SURVEY OF DEVELOPMENTS IN ROCKET PROPULSION

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ABSTRACT

Rocket propulsion has progressed rapidly from the relatively crude liquid rocket engines of a few short years ago to the sophisticated electrical propulsion devices of tomorrow. By faithful application of certain basic principles of propulsion to chemical systems, nuclear systems and conducting-gas devices, our ability to efficiently propel a great variety of space vehicles has improved at an encouraging rate.

Before we launch into discussions concerning development of lubricants for use in rocket engines, it would be wise to survey our present status in the various types of rocket propulsion engines. From this survey we can then assess the validity of possible development trends and speculate as to the directions future propulsion will take us.

Rocket engines are self-contained systems, that is, they differ from air-breathing turbojet engines in that they carry their own oxidizers, thus operating completely independent of the environment through which they pass. There are two broad classifications of rocket engines; the conducting-gas devices which, as we will see shortly, are best suited for advanced space propulsion, and the nonconducting gas devices, the best known of which is the chemical engine.

The chemical engine is so named because it is a chemical process which is the source of energy; the combustion of fuel and oxidizer resulting in a gaseous product which can be expanded through a nozzle to produce thrust. The propellants can be in either solid or liquid form; and we shall now examine both of these chemical engine types in more detail, placing emphasis on the major sub-systems, for it is here in the basic components where difficulties are encountered and where system limitations are defined.

Looking first at a functional diagram of a liquid rocket engine (Fig. 1), we note an oxidizer supply and associated plumbing, a fuel supply with its piping, some type of feed system to move the propellants from the tanks to the combustion chamber, an injector to properly mix and regulate the flow and finally the converging-diverging or DeLaval nozzle through which the hot gas is expanded.

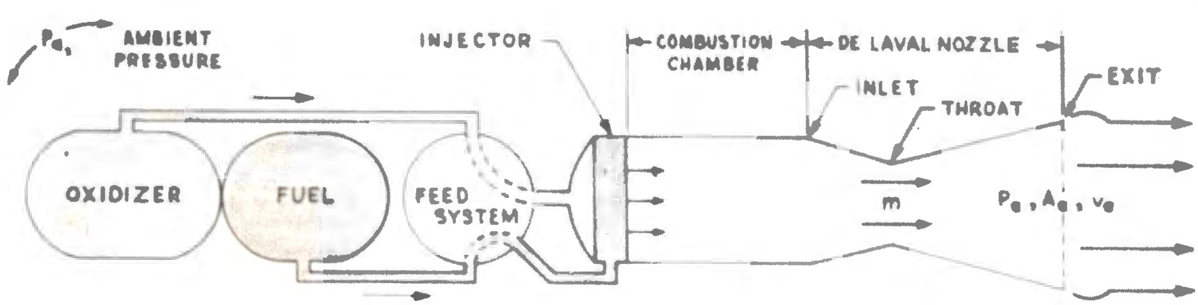

FIGURE 1. SIMPLIFIED LIQUID ROCKET ENGINE

Early engines used the pressure feed system illustrated in Figure 2. It consisted merely of a supply of high-pressure gas vented to the top of the propellant tanks so as to force propellant into the combustion chamber. This was a very simple, reliable system, but unfortunately the weight of gas tankage necessary became prohibitive for engines designed for long duration burning.

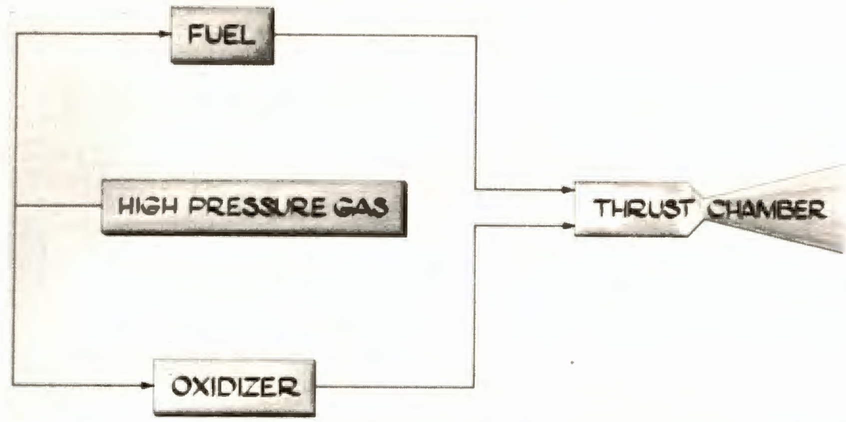

FIGURE 2. SIMPLIFIED DIAGRAM OF PRESSURE FEED SYSTEM

Also, the pressures required for high-performance were not available from a pressure feed system, so the trend took us to a utilization of a pump feed system (Fig. 3). To trace a cycle of operation, let us start with the gas generator, which is a spherical combustion chamber used to generate gas for driving this turbine. The turbine in turn is geared to the propellant pumps which move the liquids to the combustion chamber. The propellant lines are usually tapped prior to the injector, and through what is called a bootstrap system, a small portion of fuel and oxidizer is brought back to the gas generator to sustain burning, which was initially started by a separate liquid propellant supply or a solid propellant charge. The hot gas, after leaving the turbine is dumped overboard after going through a heat exchanger which gasifies helium used for pneumatic systems in most of our ballistic missiles. The thrust is controlled by regulating the propellant pressure by pump speed which in turn depends on the output of the gas generator. Gas generator operation is directly controlled by regulating propellant flow into the generator by valves in the bootstrap line.

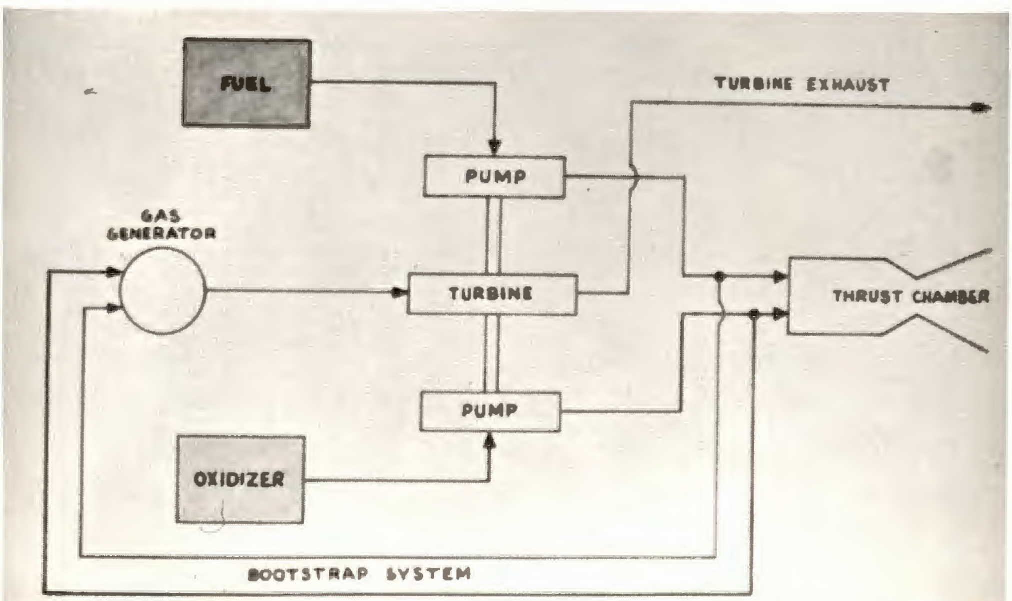

FIGURE 3. SIMPLIFIED DIAGRAM OF PUMP FEED SYSTEM

The turbopump (Fig. 4) is the most critical of the engine components in respect to both operation and performance limitation. The gas generator exhaust enters through the lower part driving the turbine, the gear train, and finally the propellant pumps. The fuel enters here on the right and is discharged above while the LOX goes through a similar route on the left side. Pump outlet pressures are in the neighborhood of 800 to 1000 psi and are achieved by the pump running at 33,000 rpm. The pump can attain this speed from a standing start in 1/2 second; lubricating oil is sprayed on the gears as they break away. If the gears were lubricated prior to meshing, close tolerances would generate an additional heat load. Turbopumps on our early engines were a trouble source, due primarily to improper lubrication caused by oil puddling in the bottom upon missile acceleration and oil frothing as pressure decreased at altitude. Both of these conditions were corrected when the oil ports were relocated and the entire crankcase pressurized.

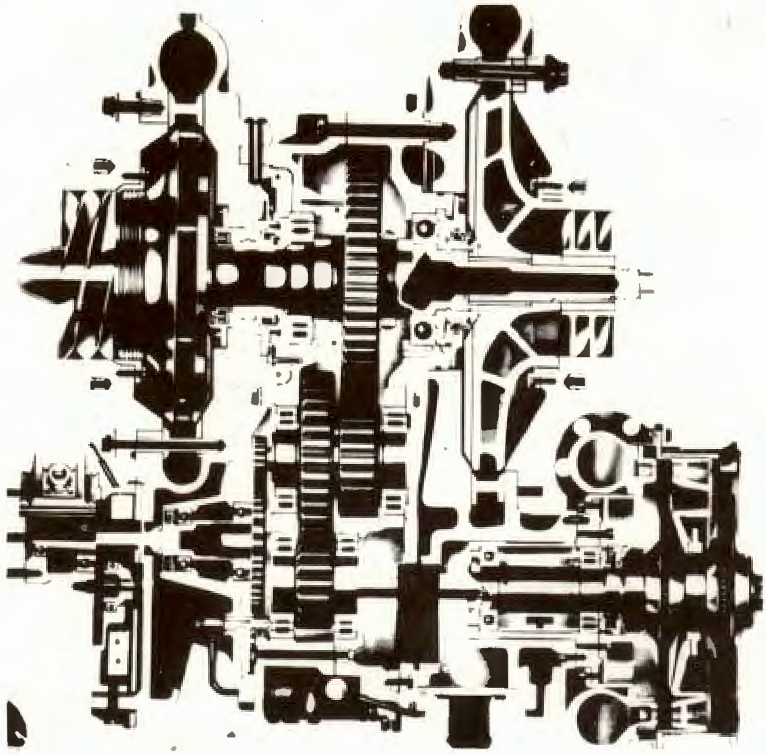

FIGURE 4. TURBOPUMP ASSEMBLY

Initial ignition takes place in the center of the injector and can be electrical, a powder squib or injection of a hypergolic propellant. A hypergolic is a material which ignites spontaneously upon contact with another propellant, such as tri-methyl aluminate igniting when it contacts liquid oxygen. The hypergol slug is forced through the injector by fuel pressure. After initial ignition, burning is self-sustaining with temperatures in the combustion chamber and nozzle ranging from 3000 to 8000°F and pressures as high as 3000 psi.

The manner in which the fuel and oxidizer is directed into the combustion chamber through the injector is varied, as evidenced by the different flow patterns shown in Figure 5. The two principal objectives of the injector are to thoroughly mix the propellants so optimum burning is maintained and to provide an even flow pattern to minimize local hot spots. Four of the diagrams illustrate types of impinging patterns. In two diagrams, the fuel and oxidizer are merely fed straight through the injector. One uses a splash plate for mixing, and two utilize a mixing chamber prior to injection. Experience has indicated that in most cases, the simpler the pattern, the more effective the burning. Current engines use essentially some combination of the first three impinging patterns. It should also be noted that certain patterns work better with certain propellant combinations.

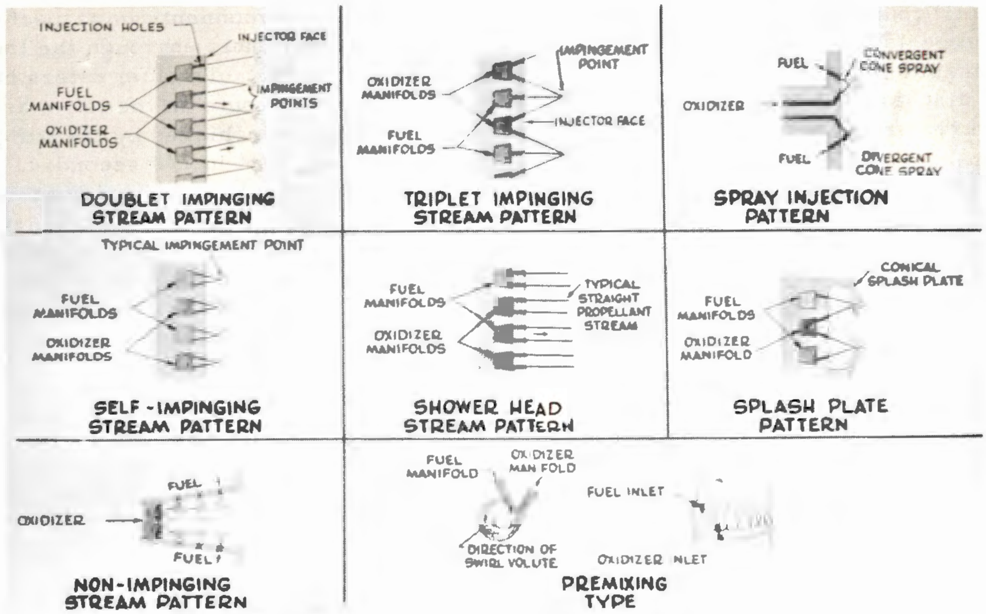

FIGURE 5. SCHEMATIC DIAGRAM OF SEVERAL INJECTOR TYPES

We can use the typical engine shown in Figure 6 to more or less tie together the various components we have discussed so far. To run through a cycle again, we have the gas generator sending hot gas through the turbine then exhausting out through the heat exchanger. The turbine drives the pumps pulling in LOX and fuel, and we see the bootstrap lines routing propellant back to the gas generator.

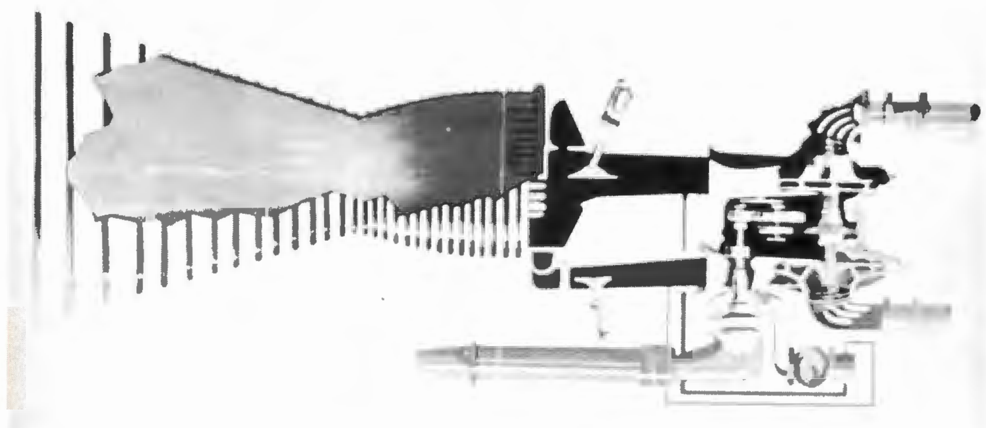

FIGURE 6. TYPICAL LARGE ROCKET ENGINE

We have mentioned temperatures in thrust chamber of about 5000°F, so obviously there must be a cooling system. On early liquid rocket engines, the heat load imposed by combustion was simply absorbed by the material out of which the nozzle was fabricated. As performance improved and heat loads increased, this method could not be used due to the weight penalties imposed by the heavy metallic structure which would be required. What is used now is a technique called regenerative cooling (Fig. 7). The fuel comes from the pump and is allowed to enter alternate nickel-steel tubes which are brazed together and form the wall of the thrust chamber. The fuel then flows the length of the chamber, manifolding at the aft end and flowing back the adjacent tubes. Upon reaching the forward end, the fuel is reversed back through the injector into the combustion

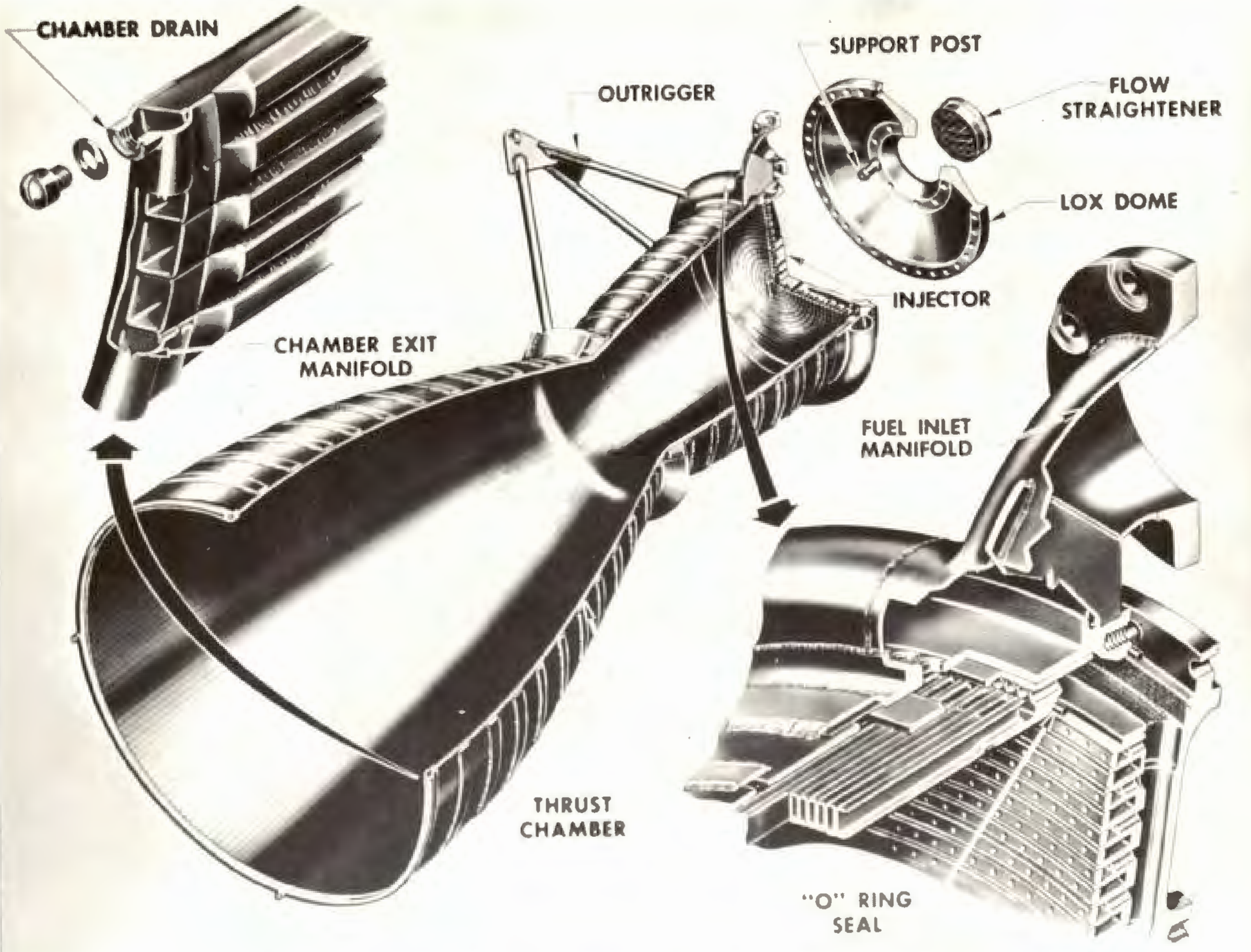

FIGURE 7. THRUST CHAMBER ASSEMBLY

chamber. During this process, a tremendous amount of heat is transferred to the fuel, reducing the interior temperature of 5000°F down to about 150°F on the outside of the chamber wall. In addition, we benefit by obtaining a relatively hot fuel for subsequent burning which, of course, improves combustion characteristics.

In Figure 8, we see the actual construction of this type of combustion chamber. Until about a year ago, tubes had been hand brazed in place, but this has become a machine operation for the most part.

There are two directions in which we can proceed to obtain greater thrust from liquid rocket engines. First, we can build bigger and bigger single thrust chambers such as the Aerojet engine shown in Figure 9. This power plant will produce about 1.5 million pounds of thrust at sea level using about an 18:1 expansion ratio. Rocketdyne, with their F-1 Engine, has produced a comparable performer in this large chamber field.

The second course of action to obtain more thrust is the concept of engine clusters. This idea is illustrated by the Saturn Booster (Fig. 10), which is composed of a cluster of eight H-1 engines, each rated at about 190,000 pounds of thrust. Proponents of this concept argue that an increased reliability is offered by multiple engines versus a single power plant.

When choosing a propellant or propellants for a liquid rocket engine system, the designer looks for certain desirable qualities such as high energy per unit mass, high density and good

FIGURE 8. CONSTRUCTION OF COMBUSTION CHAMBER

FIGURE 9. SINGLE-CHAMBER ROCKET ENGINE

FIGURE 10. THE SATURN BOOSTER

handling and storage characteristics. That is, the propellants should not be toxic, poisonous or corrosive, or if so, to a very limited degree.

Propellant systems can be either monopropellant or bipropellant. A monopropellant system uses a substance which contains both a combustible ingredient, the fuel, and an oxidizing agent. Monopropellants are usually stable at ordinary atmospheric conditions but decompose readily yielding hot combustion gases when heated or pressurized. A bipropellant system contains separate fuel and oxidizer which are not mixed before being admitted to the combustion chamber. The bipropellant systems are used almost exclusively in today's rocket engines due to their higher performance characteristics.

Table 1 compares the performance of some of the bipropellant systems in use today. RP-1 fuel is a kerosene-base product, a refined JP-4. Hydrazine and UDMH, termed storable fuels, are both of the same family, slightly toxic and corrosive. Oxidizers are either cryogenic, that is, they must be kept refrigerated to remain in usable form, or they are storable. Liquid fluorine would yield better performance as measured in seconds of specific impulse than liquid oxygen, but it is difficult or impossible to handle this highly toxic and corrosive material. The present performance of nitrogen tetroxide and chlorine trifluoride as oxidizers is considerably less than liquid oxygen, but they are constantly being improved and their use will ultimately phase out cryogenic oxidizers. The combination of nitrogen tetroxide and a mixture of UDMH and hydrazine is currently programmed for the Titan II weapon system.

TABLE 1. LIQUID PROPELLANT PERFORMANCE

Values of Specific Impulse (Isp) where

$$\text{Isp (Seconds)} = \frac{\text{Pounds of Thrust}}{\text{Pounds of Propellant/Sec}}$$

	Oxidizers			
	Cryogenic		Non-Cryogenic (Storable)	
Fuels	Liquid Oxygen	Liquid Fluorine	Nitrogen Tetroxide	Chlorine Tri Fluoride
RP-1	286	295	263	251
Hydrazine (N_2H_4)	301	334	283	279
UDMH	295	308	274	269

·Above values of Isp for chamber pressures of 1000 psi

The other major type of chemical rocket is the solid propellant engine (Fig. 11). Where in a liquid engine the propellants are pumped from storage tanks and injected into the combustion chamber for burning, solid propellants are burned "in place," that is, the propellant container or case also serves as the combustion chamber. It is cylindrical in section and can be constructed of any material compatible with weight limitations and strength requirements for high heat loads. Currently, light-weight steel alloys are being used but much research is being directed toward ceramics and plastics as materials for case construction. The burning is initiated by the igniter assembly with electrical ignition of a powder charge which sprays hot gas down the port area igniting the grain.

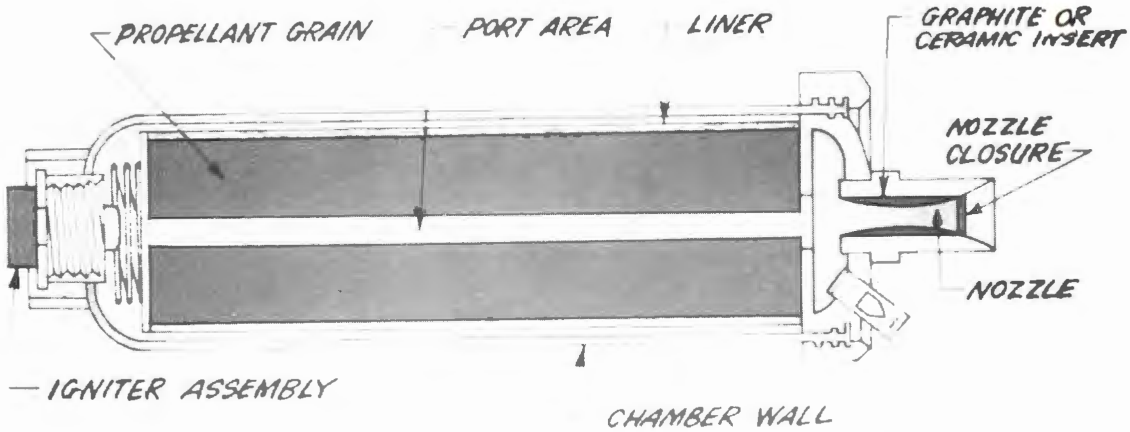

FIGURE 11. SOLID PROPELLANT ROCKET ENGINE

The grain is consumed in a direction normal to the burning surface. The grain material and internal configuration determine the chamber conditions and the burning characteristics. Bonding the grain to the case and also serving as an insulation, is the case liner which is usually a near-inert material.

Solid propellant grains for large engines are the composite type, where the fuel and oxidizer are physically rather than chemically mixed. About 80% of the grain is oxidizer, usually a nitrate, chlorate or perchlorate. The remainder is a fuel such as the rubber polymer, polyurethane, which must provide, in addition to combustion properties, the physical strength or rigidity of the grain. Included in the fuel is usually an additive (about 5%) to increase burning rate such as powdered aluminum or magnesium.

Performance of a solid propellant grain depends primarily on the combustion gas pressure. This pressure in turn depends on rate of burning and the area exposed to burning which is configured by core design and the use of inert inhibitors on some of the burning surfaces. Figure 12 shows just four of many possible internal configurations. The objective in core design is to achieve various directional burning times so as to consume the entire grain evenly and simultaneously.

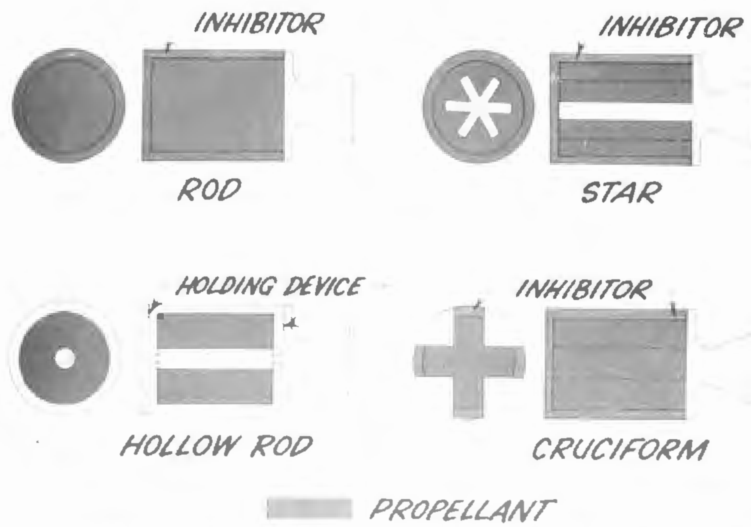

FIGURE 12. PROPELLANT GRAIN CONFIGURATIONS

In addition to the chemical engine, another type of nonconducting gas device is a nuclear fission engine. This engine differs from a chemical engine in that energy is transmitted to the working fluid by heat transfer rather than chemical reaction. The heat source in this case is a nuclear reactor, gas cooled and designed to generate heat by controlled fission of U-235. A typical fission engine system is similar to a conventional liquid chemical engine, even to physical size (Fig. 13).

The propellant is pumped at high pressure from a storage tank and divided into two streams. One stream flows through the nozzle cooling jacket before entering the reactor core and the other stream performs a cooling function in flowing over control rods, structural supports, etc. The combined flow then passes through the core where it is heated to about 5000°F. This heated gas is then expanded through the converging-diverging nozzle, producing thrust in the familiar manner. A portion of the gas cooling the control rod is diverted before entering the core and, having had a sufficient amount of heat transferred to it, is allowed to expand through a turbine which drives the propellant pump and an auxilliary power unit.

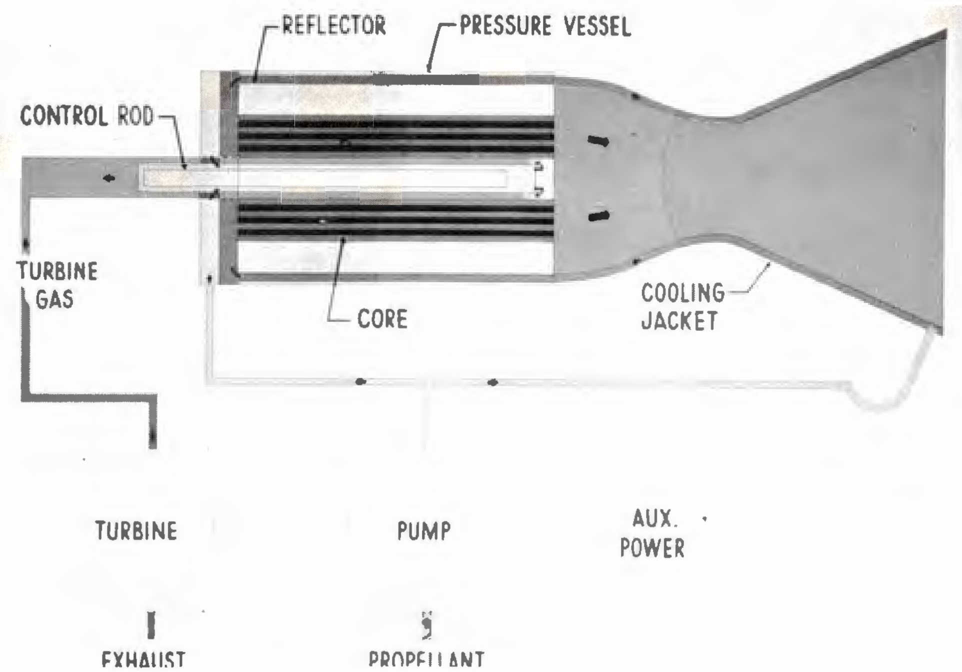

FIGURE 13. TYPICAL NUCLEAR ENGINE

The performance yardstick which I discussed in connection with chemical engines, specific impulse, is directly proportional to the square root of the temperature and inversely proportional to the square root of the molecular weight of the propellant. Since the reactor temperature is limited by reactor materials, maximum performance is achieved by using a working fluid with the lowest molecular weight possible, hence the use of hydrogen with a molecular weight of two.

Even though a nuclear engine with a single propellant reduces complexity, the bulky nuclear reactor necessitates an engine design of about the same size as a chemical engine (Fig. 14). However, the low molecular weight of propellant, be it hydrogen, helium or ammonia, results in a smaller gross weight and mass ratio, the ratio of launch weight to burnout weight. All of these factors produce a specific impulse of about 800 seconds or about double that of a chemical engine of like size.

The development of a nuclear reactor rocket engine poses several problems peculiar to this method of propulsion. The reactor must be designed so it will come up to operating power in a matter of seconds and operate at high temperatures without erosion, corrosion or change in physical characteristics. Methods must be provided to reduce heat in engine components and tankage from neutron and gamma radiation, plus danger from induced radio-activity. During the engine development stage, radiation shielding will probably be a compromise between heavy shielding and distance between the reactor assembly and the rest of the vehicle system. Fission fragments

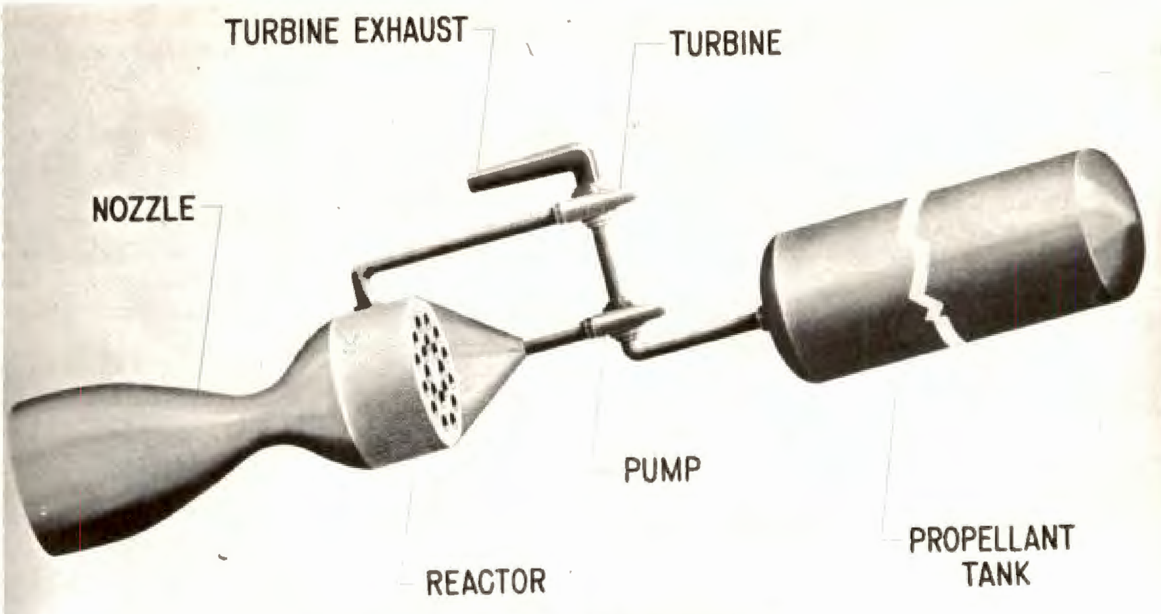

FIGURE 14. NUCLEAR ROCKET ENGINE

produced from reactor processes are also a problem, since they are carried by the propellant in the exhaust jet which could result in contamination of the test stand and nearby areas.

The second broad classification of rocket engines, the conducting-gas device, derives its energy from ionization within the atom in forming a plasma of positively-charged ions and negatively-charged electrons. There are three basic types of devices utilizing this energy source, the electrostatic, the electromagnetic and the electrothermal.

The electrostatic device separates the ions from the electrons and discharges them by means of voltage electrodes. Typical of an electrostatic device is the ion engine (Fig. 15). In this system a nuclear reactor provides thermal energy to heat a working fluid such as liquid sodium. This fluid is vaporized and allowed to flow across a gas turbine which drives an electrical power generator and a pump. The vapor after leaving the turbine is condensed in a radiator and pumped back to the reactor where the working fluid cycle is repeated.

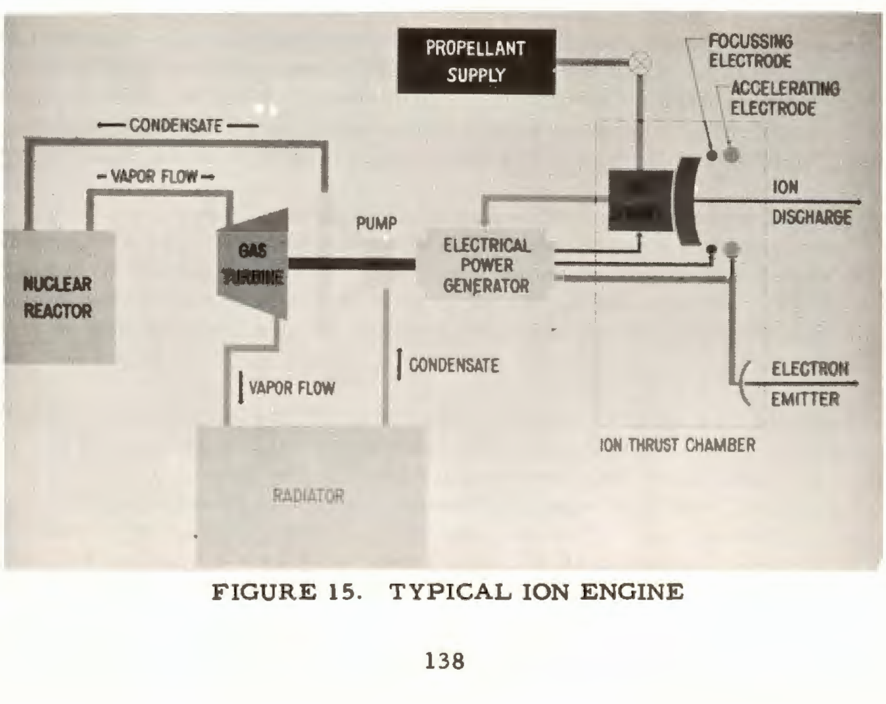

FIGURE 15. TYPICAL ION ENGINE

The propellant or ion source is fed into the ion thrust chamber where power from the electrical power generator forms a plasma. By means of focusing electrodes, the ions are extracted from the plasma and accelerated into a high velocity jet which provides most of the engine thrust. The electrons are meanwhile collected at the ion source and flow back through the generator to an electron emitter. Here the electrons are ejected at a rate calculated to electrostatically balance the ion jet.

The ions and electrons must be separated in this type of device for, in order to obtain a net force, the accelerating field (between electrodes) must act on a region where only one type of charge is present. Therefore, only positive ions are allowed to flow through the accelerating field. If only the positively charged ions are ejected, however, the vehicle would soon attain a large negative charge which would hinder and ultimately prevent further expulsion of ions. This condition is circumvented by the emission of electrons which unite downstream with the ions to once again form a neutral plasma.

The ion source or propellant can be one of several materials ranging from gases such as helium, hydrogen and nitrogen to metallic solids such as cesium or rubidium. The choice of ion source material is directly influenced by the method of ionization whether it be by electron bombardment within an electromagnetic field as in a discharge, or by the interaction between a low ionization potential propellant and a heated metallic plate with a high potential such as tungsten. The first NASA engine of this type will generate about 30 kw, develop about 0.1 lb of thrust with 5000 seconds of specific impulse, and will operate for about a year. The major problems at present are electrode spacing and beam neutralization.

An electromagnetic device forms a plasma but retains it in neutral form, that is, does not separate the ions from the electrons. The neutral plasma is then accelerated electromagnetically to obtain thrust. Current electromagnetic devices are called plasma or magnetohydrodynamic engines.

A sort of hybrid technique is employed in an electrothermal device, where electrical energy in the form of an arc discharge heats a propellant which is then expanded through a conventional converging-diverging nozzle. Arc jet engines are typical of this category with about 2000 to 10,000 seconds expected from the early engines.

Electrical propulsion devices in general can be characterized by large weight, bulky external configuration and low thrust. Because the specific impulse varies inversely as the mass flow, the electrical engines with plasma as propellant achieve about the ultimate in low mass discharge, hence the high specific impulse.

In reviewing the rocket engine performance spectrum, we have first the chemical engine which will yield a maximum of about 400 seconds of impulse using hydrogen and fluorine as propellants. The liquid chemical engine with its higher performance and controllability relative to the solid chemical engine can best be adapted to space missions, where the primary requirement is to lift large vehicles from the earth's surface. Because of the instant readiness and minimum maintenance characteristics inherent in solid chemical engines, they probably have the greater potential for weapon systems such as ICBM's and IRBM's.

Nuclear fission engines expect to double chemical engine performance, producing about 800 to 1000 seconds of impulse. Since the thrust of these engines can be varied up to 75,000 pounds, they will be suitable for both booster and space application. Finally, the electrical devices with their high impulse and low thrust and long operating time, can be used only in outer space missions.