and to remain separated from the gaseous environment.
- **Viscous forces**. Stresses generated from differences in velocity between the jet and the surrounding gas or through multiple jets interaction.

Although it is convenient, in terms of performace, to have a fast atomization of reactants, it is not always wanted to have combustion too close to the injection plate. This will result in the design of the injector to slightly *delaying* the atomization, mixing and evaporation of droplets process in order to move the high temperature gradients away from the injectors' nozzle. At the same time it is worth considering the huge stress the injectors go through. This highlights, overall, the importance of finding a balanced configuration in order to maintain performance and the reliability of the engine for all possible working conditions [20, Ch. 6].

7.9.2 Injectors and Atomization

As noted earlier, different types of injectors exist. Their task is the same but they work in a totally different ways. The main type in use are: straight injectors and centrifugal injectors. The reason they are used is good performance and years of experience.
They can be subdivided into two categories based on the velocity of the reactants [21, Ch. 5]: pressure-atomizing and air-assisted atomizers, Figure 7.13.

[3] $SMD = \frac{\sum_i N_i D_i^2}{\sum_i N_i D_i^3}$, where N_i refers to the number of particles with a defined diameter D_i.

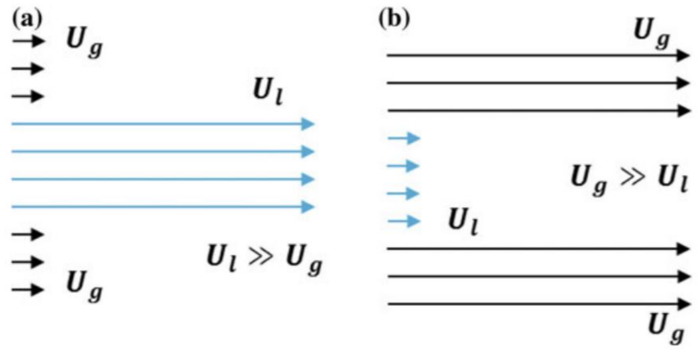

Figure 7.13: Typical gas- and liquid-phase velocity profile in pressure-atomizing and air-assist atomizers [21, Ch. 5].

Here the keypoint is to convert pressure energy into kinetic energy and/or using fluid interaction in order to atomize the fluid. Having increased the velocity of the fluid at the nozzle, viscosity plays a huge role because the presence of shear layers.
No matter what type of injector is used, it is possible to subdivide every atomization process in 2 parts [19, Ch. 2]:

- **1st breakup**. The jet or liquid sheet starts to generate moderate dimensions droplets. The size of droplets, however, is not enough to have a fast and performant combustion.
- **2nd breakup**. It consists pf a complicated interaction of the medium size droplets with the environment. The result of this interaction is the bursting of the droplet and the generation of smaller droplets. After this interaction, the tiny droplets go through evaporation and combustion [4].

In the following subsections it is described the breakup procedure for a centrifugal injector. The choice of describing this kind of injector is because it is one of the most used injector in the liquid propellant engines[5] and it can be seen quite easily the influence of working condition variation on droplets diameter and atomization performance in general [22]. It is worth keeping in mind that with this type of injector we are considering conical liquid sheet behaviour (and not about liquid jets).

1st Breakup

As with all the injectors' behaviour, the pattern the liquid shows after being expelled from the injector nozzle depends *also* on the pressure drop inside the injector (due mainly to acceleration mechanisms). For a swirler injector the pressure drop inside the injector can generate 4 different patterns (Figure 7.14).

- **Stage 1**. Due to low pressure drop, the liquid doesn't present any effective swirl motion.
- **Stage 2**. The kinetic energy is higher but it is not enough to counteract the surface tension action of the liquid. The flow will show a bulb shape configuration.

[4] Each of the two breakup processes face evaporation and combustion, but the intensity is really small compared with the situation after the 2nd breakup.

[5] Copenhagen suborbitals BPM5, P&W RL10-A-3 (although all the previous versions were made with straight injectors), etc.

- **Stage 3**. This time the kinetic energy is high enough to overcome the surface tension action. The droplet formation is the result of perturbation propagations inside the liquid sheet. The problem of this configuration is the medium-large dimension of the formed droplets.

- **Stage 4**. In this case, the liquid sheet is extremelly unstable. The result is a really fast breakup.

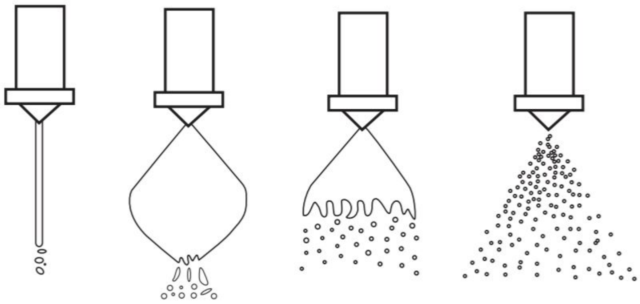

Figure 7.14: Flow patterns at different pressure drops [19, Ch. 2].

It is possible to divide also the atomization process for a swirl injector in different steps, Figure 7.15:

- **Complete liquid region**. In this region instabilities are growing up but there isn't any kind of sheet breakup. The liquid sheet expands.

- **Liquid lump region**. Instable phenomena explode and the first droplets are created.

- **Liquid filament and dense atomization region**. These droplets undergo mechanical stresses that deform the particles.

- **Sparse atomization region**. From the elongated filaments there is an internal burst of the droplet that gererates smaller particles[6].

2nd Breakup

The 2nd breakup process is a very complex subject to study. Although the flow inside the system is in a turbulent state, it is worth studying the breakup process of the droplet in laminal flow conditions.
Lane [23] suggested that, in a laminar flow, the behaviour of a droplet is different if it accelerates or if it suddenly enters inside a high speed airflow. In rocket propulsion the liquid droplet accelerates in the combustion chamber (herein this condition is studied).

This case can be subdivided in many steps: in every of these steps there is the deformation of the droplet shape and the changing of the airflow around the deformed droplet. At the same time, it is necessary to be able to understand the limit where the droplet can remain in one piece (essentially, being able to model in a proper way the surface tension forces that are related to droplet geometry).
When the droplet is accelerated, it starts to flatten, Figure 7.16. Once the thickness of

[6] After this step 2nd breakup mechanism starts.

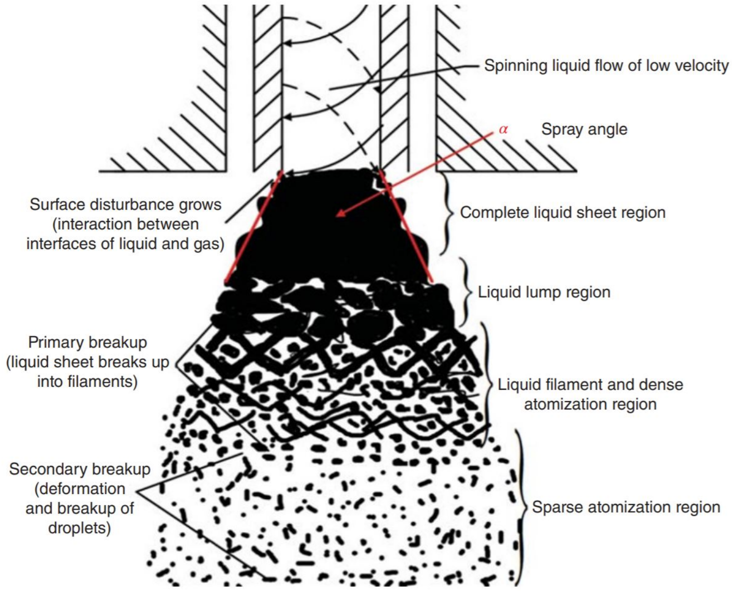

Figure 7.15: Atomization process in a swirler injector [19, Ch. 2].

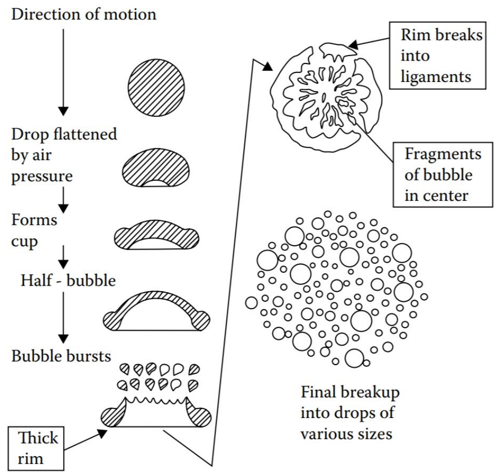

Figure 7.16: Liquid droplet breakup scheme in laminar flow conditions [19, Ch. 2].

the central part of the bowl shaped droplet is small enough, the droplet bursts and there is the creation of many smaller droplets (from the exploded central part). Although this explosion generates a lot of small dimension droplets, the original droplet makes around $60-70\%$ of the total volume. At the end of the first explosion, around the $30-40\%$ of the original droplet is disintegrated in smaller droplets. The remaining part of the original droplet has a ring-like shape and it goes under a second explosion where there are formed many large droplets. These large droplets go through new explision cycles until there is the total evaporation of the propellant.
For a droplet placed inside a fluid in motion there is a sort of generation of string-like shapes of the droplets: in some sense, the droplet tends to have the same shape/orientation as the streamlines.

Pressure Influence on Atomization

It is interesting to understand the position of liquid sheet breakup with respect to the combustion chamber pressure and injector pressure drop.

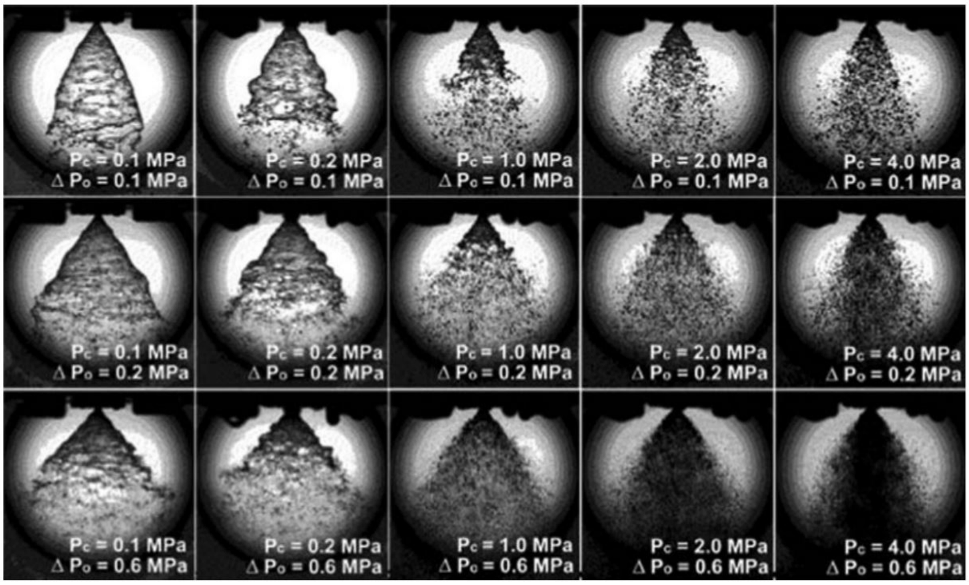

Figure 7.17: Effect of combustion chamber pressure and injector pressure drop [22, KSLV-I swirl injector test].

It is important to bear in mind that in the atomization process it is necessary to avoid injector plate damage, high thermal stresses and to generate a compatible spray geometry with respect to the combustion chamber geometry. As Figure 7.17 shows, the greater the pressure drop, the greater the instabilities and so really fast atomization. In addition, the greater the chamber pressure the closer the atomization to the injection plate. As this suggests, it is necessary to find the best way to satisfy mechanical and combustion needs. Engineers have achieved the best performance using a pressure drop around $20 - 30\%$ of chamber pressure.

7.10 Combustion Instabilities

Combustion instabilities are one of the main problems that affect liquid rockets. They are mainly caused by an oscillatory behaviour due to the coupling of fluid dynamics and combustion phenomena. When these fluctuations interact with the natural frequencies of the engine, periodic superimposed oscillations occur. It has been studied that these pressure fluctuations do not exceed 5% of the mean chamber pressure [2, Ch. 9].
Combustion instabilities can cause excessive pressure vibration forces which can exceed the mechanical properties of various engine parts or even, excessive heat transfer. The aim to study combustions instabilities is to prevent them and to keep a reliable operation of the engine.
Instabilities found in combustion create large concentrations of vibratory energy occurring in a fairly well-defined intervals with a pressure peak that might be mantained, increased or decreased.
Four instabilities can be studied:

- Chugging instability: from 10 Hz to 40 Hz.

- Buzzing instability: from 400 Hz to 1000 Hz.
- Screaming instability: at frequencies higher than 1000 Hz.

7.10.1 Chugging

This instability arises mainly from the elastic nature of feed systems and structure of the vehicle and it may appear at low chamber pressure engines (100-500 psia), due to propellant pump cavitation, tank pressurization control fluctuations and vibration of long engine components such as feed lines.
There is an amplification of the oscillations when the structure and feed system have the same natural frequency producing a coupling of the forces with a usual frequency of 10 to 50 Hz which produces a low-frequency longitudinal motion of vibration, hence, this instability sometimes can be referred as pogo instability.

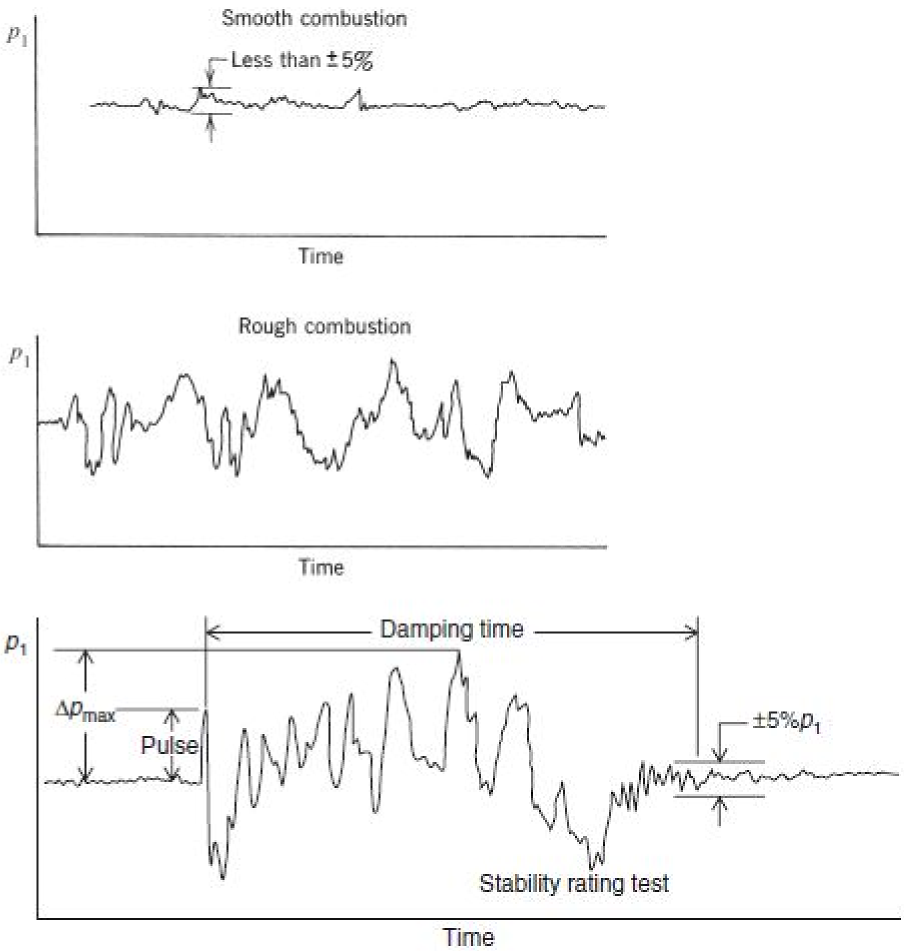

Figure 7.18: Oscillograph traces of chamber pressure [2, Ch. 9].

In Figure 7.18 the fluctuations of the pressure oscillations in time focusing on the increment of each perturbation with respect to the nominal pressure which is around a 5%. We can also compare the oscillograph or a smooth and a rough combustion which is crearly visible due to its high peaks presented.

7.10.2 Buzzing

Buzzing instabilities has pressure perturbation above the 5% of the mean pressure in the combustion chamber, henceforth, it does not have large vibratory energy but is more noisy and annoying than damaging [2, Ch.9]. It is also a characteristic of the coupling between the combustion and flow in a portion of the feed system.
The buzzing instability is more frequently presented in medium-size engines which produces from 2000 to 250,000 N of thrust.

7.10.3 Screeching

This instability which is common in new engines developed is also called screaming due to the fact that has high frequency, thus, high energy which at the end is the most destructive kind of instability willing to destroy an engine in less that 1 second. Inside the combustion chamber the boundary layer length seems to reduce significantly causing the rate of heat transfer to increase by an order of magnitude causing the metal to melt. This increase in heat transfer is similar as with detonation.
It has been studied that high frequency instabilities produce a longitudinal and a transverse mode, both decoupled from each other.
Firsty, the longitudinal mode propagates only along axial planes of combustion chamber and pressure waves are reflected at the injector face and the converging nozzle cone [2, Ch.9].
Contrary to longitudinal mode, pressure oscillations of transverse mode propagate along the perpendicular planes to the chamber axis and can be broken down into tangential and raidal modes. This mode in particular is presented in large liquid rockets near the injector.

7.10.4 More about Instabilities

At steady-state conditions of hypergolic propellants, a random high-amplitude pressure disturbance may occur called popping which is similar to a detonation wave with a rise of the pressure of just a few microseconds. The pressure ratio across the wave can reach a magnitude of 7:1. A good design of the injector is desired to eliminate this kind of high-amplitude perturbation.
Additionally, cavitation phenomena in the turbomachinery inside the feed system may induced some instabilities.
Semiempirical techniques exist for artificially disturbing combustion process or introducing shock waves in a thrust chamber during testing of instabilities:

- Nondirectional bombs (common)
- Oriented explosive pulses (common)
- Directed flows of inert gas
- Momentary operation while changing the mixture ratio
- Introduction of slugs of inert gas
- Hard starting of engine by introducing a quantity of unreacted propellant.

With all this techniques, what is measured is the time interval that takes to the motor to return, or recover from the instability to steady state.

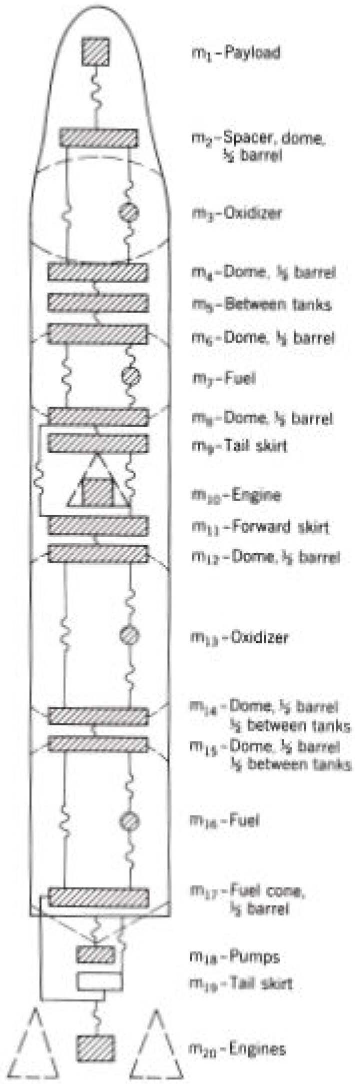

Figure 7.19: Typical spring-mass model used in analysis of pogo instabilities [2, Ch. 9].

7.10.5 Control of Instabilities

In general, the area in charge to control instabilities relay on a-priori acquired experience with similar engines. Analytical tools are, nowadays, available to evaluate the combustion process; even though, most of tests have to be done on a full-scale engine.
Techniques for damping pogo instability include the use of energy absorption devices in fluid lines, perforated tank liners, special tank supports and properly designed engine structures to avoid resonance in any case where components in feed systems are free to oscillate.
In general, the designer can change many injector details such as injector hole pattern, injector size and/or injection pressure. However, minimal changes in injector geometry can increase or reduce instabilities
Alternatively, injector face baffles (deflectors) are introduced as a way to acoustically (frequencies below 4000 Hz) damp the combustion chamber minimizing the coupling and amplification of gas dynamics forces within the chamber. These were widely accepted in the 1960s to prevent high-frequency instability.

According to Sutton [2, Ch.9], the principal damping source affecting propagation of transverse oscillations is the combustion itself due to the huge volumetric change and

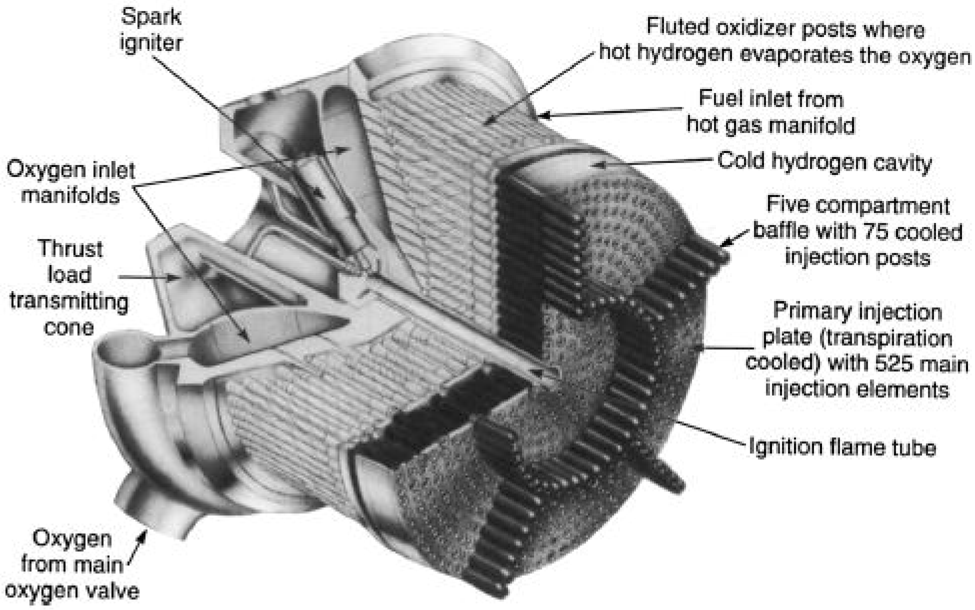

Figure 7.20: Space Shuttle Main Engine where baffles are visible to damp high frequency perturbations [2, Ch. 9].

transfer of momentum where they take some energy from high instantaneous local pressures.
Some acoustical absorbers are applied as discrete cavities in the wall of combustion chamber near the injector end to remove energy from the vibratory system.
Back in the space race, when the F-1 engine for Saturn V was being developed, some oscillations of 4 kHz were observed, hence, the designers developed a diagnostic technique which consisted in the implementation of small bombs outside the combustion chamber just to induce variations in pressure when the engine was firing in order to determine how to eliminate these oscillations. To sum up, there is already a lot of information and

Figure 7.21: Test firing of the F-1 engine. [24]

data to establish key design parameters and estimate the possible resonances of engine.

In addition, it is a must-do to design the structure to avoid the coupling of each of the forces and to create a robust feed system which will be able to provide a good mixing and dispersion of the reactives while the systems is disturbed.

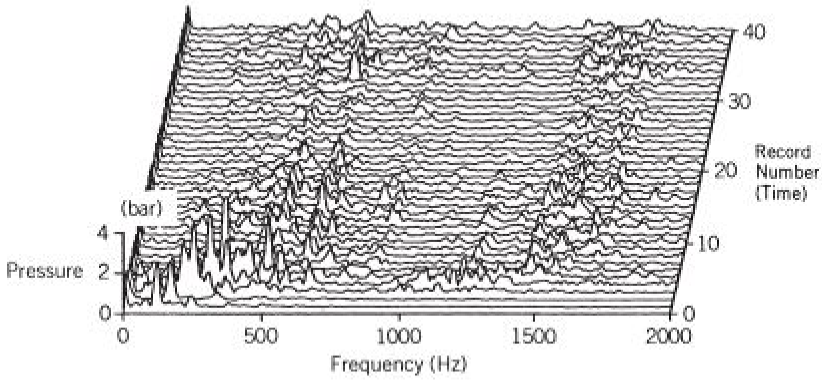

Figure 7.22: 40 superimposed frequency amplitude diagrams taken 0.2 s apart during start phase of the Vulcan HM 60 thrust chamber operating at 109 bar with an O/F of 6.6 [2, Ch. 9].

Bibliography

[1] Pablo Rachov. *Electric Feed Systems for Liquid Propellant Rocket Engines*. 2010.

[2] George P Sutton and Oscar Biblarz. *Rocket propulsion elements*. John Wiley & Sons, 2016.

[3] *Gas-generator cycle*. Apr. 2021. URL: https://en.wikipedia.org/wiki/Gas-generator_cycle.

[4] *Expander cycle*. June 2021. URL: https://en.wikipedia.org/wiki/Expander_cycle.

[5] *Staged combustion cycle*. June 2021. URL: https://en.wikipedia.org/wiki/Staged_combustion_cycle.

[6] Stefan Schlechtriem and Armin Herbertz. *Triebwerkszyklen*. Oct. 2019.

[7] *Combustion tap-off cycle*. May 2020. URL: https://en.wikipedia.org/wiki/Combustion_tap-off_cycle.

[8] Eduardus Josephus Vermeulen, Bernd Mathias Jozef Brauers, and Wilhelmina Helena Maria Welland-Veltmans. *Igniter for a rocket engine, method for ignition of a rocket engine*. US Patent 8,839,600. Sept. 2014.

[9] Ann Grimm Bob Cannon Mary Roberts and Kathleen Greer. “Igniters and Their Use”. In: (2012). URL: https://studylib.net/doc/18765556/igniters-and-their-use.

[10] Robert W Marable. *Ignitor for liquid fuel rocket engines*. Oct. 2002.

[11] Bill Greene. *Inside the LEO Doghouse: Light My Fire!* 2014. URL: https://blogs.nasa.gov/J2X/tag/ignition/.

[12] Thomas Matthew E., Bossard John A., Early Jim, Trinh Huu, Dennis Jay, Turner James. *Laser Ignition Technology for Bi-Propellant Rocket Engine Applications*. 2001. URL: https://ntrs.nasa.gov/citations/20020051003.

[13] Thomas Matthew E., Bossard John A., Early Jim, Trinh Huu, Dennis Jay, Turner James. *Laser Ignition Technology for Bi-Propellant Rocket Engine Applications*. Ed. by NASA. 2003. URL: https://ntrs.nasa.gov/citations/20020022186.

[14] Martin Adam, Eskridge Richard. *Plasma Igniter for Reliable Ignition of Combustion in Rocket Engines*. Ed. by NASA. 2011. URL: https://ntrs.nasa.gov/citations/20110002983.

[15] Wang Si-Bo, Yu Jin-Lu, Ye Jing-Feng, Li Guo-Hua, Chen Zhao, Jiang Lu-Yun, Gu Chen-Li. *Ignition characteristics of pre-combustion plasma jet igniter*. Ed. by Project supported by the National Natural Science Foundation of China. 2019. URL: http://cpb.iphy.ac.cn/article/2019/2012/cpb_28_11_114702.html.

[16] Wang Si-Bo, Yu Jin-Lu, Ye Jing-Feng, Li Guo-Hua, Chen Zhao, Jiang Lu-Yun, Gu Chen-Li. *Ignition characteristics of pre-combustion plasma jet igniter*. Ed. by Project supported by the National Natural Science Foundation of China. 2019. URL: http://cpb.iphy.ac.cn/article/2019/2012/cpb_28_11_114702.html.

[17] Davidson Center. *dsc80292*. URL: http://heroicrelics.org/ussrc/engines-f-1-dcse/dsc80292.jpg.html. (accessed: 14.05.2021).

[18] Stephen D Heister et al. *Rocket Propulsion*. Cambridge University Press, 2019.

[19] Zhen-Guo Wang. *Internal combustion processes of liquid rocket engines: modeling and numerical simulations*. John Wiley & Sons, 2016.

[20] Sung-Piau Lin. *Breakup of liquid sheets and jets*. Cambridge university press, 2003.

[21] Saptarshi Basu et al. *Droplets and Sprays: Applications for Combustion and Propulsion*. Springer, 2017.

[22] Youngbin Yoon and In-Seuk Jeung. *Effects of Ambient Gas Pressure on the Breakup of Sprays in Like-Doublet and Swirl Coaxial Injectors*. Tech. rep. SEOUL NATIONAL UNIV (REPUBLIC OF KOREA), 2004.

[23] WR Lane. “Shatter of drops in streams of air”. In: *Industrial & Engineering Chemistry* (1951).

[24] NASA Marshall Space Flight Center. *F-1 Engine Test Firing*. 1960. URL: https://commons.wikimedia.org/wiki/File:F-1_Engine_Test_Firing.jpg (visited on 05/14/2021).