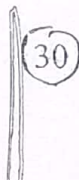


## LIQUID PROPELLANT ROCKET ENGINES

Rocket engines in which liquid fuels and oxidizers are used are known as liquid propellant rocket engines. In 1926, Robert Goddard tested the first liquid Propellant rocket engine. He used gasoline and liquid oxygen as propellant. The propellants are fed into the thrust chamber (combustor or combustion chamber) from their containers for combustion or chemical reaction. For better mixing and efficient combustion the fuel and oxidizer are atomized through the injectors. The feed system regulates the optimum mixture ratio for a given set of propellants.

A liquid propellant rocket engine consists of one or more thrust chambers, one or more tanks to store the propellants, a feed mechanism to force the propellant into the thrust chamber, a power source to furnish the energy required by feed mechanism.



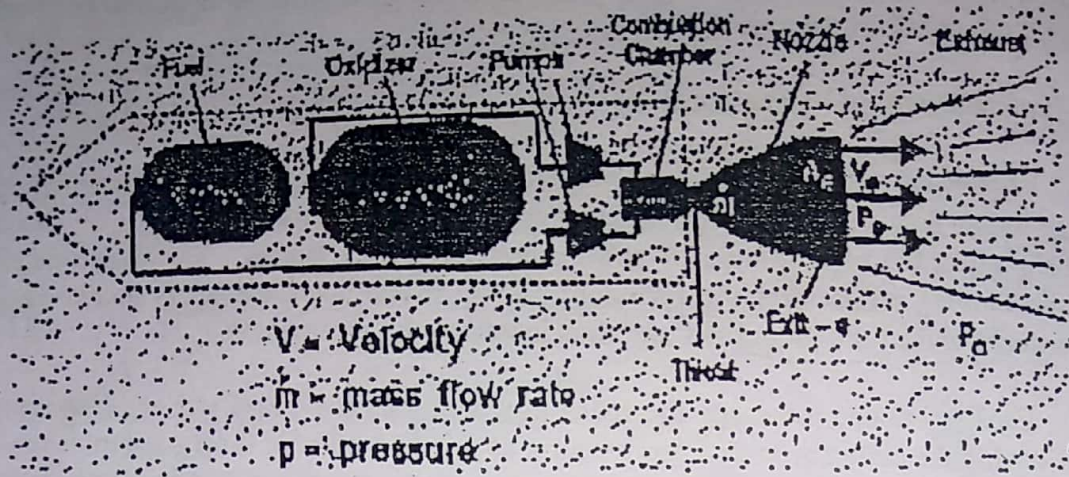


Fig 2. A Liquid Propellant Rocket Engine

In a liquid propellant rocket, the fuel and oxidizer are stored in separate tanks, and they are fed through the combustion chamber and pumps. In combustion chamber the fuel and oxidizer are mixed and burned, then they release the thermo chemical energy, this develops high temperature and high pressure in the combustion chamber. Then the highly heated products are expanded in the nozzle creating the thrust.

#### Liquid propellants:

The term liquid propellant embraces all the various used and may be one of the following

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

#### Monopropellants:

A liquid propellant, which contains both fuel and oxidizer in a single chemical, is known as "monopropellant"; this is stable at normal ambient conditions and liberates thermo-chemical energy on heating or igniting. A monopropellant should decompose easily and rapidly on injection and ignition into the combustion chamber. The number of chemicals, which satisfy the aforementioned properties, is small. Hydrogen peroxide ( $H_2O_2$ ), hydrazine ( $N_2H_4$ ), nitroglycerine [ $C_3H_5(O NO_2)_3$ ], nitro methane ( $CH_3NO_2$ ) are some of the monopropellants used for rocket propulsion.

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Monopropellants, which decompose in the presence of solid or liquid catalyst, is called Katergols. E.g. Hydrogen peroxide ( $H_2O_2$ ) in the presence of Silva, Hydrazine in the presence of Iridium.

Monopropellants, which decompose when ignited, are called Monergols. E.g. Nitroglycerine, Picric acid, Nitro methane.

**Bipropellants:**

A liquid propellant, which contains the fuel and oxidizer in separate units, is known as bipropellants. I.e. A bipropellant rocket unit has two separate liquid propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of liquid propellant rockets have been manufactured for bipropellant applications. Some of them are in the following table.

S.No	Fuel	Oxidizer
1.	Hydrazine LH <sub>2</sub> Gasoline UDMH	Liquid Oxygen (LOX)
2.	Ethanol Hydrazine	Hydrogen Peroxide ( $H_2O_2$ )
3.	Hydrogen UDMH Hydrazine	Nitrogen tetroxide ( $N_2O_4$ )

**Cryogenic Propellants:**

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

Because of the low temperatures of cryogenic propellants, they are difficult to store over long periods of time. For this reason, they are less desirable for use in military rockets, which must be kept, launch ready for months at a time. Also, liquid hydrogen has a very low density (0.59 pounds per gallon) and, therefore, requires a storage volume many times greater than other fuels. Despite these drawbacks, the high efficiency of liquid hydrogen/liquid oxygen makes these problems worth coping with when reaction time and storability are not too critical. Liquid hydrogen delivers a specific impulse about 40% higher than other rocket fuels.

Liquid hydrogen and liquid oxygen are used as the propellant in the high efficiency main engines of the space shuttle. LH<sub>2</sub>/LO<sub>2</sub> also powered the upper stages of

the Saturn V and Saturn IB rockets as well as the second stage of the Atlas/Centaur launch vehicle, the United States' first LH<sub>2</sub>/LO<sub>2</sub> rocket (1962)

### Hypergolic propellants:

Hypergolic propellants are fuels and oxidizers, which ignite spontaneously on contact with each other and require no ignition source. The easy start and restart capability of hypergolics make them ideal for spacecraft maneuvering systems. Also, since hypergolics remain liquid at normal temperatures, they do not pose the storage problems of cryogenic propellants. Hypergolics are highly toxic and must be handled with extreme care.

Hypergolic fuels commonly include hydrazine, monomethyl hydrazine (MMH) and unsymmetrical dimethyl hydrazine (UDMH). The oxidizer is typically nitrogen tetroxide (N<sub>2</sub>O<sub>4</sub>) or nitric acid (HNO<sub>3</sub>). UDMH is used in many Russian, European, and Chinese rockets while MMH is used in the orbital maneuvering system (OMS) and reaction control system (RCS) of the Space Shuttle orbiter. The Titan family of launch vehicles and the second stage of the Delta use a fuel called Aerozine 50, a mixture of 50% UDMH and 50% hydrazine.

### Cold gas propellants:

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

### Storable propellants:

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

### Gelled propellants:

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

## Oxidizers

### 1. Liquid oxygen (LOX)

This is a cryogenic propellant (Boiling point = 90 K at atmospheric pressure); therefore, it requires proper insulation of containers and feed lines to prevent evaporation. It is non-corrosive and nonoxidant, but harmful to personnel. It can explode under impact

when mixed with oils and organic substances. Its specific gravity is 1.14, molecular weight 32.0 and vapour pressure 51.7 mbar (at 88.8 K). It is widely used as oxidizer and burns with a bright, white yellow flame with most hydrocarbon fuels. It has been used in combination with alcohols, jet fuels, gasoline and hydrogen.

## 2. Liquid Fluorine

In combination with most fuels, liquid fluorine affords higher values of performance and high specific gravity (1.5). But it is toxic, corrosive and reactive. It has been used in experimental rocket engines, but not in production engines.

## 3. Hydrogen peroxide

It is highly reactive requiring special materials and handling systems. Besides some catalysts such as manganese dioxide, calcium permanganate and potassium permanganate many impurities in the containers or feed system act as catalysts decomposing it into steam ( $H_2O$ ) and oxygen in an exothermic reaction. Highly concentrated solution of hydrogen peroxide is used in rocket engines; this is harmful to the skin of the personnel and suffers from hazards of fire and explosion.

It boiling point is 423.5 K, vapor pressure 17.23 mbar (327.78 K) and molecular weight 34. It was used in rocket applications between 1938 and 1965, but is no longer used today, primarily because of its storage stability problems.

## 4. Nitrogen tetroxide

It is a storable propellant (B.P = 294.4 K) used in combination with a number of rocket fuels such as hydrogen, unsymmetrical dimethyl hydrazine etc. Its molecular weight is 92.0.

It is a high density yellow brown liquid. Its specific gravity is 1.44. It is hypergolic with many fuels. The fumes are reddish-brown and are extremely toxic. Its liquid temperature is narrow and it is easily frozen or vaporized. The extremely low temperature makes the problem of choosing suitable tank and piping materials difficult, because many metals lose their strength at low temperatures.

## 5. Nitric acid

RFNA is the most common type. RFNA consists of concentrated nitric acid that contains 5-20% dissolved nitrogen dioxide. The red fumes are exceeding annoying and poisonous. Two important types among a large variety of nitric acids used as oxidizers in rockets are the red fuming nitric acid (RFNA) and white fuming nitric acid (WFNA); they are used a large number of rocket fuels. Nitric acid is preferred in many applications on account of its high specific gravity. It is highly reactive, corrosive and poisonous causing storage and handling problems.

## 6. MMH (Monomethylhydrazine)

Its chemical formula is  $\text{CH}_3\text{NHNH}_2$ . MMH has been used extensively as a fuel in rocket engines, particularly in small attitude control engines, usually with  $\text{N}_2\text{O}_4$  as oxidizer. MMH is the most toxic among hydrazines.

## Fuels

Some fuels are described below:

### 1. Hydrocarbon Fuels

Petroleum derivatives encompass a variety of different hydrocarbon chemicals such as gasoline, kerosene, diesel oil and turbojet fuel. RP1 is a specially refined petroleum product particularly suitable as a rocket fuel.

### 2. Liquid hydrogen

This is a cryogenic propellant (B.P. = 20 K). Its low specific gravity (0.07) and extremely low temperatures present several technological problems. It requires larger and well-insulated storage tanks. Besides, this low temperature creates material and strength problems. It's mixture with air due to leakage can explosion. Its vapour pressure is 82.7 mbar (at 13.9 K). Liquid hydrogen when burned with liquid fluorine or liquid oxygen gives high performance. It is an excellent regenerative coolant. It is the lightest and the coldest and a boiling point of about 20 K.

### 3. Ethyl alcohol or ethanol

Ethyl alcohol or ethanol [ $\text{C}_2\text{H}_5\text{OH}$ ] along with some water is used as a rocket fuel with several oxidizers such as LOX,  $\text{H}_2\text{O}_2$ ,  $\text{N}_2\text{O}_4$ , RFNA etc. it gives lower combustion temperature with slightly lower performance. Its molecular weight is 46.06 and specific gravity 0.79.

### 4. Hydrazine

It is toxic, colorless, liquid with a high freezing Point (274.3K). It is an excellent monopropellant when decomposed by a suitable catalyst such as Iridium. It can also be used as a monopropellant; it is used in gas generators or in space engine altitude control rockets. Hydrazine reacts with many materials. Compatible materials include stainless steel, nickel and series of aluminum. Iron, copper and its alloys, Monel metal magnesium. Zinc and some type of aluminum alloys must be avoided. This also a storage rocket fuel widely used with several oxidizers. On leakage it form explosive mixture with air leading to fires and accidents. It is harmful to personnel. A mixture of hydrazine and UDMH is also employed as rocket fuel. Hydrazine has a molecular weight of 32 and specific gravity of about unity.

## 5. Unsymmetrical dimethyl-hydrazine

Aeronautical

This is also known as UDMH. It is a derivative of hydrazine. Its chemical formula is  $(\text{CH}_3)_2\text{NNH}_2$  with a molecular weight of 60. As stated before it is used as a 50:50 mixture with hydrazine in combinations with the oxidizer nitrogen tetroxide or Red fuming nitric acid. Its specific gravity is 0.66. Its boiling point is 336.7 k.

### Properties of liquid propellants

1. Energy released during the combustion per unit mass of the propellant combination should be high.  
*Combustion*
2. High-density propellants are preferred; they require smaller tanks and structures offering lower values of the mass ratio and aerodynamic drag.
3. The propellants should have lower freezing point; this makes the handling of propellants, particularly cryogenic propellants easier in various rocket components. Low freezing point rules out any possibility of freezing of the propellants at high altitudes and in cold weather.
4. They must be non-corrosive so that their handling is easier and less demanding on materials used in various parts of the rockets.
5. They should be chemically stable. Their properties should not deteriorate with time. They should not absorb moisture. Small amounts of impurities should not change their chemical properties.
6. Propellants should have low values of vapour pressure and viscosity. Lower vapour pressure avoids cavitations problems in the feed pumping system. Lower viscosity decrease the power required for pumping.
7. Propellants must have higher specific heat and thermal conductivity for better performance.
8. They should not be poisonous and hazardous. Some propellants are poisonous; they are dangerous for personnel if inhaled or touched.
9. Products of combustion must have lower values of molecular weight and gas constant.

## PROPERTIES OF LIQUID ROCKET PROPELLANTS

Compound	Chemical Formula	Molecular Weight	Density	Melting Point	Boiling Point
Liquid Oxygen	O <sub>2</sub>	32.00	1.141 g/ml	-218.8°C	-183.0°C
Nitrogen Tetroxide	N <sub>2</sub> O <sub>4</sub>	92.01	1.45 g/ml	-9.3°C	21.15°C
Nitric Acid	HNO <sub>3</sub>	63.01	1.55 g/ml	-41.6°C	83°C
Liquid Hydrogen	H <sub>2</sub>	2.016	0.071 g/ml	-259.3°C	-252.9°C
Hydrazine	N <sub>2</sub> H <sub>4</sub>	32.05	1.004 g/ml	1.4°C	113.5°C
Methyl Hydrazine	CH <sub>3</sub> NHNH <sub>2</sub>	46.07	0.866 g/ml	-52.4°C	87.5°C
Dimethyl Hydrazine	(CH <sub>3</sub> ) <sub>2</sub> NNH <sub>2</sub>	60.10	0.791 g/ml	-58°C	63.9°C
Dodecane (Kerosene)	C <sub>12</sub> H <sub>26</sub>	170.34	0.749 g/ml	-9.6°C	216.3°C

### PROPELLANT SELECTION AND DESIRABLE PROPERTIES:

#### 1. Economic factors:

Availability in large quantity and a, low cost (impulse/unit cost must be as large as possible) are very important considerations in the selection of a propellant. In military applications, considerations have to be given to logistics of production supply, and other possible military uses. The production process should be simple, requiring only ordinary chemical equipments and available raw materials.

#### 2. Performance of propellants:

The performance can be compared on the basis of the specific impulse, the effective exhaust velocity, the characteristic velocity, the specific propellant consumption, the ideal exhaust velocity or other engine parameters. For high performance, a high content of chemical energy is desirable because it permits a high combustion temperature. A low molecular weight of the product gases is also desirable

$$I_{sp} \propto \sqrt{(T_c/M)}$$

Where  $T_c$  = combustion temperature  
 $M$  = Molecular weight.

### 3. Common physical hazards:

Corrodibility should low. Properties like toxicity, fire and explosion hazards must be minimum.

### 4. Desirable Physical Properties:

#### a. Low freezing point:

Freezing point should be low. This permits operation of rockets in cold weather.

#### b. High specific gravity:

In order to accommodate a large mass of propellant in a given vehicle tank size, a dense propellant is required. It permits a small vehicle construction and consequently a relatively low structural weight and low aerodynamic drag.

#### c. Stability:

Should be chemically and physically stable. A good liquid-propellant should also have negligible chemical reaction with piping, tank walls, valve seats and gasket materials, even at relatively high ambient temperatures. It should have minimum reaction with the atmosphere.

#### d. Heat transfer properties:

High specific heat, high thermal conductivity and a high boiling or decomposition temperature are desirable for propellants that are used for thrust chamber cooling.

#### e. Pumping properties:

A low vapor pressure permits handling of the propellants and a more effective pump design where the propellant is pumped. If the viscosity of the propellant is too high, then pumping end engine-system calibration become difficult.

f. **Temperature variation:**

The temperature variation of the physical properties of the liquid propellant should be small.

g. **Ignition, combustion and Flame properties:**

All rocket propellants should be readily ignitable and have a small ignition time delay in order to reduce explosion hazard during starting.

Propellant should be smooth without combustion instabilities.

Reaction rate should be fast so that its residence time is minimum. This reduces the length of combustion chamber and hence the cost.

Smoke and brilliantly luminous exhaust flames are objectionable in certain military applications because they can be easily detected.

3- **PROPELLANT FEED SYSTEMS**

16. Liquid propellants are required to be injected at a pressure slightly above the combustion pressure. Two types of feed systems can be employed;

- They are
1. Gas pressure feed systems
  2. ~~The~~ pump feed systems.

The former is much simpler and widely used of low thrust and short-range operations. The later is used in large engines.

The propellant feed system has two principal functions; to raise the pressure of the propellants and to feed them to one or more thrust chambers. The energy for these functions comes either from a high pressure gas, centrifugal pumps, or a combination of the two. The selection of a particular feed system and its components is governed primarily by the application of the rocket, duration, number of type of thrust chambers, past experience, mission and by general requirement of simplicity of design, ease of manufacture, low cost, and minimize inert mass. All feed systems have piping, a series of valves, provisions for filling and removing (draining and flushing) the liquid propellants, and control devices to initiate, stop and regulate their flow and operation.

In general, a gas pressure feed system gives a vehicle performance superior to a turbopump system. When the total impulse or the mass of propellant is relatively low, the chamber pressure is low, the engine thrust to weight ratio is low (usually  $< 0.6$ )

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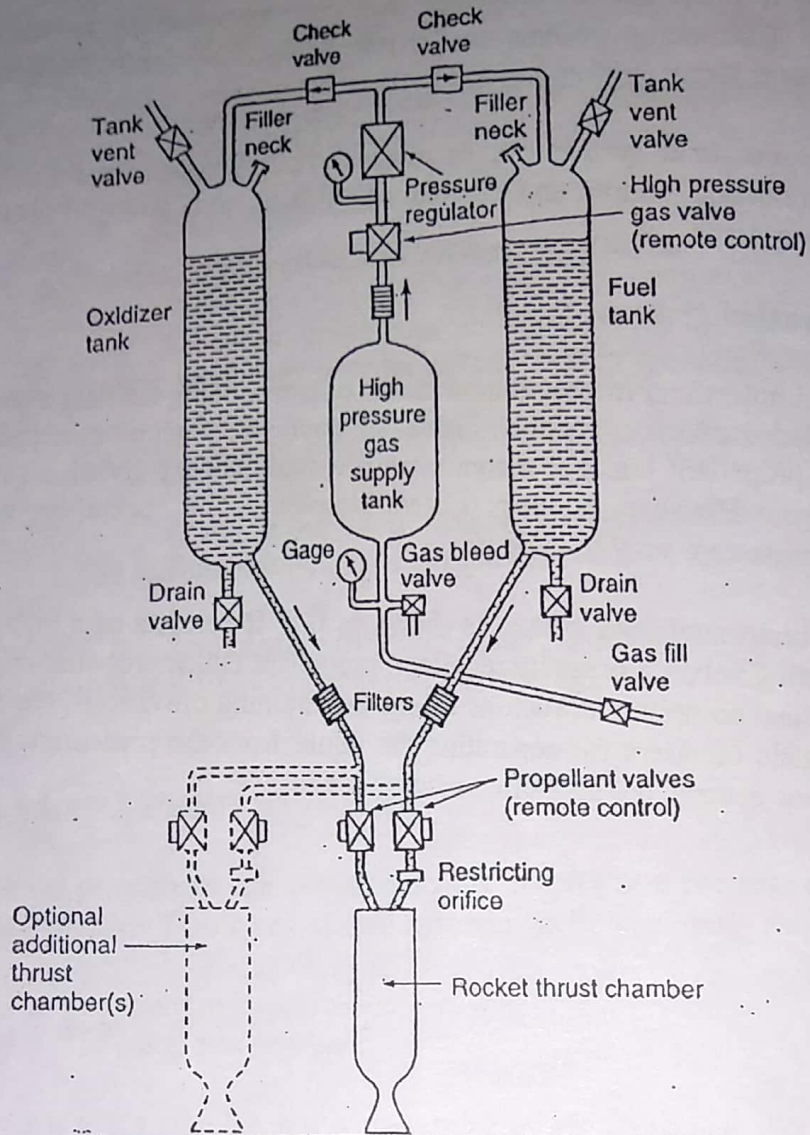
In turbopump feed system the propellant tank pressure are much lower (by a factor of 10 to 40). Turbopump systems usually give a superior vehicle performance when the total impulse is larger (higher  $I_{sp}$ ) and the chamber pressure is higher.

The gas pressure feed system can be relatively simple, such as for a single operation, factors preloaded, simple unit or quite complex as with multiple restartable thrusters or reusable systems.

#### 1. Gas pressure feed systems:

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

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



**A Liquid Propellant Rocket engine with Gas pressure feed system**

An inert gas is separately carried at a pressure much higher than the injection pressure, this is used to exert the required pressure in the propellant tanks. The pressurizing gas is chosen on the basis of its chemical properties density, pressure and the total weight of the gas and the tank. Nitrogen, Helium and air have been used for pressurization. The propellants under high pressure are forced to flow into the thrust chamber through valves, feed lines and injectors.

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

In this method no moving parts such as pumps and turbines are used. Therefore the system is considerably simpler. However the pressurization of the propellant tanks requires them to be comparatively much heavier and introduces a weight penalty besides other problems. Therefore this system is unsuitable for large rocket engines and long range missions. With monopropellants the gas pressure feed system becomes simpler, since there is only one propellant and not two, reducing the number of pipes, valves and tanks.

Pressure for injection can also be generated within the propellant tank or tanks by introducing a small quantity of a gas which reacts exothermally with the propellant, this produces the high pressure gas requires to force the propellant into the combustion chamber.

**Turbo pump feed system:**

For high pressure and high thrust liquid engines, the turbo pump feed system is preferred. Here the propellants are pressurized by means of pumps, which in turn are driven by turbines. These turbines derive their power from the expansion of hot gases.

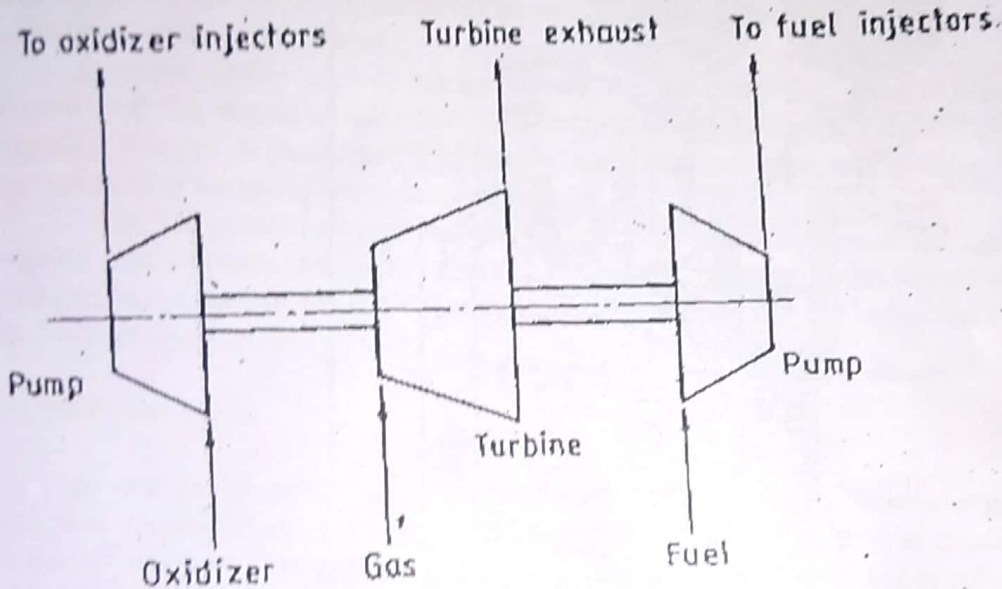


Fig 4. Turbo pump feed system for Liquid propellant Rocket Engine

The turbine or turbines work on high pressure and temperature gas generated separately or tapped out from the main combustor. (Fig 4) depicts a general arrangement of a turbo pump system. Here both the fuel and oxidizer pumps are driven by a single turbine in order to achieve flexibility in choosing the design and operating parameters. The fuel and oxidizer pumps can be driven separately by their turbines.

Fig (5) depicts a turbo pump feed systems employing a single turbine driving the fuel and oxidizer pumps through a reduction gear. The turbine operates on a separate gas stream generated from the propellants in an independent gas generator as shown. A pressurizing

gas can be used to increase the pressure of the propellants at the pump suction to avoid cavitation and the resulting instability in pump operations. Generally turbine speeds are high; therefore propellant pumps can be driven at optimum speeds through reduction gears with an additional weight penalty. The working gas for the turbine can also be generated at the optimum temperature and pressure. The gas generator has also its own injection and ignition systems. The flow of propellants of gas generator occurs (in the system as shown in Fig 5) due to action of pressurizing gas. If the gas pressurization is not employed the propellants can be bled from the delivery lines of the pumps. The propellants flow required for driving the turbine is of order of 1.5 to 5 percent in of main inflow. The turbine exhaust is also expanded through an exhaust nozzle to provide additional thrust as shown in the figure.

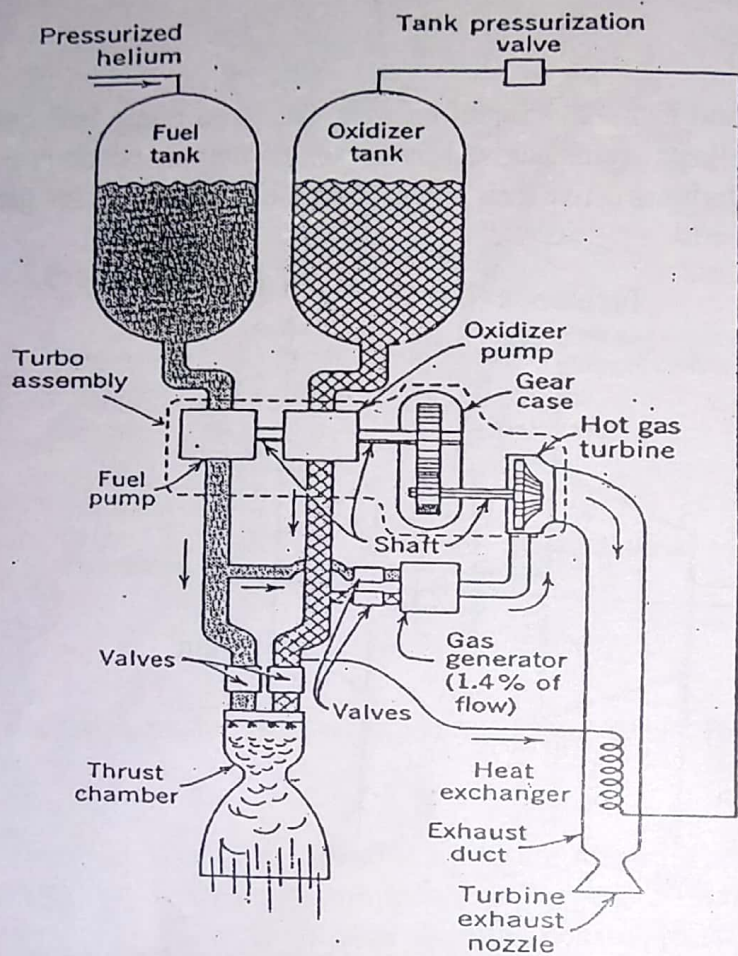


Fig 5. Schematic diagram of a Liquid propellant Rocket Engine with a turbo pump feed system

An auxiliary power unit is also needed in a rocket engine. A single turbine can develop sufficient power to drive the propellant pumps as well as electric generator. Besides working on high energy gases bled from the main thrust chamber or combustor it can also employ its own combustor with a gas pressure feed systems.

There are enormous temperature differences within a turbo-pump unit. The high-pressure gas at the turbine inlet is at a high temperature of order of 15000 K whereas the

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temperature of some of the cryogenic propellants may be as low as 100 K at the pump inlet. Many propellants are highly reactive. Therefore, the sealing arrangement in propellants should be perfect and resistant to corrosion.

Both positive displacement and turbopumps can be used for delivering propellants from tanks to the combustion chamber. However, centrifugal pumps are widely used. For a given peripheral speed of the pump impeller it is preferable to employ higher rotation speed in order to restrict the size of the pump.

Engines with turbopumps are preferred for booster and sustainer stages of space launch vehicles, long-range missiles, and in past also for aircraft performance augmentation. These are usually lighter than other types for these high thrust, long duration applications.

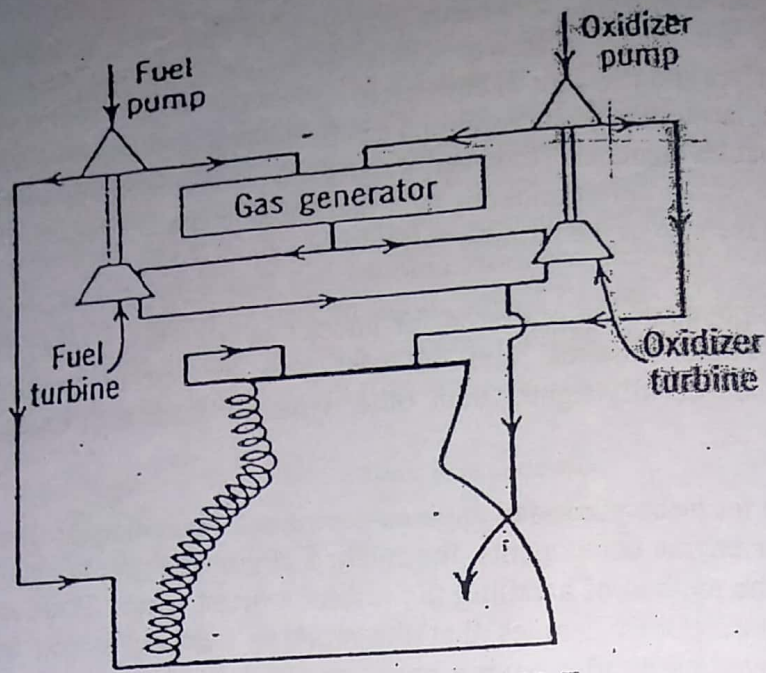
An engine cycle for turbo pump fed engines describes the specific propellant flow paths through the major engine components, the method of providing the hot gas to one or more turbines, and the method of handling the turbine exhaust gases. There are open cycles and closed cycles. Open denotes that the working fluid exhausting from the turbine is discharged overboard, after having been expanded in a nozzle of its own, or discharged into the nozzle of the thrust chamber at a point in the expanding section for downstream of the nozzle throat.

In closed cycles or topping cycles all the working fluid from the turbine is injected into the engine combustion chamber to make the most efficient use of its remaining energy. In closed cycles the turbine exhaust gas is expanded through the full pressure ratio of the main thrust chamber nozzle.

The gas generator cycle and the staged combustion cycle can use most of the common liquid propellants. The expander cycle works best with vapourized cryogenic hydrogen as the coolant for the thrust chamber, because it is an excellent heat absorber and does not decompose.

#### Gas generator cycle:

In the gas generator cycle the turbine inert gas comes from a separate gas generator. Its propellant can be supplied from separate propellant tanks or can be bled off the main propellant feed system. This cycle is relatively simple; the pressures in the liquid pipes and pumps are relatively low. It has less engine specific impulse than an expander cycle or staged combustion cycle. The pressure ratio across the turbine is relatively high, but the turbine or gas generator flow is small of total propellant if compared to closed cycles. Some early engines used a separate monopropellant for creating the generator gas.

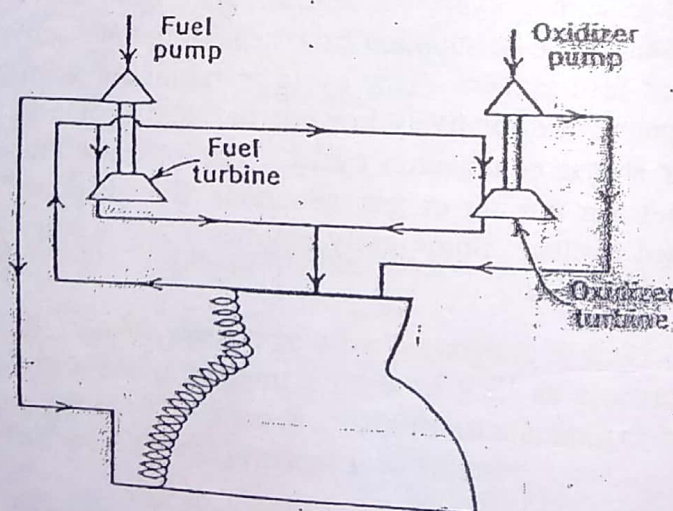


**GAS GENERATOR CYCLE**

The RS-68 rocket engine has a simple gas generator cycle. This engine is the largest liquid hydrogen/liquid oxygen rocket engine built to date.

**Expander cycle:**

In the expander cycle most of the engine coolant (usually hydrogen fuel) is fed to low pressure ratio turbines after having passed through the cooling jacket where it picked up energy. Part of the coolant, perhaps 5 to 15% bypasses the turbine and rejoins the turbine exhaust flow before the entire coolant flow is injected into the engine combustion chamber where it mixes and burns with the oxidizer.



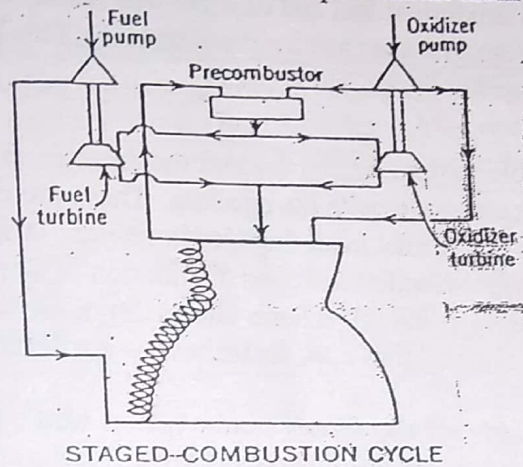
**EXPANDER CYCLE**

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The primary advantages of the expander cycle are good specific impulse, engine simplicity, and relatively low engine mass. In the expander cycle all the propellants are fully burned in the engine exhaust nozzle. This cycle is used in the RL10 hydrogen/oxygen rocket engine. A recent modification of this engine, the RL10B-2. It delivers the highest specific impulse of any chemical rocket engine to date.

### Staged combustion cycle:

In the staged combustion cycle, the coolant flow path through the cooling jacket is the same as that of the expander cycle. Here a high pressure pre-combustion chamber where is burns with the remaining oxidizer. This cycle tends itself to high chamber pressure operation, which allows a small thrust chamber size. The staged combustion cycle gives the highest specific impulse, but it is more complex and heavy. A variation of the staged combustion cycle is used in the space shuttle main engine.



### Combustion

Combustion of a liquid propellant (fuel and oxidizer mixture) in the combustor or thrust chamber of a rocket engine requires the following basic processes-injection, atomization, mixing, vaporization, ignition and exothermic chemical reaction between the fuel and oxidizer. Mixing and chemical reactions may also take place before vaporization.

Complete combustion depends besides other factors, on the oxidizer-fuel ratio, combustion temperature and time and space available, degree of atomization and mixing. In contrast to air breathing engines the oxidizer-fuel ratio in liquid propellant rockets is lower than the stoichiometric value to give a fuel rich mixture. The combustion temperature corresponding to stoichiometric ratios are very high.

The propellants are injected into thrust chamber through fine orifices for proper atomization. Various methods are employed to atomize and mix the fuel and oxidizer.

# Liquid Propellant Injectors:

## Liquid propellant Rocket Injectors

The injector as the name implies injects the propellants into the combustion chamber in the right proportions and the right conditions to yield an efficient, stable combustion process. Placed at the forward or upper end of the combustor, the injector also performs the structural task of closing off the top of the combustion chamber against the high pressure and temperature it contains.

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

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

The higher passage velocity causes a uneven flow through the different identical injection holes and thus a poorer distribution and wider local gas composition variation. Dribbling result in after burning, which is an insufficient irregular combustion that gives a little "cut off" thrust after valve closing. For application with very accurate terminal vehicle velocity requirements, a cut off impulse has to be very small and reproductive and often walls are built into the injector to minimize passage volume.

The various types of injectors in liquid propellant rocket engines are as follows

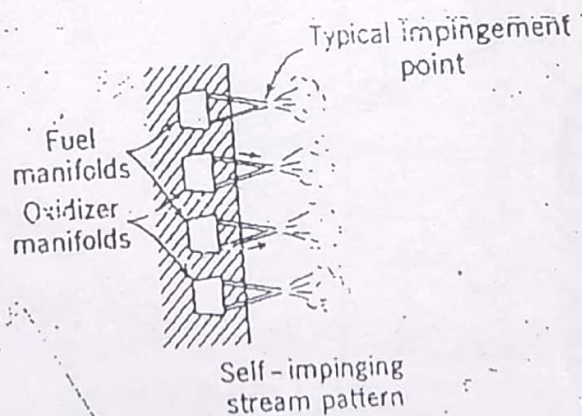
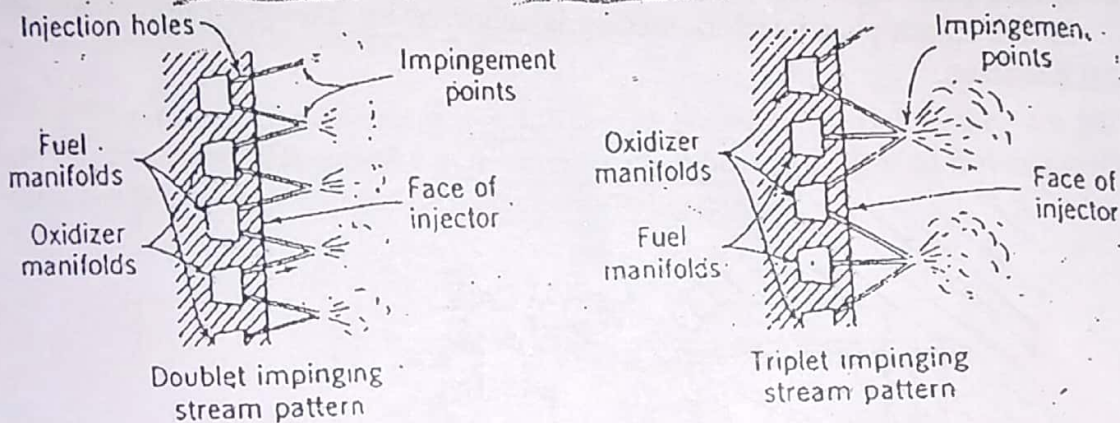
1. The impinging stream type injector
2. The non-impinging or shower type injector
3. The splash plate injector
4. Sheet or spray type injector
5. The pre-mixing injector
6. The co-axial hollow post injector

## 1. The impinging stream type injector:

In the impinging stream type injector, the propellants are injected in such a manner that the fuel and oxidizer streams impinge upon each other. A conical outlet housing the jet is also employed to collect the streams and supply a well-mixed propellant charge to the combustor. This will aid the atomization of liquid into droplets. Majority of rocket engine uses impinging type injector.

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

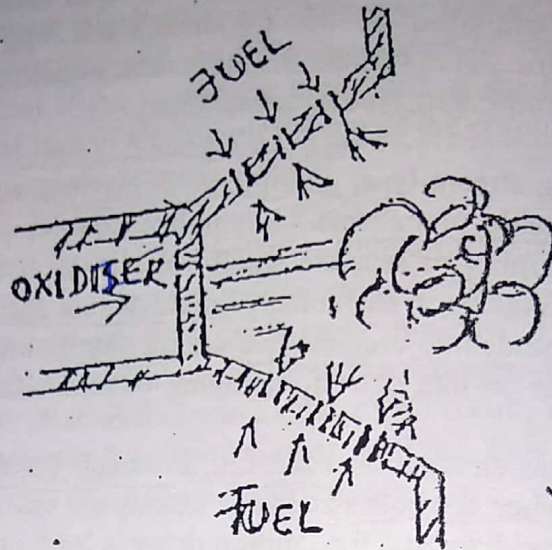
The two liquid streams then form a fan which breaks up into droplets. Unlike doublets work best when the hole size (more exactly the volume flow) of the fuel is about equal to that of the oxidizer and the ignition delay is long enough to allow the formation of fans. For uneven volume flow the triplet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



## 2. The non-impinging type or shower head injector:

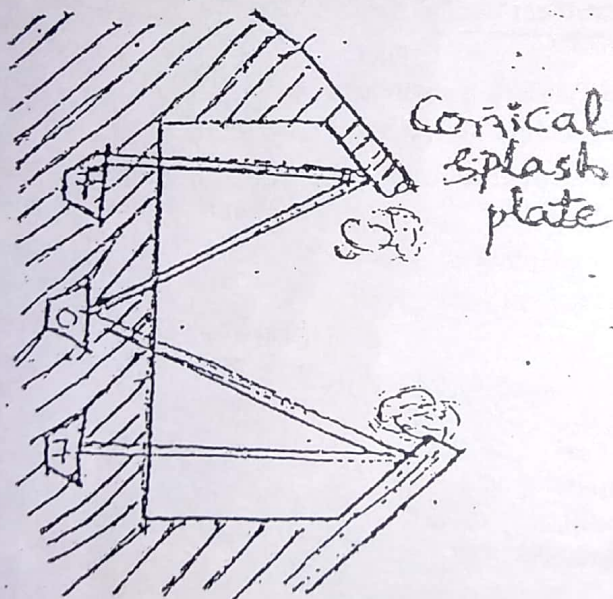
The non-impinging or shower head injector is one in which the oxidizer and fuel do not impinge at any specific point, but they are mixed by the turbulence within the chamber. The shower head injector employs non-impinging streams of propellant usually

emerging normal to the face of the injector. It relies on turbulence and diffusion to achieve mixing. The German world war II V-2 rocket used this type of injector. This is not used, because it requires a large chamber volume for good combustion.



3. The splash plate injector:

In the splash plate injectors mixing is achieved by impinging propellant streams against a surface.

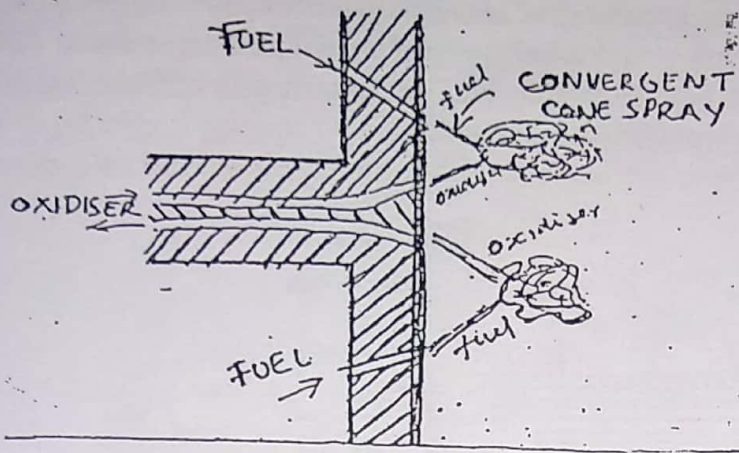


SPLASH PLATE PATTERN.

4. Sheet or spray type injector:

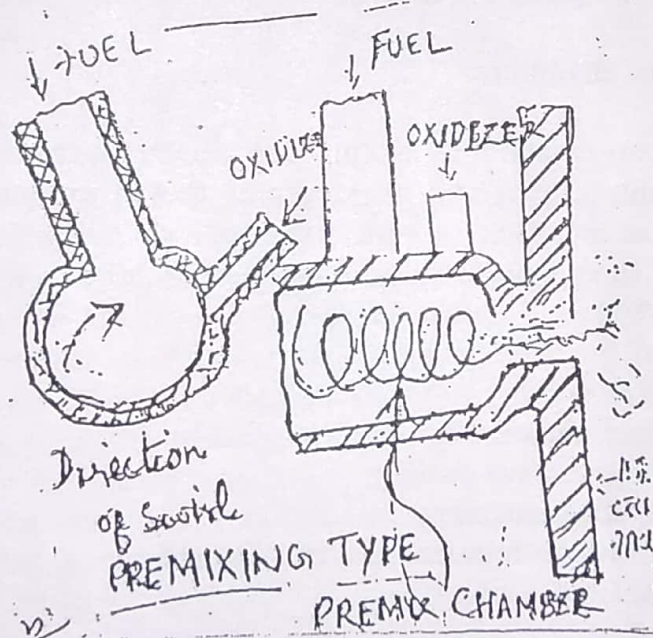
A Sheet or spray injector has oxidizer and fuel holes arranged in circles, so as to produce conical or cylindrical spray patterns that intersect within the chamber. spray type injector give cylindrical, conical or other types of spray sheets; these spray

generally intersect and thereby promote mixing and atomization. By varying the width of the sheet (through an axially movable sleeve) it is possible to throttle the propellant flow over a wide range without excessive reduction in injector pressure drop. This type of variable concentric tube injector was used on the descent engine of the lunar excursion module and throttled over a range of flow with only a very small in mixture ratio.



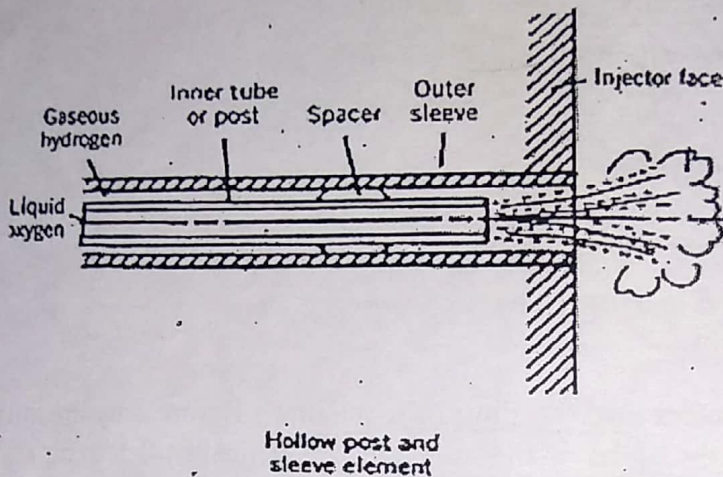
### 5. The pre-mixing injectors:

The pre-mixing injectors mix the liquid propellants. Before they are introduced into the chamber. The lengths of the premix chamber are critical so that explosion of the premixed propellant does not take place inside the engine.



## 6. The coaxial hollow post injector:

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



## 5. 8m ✓ COOLING IN LIQUID ROCKETS:

### Cooling of thrust chambers:

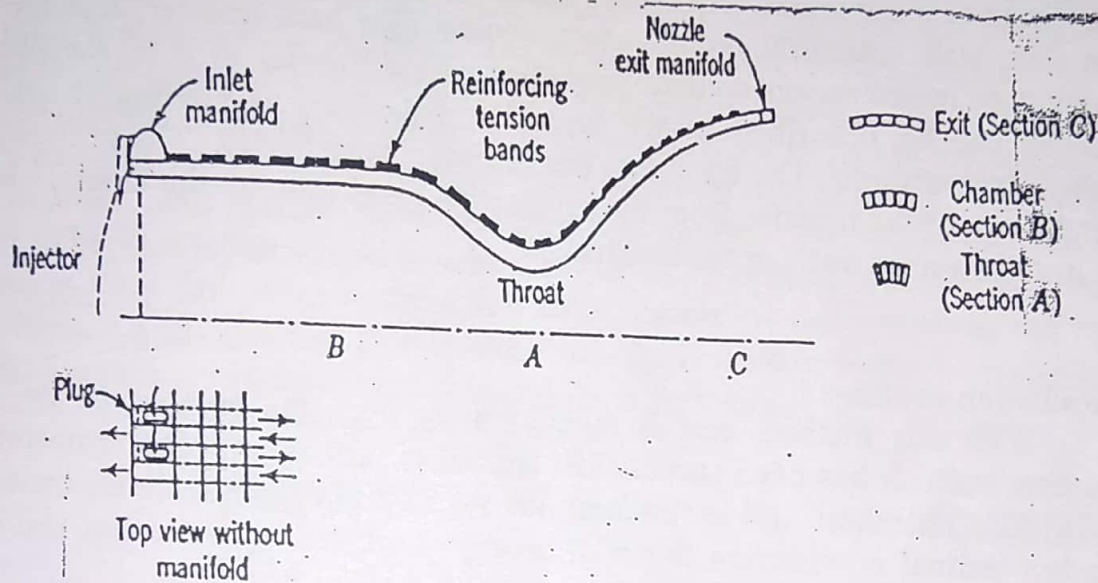
The primary objective of cooling is to prevent the chamber and nozzle walls from becoming too hot, so they will no longer be able to withstand the imposed loads or stresses, thus causing the chamber or nozzle to fail. Cooling reduces the wall temperatures to an acceptable value. Cooling also helps to reduce the oxidation of the wall material.

### Cooling methods:

1. Steady state method
  - a. Regenerative cooling
  - b. Radiation cooling
2. Unsteady state or transient heat transfer method
  - a. Ablative cooling
3. Film cooling
  - a. Transpiration cooling
4. Dump cooling
5. Thermal insulation

## 1. Steady state method

In this method the heat transfer rate and the temperature of the chamber reach thermal equilibrium. Cooled thrust chambers have provisions for cooling some or all metal parts coming into contact with hot gases, such as chamber walls, nozzle walls and injector faces. Internal cooling passages, cooling jackets can consist of separate inner and outer walls or of an assembly of contoured, adjacent tubes. The nozzle throat region is usually the location that has the highest heat transfer intensity and is therefore the most difficult to cool. For this reason cooling jacket is designed so that the coolant velocity is higher at the critical regions by restricting the coolant passage cross section. The steady state method includes regenerative cooling and radiation cooling.



### a. Regenerative cooling:

It is done by building a cooling jacket around the thrust chamber and circulating one of the liquid propellants (usually the fuel) through it before it is fed to the injector. This cooling technique is used primarily with bipropellant chambers of medium to large thrust. *The heat which is taken away by way of cooling is picked up by a propellant and fed back to the combustion chamber, so it is not lost* that why the name regenerative cooling. A long duration liquid propellant rocket motor requires regenerative cooling. It has been effective in applications with high chamber pressure and high heat transfer rates. Also most injectors use regenerative cooling.

It is most widely used method which utilizes one or possibly both of the propellants fed through passages in the thrust chamber wall for cooling before being injected into the combustion chamber.

**Advantages:**

1. No performance loss (thermal energy absorbed by the coolant is returned to the injector)
2. No change in the wall contour as a function of time, indefinite firing duration, and relatively light weight construction.

**Disadvantages:**

1. Limited throttling with most coolants, reduced reliability with some coolants (eg. Hydrazine)
2. High-pressure drops required at high heat flux levels, and thrust levels.
3. The mixture ratios or nozzle area ratios possibly limited by maximum allowable coolant temperature.

The design of a regeneratively cooled thrust chamber involves consideration of gas side heat flux, wall structural requirements, coolant side heat transfer, and the effects of temperature increases on coolant properties. Heat transfer in a regeneratively cooled chamber can be described as the heat flow between two moving fluids, through a multiplayer partition. The fig shows this process schematically. The general steady state correlation of heat transfer from the combustion gases through the layers, which include the metal chamber walls to the coolant.

**b. Radiation cooling:**

With this method heat is radiated away from the surface of the outer thrust chamber wall. It has been successfully applied to very small, high temperature material combustion chambers and to low heat flux regions, such as nozzle extensions. When it reaches thermal equilibrium this wall usually glows red or while hot and radiates heat away to the surrounding or the empty space.

Radiation cooling is used with monopropellant thrust chamber, bipropellant and monopropellant gas generators and for diverging nozzle sections beyond an area ratio of about 6-10. A few small bipropellant thrusters are also radiation cooled. This cooling scheme has worked well with lower chamber pressures (less than 250 Psi) and moderate heat transfer rates.

**2. Unsteady state or transient heat transfer method:**

This method is also called heat sink cooling. This method has mostly been used with low chamber pressure and low heat transfer rates. Heat sink cooling of thrust chambers can be done by absorbing heat in an inner liner made of an ablative material, such as fiber reinforced plastics. Ablative materials are used extensively in solid propellant rocket motors.

**a. Ablative cooling:**

(In this process combustion gas side wall material is sacrificed by melting, vaporizing and chemical changes to dissipate heat. As a result, relatively cool gases flow over the wall surface, thus lowering the boundary layer temperature and assisting the cooling process. In addition, the ablative material is usually a good thermal insulation.)

Ablative cooling was first used and still used extensively with solid propellant rocket motors. It has been successfully applied to liquid propellant thrust chamber, particularly at low chamber pressure, short duration and also in nozzle extensions for both large and small thrust chambers where the static gas temperatures are relatively low.

An ablative material usually consists of a series of strong, oriented fibers (such as glass, Kevlar or carbon fibers) engulfed by a matrix of an organic material (such as plastics, epoxy resins or phenolic resins)

### 3. Film cooling:

Here, exposed chamber wall surfaces are protected from excessive heat by a thin film of coolant or propellant introduced through orifices around the injector periphery or through manifolded orifices in the chamber wall near the injector and sometimes in several more planes toward the throat. The method has been used particularly for high heat fluxes either alone or in combination with regenerative cooling.

This method is applied to the complete thrust chamber or just to the nozzle, where the heat transfer is the highest. In this method a relatively cool thin fluid film covers and protects exposed wall surfaces from excessive heat transfer. Fig shows the film cooled chambers. The film is introduced by injecting small quantities of fuel or an inert fluid at very low velocity through a large number of orifices along the exposed surfaces in such a manner that a protective relatively cool gas is formed. A coolant with high heat of vaporization and a high boiling point is desirable.

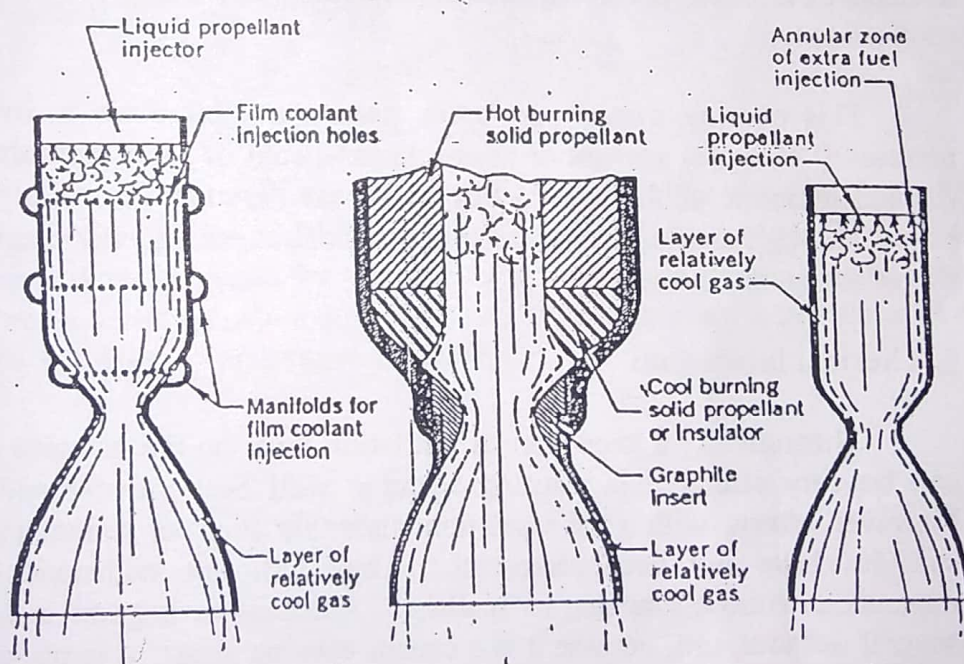


FIGURE 8-11. Simplified diagrams of three different methods of forming a cool boundary layer.

In liquid propellant rocket engines extra fuel can also be admitted through extra injection holes at the outer layers of the injector. This differs from film cooling or

transpiration cooling manifolds. In solid propellant rocket engines this can be accomplished by inserting a ring of cool burning propellant upstream of the nozzle or by wall insulation material whose ablation will release cool gases into the boundary layer.

Turbine discharge gas ( $700 - 1100^{\circ}\text{C}$ ) has also been used as film coolant for uncooled nozzle exit sections of large liquid propellant rocket engines.

#### a. Transpiration cooling:

A special type of film cooling sweat or transpiration cooling uses a porous wall material which admits a coolant over the surface. This technique has been successfully used to cool injector faces in the upper stage of the moon launch vehicle and space shuttle Main engine (SSME) with Hydrogen fuels.

In this method introduces a coolant (either gaseous or liquid propellant) through porous chamber walls at a rate sufficient to maintain the desired temperature of the combustion gas side chamber wall.

#### 4. Dumb cooling:

With this principle a small percentage of the propellant, such as the hydrogen in  $\text{LO}_2/\text{LH}_2$  engine, is fed through passages in the thrust chamber wall for cooling and is subsequently dumped overboard through openings at the rear end of the nozzle skirt. Because of inherent problems, such as performance losses, this method has only limited applications.

This cooling system may prove particularly effective in hydrogen fueled, low pressure ( $<100\text{psia}$ ) systems or in nozzle extension of high pressure hydrogen systems. A small amount of the total hydrogen flow is directed from the main fuel feed line, passed through cooling passages and ejected. The heat transfer mechanism is similar to that of regenerative cooling.

#### 5. Thermal insulation:

Theoretically a good thermal insulation layer on the gas side of the chamber wall can be very effective in reducing chamber wall heat transfer and wall temperatures. However efforts with good insulation materials such as refractory oxides or ceramic carbides have not been successful. They will not withstand differential thermal expansion without cracking or spalling. Asbestos is a good insulator and was used several decades ago, because it is a cancer causing agent, it is no longer used. Coating development efforts with rhenium and other materials are continuing.