Rocket propulsion

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ROCKET propulsion is one application of the general principle of jet propulsion, which involves forward motion caused by rearward ejection of matter from a propelled body in the form of a high velocity fluid jet. Jet propulsion is fundamentally based on Newton's Third Law of Motion: For every action, there is an equal and opposite reaction. In rocket or jet engines, the force of the momentum of the rearward ejection of the jet stream (generally hot gas molecules streaming from a nozzle) imparts a reverse (i.e., forward) motion or thrust to the device. A very simple experiment illustrating jet propulsion can be conducted by blowing up an ordinary rubber balloon and observing its forward motion when released.

The jet can be generated in two ways: (1) By burning compressed air taken from the atmosphere with hydrocarbon fuel in a combustion chamber, and ejecting the combustion products through a nozzle (*air-breathing engine*, of which the turbojet and ramjet are examples); or (2) by reacting chemicals (propellants), entirely carried, in a similar manner (*rocket engine*, such as liquid bipropellant, liquid monopropellant, or solid propellant system). Both processes produce gases at high temperatures and pressures. This heat energy is subsequently converted to the kinetic energy of a high velocity gas stream by expansion in a nozzle. The chemical rocket engine, not dependent upon air as its oxidizer source, can thus propel a spaceship, which moves in a virtual vacuum, except when landing on a planet.

Let's look further into some rocket propulsion fundamentals. The momentum thrust of the gas stream is given in simplified form by the equation, $F = mv_e$, where m is the mass flow rate of gas and v_e is the exhaust gas velocity relative to the rocket. Conversion to engineering units gives the expression, $F = \dot{w}/g\ v_e$, where F is in pounds of force, w is in lb/sec, and v_e is in fps.

If the nozzle exhaust gas pressure is greater than the atmospheric pressure, additional pressure thrust is created, amounting to $A_e\ (p_e - p_0)$, where p_e and p_0 are nozzle exit and atmospheric pressures respectively, and A_e is the cross-sectional area of the nozzle. One can see from this term why a rocket engine designed for operation in the lower atmosphere will deliver more thrust at high altitudes.

The thrust expression then would be, $F = \dot{w}/g\ v_e + A_e\ (p_e - p_0)$. This equation leads to a term called *effective exhaust velocity* (c):

$F = c\ (\dot{w}/g)$, where $c = v_e + A_e\ (p_e - p_0)g/\dot{w}$. Maximum thrust is obtained for any given set of chamber conditions when $p_e = p_0$; this is referred to as the "optimum expansion ratio," and under such conditions, by definition, $v_e = c$.

It is possible to calculate the exhaust velocity (v_e) from the properties of the working fluid. It is equal to the enthalpy change of the gas during adiabatic expansion, expressing the change of heat energy into kinetic energy: $v_e = \sqrt{2J\ g\ (H_c - H_e)}$, where J represents the mechanical equivalent of heat, and H_c and H_e the enthalpy of the chamber gas and nozzle exhaust gas, respectively. Since enthalpy is largely a function of gas temperature and the *number* of molecules of gas present per unit weight (which would be a maximum for a low-density gas), one can see qualitatively the importance of achieving a high combustion temperature and low molecular weight gas product.

By assuming isentropic flow in the nozzle one can also calculate v_e in terms of nozzle inlet and exit conditions:

$$v_e = \sqrt{2\ g\ RT_c\left(\frac{k}{k-1}\right)\left[1 - \left(\frac{p_e}{p_c}\right)^{(k-1)/k}\right]}$$

where T_c is the chamber temperature, R is the gas constant, k the ratio of specific heats (at constant pressure and volume) c_p/c_v, and p_e and p_c the exit and chamber pressure, respectively. This expression is based on a number of assumptions, including perfect gases, no heat or friction losses, relatively small chamber velocity, and chemical equilibrium in the chamber, unchanging during expansion.

Although efficiencies are not commonly used in designing rocket motors, they are useful in evaluating energy conversions in the system. Of interest are the following:

(1) Combustion efficiency—The ratio of the actual to the ideal heat of combustion per unit weight of propellant.

(2) Thermodynamic efficiency—The last equation shows that the exhaust velocity (v_e) will be a maximum when the exhaust pressure (p_e) is zero, a situation present only with the ideal case of infinite expansion ratio. Obviously this cannot be realized, since it would require the complete conversion of the enthalpy or thermal energy of the exhaust gases into kinetic energy. However, if such conversion is taken to represent 100 per cent thermodynamic efficiency, the efficiency (E) can be expressed as:

$$E = \frac{v_e^2}{v_{e\max}^2} = 1 - \left(\frac{p_e}{p_c}\right)^{k - 1/k}$$

(3) Propulsion efficiency—The ratio of useful work put into propelling the vehicle to total kinetic energy of the exhaust jet. The useful work is $F\ v_v$, where F is the thrust in pounds, and v_v is absolute velocity in fps. The propulsion efficiency then equals vehicle energy divided by vehicle energy plus residual kinetic jet energy:

$$E = \frac{Fv_v}{F\ v_v + {}^1/_2\ \dot{w}/g\ (c - v_v)^2} = \frac{2v_v/c}{1 + (v_v/c)^2}$$

A plot of propulsion efficiency versus v_v/c shows that the former is a maximum when $v_v = c$, in other words, when vehicle velocity is equal to exhaust gas velocity.

Comparison of Rocket Types

Type	Energy Source	Working Fluid	Means of Acceleration	Exhaust Velocity (fps)
Chemical	Chemical	Combustion products	Thermal expansion	12,000
Nuclear	Nuclear fission	Heated gas	Thermal expansion	25,000–30,000
Solar	Solar	Heated gas	Thermal expansion	25,000–35,000
Arc	Nuclear	Plasma	Thermal expansion and/or electromagnetic field	40,000–50,000 60,000–150,000
Ionic	Nuclear	Ions	Electromagnetic field	150,000–500,000
Fusion	Nuclear	Fusion products	Thermal expansion, electromagnetic field	1,800,000

Two other terms widely used in the rocket literature are *mass ratio* and *burnout velocity*. The mass of the rocket (M) consists of two parts, propellant (m_p) and inerts (m_i); $m_p/m_p + m_i$ is called the mass ratio. During rocket flight, the inert mass remains constant but the propellant mass decreases at a rate equal to the loss of propellant gases through the nozzle. The acceleration of the rocket can be expressed as $d_v/dt = -c\ dM/dt$, which integrated over the rocket burning time gives $v_b = c \ln (1 + m_p/m_i)$, where v_b is the terminal velocity at burnout. Achievement of proper burnout velocity is, of course, very important for successful satellite orbiting or escape from the earth's atmosphere.

Specific Impulse

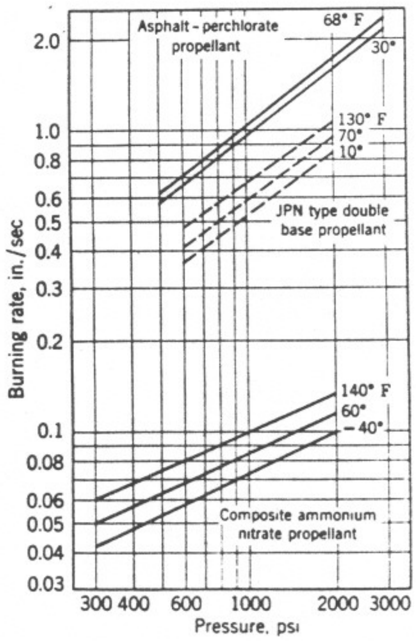
It is convenient for rocket motor designers to use as a performance parameter the thrust delivered per unit weight of propellant consumed per second. This ratio, specific impulse, is defined as: $I_{sp} = F/\dot{w} = c/g$. When the rocket exhaust gases expand fully to the pressure of the surrounding atmosphere, $I_{sp} = v_e/g$. Total impulse delivered is the integral of thrust (F) over a given time (t), and can be expressed as: $I_t = \int I_s\ w\ dt$. For constant thrust, $I_t = I_s W$, where W is the total weight of propellant consumed.

Based on the equation $I_{sp} = v_e/g$ and the equation for v_e on page 7, one can calculate a theoretical value for specific impulse in terms of the properties of the combustion gases:

$$I_s = 9.8\sqrt{\frac{k}{k-1}\left(\frac{T_c}{M}\right)\left[1-\left(\frac{p_e}{p_c}\right)^{k - 1/k}\right]}$$

where M is the average molecular weight of the gas mixture and other terms are as previously noted. Here, again, we can see the importance of the highest possible gas temperature and lowest possible gas molecular weight, as well as high chamber pressure, for maximum values of specific impulse. (In other words, hydrogen gas taken to high pressure and temperature and then expanded through a nozzle would provide an ideal rocket system.) Such conditions as high chamber temperature and pressure would require heavy chamber walls, with consequent low mass ratio; obviously the optimum combination must be selected. A consideration of the ratio total impulse/total motor weight will show, for example, that a system with an I_{sp} of 200 and mass ratio or

Burning Characteristics of Some Typical Solid Propellants



0.5 is equivalent on a weight basis to a system of lower I_{sp} of 175 but higher mass ratio of 0.575.

The energy in the hot chamber gases is derived from the breaking of initial chemical bonds present in the starting propellant components, and the formation of new and stronger bonds in the product gases. The more extreme these differences are, the better the energy release. For example, in the reaction $H_2 + F_2 \rightarrow 2HF$, the relatively weak H–H and F–F molecular bonds are broken, and the very strong H–F bond formed; in the process considerable energy is released. This fact, coupled with the low molecular weight of HF, makes the hydrogen-fluorine combination one of the best available, from an impulse standpoint.

The above calculation for specific impulse is based on frozen equilibrium (or constant-composition) basis, which assumes that no further chemical reaction occurs during the expansion because of the extremely short contact times in the nozzle (of the order of 0.1 milliseconds). On the other hand, the shifting equilibrium basis assumes that the gases achieve thermal and chemical equilibrium during the expansion process, so that there is partial reassociation of the dissociated molecules (due to the lower temperatures), with consequent further energy release. Therefore the shifting basis yields a higher value for specific impulse.

The liquid bipropellant system consists of a suitable oxidizer-fuel combination, which would ideally have the following properties:

(1) High specific impulse (i.e., high energy per pound of material reacted; low-molecular-weight exhaust products).

(2) Boiling point greater than 90 C (to avoid insulation and/or heavy-walled storage tanks).

(3) Freezing point below −60 C, so that propellants may be handled as liquids over a wide temperature range.

(4) High viscosity index (i.e., slight viscosity changes over a wide temperature range).

(5) Density greater than 0.9 grams/cc (to minimize storage tank weight).

(6) Ease of ignition (a fuel-oxidizer combination which ignites spontaneously on contact is called "hypergolic").

(7) Nonsensitivity to mechanical and thermal shock.

(8) Thermal properties suitable for use as a motor-wall coolant (good thermal stability, high specific heat, and thermal conductivity).

(9) Nontoxic to humans.

(10) Noncorrosive to common materials of construction.

(11) Low cost and plentiful supply (i.e., good logistics).

(12) Possess good long-term storage characteristics (no chemical decomposition, reaction, or gas-buildup).

Ideal Not Yet Realized

Obviously the ideal propellant combination is yet to be found; all systems represent compromises of the above. The table at the bottom of page 10 lists some common propellant combinations.

A liquid monopropellant is a single liquid capable of sustained autodecomposition under the proper conditions. It may be a mixture of several components, such as nitric acid and aniline, or a single compound, such as nitromethane. Obviously the handling of one liquid rather than two has some attractive features, such as fewer valves and less plumbing. The table here lists the performance of several common monopropellants for $p_c/p_0 = 600/14.7$, optimum expansion, and frozen equilibrium.

Material	Theoretical I_{sp}, (sec)
Ethylene oxide	170
90% hydrogen peroxide	144
Ethyl nitrate–propyl nitrate (60/40)	195

The liquid bipropellant system in simple form involves a means for bringing fuel and oxidizer from the storage tanks to the injectors, flow control, and a combustion chamber with expansion nozzle. The storage tanks are generally kept under moderate gas pressure, to maintain a constant head on the pump and to minimize pump cavitation. If the pressure is sufficiently high, this can be used to force the propellants to the chamber; more commonly a turbopump is used (i.e., a centrifugal pump driven by a turbine which is in turn powered by gases from a special gas generator).

These operating principles of a liquid bipropellant motor can be illustrated with reference to the first stage of the Vanguard satellite-launching vehicle, which is shown diagrammatically below. Using kerosene and liquid oxygen, it had a nominal burning time of 150 sec and a thrust rating of 27,000 lb at sea level. The steel thrust chamber was regeneratively cooled by kerosene flowing through helical passages between the chamber's inner and outer shells, on its way to the injector head. The chamber, mounted on gimbals, was hydraulically activated so that the direction of the thrust vector could be changed to provide pitching and yawing moments which controlled the vehicle's attitude during first stage flight. The turbopump delivered propellants as desired and also drove the hydraulic pump for the control system. Power for the turbopump was derived from high-temperature steam, provided by the decomposition of 90 per cent hydrogen peroxide as it passed through a silver-screen catalyst chamber. The second stage of Vanguard was a white fuming nitric acid (WFNA) and unsymmetrical dimethyl hydrazine (UDMH) bipropellant system; the third stage was a solid propellant rocket. The illustration on page 10 shows the assembled Vanguard vehicle—71 ft long, 45 in. in diam (first stage), and 22,600 lb loaded for launching.

Solid Propellants

Solid propellants are solid mixtures (which may be homogeneous or heterogeneous in composition) of an oxidizer and fuel capable of sustained combustion if initiated. In use today are two general types: *double-base* (homogeneous), consisting of cellulose nitrate polymer plasticized with high-energy liquid (such as glycerine trinitrate), where the nitrate grouping is oxidizer and the $-CH_2-$ moiety the fuel, and *composite* (heterogeneous), consisting essentially of a polymeric binder matrix, such as a plastic or rubber (which also serves as the fuel), and a crystalline oxidizer, such as ammonium nitrate or perchlorate. Both types of solids usually contain a number of special ingredients for controlling burning rate, physical properties, etc.

After suitable mixing of ingredients, followed by certain other processing steps, the double-base propellants are processed into the desired grain shape by either mechanical extrusion or casting followed by a thermal curing or "setting" period.

Composite propellants are the oldest form of rocket fuel. Black powder, used for centuries, consists of potassium nitrate oxidizer with sulfur used as both binder and (with charcoal) as fuel. (It is still used as an igniter material today.) Oxidizers which have been used include sodium nitrate, potassium nitrate, ammonium nitrate, potassium perchlorate, ammonium perchlorate, and lithium perchlorate. The two most commonly used are ammonium nitrate and ammonium perchlorate; the nitrate is best as regards price and availability (it is a basic fertilizer ingredient), but the perchlorate possesses marked advantages in density, available energy and processing capability.

Many types of polymers have been used as binder materials, e.g., synthetic rubbers of various sorts, polystyrene, and polyvinyl chloride. Desirable properties of a binder include high energy, which will give low-molecular-weight gaseous combustion products; proper processing characteristics; and proper physical properties so that resulting propellant meets stringent physical handling requirements.

Although composites can also be extruded into proper grain shape, a semi-liquid polymer that can be mixed with oxidizer and cast as a slurry offers many advantages, such as lower cost and larger motor size. Ideally, such a slurry should set or cure to a solid within a reasonable time-temperature combination.

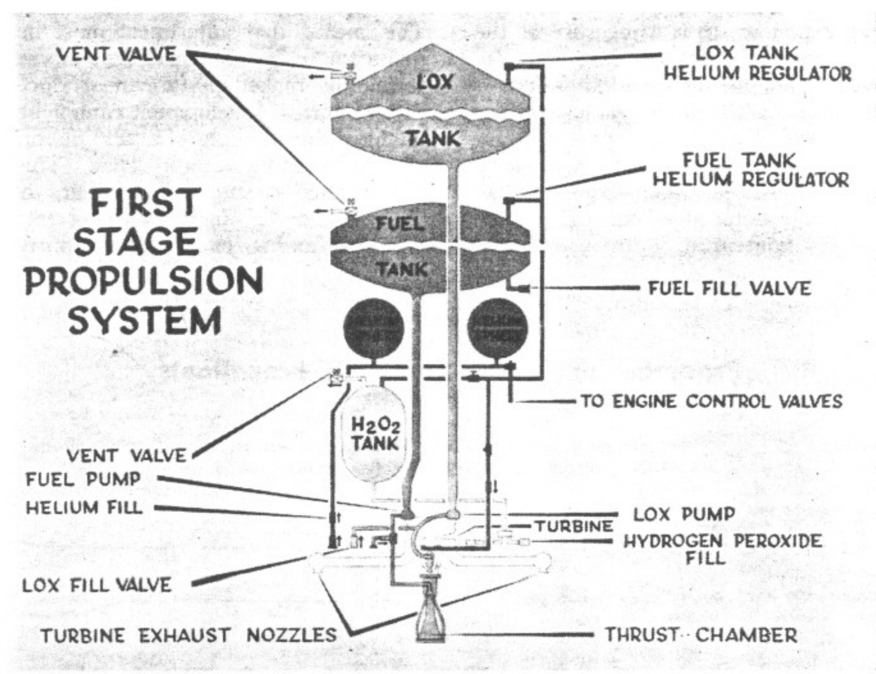
A typical composite would contain 75 per cent ammonium perchlorate, 20 per cent polymer binder and 5 per cent of various additives, which might consist of liquid plasticizer, stabilizers against storage deterioration, and ballistic modifiers for burning rate control. The essential operations of preparing a cast propellant therefore include oxidizer grinding, weighing, conveying, mixing, casting, and curing.

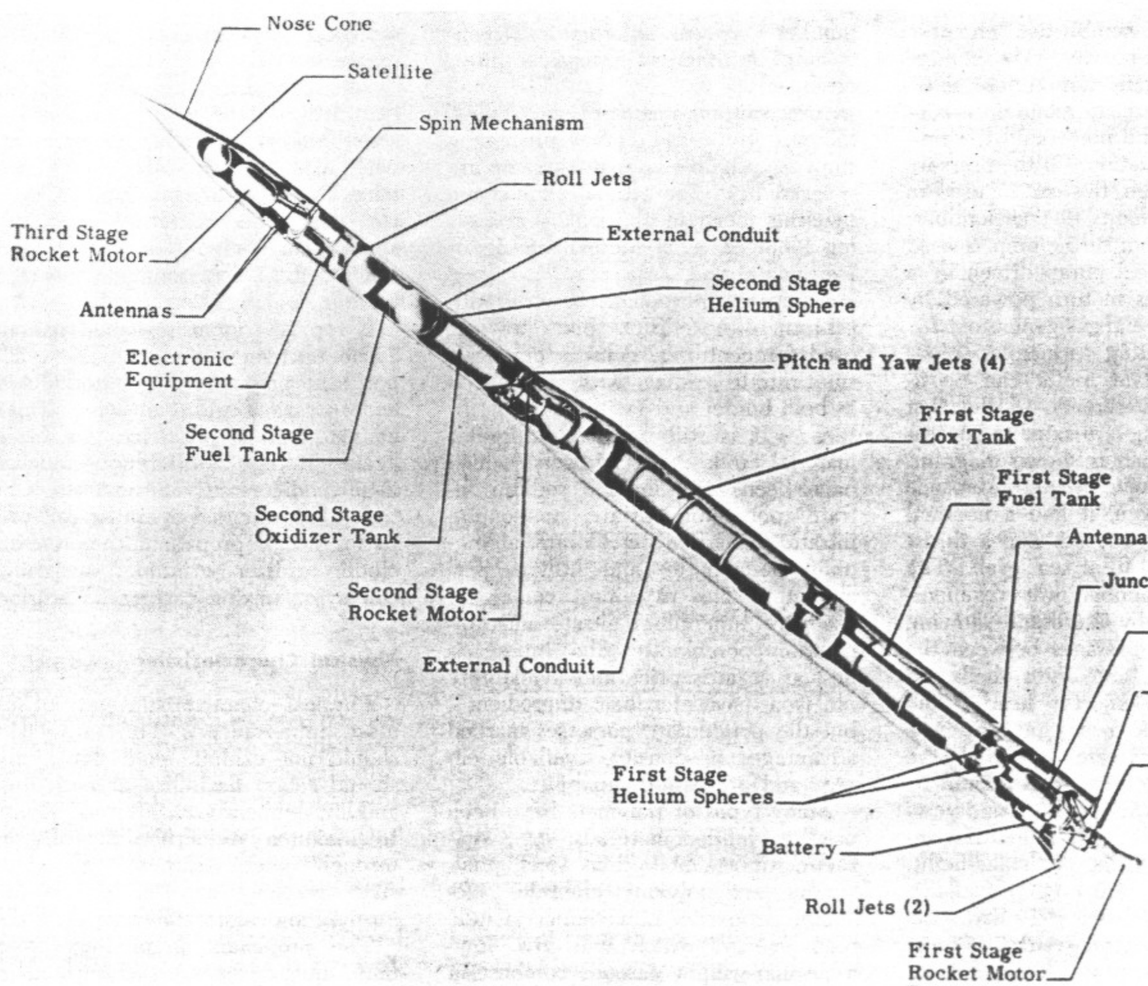
Physical Characteristics

Physical characteristics are of utmost importance. The propellant should not exhibit "cold flow," and should retain flexibility at both high and low temperatures. It must be able to maintain its structural integrity through wide temperature cycles, since cracking can lead to erratic burning and motor blowups.

The propellant grain burns very uniformly on the exposed surfaces at a rate which is primarily a function of the pressure in the combustion chamber, and the temperature of the pro-

Diagram of Vanguard First-Stage Engine





VANGUARD

pellant grain. The rate is expressed as inches (of surface burned) per second (ips), and for most propellants can be approximated by the empirical equation, $r = ap_c^n$, where p_c is the chamber pressure, and a and n are constants for a given temperature.

For practical propellants, r can have values ranging from 0.03 to 2.5 ips. The exponent, n, is a measure of the sensitivity of the burning rate to changes in pressure, and therefore is an important characteristic of any solid propellant. It is desirable for motor design purposes to have n as low as possible. The exponent can be readily determined for any propellant by measuring burning rates of strands of propellant in a pressurized bomb at various pressure levels, and determining the slope of the logarithmic plot of r vs. p_c.

The graph on page 8 shows such plots for several representative propellants. As can be seen, the burning rate of a particular composition is also quite dependent on temperature. This means that adjustment must be made for temperature changes in determining rocket performance, since rate of thrust development cannot be changed during flight of a solid motor, as it can with liquid propellants. This rate change is shown by changes in the value of "a" in the burning rate equation, and is reflected in the burning rate temperature coefficient at constant pressure:

$$\sigma_p = \left(\frac{\partial \ln r}{\partial T}\right)_p$$

Since burning rate is affected by temperature, this is reflected in the equilibrium chamber pressure change with temperature (at constant k, where k is the ratio of the burning surface area to the nozzle throat area), and is expressed by:

$$\pi_k = \left(\frac{\partial \ln p}{\partial T}\right)_k$$

These coefficients are normally expressed as % change/F. The table on page 11 gives appropriate data for several representative propellants.

A solid propellant motor is essentially a propellant-filled pressure vessel with a nozzle and a means for ignition. The critical design parameters are the chamber pressure and the tensile strength of the metal wall. The equilibrium chamber pressure depends upon the rate at which gas is being generated and the rate at which it is escaping through the nozzle. As an approximation, gas-generation rate is a function of propellant burning area (A_b) and gas escape is a function of

Properties of Typical Cast Solid Propellants

	Composite 1	Composite 2	Double Base
Oxidizer	Ammonium perchlorate	Ammonium perchlorate	Cellulose nitrate
Fuel	Polybutadiene-acrylic acid plus aluminum	Polyurethane	with glycerine trinitrate
Theoretical specific impulse at sea level with 1000 psi chamber pressure	250	238	219
Burning rate, r, ips	0.467	0.227	0.45
Burning rate exponent, n	0.236	0.5	0.61
Temp coefficient of pressure, π_k, %/F	0.115	0.13	...
Density lb/cu in.	0.063	0.062	0.057

nozzle throat area (A_t). Therefore we can say:

$$p_c \cong \left(\frac{A_b}{A_t}\right)^{1/1-n} = k^{1/1-n}$$

To develop high thrust, both p_c and A_t should be large, which means that the burning rate area should also be large.

This can be accomplished with an *internal grain burning* by having a central perforation to give large surface area. This perforation is often in the shape of a star, since a star grain can be designed so that the burning area remains constant during the entire run. The figure below shows a cross-sectional view of a typical star grain.

Another very important improvement in solid propellant design has been *case-bonded* propellant. With the propellant bonded directly to the chamber wall, and the burning progressing from the interior toward the chamber wall, the wall is effectively insulated from the hot gases by the propellant itself. Therefore "cold-strength" design can be used, with resulting lightweight chambers. For this reason, and the natural simplicity of the solid propellant system, high mass ratios can be achieved (93 per cent has been suggested as feasible).

It must be stated emphatically that both liquid and solid propellant rockets have their respective area of clean-cut superiority, and both will continue to be used. We may briefly summarize the main advantages of each:

Solids: Lower cost, simplicity of design (and therefore higher reliability), ease of handling and start-up (and therefore maximum "readiness").

Liquids: Higher potential specific impulse, ability to be stopped, restarted and throttled (and therefore easier to "program" a specific mission).

Many rocket propulsion schemes have been suggested which do not depend on the energy of a chemical reaction for thrust generation. These can be classified as follows.

Nuclear Fission: A suitable working fluid (such as hydrogen or ammonia gas stored in the rocket) is heated up by passage through a nuclear reactor, and then expanded through a nozzle.

Solar: Solar radiation is reflected by spherical mirrors to a heat exchanger where it heats up the working gas.

Nuclear Fusion: The use of a thermonuclear power plant as an energy source.

Ionic: The generation of ions and electrons by heating or irradiating a suitable surface; the ions are then accelerated by an electromagnetic field. The electric power required is generated by a nuclear reactor and suitable generating equipment. This process can only be carried out in a vacuum; outer space would provide such an environment.

Free Radicals: If a free radical such as $H^{\bullet}$ could be stabilized and stored, and at the appropriate moment of nozzle expansion, give up its rather considerable energy of recombination, very high specific impulse values could be achieved. At the moment, the prospects of such stabilization in a practical system seem remote.

Arc: Creation of a "plasma" of hot ionized gases, by means of an electric arc, and subsequent expansion.

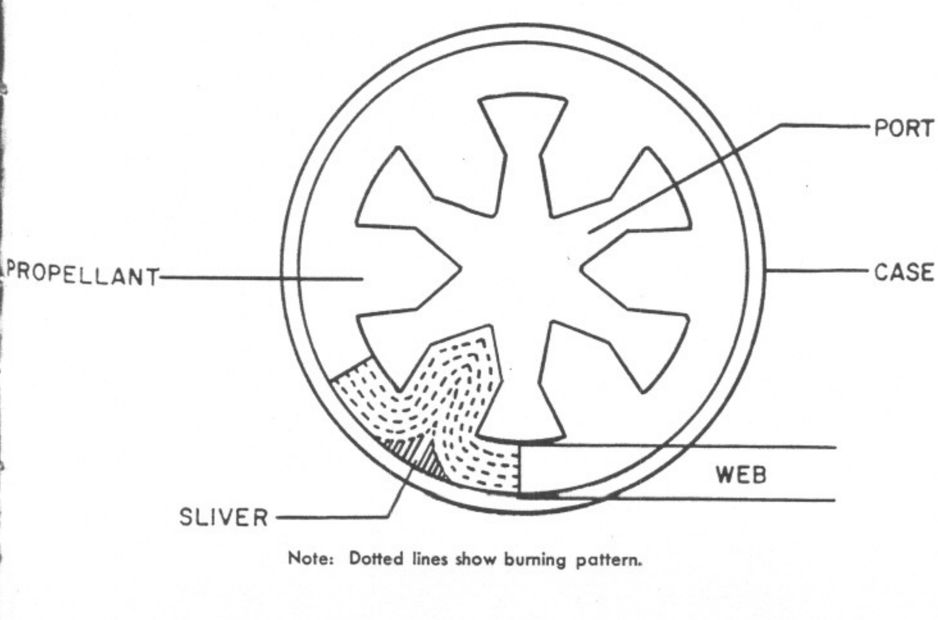
The table on page 7 compares these systems. It should be noted that despite the high exhaust velocities realized, the weight of power plant per pound of thrust realized is very high for these systems. For example, the thrust to engine weight ratio for a chemical or nuclear rocket is about 10–80; for an ionic rocket it is in the order of 5×10^{-3} to 5×10^{-5}.

Theoretical Performance of Several Liquid Propellants

Oxidizer	Fuel	Mixture Ratio, Oxidizer/Fuel	Theoretical Combustion Temp (Deg F)	I_{sp} (sec)
RFNA (22% NO_2) (red fuming nitric acid)	JP-4 (Jet Fuel)	4.1	5150	238
	UDMH (Unsymmetrical dimethyl hydrazine)	2.6	5200	249
N_2O_4 (nitrogen tetroxide)	Hydrazine	1.1	4950	263
Hydrogen peroxide (90%)	Hydrazine	1.5	4170	252
Oxygen (lox)	JP-4	2.2	5880	255
	Ammonia	1.3	4940	232
	UDMH	1.4	5650	261
	Hydrazine	0.75	5370	279
	Hydrogen	3.5	4500	363

Conditions: $p_c/p_0 = 500/14.7$; optimum expansion; frozen equilibrium.

Cross-sectional View of Solid Propellant Grain



Note: Dotted lines show burning pattern.

Suggested Reading

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