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FOREIGN TECHNOLOGY DIVISION

AIR FORCE SYSTEMS COMMAND

INTRODUCTION TO BALLISTICS
ROCKETS AND ROCKET BALLISTICS (SELECTED PARTS)

By

Waldemar Wolff

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EDITED TRANSLATION

INTRODUCTION TO BALLISTICS
ROCKETS AND ROCKET BALLISTICS (SELECTED PARTS)

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3.1. THE CHEMICAL PROPULSION SYSTEM

In the chemical propulsion system, energy liberated by an exothermal chemical reaction between two substances, i.e., an oxidizer and a fuel, is converted. The term chemical propellant is applied to pairs of liquids or solids that produce the desired reaction. In general, the term oxidizer implies only oxygen. However, the scope of the terms oxidizer and fuel must be expanded beyond the limits of other definitions. To expand these conceptions, we use the part of the periodic system indicated below as a point of departure.

H	1 Brennstoff				2 Oxydatoren	
Li	Be	B	C	N	O	F
Na	Mg	Al	Si	P	S	Cl

1) Fuels; 2) oxidizers.

The elements to the right of neutral nitrogen and their compounds will be called oxidizers here; those to the left and their compounds will be known as fuels.

The oxidizers of greatest importance at the present time are liquid oxygen and compounds with high oxygen content, such as nitric acid and perchlorates. Fluorine has recently been attracting increasing interest. The fuels used most extensively at the present time are hydrocarbons. Compounds of nitrogen and hydrogen, such as ammonia, hydrazine and others, are also to be taken into consideration. More and more interest attaches to the use of boron and lithium compounds. The chemical propellants are characterized by very high potency. Large quantities of energy are liberated in short times. In this respect, the chemical propellants are superior to all others. The combustion products are heated by the heat liberated in the chemical reaction. They flow out through a nozzle. The heat liberated (disordered motion of the molecules) is for the most part converted into directional motion (the kinetic energy of the gas stream). If a small amount of gas ΔM flows at the velocity v_e through the exit cross section of the nozzle, its kinetic energy is

$$\Delta E_{kin} = \tfrac{1}{2} \Delta M v_e^2.$$

Since a quantity m flows out in a unit of time, the power N is

$$N = \tfrac{1}{2} m v_e^2. \qquad (1)$$

The relationship

$$S = m v_e$$

applies for the thrust. We can therefore write (1) in the form

$$N = \frac{S v_e}{2}. \qquad (2)$$

Example: What is the power output of a rocket propulsion system with a thrust of 68 Mgf and an exhaust velocity v_e = 2200 m/sec? We obtain

$$N = \tfrac{1}{2} \cdot 9{,}81 \cdot 68000 \text{ kg m/s}^2 \cdot 2200 \text{ m/s}$$
$$N = 7{,}34 \cdot 10^8 \text{ W}$$
$$N = 734 \text{ MW}$$
$$(1 \text{ MW} = 10^6 \text{ W})$$

Powerplants with 68 Mgf of thrust are used in the Atlas rockets. The example shows us that the outputs of the larger power plants are of the same order as those of modern megawatt electric power plants. The Klingenberg power plant (Berlin) has an output of 270 Mw; Krasnoyarsk produces 4000 Mw.

An output of 20,000,000 hp was indicated for the booster rocket of the "Wostok" spacecraft with which Gagarin first orbited the earth. Since 1 hp = 735.5 w, this corresponds to

$$N = 2 \cdot 10^7 \cdot 735{,}5\ \mathrm{W} = 1{,}471 \cdot 10^{10}\ \mathrm{W}.$$

If v_e is known, the thrust S can be determined from the above on the basis of (2). We have

$$S = \frac{2N}{v_e}.$$

Assuming v_e = 2500 m/sec, we obtain

$$S = \frac{2 \cdot 1{,}47 \cdot 10^{10}\ \mathrm{kg\,m^2/s^3}}{2500\ \mathrm{m/s}} = 1{,}18 \cdot 10^7\ \mathrm{kg\,m/s^2}$$

or

$$S = 1{,}20 \cdot 10^6\ \mathrm{kp}.$$

If we assume further that the thrust is 1.5 times the launch weight of the entire booster rocket, we obtain $M_0^{(1)}$ = 800,000 kg. The mass of the spacecraft, including that of the pilot, comes to 4725 kg. Accordingly, $M_0^{(1)}/M_N$ = 800,000 kg/4725 kg = 170 a favorable figure. Thus we have confirmed an earlier conjecture. If we had assumed a thrust-to-launch-weight ratio larger than 1.5, the value of $M_0^{(1)}/M_N$ would have been much smaller. The high potency of chemical propellants is not seen only in rockets. Chemical propellants are also used for artillery and small arms. Since the combustion processes unfold considerably more rapidly in these examples, the power outputs are even higher. In [2], page 20, it is calculated by way of example that the power output of a 150-mm fieldpiece is 774,000 kw. High outputs are characteristic of all chemical propellants. This does not imply, however, that this is an advantage in all cases. We shall discuss this point in greater detail in the sections devoted to nonchemical propulsion systems.

3.1.3. Solid-Fuel Propulsion

3.1.3.1. The characteristic properties of solid rockets

Black powder was used in China as a propellant for rockets several centuries before our era. Black powder is a mixture of sulfur, saltpeter and charcoal. The saltpeter is the oxidizer and the charcoal and sulfur are the fuels. The function of the sulfur is to make the combustion as uniform as possible. Artillery rockets with black-powder cartridges were used extensively during the 19th Century. The artillery rockets built by William Congreve were used by the English in 1807 during the bombardment of Copenhagen. Particular attention was devoted to the development of rocket artillery in Russia during the 19th Century. The works of General A.D. Saziadko (1779-1837) and those of the Artillery General K.J. Konstantinov (1819-1871) are worthy of particular attention. The latter conducted penetrating scientific investigations that enabled him to improve range and accuracy considerably. In his 1856 book entitled "Ueber Kampfraketen" [Battle Rockets], he even referred to the possibility of improving the accuracy of the fin-stabilized rocket by causing it to spin around its longitudinal axis. The Russian Army had special rocket troops. Then the introduction of rifled cannon, which were superior to the artillery rockets as regards range and accuracy, led to the gradual disappearance of rockets from the arsenal. Only during the Second World War did the development of the solid artillery rocket resume its advance. This was occasioned by its mobility and its firepower and the possibility of using it from aircraft. Solid-fuel rockets are, moreover, suitable for use in assisting aircraft, and combat aircraft in particular, at takeoff.

The barrel of a fieldpiece must withstand a high gas pressure, and its carriage must be able to absorb the recoil. The rocket launcher, on the other hand, is simply an aiming device. Thus the mass of the launcher does not differ appreciably from that of the rocket, while the mass of a 150-mm fieldpiece is about 200 times that of its projectile. By virtue of their low weight, antitank rockets can be carried up to the very front line by the infantry. Rockets of 200-mm and larger caliber can be fired from aircraft. An artillery piece with a mass of about 14 tons would be required for a projectile of the same caliber.

The principal respect in which the solid rocket differs from the liquid rocket is the greater simplicity in manufacturing and handling it and the resulting higher reliability. In a solid rocket, all of the propellant is accommodated in the combustion chamber in the form of one or more powder grains. No propellant-handling system is required.

The advantages of simplicity and reliability are partly offset by certain disadvantages, such as temperature sensitivity (dependence of the combustion process on the temperature of the propellant charge) and short burn times (1 sec and shorter up to 30 sec). When the rocket is ignited, the combustion process lasts until the propellant charge has been consumed. Further disadvantages are found in the facts that it is difficult to produce large propellant cartridges and the production of solid propellants costs 30 to 50 times as much as the average for liquid propellants. In spite of these disadvantages, solid propellants are being used more and more extensively, not only for artillery rockets, but quite generally. Solid-fuel units are used in the upper stages of intercontinental ballistic missiles and satellite boosters. The development of solid-fuel systems is a focus of attention today. In general, solid rockets have higher combustion-chamber pressures than liquid rockets. They range from 30 kgf/cm^2 to 150 kgf/ /cm^2. However, the higher pressure requires thick combustion-chamber walls. This disadvantage is relieved to some extent by the short burning times, which enable us to dispense with the special cooling systems required for liquid rockets. Solid-fuel propulsion systems generally have exhaust velocities somewhat lower than those obtained with liquid propellants. However, since their densitites are 20 to 80% higher, this is no disadvantage.

3.1.3.2. The combustion process

The combustion process is treated in detail in [2]. We make particular reference to Sections 5.31, pages 40 to 47 and 5.34, pages 66 to 68 in that source. Here we shall simply repeat the statements made in connection with the concept of linear combustion rate. It is customary to denote this by r in the technical literature on rockets. Solid-fuel elements burn in closed spaces, since they contain the oxygen required for combustion. If a solid-fuel grain is ignited in a closed space, it will at first burn only at the place at which ignition occurred. Now this results in the formation of hot combustion gases, which expand to fill the entire space. Under the influence of these gases, combustion is initiated over the entire exposed surface of the grain. Now the fuel burns in layers perpendicular to its surface. Assume that a layer of thickness e is burned away during a time t, i.e., a layer r = e/t thick burns per unit of time. This is the linear combustion rate. If the combustion rate is not constant, the differential quotient de/dt appears in place of e/t. That is, we have

$$r = \frac{de}{dt}. \qquad (1)$$

r depends on the chemical nature of the powder and the instantaneous pressure at which combustion is taking place.

Vielle derived the following relationship, which is also known as the law of combustion:

$$r = \frac{de}{dt} = ap^n \qquad (2)$$

Here, a and n are constants that depend on the chemical nature of the powder. Vielle worked on the developed of smokeless powder and assumed n = 2/3. Later, Schmitz established as a result of detailed investigation that it would be more accurate to set n = 1. Then Muraour and Aunis proposed the following combustion law on the basis of theoretical considerations:

$$\frac{de}{dt} = a + bp. \qquad (3)$$

Here a and b are constants; a takes conductive heat transfer into account, while b is a correction for energy transfer by molecular collisions.

The two combustion laws (2) and (3) were designed for small arms and artillery pieces in which passage of the projectile through the barrel takes 0.001 sec to 0.060 sec, depending on caliber. The time required for combustion of the charge is considerably shorter, representing only a fraction of the time in the barrel. In solid-fuel rockets, on the other hand, the combustion times range from 1 sec or less to about 30 sec. In spite of this, the combustion laws set forth above are used for rockets even today. A giant cannon with a caliber of 800 mm and a projectile mass of 7200 kg was developed in Germany in 1938-1941. v_0 was 700 m/sec and the charge mass was 1810 kg. The time taken by the projectile to pass through the rifled part of the barrel, which was about 42 m long, was 0.29 sec, or much longer than for a normal weapon. It is noteworthy that the results of interior-ballistic calculation deviated sharply from test results. Even at that time, the author noted that this was a result of unjustified transfer of the combustion law. (It was assumed that $r = ap$.) It follows clearly from Soviet experiments that the combustion law $r = ap$ is valid only for processes in normal artillery pieces and small arms. A discussion of this may be found in [2], 5.34, pages 66 to 68. In combustion processes of greater duration, such as those occurring in solid rockets, the temperature of the combustion gases works on the remainder of the powder grain and thus increases the value of a in (2). Accordingly, a is no longer a constant. As was shown in [2], the error can be compensated by taking $n < 1$ instead of setting $n = 1$. On the basis of stability investigations for the combustion process, we obtain the condition $n < < 1$ for rocket propulsion elements if we use (2) as a basis. Now, however, the invalid assumption that the exponent n and the coefficient a depend only on chemical composition, i.e., are constants, is generally adapted. Naturally, the thickness of the powder grain and the duration of the combustion process also affect these quantities. A similar statement also applies to (3). Hence the exponent n must be determined experimentally, if possible by testing propellant charges with the proposed dimensions.

As extreme limits for the linear combustion rates, which depend on pressure and composition, we may take 0.025 cm/sec and 25 cm/sec. Commonly encountered combustion rates lie between 0.1 and 5 cm/sec.

Of particular significance for rocket engineering is the fact that it has been possible to develop solid propellants whose combustion rates are independent of pressure in a certain range. Here we are concerned with the so-called composite propellants. As an example, there is a propellant whose combustion rate is 0.4 cm/sec at 20 kgf/cm^2 and increases with rising pressure to reach 0.6 cm/sec at 70 kgf/cm^2 and then remains constant as the pressure increases further.

3.1.3.3. Solid propellants

It has already been pointed out that up to the end of 19th Century, black powder was the only propellant available for rockets. The development of modern smokeless powders began in the second half of the 19th Century. Black powder only was used in artillery pieces and small arms up to about 1885. Then there was a gradual transition to smokeless powder ([2], 4., pages 21 to 24).

Nitrocellulose is the basis for any modern powder for use in artillery and small arms. Although nitrocellulose is an explosive, it can be converted by solution in a mixture of ether and alcohol into a uniform mass that does not decompose explosively and hence can be used as a propellant after the solvent has been driven off. This propellant is known as nitrocellulose powder and is used in some armies. If the nitrocellulose is dissolved in the liquid explosive nitroglycerine instead of in ether and alcohol, we likewise obtain a propellant, nitroglycerine powder. The nitroglycerine remains in the propellant; essentially, therefore, nitroglycerine powder is a mixture of nitrocellulose and nitroglycerine. The smokeless powders have an important advantage in their formability. Attempts have also been made to shape black powder by pressing. It is has been found, however, that the powder grains crumble, so that it is impossible to regulate the combustion process.

Rocket propellant charges were developed on the basis of nitroglycerine powder, which, as we have already noted, was a mixture of nitrocellulose and nitroglycerine. The oxidizer and fuel are parts of the same molecule. Such propellants are known as homogeneous or double-base propellants (monergolic propellants). Diethyl phthalate is added to the mixture to increase its plasticity. All nitrogen-based propellants tend to decompose. This can be avoided by addition of a substance known as a stabilizer – for example, diphenylamine or centralite. A typical composition for such a propellant is:

Nitrocellulose	51.4%
Nitroglycerine	43.3%
Diethyl phthalate	4.0%
Potassium nitrate	1.1%
Diphenylamine	0.1%
Nigrosin	0.1%

The development of propellants took a different direction in Germany after the First World War. Glycerine is a component of fat and for all practical purposes can be obtained only from that material. The result was a catastrophic fat shortage in Germany during the First World War. In the development that was now undertaken, a substitute for nitroglycerine was found in diglycol, the raw materials for which are coal and chalk. The new propellant came to be known as diglycol powder. It was found that this powder has a number of attractive properties and that it is also particularly suitable for use in rockets. All solid propellant charges used in the Second World War were (homogeneous) double-base propellants. Even today, such propellants are used in small rockets. Their specific impulse lies at about 200 to 240 kgf-sec/kg, which may be regarded as high values. The combustion rate is relatively high, about $r = 2.2$ cm/sec at 100 kgf/cm^2. The pressure required for uniform combustion is also high. The minimum required pressure is about 40 kgf/cm^2. This, however, imposes narrow limits on designs. The double-base propellants have, at the present time, reached the limit of their development.

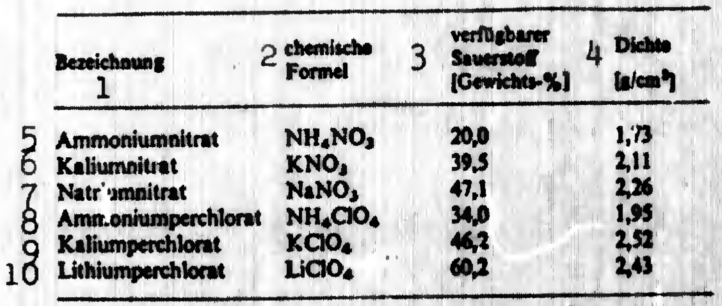
The year 1940 saw the beginning of development work on composite propellants (propergols). They consist of an oxidizer and a so-called fuel binder, which is thus a fuel that simultaneously has the function of cementing the propellant charge into a solid mass and conferring the required mechanical properties on it. The fuel binder is frequently laced with metallic additives, which increase the energy content. With composite propellants, it becomes possible to vary the components within wide limits and obtain compositions with good thermodynamic and ballistic properties.

The oxidizer makes up 70 to 80% of the mixture, while the fuel, which may consist in part of metals, represents 20 to 30%.

The most important solid oxidizers are listed in the table that follows.

TABLE 3.6

The Most Important Solid Oxidizers



	Bezeichnung 1	2 chemische Formel	3 verfügbarer Sauerstoff [Gewichts-%]	4 Dichte [g/cm³]
5	Ammoniumnitrat	NH_4NO_3	20,0	1,73
6	Kaliumnitrat	KNO_3	39,5	2,11
7	Natriumnitrat	$NaNO_3$	47,1	2,26
8	Ammoniumperchlorat	NH_4ClO_4	34,0	1,95
9	Kaliumperchlorat	$KClO_4$	46,2	2,52
10	Lithiumperchlorat	$LiClO_4$	60,2	2,43

1) Name; 2) chemical formula; 3) available oxygen [% by weight]; 4) density [g/cm³]; 5) ammonium nitrate; 6) potassium nitrate; 7) sodium nitrate; 8) ammonium perchlorate; 9) potassium perchlorate; 10) lithium perchlorate.

The thrust is proportional to specific impulse, i.e., to exhaust velocity, and to density. Hence high density is desirable.

A large number of synthetic resins, rubbers and waxes come into consideration as fuel binders. A number of new synthetics are acquiring greater and greater importance. Among these, special mention should be made of polyurethane, which is distinguished by high density as well as by the fact that it is highly elastic and form-stable.

Initially, the specific impulses of the composite propellants were relatively low. The first propellants of this type, which consisted of asphalt and potassium perchlorate, had a specific impulse of 185 kgf-sec/kg. It has recently been possible to raise the specific impulse to 240 kgf-sec/kg and higher. The mixing ratio has a strong influence on the specific impulse and on combustion temperature. The addiabatic exponent γ likewise depends on the mixture ratio, but only slightly.

Of particular importance for design is the fact that a rather wide range of combustion rates is available. Propellants using ammonium nitrate as the oxidizer have low combustion rates. Propellants with ammonium perchlorate have medium and those with potassium chlorate have high combustion rates.

Although the double-base propellants have been used to make grains only for small rockets, the composite propellants served as a basis for the development of very large solid rocket propulsion systems. The largest American solid rocket engine tested thus far delivers a thrust of 113 Mgf for 80 sec, i.e., a total impulse of about 9,000,000 kgf-sec. The V2 liquid rocket had a total impulse of 1,650,000 kgf-sec. The propellant load has a mass of 36 tons. Such large masses impose high requirements as to such mechanical properties as strength, elasticity, ductility and storability. Heavy loads arise at the high accelerations operating during flight, and the pressure of the powder gases also acts on the propellant grain. Flowing at high speed over the propellant charge, the combustion gases produce shearing forces. Any cracks that form break up the grain into smaller pieces during burnoff, and the resulting increase in the burning area raises the pressure in a manner that may be fatal to the rocket.

As the size of the propellant grains increases, the difficulties encountered in manufacture, testing and transport are aggravated. The propellant grains are either excluded or cast. Both methods have limitations as regards the size of the grains that can be produced. In casting, the amount of time during which the mixture retains its casting property is a decisive factor.

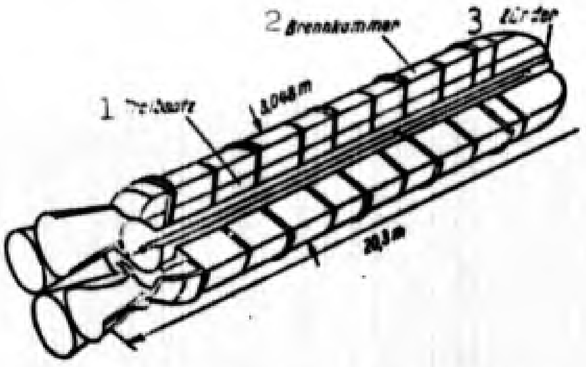


Fig. 3.8. Segmented solid propellant charge with total impulse of 45,000,000 kgf-sec. 1) Propellant charge; 2) combustion chamber; 3) igniter.

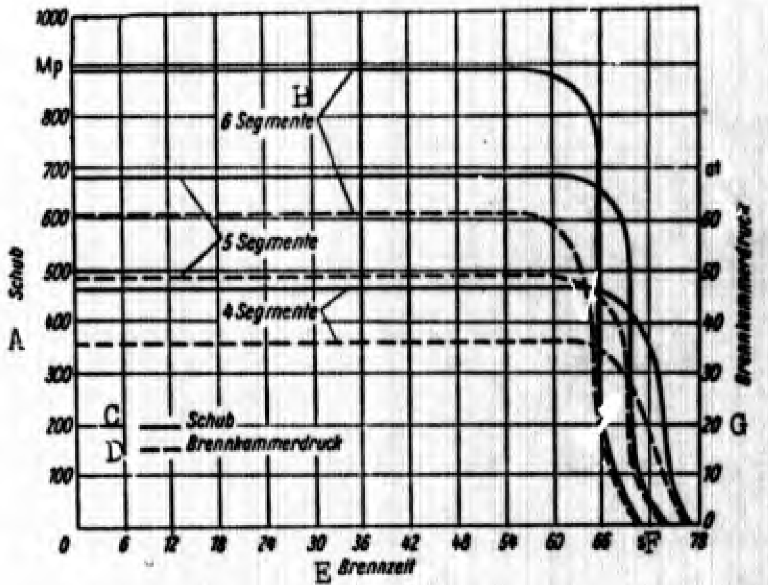


Fig. 3.9. Thurst and combustion-chamber-pressure curves as functions of burning time in a segmented solid propellant charge with various numbers of segments. A) Thrust; B) segments; C) thrust; D) combustion-chamber pressure; E) burning time; F) seconds; G) combustion-chamber pressure.

The shipping difficulties in particular are not to be underestimated. There are three possible ways of circumventing them:

1. Lashing several small powerplants together,
2. Charging or manufacturing solid rockets at the launching pad,

TABLE 3.7.

Comparison of Two Liquid Propellant Combinations with a Solid Propellant

Treibstoff A	B Zustand des Treibstoffs	Dichte [kg/m³] C	D spez. Impuls [kp s/kg]	E Impulsdichte [kp s/m³]
F Alkohol + flüssiger Sauerstoff	G flüssig	990	239	236,6 · 10³
H iger Wasserstoff + Fluor	flüssig G	221	357	78,9 · 10³
I 80% Ammoniumperchlorat + 20% Brennstoffbinder und Zusätze	fest J	1660	224	371,8 · 10³

A) Propellant; B) state of propellant; C) density [kg/m^3]; D) specific impulse [kgf-sec/kg]; E) impulse density [kgf-sec/m3]; F) alcohol + + liquid oxygen; G) liquid; H) liquid hydrogen + fluorine; I) 80% ammonium perchlorate + 20% fuel binder and additives; J) solid.

3. Use of the "building-block" principle.

E.R. Roberts has reported on application of the third principle.

Figure 3.8 shows a design for a segmented solid propulsion system with a total impulse of 45,000,000 kgf-sec. The rocket is assembled at the launching pad. The combustion chamber, 17 m long, consists of five similar parts 3.4 m long with a mass of 39,500 kg. The production of such units is possible with available machinery. Testing and shipment present no particular difficulties. The output of such a rocket can be varied by increasing or decreasing the number of segments by 1. The pressure and thrust then vary accordingly, while the combustion time remains approximately constant (Fig. 3.9).

The solid propellants are far superior to liquid propellants in one aspect that is of particular importance for the design of rockets. Here we refer to the volume-specific impulse or the impulse density. We use the symbol I_{vs} for this quantity. The impulse density is the impulse obtained per m^3 of a propellant, and is thus expressed in kgf-sec/m^3. If we expand this with kg, we obtain

$$\text{kgf-sec/kg} \cdot \text{kg/m}^3.$$

Accordingly, the impulse density is equal to the product of the specific impulse by the propellant density. Propellants with high impulse densities are particularly attractive insofar as they occupy little space, i.e., permit a weight reduction. Table 3.7 presents a comparison of two liquid propellants and a solid propellant. The comparison shows the clear superiority of the solid propellant. The liquid-hydrogen/flourine combination has not as yet found any practical application.

Although it would indeed produce a high specific impulse, it falls quite far short of the other propellants, and the solid propellants in particular, as regards impulse density. Even when we remember that a certain additional free volume is required from the beginning for the solid propellant charges, solid rockets still remain far superior as regards output density.

3.1.3.4. Interior ballistics of solid rockets

The task of interior ballistics consists in determining the required mass of propellant charge, the shape and dimensions of the powder grains and the size of the required combustion chamber on the basis of measurements and calculations. Of prime importance here is the pressure curve, which can be determined experimentally and theoretically.

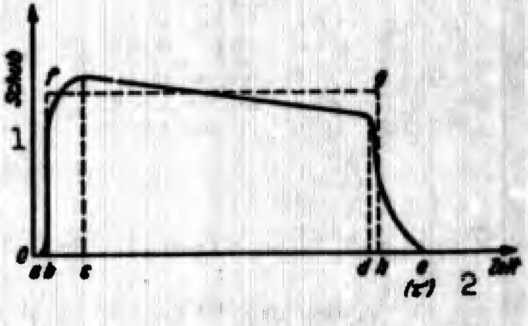


Fig. 3.10. Time curve of the thrust of a solid rocket. 0a) Ignition delay; ac) time allowed for pressure buildup; cd) time of equilibrium pressure; ae) actual combustion time (τ'). ------ = Rectangle of equal total impulse; bf = hg is the effective thrust and bh is the effective combustion time (τ). 1) Thrust; 2) time.

The shape of the pressure curve for a rocket is essentially different from that in an artillery piece or a hand gun. In the rocket, an attempt is made to maintain pressure constant — something that cannot be done in a firearm and would even be disadvantageous in certain respects [2], 6.43. In rockets, whose interior ballistics are simpler than those of a firearm, the thrust is characteristically proportional to pressure. The time curves of thrust and pressure have the same appearance. Figure 3.10 presents a typical thrust-versus-time curve for a rocket. The time 0a is the ignition delay, ac is the time during which the pressure builds up, cb is the time of equilibrium pressure, ae is the actual burning time τ'. The area of rectangle bfgh is equal to the area between the thrust curve and the axis of abscissas, and is therefore equal to the total impulse.

We have

$$I = \int_0^{\tau} S\,dt. \qquad (1)$$

bf = hg is the effective thrust and bh the effective burning time τ. Now the question as to which quantities determine the equilibrium pressure must be answered. This pressure is not there at once, but must first be built up (time segment ac). We assume that the ignition process has advanced to the point at which the entire surface of the powder grain or grains is burning. Part of the gases formed remains in the combustion chamber and raises the pressure in it, while the rest flows out of the nozzle. If F_0 is the total surface of the powder grains, the mass of the gases formed per unit of time is $rF_0\varrho_p \cdot \varrho_p$ is the density of the propellant. If ρ_0 is the density of the combustion gases, then $\rho_0 V_0$ is the mass of the gases present in the free combustion space. V_0 is the free volume, i.e., the combustion chamber volume reduced by the volume of the propellant grain. $d/dt(\varrho_0 V_0)$ is the time change in this mass, i.e., the mass increase during the pressure buildup. If we again denote by m the mass of the gases flowing out through the nozzle per unit of time, the entire process will be described by the relationship: mass of gases formed = change in total mass in combustion space + mass of exhaust gases, i.e.,

$$rF_0\varrho_p = \frac{d}{dt}(\varrho_0 V_0) + m. \qquad (2)$$

Differentiating and inserting the value of m according to (20) in 3.1.1., we obtain

$$rF_0\varrho_p = \frac{d\varrho_0}{dt}V_0 + \varrho_0\frac{dV_0}{dt} + \frac{\Gamma}{\sqrt{RT_0}}F_s p_0. \qquad (3)$$

Finally, the process reaches a condition in which the pressure and density in the combustion pressure no longer change.

Then

$$\frac{d\varrho_0}{dt} = 0. \qquad (4)$$

The free volume V_0 still changes, since the propellant grain continues to burn off. We have

$$\frac{dV_0}{dt} = rF_0. \qquad (5)$$

Applying (4) and (5), we obtain from (3)

$$rF_0\varrho_p = rF_0\varrho_0 + \frac{\Gamma}{\sqrt{RT_0}}F_s p_0 \qquad (6)$$

and from this

$$p_0 = r\frac{F_0}{F_s}(\varrho_p - \varrho_0)\frac{\sqrt{RT_0}}{\Gamma}. \qquad (7)$$

F_0/F_s is the ratio of propellant-charge surface to the narrowest cross section of the nozzle. It is known as the initial area ratio. We shall denote it by the symbol K, i.e.,

$$K = \frac{F_0}{F_s}. \qquad (8)$$

For the internal pressures to be considered for solid rockets, ρ_0 is smaller than 1 to 2% of ρ_p, and we may therefore disregard it. Then Eq. (7) will read

$$p_0 = rK\varrho_p \frac{\sqrt{RT_0}}{\Gamma}. \tag{9}$$

r is a function of the pressure p_0. In accordance with (2) in 3.1.3.2., we set

$$r = ap_0^n.$$

Substituted in (9), this gives

$$\frac{p_0}{p_0^n} = p_0^{(1-n)} = \left[a\varrho_p \frac{\sqrt{RT_0}}{\Gamma}\right] K \tag{10}$$

or

$$p_0 = \left[a\varrho_p \frac{\sqrt{RT_0}}{\Gamma}\right]^{\frac{1}{1-n}} \cdot K^{\frac{1}{1-n}}. \tag{11}$$

All values in the square brackets of Eq. (11) are constants for a given propellant. Accordingly, the equilibrium pressure depends only on the initial area ratio. If we assume that n = 0.8, p_0 is proportional to

$$K^{\frac{1}{1-0.8}} = K^5.$$

It follows from this, however, that any change in the initial area ratio is of incisive importance. If we assume that discontinuities occur due to inadequate strength in the propellant grain and increase its area by only 10%, the pressure will rise by 61% according to (11). The rise would be smaller for small n. It is 21% for n = 0.5. Thus we should strive to secure the lowest possible n.

In 3.1.2., Formula (4), we introduced the characteristic rate c*. It is

$$c^* = \frac{F_t p_0}{m}.$$

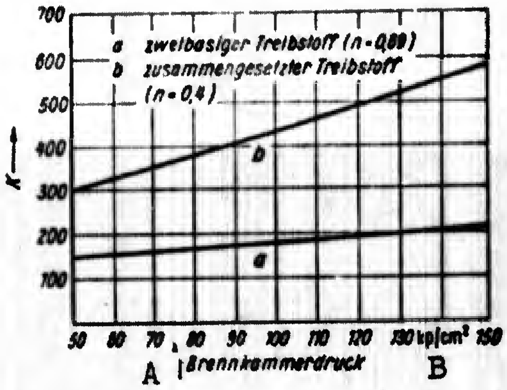


Fig. 3.11. Initial area ratio K as a function of combustion chamber pressure. a) Double-base propellant (n = 0.69); b) composite propellant (n = 0.4). A) Combustion chamber pressure; B) kgf/cm².

c* is a function only of the propellant and can be determined directly by experimental means. There is another relationship that we shall not derive here:

$$c^* = \frac{\sqrt{RT_0}}{\Gamma}. \qquad (12)$$

If this relation is substituted in (11), we obtain

$$p_0 = [a\varrho_p c^*]^{\frac{1}{1-n}} K^{\frac{1}{1-n}}. \qquad (13)$$

It follows from (13) for the initial area ratio that

$$K = \frac{1}{a\varrho_p c^*} p_0^{1-n}. \qquad (14)$$

Figure 3.11 shows the initial area ratio as a function of combustion chamber pressure, line a representing a double-base propellant with n = 0.69 and b a composite propellant with n = 0.4. We may regard 100 and 2000 as the extreme limits for the initial area ratio.

Initial area ratio is the most important quantity for design. Calculations for an end- or cigarette burner will be illustrated below.

In an end-burner, combustion is restricted to the cross section of the combustion chamber. Such rockets deliver constant thrust over a rather long period of time.

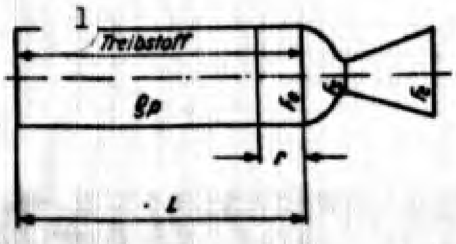


Fig. 3.12. Initial area ratio. F_0 = end area; F_s = narrowest cross section; F_e = end cross section; r = part of propellant grain burned per unit time; L = length of propellant grain; ρ_p = density of solid propellant. 1) Propellant.

These rockets have an optimum charge density, but nevertheless deliver only relatively small thrusts. Because of the long burning time, the walls become very hot and must be comparatively thick if they are to be strong enough. A particular difficulty consists in the fact that as a result of severe heating, there is a danger that the propellant grain will be ignited prematurely from the wall, so that the combustion surface becomes conical and, consequently, considerably larger. End-burners require particular care in manufacture. The propellant grain must be thoroughly insulated from the wall and sealed off.

The main reason why large thrusts cannot be obtained with such rockets is that combustion is restricted to the cross section. Large thrusts would require disproportionately large cross sections.

Calculations for an end-burner

Ballistic requirements:

Thrust at ground	$S = 600$ kgf
Burning time	$\tau = 15$ sec
Propellant:	composite propellant consisting of 76.5% $KClO_4$ and 23.5% asphalt and oil $r = 0.10\ p^{0.745}$ cm/sec, with p supplied in kgf/cm^2 $c^* = 1051$ m/sec $\gamma = 1.27$ $\rho_p = 1770$ kg/m^3

Desired combustion chamber pressure: $p_0 = 130$ kgf/cm^2

a) narrowest cross section

It follows from $S = \zeta_0 F_s p_0$ that

$$F_s = \frac{S}{\zeta_0 p_0}$$

$$\zeta_0 = \Gamma \sqrt{\frac{2\gamma}{\gamma - 1}\left[1 - \left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

$$= 0{,}6618 \sqrt{9{,}41 \left[1 - \left(\frac{1}{130}\right)^{0{,}221}\right]}$$

$$= 1{,}648$$

$$F_s = \frac{600\ \mathrm{kp}}{1{,}648 \cdot 130\ \mathrm{kp/cm^2}} = 2{,}801\ \mathrm{cm^2}.$$

b) burning area

Since a is circumstantial as regards units, we transform (14), setting

$$a = \frac{r}{p_0^n}$$

We then obtain

$$K = \frac{1}{r \varrho_p c^*} p_0. \tag{15}$$

We have

$$r = 0{,}10 \cdot 130^{0{,}745}\ \mathrm{cm/s} = 3{,}757\ \mathrm{cm/s},$$

$$K = \frac{130 \cdot 10^4\ \mathrm{kp/m^2}}{3{,}757 \cdot 10^{-2}\ \mathrm{m/s} \cdot 1770\ \mathrm{kg/m^3} \cdot 1051\mathrm{m/s}}$$

$$= \frac{1{,}30 \cdot 9{,}81 \cdot 10^6\ \mathrm{kg/m\,s^2}}{3{,}757 \cdot 10^{-2} \cdot 1{,}770 \cdot 10^3 \cdot 1{,}051 \cdot 10^3\ \mathrm{kg/m\,s^2}} = 182{,}5,$$

$$F_0 = KF_s = 182{,}5 \cdot 2{,}801\ \mathrm{cm^2} = 511{,}2\ \mathrm{cm^2}.$$

c) length of propellant grain

$$L = 3{,}75\ \mathrm{cm/s} \cdot 15\ \mathrm{s} = 56{,}36\ \mathrm{cm}.$$

d) specific impulse.

The propellant grain is a cylinder with the diameter d = 25.52 cm and a height h = 56.36 cm.

Its volume is

$$V = 511{,}2\ \mathrm{cm^2} \cdot 56{,}4\ \mathrm{cm} = 28832\ \mathrm{cm^3} = 0{,}028832\ \mathrm{m^3}.$$

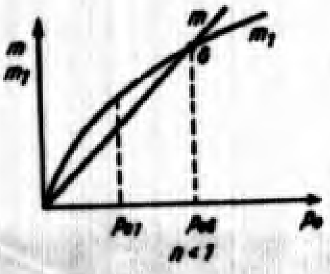


Fig. 3.13. Stable combustion. m = mass of gas leaving nozzle per unit time as a function of internal pressure p_0; m_1 = mass of gas formed by combustion per unit of time; G = equilibrium state; p_{0G} = constant stable pressure.

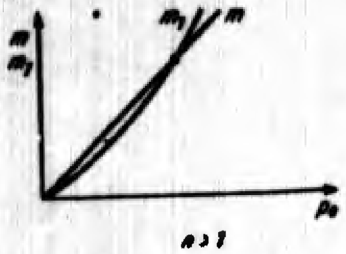


Fig. 3.14. Unstable combustion.

The mass of the propellant is

$$M_{Tr} = 1770\,\text{kgm}^{-3} \cdot 0{,}028832\,\text{m}^3 = 51{,}032\,\text{kg}.$$

The total impulse

$$I = 600\,\text{kp} \cdot 15\,\text{s} = 9000\,\text{kps}.$$

Accordingly, the specific impulse

$$I_s = \frac{9000\,\text{kps}}{51{,}032\,\text{kg}} = 176{,}4\,\frac{\text{kps}}{\text{kg}}$$

and the effective exhaust velocity

$$c_e = 176{,}4 \cdot 9{,}81\,\frac{\text{m}}{\text{s}} = 1730\,\text{m/s}.$$

Now we come to the question as to the requirements that must be met for adjustment to equilibrium pressure. This state intervenes when the mass of the gases flowing out of the nozzle per unit time is equal to the mass of newly formed gas per unit of time. For the gases flowing out of the nozzle per unit time, we have according to (18) in 3.1.1.

$$m = \frac{\Gamma}{\sqrt{RT_0}} F_* p_0 = b p_0. \qquad (16)$$

b is a constant and m is porportional to p_0. Figure 3.13 shows m plotted as a function of p_0. $m = bp_0$ is represented by a straight line passing through the origin. The mass of the gases formed by combustion per unit of time is

$$m_1 = rF_0\varrho_p = ap^n F_0\varrho_p. \qquad (17)$$

If we assume that F_0 is constant, as it is in most cases, we may write

$$m_1 = cp^n \qquad (18)$$

We assume $n < 1$; then m_1 is represented by a parabola, as in Fig. 3.13. If, for example, the pressure is p_{01}, then more gas is formed than flows through the nozzle and the pressure rises to p_{0G}. Then an equilibrium state is established. This state is stable, i.e., even when disturbances arise the system no longer returns to point G.

Figure 3.14 shows the labile or unstable state. It intervenes when $n > 1$. Constant pressure is not possible in this case.

The so-called side-burning grains are used much more extensively than the end-burners. In this category, the lateral surfaces of the powder grain burn. The end surfaces are generally propected from combustion. If, nevertheless, they also burn, they exert little influence. In most cases, the powder grain takes the form of a hollow cylinder, and a number of powder grains of the same shape may also be used. The hollow cylinder burns from the inside as well as from the outside. As the outer surface becomes smaller, the inside surface increases correspondingly, with the result that the total burning surface remains constant (for exact relationships, see [2], 5.32). Poweder cylinders, which are known as tubular grains, are also used in artillery shells.

Figure 3.15 shows a schematic representation of a tubular grain. The combustion gases flow through the powder grain to the nozzle. Since

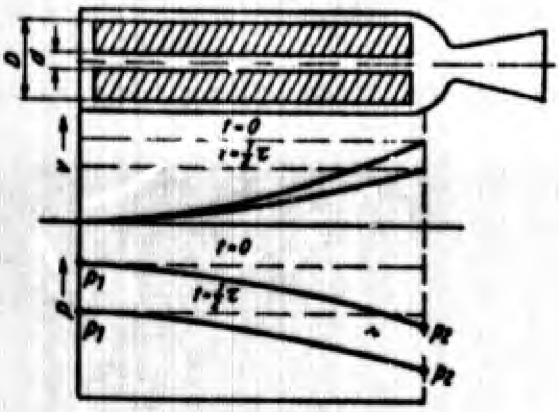


Fig. 3.15. Curves of gas pressure p and flow velocity v in a tubular grain at the beginning of combustion (t = 0) and after half of the combustion time (t = ½τ).

new gases are being formed, the flow velocity in the direction of the nozzle increases, with the result that the pressure falls off. Figure 3.15 shows the curves of the velocity v and pressure p, for the beginning of combustion and for the point in time halfway through combustion, τ/2. Although conditions are more complicated than in the case of an end-burner, we may assume that the linear rate of combustion varies only slightly. Although the pressure drops in the direction toward the nozzle, the linear combustion rate is raised again accordingly by the higher velocity of the combustion gases, so that if the rocket has been designed properly, the powder grain will burn off uniformly. Prime requirements here are that the powder grain not be too long nor the cavity too narrow, since dead zones in the gas must be avoided.

The mean between the pressure p_1 at the beginning and the pressure p_2 at the end is taken as the combustion chamber pressure p_0, to which all calculations are referred.

In ordinary rockets p_1 is about 11% greater than p_2, but in high-performance rockets, the excess may range up to 40%.

The dimensions of such a hollow propellant cylinder are determined by the thrust, the burning,time, the combustion chamber pressure and the characteristic values of the propellant. Let us illustrate this with an example.

Example: It is required that S = 20,000 kgf and τ = 10 sec. The specific thrust is I_s = 200 kgf-sec/kg. For the linear combustion rate, we have $r = 0.089 \times p^{0.69}$ cm/sec. The combustion chamber pressure is p_0 = 100 kgf/cm^2. The density of the propellant is ρ = 1620 kg/m^3. For the mass throughput per second, we have the relationship

$$m = \frac{S}{I_s};$$

since, written in dimensional form,

$$\frac{\text{kp}}{\text{kps/kg}} = \frac{\text{kg}}{\text{s}},$$

or

$$m = \frac{20000\,\text{kp}}{200\,\text{kps/kg}} = 100\,\text{kg/s}.$$

If F_0 is the lateral area and ρ the density of the propellant, we have for the mass of gas formed per unit time

$$m = F_0 r \rho.$$

The influence of the end surfaces can be disregarded, since they are small compared to the lateral area. In general, they are prepared in such a way that they do not participate in the combustion process. For the linear combustion rate, we obtain

$$r = 0{,}089 \cdot 100^{0{,}69}\,\text{cm/s} = 2{,}135\,\text{cm/s}$$

and thus for the lateral area

$$F_0 = \frac{m}{r\rho} = \frac{100\,\text{kg/s}}{2{,}135 \cdot 10^{-2}\,\text{m/s} \cdot 1620\,\text{kg/m}^3} = 2{,}9\,\text{m}^2.$$

If D is the outside diameter of the hollow cylinder and d its inside diameter, we have for the combustion time τ (since the cylinder burns both from the inside and from the outside)

$$\tau = \frac{D - d}{2r}$$

and, accordingly, for the wall thickness

$$D - d = \tau \cdot 2r = 10\,\mathrm{s} \cdot 2 \cdot 2{,}135\,\mathrm{cm/s} = 42{,}7\,\mathrm{cm}.$$

If the inside diameter d = 10 cm, then D = 52.7 cm.

If L is the length of the hollow propellant cylinder, we have

$$F_e = \pi L(D + d).$$

From this we obtain

$$L = \frac{F_e}{\pi(D + d)} = \frac{2{,}9\,\mathrm{m}^2}{\pi \cdot 0{,}627\,\mathrm{m}} = 1{,}48\,\mathrm{m}.$$

Accordingly, the propellant grain is a hollow cylinder the length of 1.48 m, an outside diameter of 0.527 m and an inside diameter of 0.10 m. The inside diameter may not be too small unless we are willing to accept erosion phenomena, as will be seen later.

If the rocket were built as an end burner, the same combustion area would be necessary, i.e., the diameter would have to be

$$D = \sqrt{\frac{4F_e}{\pi}} = 1{,}92\,\mathrm{m}$$

and the length

$$L = r\tau = 0{,}214\,\mathrm{m}$$

The propellant grain would have to be a flat cylinder. It is probably unnecessary to stress that such a rocket would be impossible.

Powder grains like the hollow cylinder described above, which burns from the inside and outside, are referred to as unrestricted-combustion powder grains. When they are used, the hot powder gases also flow along the combustion-chamber walls and heat them. This is a disadvantage, particularly in rockets with long burning times, since it necessitates making the combustion chamber walls very thick.

For this reason, restricted-combustion powder grains, which burn only from the inside, are used. The cylindrical outer surface is applied to the combustion-chamber walls and protects them from heating.

Consequently, the combustion chamber wall can be made considerably thinner and lighter. Needless to say, measures must be taken to prevent the hot gases from penetrating between the wall and the powder grain. For this purpose, a material known as an inhibitor is applied to the outer surface of the powder grain to prevent combustion on this surface.

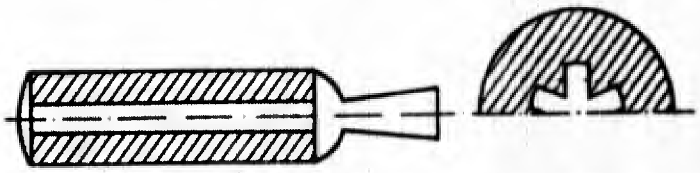


Fig. 3.16. Longitudinal and transverse sections through a rocket with restricted combustion.

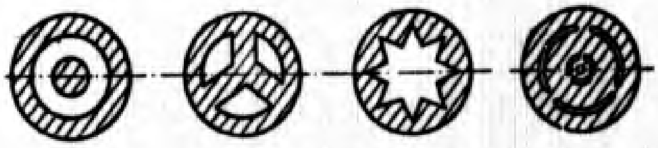


Fig. 3.17. Cross sections through powder grains with constant burning areas.

Figure 3.16 shows longitudinal and transverse sections through such a rocket. If the cavity were cylindrical, the combustion area would no longer be constant, but would become larger during the course of the process, and the initial area ratio would also increase. According to (11), this would mean a pressure rise. Hence the cavity is given cross sections of the kind shown in Figs. 3.16 and 3.17. The cross sections have almost constant burning areas up to burnout, but disintegrate toward the end of combustion into individual part-grains, so that the pressure does not collapse suddenly at the end but falls off gradually.

Many authors have adopted the terms degressive and progressive for powder types from the interior ballistics of firearms ([2], pages 56 and 58). However, we do not feel that this extension is justified, since the conditions prevailing in conventional firearms are totally different.

A disadvantage of solid propellants is their sensitivity to temperature, or, more precisely, the temperature sensitivity of the propellant charge at the instant at which combustion begins. For the linear combustion rate, we make the assumption

$$r = ap^{\nu}.$$

The coefficient a depends on the initial temperature T_1. a increases with increasing temperature T_1. As a result, the propellant charge burns more rapidly. This means, as we see from (11), an increase in the combustion chamber pressure p_0, as well as shortening of the burning time. The converse is the case for a low temperature T_1. If T_1 is too low, we may have the case in which the combustion-chamber pressure does not rise high enough to ensure uniform combustion. Then the gases exit from the nozzle in a pulsating manner; this phenomenon is known as "coughing."

Temperature sensitivity is determined experimentally. If T_{10} is a standard temperature, usually +15°C, and a_0 is the corresponding coefficient, we may write the empirical relation

$$a = a_0 e^{\alpha(T_1 - T_{10})} \tag{19}$$

Here, α is the temperature sensitivity coefficient.

In the example of calculation for an end-burner, we assumed a composite propellant consisting of $KClO_4$ asphalt and oil. For this propellant, $\alpha = 0.0015 \text{ deg}^{-1}$. If the reference temperature T_{10} = + + 15°C, we obtain for T_1 = +55°C

$$\frac{a}{a_0} = e^{,0015 \cdot 40} = 1{,}062.$$

According to (11), we obtain for the pressure p_0' at T_1 = +55°C

$$\frac{p_0'}{p_0} = \left(\frac{a}{a_0}\right)^{\frac{1}{1-n}}, \qquad (20)$$

i.e., for $T_i = +55°C$ with $n = 0.745$

$$p_0' = 130\ \text{kp/cm}^2 \cdot 1{,}062^{\frac{1}{0{,}255}} = 164{,}6\ \text{kp/cm}^2.$$

With this, we obtain for the linear combustion rate

$$r = a p_0'^{n} = 1{,}062 \cdot 0{,}10 \cdot 164{,}5^{0.745}\ \text{cm/s} = 4{,}755\ \text{cm/s}.$$

The length of the propellant grain is 56.36 cm. Accordingly, we obtain for the combustion time at 55°C

$$\tau' = \frac{56{,}36}{4{,}755} = 11{,}85\ \text{s}$$

as opposed to a "normal" burning time of 15 sec at +15°C. Since the total impulse changes only slightly, the final combustion rate also changes only slightly. The powered trajectory, on the other hand, is shortened in proportion to the time.

If we assume an average constant acceleration <u>a</u>, we then have for the length of the powered segment

$$s_B = \frac{a}{2}\tau^2. \qquad (21)$$

Then

$$v_B = a\tau. \qquad (22)$$

If we substitute (22) in (21), we get

$$s_B = \frac{v_B}{2}\tau. \qquad (23)$$

If $\overline{m}$ is the average mass of the rocket, we have for the total impulse I

$$I = \overline{m} v_B. \qquad (24)$$

Substituting (24) into (23), we obtain

$$s_B = \frac{I}{2\overline{m}}\tau, \qquad (25)$$

i.e., since I and $\overline{m}$ are invariable, the length of the powered trajectory is proportional to the burning time τ. The temperature sensitivity coefficient $\alpha = 0.0015$ deg^{-1} indicated in the above example is relatively low. Many propellants have larger α. For modern propellants, $\alpha < < 0.005$ deg^{-1}. With the double-base propellants, however, values larger than 0.005 deg^{-1} occur. The example shows that the exponent n is also of decisive importance for the temperature effect. The smaller n, the weaker is the influence of temperature. By logarithmic differentiation of (11) we obtain

$$\frac{\mathrm{d}p_0}{p_0} = \frac{1}{1-n}\,\frac{\mathrm{d}a}{a}. \tag{26}$$

It follows (19) by logarithmic differentiation that

$$\frac{\mathrm{d}a}{a} = \alpha\,\mathrm{d}T_1, \tag{27}$$

and, accordingly, from (26) and (27)

$$\frac{\mathrm{d}p_0}{p_0} = \frac{1}{1-n}\,\alpha\mathrm{d}T_1. \tag{28}$$

If n differs little from 1, then 1/(1-n) is very large. For this reason, attempts are made to secure not only small α, but also small n in the development of new propellant grains.

The displacement of the combustion endpoint obtained from (23) has an effect on over-all range in the case of artillery rockets. In order to eliminate this effect, the Honest John artillery rocket (USA), for example, is shrouded by an electrically heated jacket.

Fig. 3.18. Honest John short-range rocket (USA) with electrically heated shroud.

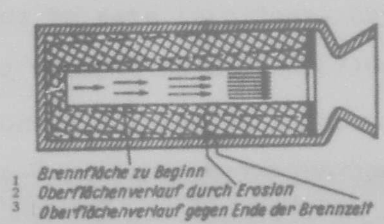


Fig. 3.19. Influence of erosion on combustion area. 1) Initial combustion surface; 2) surface shape resulting from erosion; 3) surface shape toward end of burn.

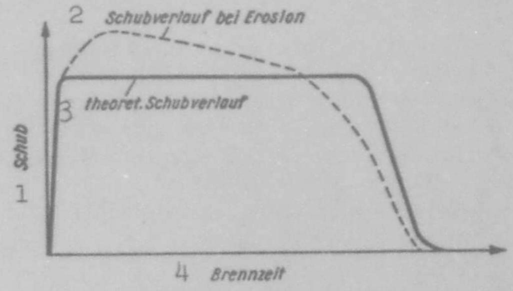


Fig. 3.20. Influence of erosion on thrust curve. 1) Thrust; 2) thrust curve with erosion; 3) theoretical thrust curve; 4) burning time.

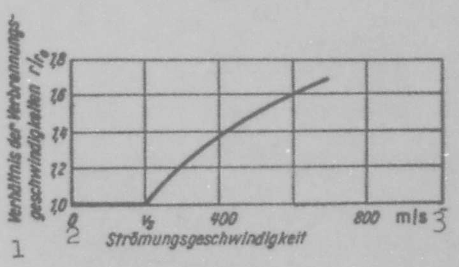


Fig. 3.21. Combustion rate as a function of gas flow velocity for ballistite. r = combustion rate elevated by erosion; r_0 = normal combustion rate; r/r_0 = erosion ratio; v_s = threshold velocity. 1) Ratio of combustion rates; 2) exhaust velocity; 3) m/sec.

We have already established that the high impulse densities of the solid propellants represent one of their particular advantages. In order to exploit this as fully as possible, the cavity volume, i.e., the hollow spaces inside the propellant grains, are made as small as possible. It has now been observed that with inside-burning grains in which the ratio of the total cross-sectional area of the gas passages to the narrowest nozzle cross section has been reduced to a value approaching 1, a phenomenon known as erosion or erosion burnoff arises. The term erosion originates from geology. It can be observed in river beds that if the flow velocity is high enough, surface friction removes thin layers from a river bed consisting of solid rock. A similar situation arises in the cavities of propellant grains. Since the gas velocity is higher toward the end of the propulsion grain, particles of the surface are ablated, with the result that the burning rate increases. The gas passage, which originally had a constant cross section, is widened toward the back end of the propellant grain.

The propellant is used up first in this zone. Toward the end of the burn, the combustion area gradually becomes smaller as a result of nonuniform consumption of the propellant, instead of dropping instantly to zero as would be the case for a constant cross section. As a result of erosion, the thrust curve is made to deviate from theory as shown in Fig. 3.20.

At the beginning of the combustion processes, a pressure rise and hence a thrust rise take place, since a high burning rate prevails in the narrow gas channels as a result of erosion.

Erosion phenomena do not appear until the velocity of the flowing gases has come to exceed a certain threshold velocity v_s. Figure 3.21 shows the results of Wimpress' investigations on JPN ballistite, a double-base propellant (51.5% nitrocellulose and 43.0% nitroglycerine). Here r_0 is the normal linear combustion rate, and $\underline{r}$ is the combustion rate elevated as a result of erosion. The threshold rate is 200 m/sec.

Two objectives must be met in igniting the solid propellant grain: 1. a sufficiently high temperature and 2. a sufficiently high pressure must be produced. During a not too short span of time, the surface tem-

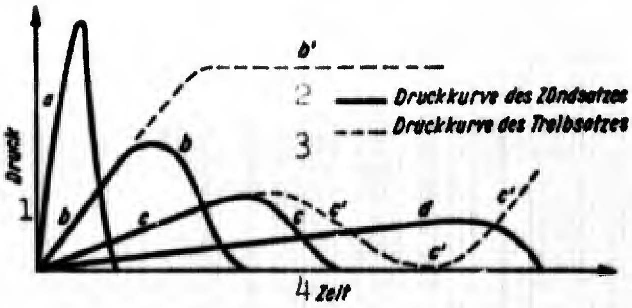


Fig. 3.22. Ignition of propellant grain as a function of ignition-charge pressure curve. a = action too short, no ignition; bb' = normal ignition process; cc' = ignition-charge pressure too low, only partial ignition of powder grain, long ignition delay; d = pressure too low, no ignition. 1) Pressure; 2) ignition-charge pressure curve: 3) propellant-charge pressure curve; 4) time.

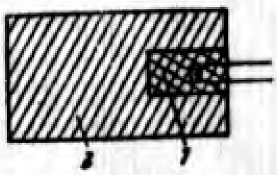


Fig. 3.23. Igniter for small rockets. 1) heat-sensitive explosive; 2) pyrotechnic mixture.

Fig. 3.24. Step diagram of thrust. 1) Thrust; 2) launch thrust; 3) sustained thrust; 4) time.

perature of the grain must be raised above the temperature of spontaneous ignition. During this process, heat is transferred to the surface by convection, but primarily by radiation and the heating effect of solid and liquid particles on the surface. Black powder is often used in igniters, since it produces a sufficient quantity of combustion products in condensed (solid and liquid) states. 1 kg of black powder yields 400 g of gases and 600 g of condensed material. The combustion temperature is 2590°K. Mixtures of metal powder (aluminum or magnesium) with an oxidizer (potasium perchlorate or nitrate) are also used quite often. To meet the second requirement, however, it is necessary that a sufficiently large quantity of gas be formed. It is a characteristic of solid propellants that they burn only beginning at a certain minimum pressure. This pressure is at least 30 kgf/cm^2. Reactions between the combustion gases of the igniter and the air present in the combustion space facilitate ignition.

A certain time is required for heat transfer to ignitie the grain. In Fig. 3.22, b represents the pressure curve of the ignition charge and b' that of the propellant in a normal ignition process. In case cc', the pressure remained too low; the propellant grain was ignited only on a small area, and the pressure first sinks and then rises only after a certain time has elapsed. Here we deal with a long ignition delay. In case a, no ignition takes place in spite of the high pressure, since the action did not continue long enough. In case d, there is again no ignition, since the pressure remained too low.

In small rockets, the igniter consists of two parts (Fig. 3.23). I contains a heat-sensitive explosive that is ignited by current flowing through a coil. The ignition charge proper is contained in II. In large rockets, a number of igniters distributed throughout the cavities are employed.

It will be seen in a later section (5.2.6) that for exterior-ballistic reasons, for example, improvement of scattering, a step curve (Fig. 3.24) is more desirable than a square one.

It is generally necessary to have a higher launching thrust and a weaker sustaining thrust. The problem might be solved by fitting the rocket with a launching booster that would then drop off after the rocket had cleared the launcher. This is not feasible for artillery rockets, since friendly troops would be endangered by the falling boosters. Moreover, two separate ignition processes are required, increasing the chances of trouble.

Thus there has been a tendency to use two propellant cylinders, with one accommodated in the cavity of the larger propellant grain (Fig. 3.25).

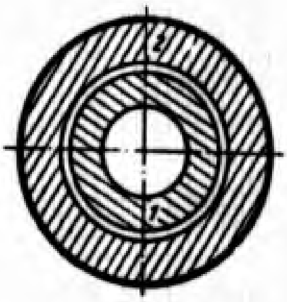


Fig. 3.25. Arrangement of propellant grain to produce a stepped diagram. 1) Propellant with high linear combustion rate; 2) propellant whose combustion rate is as independent of pressure as possible.

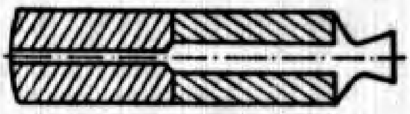


Fig. 3.26. Another arrangement of propellant grains to produce a stepped diagram.

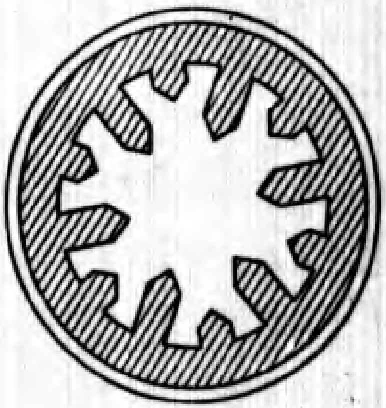


Fig. 3.27. "Wagon-wheel" propellant charge to produce a step diagram.

Grain 1 consists of propellant with a high linear combustion rate, while the r of grain 2 is small. The sharp pressure drop after combustion of grain 1 makes it necessary that the linear combustion rate of propellant grain 2 depend on pressure as little as possible, i.e., that n be small, or that the propellant grain burn completely independently of pressure in the range under consideration.

Another arrangement is shown in Fig. 3.26. Here again, two propellant grains with different linear combustion rates are employed.

It is also possible, however, to produce a stepped diagram by proper configuration of the cavity in a single propellant grain.

Figure 3.27 shows a cross section through a "wagon-wheel" propellant grain. The shapes projecting into the cavity produce a considerable increase in combustion area. After they have burned off, what is left is essentially only a hollow cylinder, and the quantity of gas formed per unit time drops off sharply as a result.

3.1.4. Liquid propulsion

.1.4.1. Liquid rocket propellants

While the oxidizer and fuel are already intimately mixed in the propellant grain in solid propulsion systems and can therefore react immediately where they are, the two components in a liquid propulsion system are held in separate tanks and must first be transferred to the combustion chamber, where they enter reaction with one another.

We distinguish hypergolic, i.e., spontaneously igniting, and non-hypergolic propellant combinations. However, this distinction represents only extreme cases. In actuality, the components always react with one another. It is simply that there are differences as regards the time elapsing between the encounter of the first droplets of the components and formation of the frame. This time, which is known as the ignition delay, is of decisive importance, since measures must be taken to prevent large quantities of the propellants from collecting in the combustion chamber, which might result in an explosion and destruction of the chamber. On the basis of ignition-delay measurement, we may decide whether a special igniting device is necessary. Here the dimensions of the combustion chamber also become a factor. In practice, the distinction between hypergolic and nonhypergolic depends on whether the ignition process depends on <u>both</u> components. An oxidizer that forms a hypergolic combination with a given fuel may be nonhypergolic with another fuel. Other factors, such as the injection system, the mixture ratio and the initial temperature are also to be considered. Thus the classification mentioned above is of only relative importance.

In addition to the liquid propellants that consists of two components, there are also <u>single-propellant systems</u> (monopropellants), in which the fuel and oxidizer are combined. Here again, we may draw a distinction between two groups:

1. Oxidizer and fuel in the same molecule, for example, methyl nitrate.

2. Oxidizer and fuel mixed, for example, nitric acid and amyl acetate.

These single-propellant systems are of subordinate importance. They do not produce high performance figures. They do, however, have the advantage that only a single tank and only a single supply system are required. Below we shall concern ourselves exclusively with the propellants consisting of two separated components.

Almost all combinations of usable oxidizers and fuels may be considered as propellants.

A number of requirements determine usefulness:

1. The components of the propellant must be stable, i.e., they may not undergo changes or decompose during storage.

2. They must be safe to handle. For example, they should not be sensitive to shock.

3. The components and their combustion products may not be toxic.

4. They may have no corrosive effects, since this gives rise to trouble in storage.

5. Low viscosity and small coefficients of surface tension are necessary to create optimum conditions for atomization and mixing of the components.

6. The components must remain liquid over a broad temperature interval (low freezing point, high boiling point), so that conditions for storage, shipment and tanking up will be favorable.

7. The components must be as inexpensive as possible and admit of domestic manufacture.

TABLE 3.8
Most Important Values for Certain Liquid Oxidizers

A Name des Oxydators	B chemische Formel	Gefrierpunkt [°C] C	Siedepunkt [°C] D	spez. Impuls [kps/kg] E	Stadium der Nutzbarmachung F
Sauerstoff	O_2	−218,9	−183	295	H wird in großen Mengen erzeugt
Ozon	O_3	−192	−111,9	368	J wird unter Laborbedingungen erforscht
K Wasserstoffsuperoxid	H_2O_2 (95%)	− 0,4	150,2	272	H wird in großen Mengen erzeugt
L rotrauchende Salpetersäure	HNO_3 (82%)	− 60,0	-	250	H wird in großen Mengen erzeugt
M weißrauchende Salpetersäure	HNO_3 (99,65%)	− 42,1	65	-	H wird in großen Mengen erzeugt
N Distickstofftetroxid	N_2O_4	− 10	22	274	H wird in großen Mengen erzeugt
O Fluor	F_2	−218	−188,3	373	P industrielle Produktion angelaufen
Q Chlortrifluorid	ClF_3	− 82,6	11,3	269	P industrielle Produktion angelaufen
R Perchlorylfluorid	ClO_3F	−110	− 46,8	290	S wird unter Laborbedingungen hergestellt

A) Oxidizer; B) chemical formula; C) freezing point; D) boiling point; E) specific impulse [kgf-sec/kg]; F) stage in development; G) oxygen; H) produced in large quantities; I) ozone; J) being investigated under laboratory conditions; K) hydrogen peroxide; L) red fuming nitric acid; M) white fuming nitric acid; N) nitrogen tetroxide; O) fluorine; P) industrial production starting; Q) chlorine trifluoride; R) perchloryl fluoride; S) produced under laboratory conditions.

It is clear that these requirements eliminate a large number of components in advance, and that compromises are still necessary.

The terms oxidizer and fuel were expanded in 3.1. They no longer apply only to oxygen and carbon and their compounds. All elements are included that belong to the first three periods of the periodic system. Because of the high atomic weights, the higher periods are not of interest; it will be remembered from 3.1.1, Formulas (11) and (12) that propellants with the lowest possible molecular weights were desirable. We shall concern ourselves first with <u>oxidizers</u>. Of the four elements that can be considered on the basis of the periodic system, however, sulfur and chlorine drop out at the outset, sulfur and its compounds because of their physical states and chlorine because of the high molecular weights of the combustion products. This leaves only oxygen and fluorine and their compounds.

Table 3.8 presents the most important quantities for certain liquid oxidizers.

Liquid oxygen (O_2), while it is admittedly the most widely used and best-known oxidizing agent, and the one first to be used, is nevertheless not an attractive rocket-propellant component because of its specific properties. The boiling point of liquid oxygen is −183°C. Because of the high vapor pressure, storage of small quantities is possible only in thick-walled containers. Thin-walled containers, like the propellant tanks of rockets, must be provided with blowoff holes. Because of the resulting losses, the rockets can be fueled only just before launching. This compromises combat readiness very very badly. Nor is immediate launching from launch silos possible, since the rockets must be raised up for fueling.

Liquid oxygen can be transported only in special containers, which again are not sealed and have high vaporization losses.

It would be desirable in the interest of high specific impulse to use ozone. Nevertheless, major difficulties stand in the way of practical application. Ozone is highly explosive and traces of organic substances result in instantaneous decomposition. Even as an additive to oxygen, it produces explosions when the mixture contains more than 1/4 ozone. Even production is difficult. Consequently, ozone has no practical significance in rocket engineering at the moment.

Hydrogen peroxide (H_2O_2) contains 94% oxygen and liberates half of it on decomposition, reforming water according to the equation

$$2H_2O_2 \rightarrow 2H_2O + O_2.$$

Nevertheless, fewer H_2O_2 decomposes explosively on shock or under illumination. For this reason, it can be used only in 70- to 80-% aqueous solutions. Thus hydrogen peroxide is worthy of attention, since it can be used not only as a component, but also as a monopropellant system. Decomposition can be accelerated by use of a catalyst (manganese compound). The specific impulse obtained using 80-per cent hydrogen peroxide as a monopropellant system is only 126 kgf-sec/kg. Decomposition takes place at a temperature of 660°C. Thus we also speak of "cold propulsion." The low temperature of decomposition is a considerable advantage.

<u>Pure nitric acid</u> (HNO_3) contains 76% oxygen. Its temperature range in the liquid state is favorable. The freezing point is −41°C and the boiling point +84°C. Its density is 1.5 g/cm^3. (The density of liquid oxygen is 1.13 g/cm^3.) On heating and exposure to light, pure nitric acid decomposes according to the reaction

$$2HNO_3 \rightarrow 2NO_2 + H_2O + {}^1/_2\,O_2.$$

Most of the brown nitrogen dioxide (NO_2) remains in solution and colors the acid red, whence the name "red fuming nitric acid." Red fuming nitric acid contains 5 to 20% excess NO_2. It is used as an oxidizer, the NO_2 also has an oxidizing effect. Technically manufactured concentrated nitric acid contains about 69% of the pure acid. The water must be removed for use as a propellant. This is done by distillation in the presence of concentrated nitric acid. White fuming nitric acid has a water content ranging up to 2%. Nitric acid is inexpensive and is produced in rather large quantities. Unfortunately, it has a strong corrosive effect on tank steels, but aluminum, on the other hand, is not attacked by concentrated nitric acid.

A particular advantage of nitrogen tetroxide (N_2O_4) consists in the fact that it does not attack container steels, i.e., it offers the possibility of storing the rockets after fueling. However, the unfavorable liquid-state range represents a disadvantage. In order to balance the advantages and disadvantages of these two last-named oxidizers to the fullest extent possible, nitric acid containing dissolved nitrogen tetroxide is used.

Recently, possibilities of using fluorine (F_2) have been under investigation, since application of this element promises very high specific impulses. The difficulties involved, however, are great. At normal temperatures, fluorine is gaseous, and is liquefied only at −188°C. It can be stored in pressurized liquid-nitrogen-cooled containers. Application is made difficult by the fact that fluorine is highly toxic and chemically strongly aggressive. Furthermore, production is expensive and complicated. The hydrogen fluoride (HF) that forms on combustion is particularly dangerous; as we know, it etches glass vigorously. A point in its favor is that fluorine ignites very quickly with hydrogen and all hydrogen-containing carbon and nitrogen compounds, i.e., is strongly hypergolic. Due to the extraordinary toxicity of hydrogen fluoride, use of flurine will be restricted to the upper stages of rockets.

Due to the difficulty of handling liquid fluorine, less dangerous fluorine-containing oxydizers have been proposed. These include chlorine trifluoride (ClF_3) and perchloryl fluoride (ClO_3F). Although they produce considerably smaller specific impulses than pure fluorine, the impulse density is nevertheless the same as for fluorine when chlorine trifluoride is used.

In summary, we may state that the number of usable oxidizers is very small.

The situation is considerably less difficult with the fuels. Table 3.9 lists the most important values for certain liquid rocket fuels. Numerous hydrocarbons and hydrocarbon derivatives are suitable for use as fuels. Of particular importance at the present time is kerosene, a mixture of various hydrocarbons obtained in the distillation of petroleum. It is very important to note that kerosene is liquid in a broad temperature range on either side of zero. Kerosene is nontoxic, chemical stable and has no corrosive effect. It can be combined with liquid oxygen and with nitric acid. The combustion temperatures obtained with nitric acid are lower than those obtained with the use of gen.

An important advantage found in kerosene, and to be looked for in any other fuel, is that it can be used as a coolant for the combustion chamber. In the USA, these fuels are designated JP (JP = jet propellant). Not only the physical and chemical properties are factors in selection of these fuels, but also their cost or, more precisely, the percentage content of these substances in crude petroleum.

Another important group of fuels are the alcohols, and ethyl alcohol in particular. It is suitable for use as a coolant and produces lower combustion temperatures in combination with oxygen than do, for example, hydrocarbons. Since ethyl alcohol mixes with water in all proportions, it is an easy matter to lower the combustion-chamber temperature. This is naturally done at the cost of a reduced specific impulse.

Methyl alcohol delivers somewhat lower performance levels than ethyl alcohol. Alcohol-oxygen mixtures have long ignition delays and thus require special ignition devices.

TABLE 3.9.

Most Important Values for Certain Liquid Rocket Fuels

Name des Brennstoffs A	B chemische Formel	Gefrierpunkt [°C] C	Siedepunkt [°C] D	Dichte [g/cm³] E	F spezifischer Impuls bei Verbrennung mit Sauerstoff [kps/kg]
JP-4*		− 60	65	0,764	-
G Hydrazin	N_2H_4	1,4	113	1,0085	264
H asymmetrisches Dimethylhydrazin	$(CH_3)_2N_2H_2$	− 52,0	63,0	0,791	275
I Wasserstoff	H_2	−259,1	−252,7	0,0738	362
J Diboran	B_2H_6	−165,5	− 92,5	0,445	293
K Pentaboran	B_5H_9	− 46,6	58,4	0,63	-
L Dekaboran	$B_{10}H_{14}$	99,7	213	-	-
M Aluminiumborhydrid	$Al(BH_4)_3$	− 64,5	44,5	0,544	276
N Lithiumborhydrid	$LiBH_4$	275	-	0,6	306
O Lithiumhydrid	LiH	680	-	0,82	268
Dimethylberyllium	$Be(CH_3)_2$	-	200	-	272
Aluminiumtrimethyl	$Al(CH_3)_3$	15	126	6,8	249

*Hydrocarbon of the kerosene type. JP = jet propellant.

A) Fuel; B) chemical formula; C) freezing point; D) boiling point; E) density; F) specific impulse in combustion with oxygen [kgf-sec/kg]; G) hydrazine; H) unsymmetrical dimethylhydrazine; I) hydrogen; J) diborane; K) pentaborane; L) decaborane; M) aluminum boranhydride; N) lithium boranhydride; O) lithium hydride.

The nitrogen-hydrogen compounds form a third group. Hydrazine (N_2H_4) and ammonia (NH_3) are to be named first here. Both yield high specific impulses with fluorine and fluorine compounds. Hydrazine is used much more extensively than ammonia, since the latter has an unfavorable liquid-state range (−78°C to −35°C) and a low density (0.6 g/cm^3).

In the USA, high hopes were at one time entertained for the boranes, i.e., compounds of boron with hydrogen. Theoretically, high specific impulses were to be expected, and, moreover, the raw material borax was available in adequate quantities. However, the production of boranes from borax is difficult. Two series of boranes are known, and of these only the series with the general formula B_nH_{n+4} might be considered for rockets. Among these, the following should be noted:

Diborane (B_2H_6), a colorless gas,
Pentaborane (B_5H_9), a liquid at normal temperature,
Decaborane ($B_{10}H_{14}$), solid at normal temperature.

Pentaborane is of particular interest, since it is liquid. Solid decaborane has been proposed as an additive to increase the heating values of other fuels.

As of this writing, the investigations that have been carried out at great cost have not produced any practical result. Although specific impulses of the order of 250 to 300 kgf-sec/kg have indeed been obtained, large quantities of boron trioxide are deposited in the combustion chambers, to the detriment of the combustion process. A great disadvantage is the enormous toxicity of the free boranes, which makes it practically impossible to handle them.

A glance at the periodic table of the elements will show that in elements such as lithium, beryllium, magnesium and aluminum and their compounds may be considered. Table 3.9 presents data for some of the compounds.

Pure hydrogen is of particular interest, since it delivers the highest chemical obtainable specific impulses with liquid oxygen or liquid fluorine as oxidizer in virtue of the high heat of combustion and low molecular weight. The use of liquid hydrogen in practical rocketry involves great difficulty. The low boiling point (-252.7°C) is one of the main difficulties. Further, the explosive oxyhydrogen gas forms on evaporation. A disadvantage is the low density in the liquid state: 0.07 g/cm^3. Nevertheless, it has been reported that H_2-O_2 powerplants have already been tested.

Mixture ratio is also of importance for the performance of a propellant combination. Let us first illustrate this by reference to an example. The term mixture ratio implies the ratio of the mass of oxidizer (m_0) to the mass of the fuel m_b. Thus we have

$$x = \frac{m_0}{m_b}.$$

Let us take liquid oxygen O_2 as the oxidizer and liquid hydrogen H_2 as the fuel. Combination of the two yields water, H_2O. The atomic weight of H is 1.008 and that of O is 16.000. Formation of 1 kmole of H_2O requires 1 kmole of H_2, i.e., 2.016 kg of H_2, and 1/2 kmole of O_2, i.e., 16.000 kg of O_2. The mixture ratio is

$$x = \frac{16{,}000}{2{,}016} \approx 8{,}0.$$

This mixture ratio is known as the stoichiometric mixture ratio. (Stoichiometry is the study of the mass ratios at which elements combine. These mass ratios are determined by the valences of the elements.)

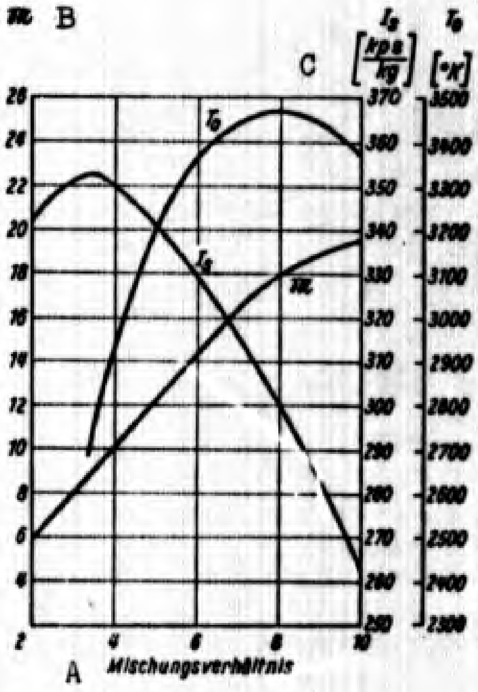


Fig. 3.28. Specific impulse I_s and combustion temperature T_0 as functions of the mixture ratio x in the combustion of H_2 and O_2. $x = m_0/$ $/m_b$; m_0 = mass of oxidizer; m_b = mass of fuel; M = average molecular weight of gases formed. A) Mixture ratio; B) M; C) [kgf-sec/kg].

In 3.1.1., Formulas (11) and (12), we saw that the exhaust velocity v, i.e., the specific impulse, is essentially proportional to $\sqrt{T_0}/M$. In the combustion of H_2 and O_2 in stoichiometric proportions, i.e., for complete combination, we obtain a combustion-chamber temperature of about 3470°K and a specific impulse of about 300 kgf-sec/kg. An excess of H_2 is present at a mixture ratio $x = 4$. Accordingly, only a part of the H_2 present burns to form H_2O. If, for example, the propellant consists of 16 kg of O_2 and 4 kg of H_2, the result is formation of about 18 kg of H_2O, and about 2 kg of H_2 remain unburned. The combustion gases are composed of 1 kmole of H_2O and 1 kmole of H_2. Accordingly, we have for 1 kmole of the combustion gases 20 kg/2 = 10 kg, i.e., an average molecular weight $M = 10$. At a mixture ratio of 8, the molecular weight of the gases formed would be 18. Although the combustion temperature does drop to 2940°K, but the molecular weight has also become smaller, the specific impulse according to Formula (11) in 3.1.1. is

$$I_{s(x=4)} = \frac{300\,\mathrm{kps}}{\mathrm{kg}}\sqrt{\frac{2940}{3470}\cdot\frac{18}{10}} = 370{,}5\,\frac{\mathrm{kps}}{\mathrm{kg}}.$$

The actual value is somewhat smaller (I_s = 350 kgf-sec/kg), since γ has also changed to some degree.

Figure 3.28 shows curves of the combustion temperature T_0, molecular weight M and the specific impulse I_s as functions of the mixture ratio x. It is seen that with diminishing mixture ratio, the specific impulse rises sharply, although the combustion temperature drops sharply. We obtain the highest specific impulse at $x = 3.5$, i.e., 353 kgf-sec/kg. The combustion temperature at $x = 3.5$ is $T_0 = 2755°K$, i.e., 715° lower than for the stoichiometric ratio. This is naturally of great importance, since a lowering of the combustion chamber temperature implies considerably gentler stressing.

TABLE 3.10

Specific Impulse I_s and Impulse Density I_{vs} as Functions of the Mixture Ratio x for Liquid Oxygen and Liquid Hydrogen ($x = O_2$ liquid/H_2 liquid)

x	I_s A [kps/kg]	ϱ [kg/m³]	B $I_{vs} = I_s \cdot \varrho$ [kps/m³]
2	342	778	$266 \cdot 10^3$
3,5	353	895	$316 \cdot 10^3$
4	350	919	$322 \cdot 10^3$
5	341	954	$325 \cdot 10^3$
6	329	980	$322 \cdot 10^3$
8	300	1012	$304 \cdot 10^3$
10	264	1035	$273 \cdot 10^3$

Liquid hydrogen $\rho_{H_2} = 70.8$ kg/m³

Liquid oxygen $\rho_{O_2} = 1131$ kg/m³

A) I_s [kgf-sec/kg]; B) $I_{vs} = I_s \cdot \rho$ [kgf-sec/m³].

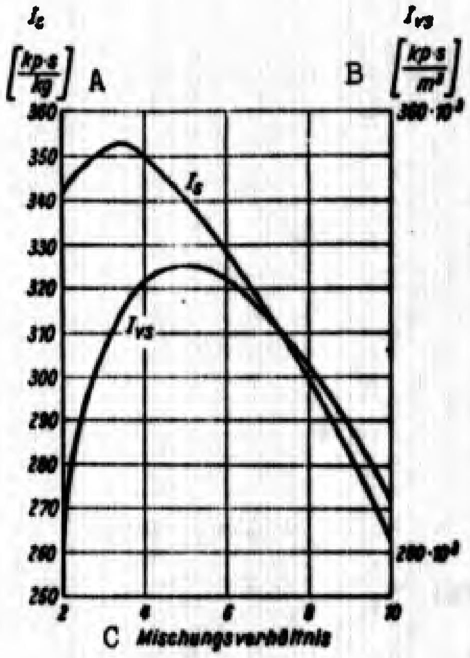


Fig. 3.29. Volume-specific impulse I_{vs} and specific impulse I_s as functions of mixture ratio in combustion of liquid hydrogen and liquid oxygen. A) [kgf-sec/kg]; B) [kgf-sec/m³]; C) mixture ratio.

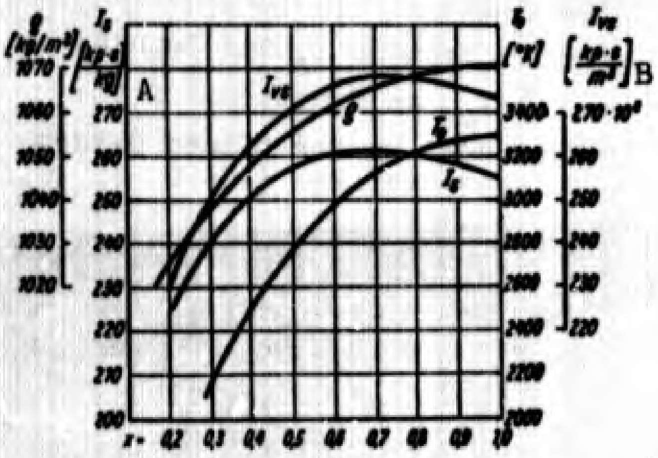


Fig. 3.30. Specific impulse I_s, volume-specific impulse I_{vs}, combustion chamber temperature T_0 and density ρ of combustion gases as functions of mixture ratio for hydrazine-liquid oxygen combination. A) [kgf-sec/kg]; B) [kgf-sec/m^3].

In 3.1.3.3., we concerned ourselves with the volume-specific impulse, which is also known as the impulse density I_{vs}, and took note of the importance of this quantity for design. Table 3.10 shows the density ρ of the liquid propellant as a function of the mixture ratio x. The density of liquid hydrogen is ρ = 70.8 kg/m^3 and that of liquid oxygen is ρ = 1131 kg/m^3. According to 3.1.3.3.,

$$I_{vs} = I_s \cdot \rho.$$

The values of I_{vs} are likewise given in Table 3.10. In Fig. 3.29, I_s and I_{vs} are plotted against the mixture ratio <u>x</u>. The maximum of I_{vs} occurs at x = 5, i.e., in a region in which the combustion chamber temperature is higher. Higher volume-specific impulse means a lower container weight. The designer will attempt to strike a balance, since he must take both the impulse density <u>and</u> the combustion chamber temperature into account. In the present case, the design of the rocket will be based on x = 4.

Figure 3.30 shows the density ρ, specific impulse I_s, volume-specific impulse I_{vs} and the combustion-chamber temperature as functions of the mixture ratio <u>x</u> for the propellant combination hydrazine plus liquid oxygen. Here again it is seen that the maxima of the specific impulse and impulse density are not reached at the highest combustion-chamber temperature.

Not only the selection of the oxidizer and the fuel, but their mixture ratio as well is of decisive importance for the physical design.

3.1.4.2. Powerplant for liquid propellants

The liquid-fueled propulsion system consits of three parts:

1. Combustion chamber,
2. The propellant-feed system,
3. Propellant tanks.

We shall be concerned for the most part only with the combustion chamber. This has the following components:

1. Injection head,
2. The combustion chamber proper,
3. The nozzle,
4. The cooling system,
5. The ignition system.

The combustion chamber is a high-output gas generator whose function is to provide the gases required for the flow process. The structural differences between combustion chambers for solid and liquid propellants stem essentially from two circumstances:

1. Liquid propulsion systems have longer burn times than solid propulsion systems.
2. In liquid propulsion systems, the two components – fuel and oxidizer – must be atomized and intimately mixed in a very short time.

Although the rocket engine is mechanically far simpler than other aerospace propulsion systems, it confronts the designer with difficult

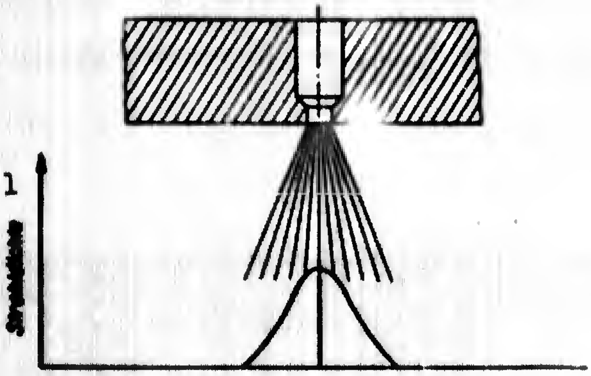


Fig. 3.31. Cylindrical injection nozzle and the corresponding spray density distribution. 1) Spray density.

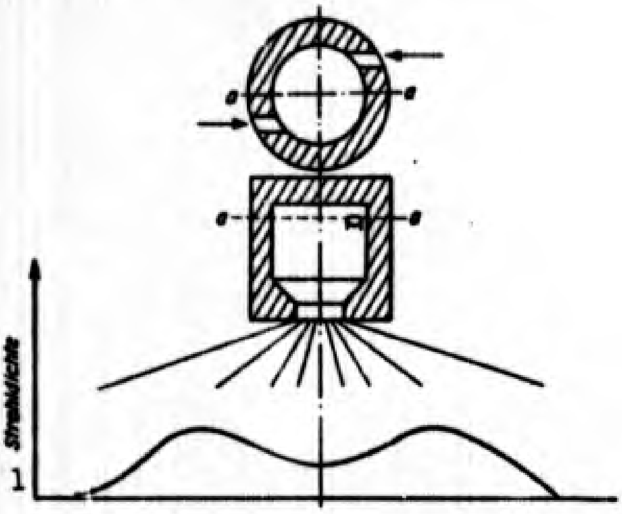


Fig. 3.32. Swirl nozzle and the corresponding spray density distribution. 1) Spray density.

metallurgical problems because of the tremendous quantities of heat transferred to the combustion chamber walls, the shock-like heating, and the strongly corrosive action of many propellants.

The successive processes in the combustion chamber, namely, atomization, vaporization, mixing and chemical reaction, are difficult to describe on a theoretical basis. We therefore resort to experiment in the development and physical design of combustion chambers. Nevertheless, we are obliged to proceed from certain fundamental conceptions in evaluating the experimental results and extending them. The construction of the injector and spray nozzles is of great significance for the processes in the combustion chamber. There is no generally optimal injector. Its mode of action depends on the propellant proponents, their physical and chemical properties, and the size and shape of the combustion chamber. We shall not go into detail here regarding the design of the injector. It consists of two parts, the manifold, which ensures uniform supply to the injection nozzles, and the injection heads, which accomodates the injection nozzles. The characteristic features of the nozzles are the throughput per second and the structure of the spray. The per-second throughput depends on pressure, or, more precisely, on the pressure gradient, as well as on the shape and dimensions of the nozzle.

Figure 3.31 shows the longitudinal section through a simple cylindrical nozzle together with the density distribution within the spray. According to this, the density of the spray is highest on the longitudinal axis and falls off rapidly as we move outward along the radius.

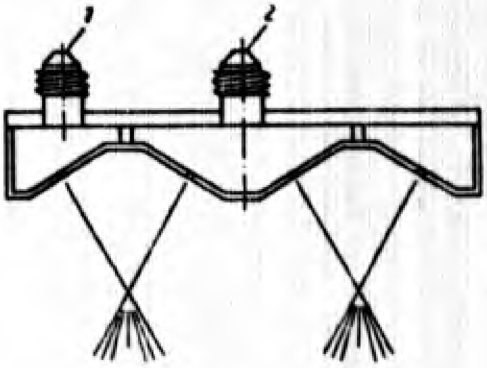


Fig. 3.33. Arrangement of impinging nozzles for oxidizer and fuel on manifold. 1) Inflow orifice for fuel; 2) inflow orifice for oxidizer.

A considerably broader density distribution is obtained by means of centrifugal or swirl nozzles (Fig. 3.32). The liquid enters laterally and is therefore set in rotation. However, not even this device

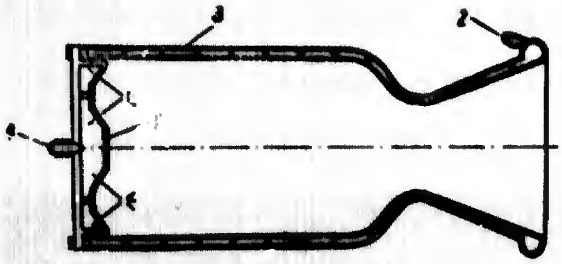


Fig. 3.34. Cylindrical combustion chamber with combustion-chamber head. 1) Combustion chamber head; 2) inflow orifice for fuel; 3) cooling jacket; 4) inflow orifice for oxidizer.

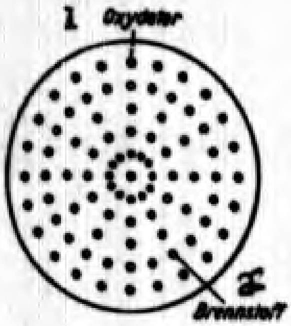


Fig. 3.35. Concentric arrangement of injection nozzles on combustion chamber head. 1) Ozidizer; 2) fuel.

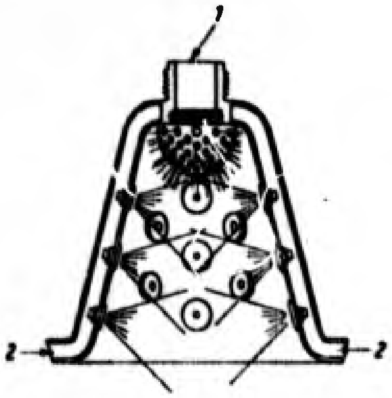


Fig. 3.36. Injection head. 1) Oxidizer; 2) fuel.

guarantees completely uniform distribution.

In order to obtain quick and thorough mixing of oxidizer and fuel and good atomization, frequent use is made of impingement nozzles, whose axes are inclined to one another in such a way that the jets meet. Figure 3.33 shows the arrangement of two impingement nozzles for oxidizer and two for the fuel on a manifold. Fuel enters at 1 and oxidizer at 2. In cylindrical combustion chambers, the injection nozzles are placed in the combustion-chamber head (Fig. 3.34). The injection nozzles are often arranged concentrically (Fig. 3.35). Other distributions are also used.

Spherical or pear-shaped combustion chambers have certain advantages, since the ratio of area to volume is particularly small, a point of significance for reasons of strength and heat handling. With these combustion chambers, however, it is difficult to place the atomizing nozzles directly on the combustion-chamber wall. For this reason, the so-called injection heads (Fig. 3.36) are employed. The oxidizer is introduced through a hemispherical atomizer head provided with a large number of orifices. The jets for the fuel are located in the side walls of the injection head.

The central problem of design is determination of the optimum combustion chamber volume. If the volume is too small, part of the propellent leaves the nozzle before the desired conversion has taken place. Excessively large volume means excessive weight and energy losses. In practice, we proceed by testing combustion chambers with various volumes for a given mass throughput $\dot{m}$ and determining the smallest volume at which the largest specific impulse is obtained. (For the per-second mass throughput, we shall henceforth use the symbol $\dot{m}$ rather than the previous $\underline{m}$.)

In order to extend such experimental results, the concept of <u>stay time</u> has been introduced. If V_0 is the volume of the combustion chamber and $\dot{V}$ the (gaseous) volume of the propellant injected per second, the stay time is

$$\tau = \frac{V_0}{\dot{V}}. \tag{1}$$

If ρ_0 is the average density of the gases, we have

$$\dot{V} = \frac{\dot{m}}{\varrho_0} = \dot{m}\frac{RT_0}{p_0} \tag{2}$$

and hence

$$\tau = \frac{V_0 p_0}{\dot{m} R T_0}. \tag{3}$$

We now assume that the stay time τ is a fixed value for a given propellant combination and a given injector design). Then it follows from (3) that as the combustion chamber pressure p_0 rises, the combustion

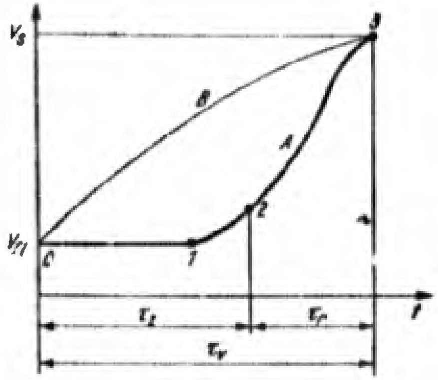


Fig. 3.37. Variation of the specific volume V_s in the combustion chamber as a function of time for two extreme cases. A) Nonhypergolic propellants; B) hypergolic propellants: 1) Volume change begins due to heating; 2) onset of chemical reaction; 3) end of chemical reaction. τ_z) Ignition delay; τ_r) time of chemical reaction; τ_v) stay time; V_{fl}) volume in liquid state.

chamber volume V_0 may be made smaller.

The actual stay times are considerably longer than would be indicated theoretically by (1); this is because the volume $\dot{V}$ of the initially liquid propellant is initially very small.

Figure 3.37 shows the variation of the specific volume V_s in the combustion chamber as a function of time for two extreme cases A and B. V_{fl} stands for the volume in the liquid state. Case A refers to nonhypergolic propellants. At the outset, the propellant is introduced in the liquid state (point 0). It is atomized, the ozidizer and fuel are mixed, and we may assume that a certain time elapses before vaporization and the chemical reaction begin. The volume change begins at point 1 as a result of heating in the combustion chamber. The chemical reaction begins at 2 and ends at 3. Now the gases enter the nozzle. The time τ_z is the ignition delay time and τ_r is the time of the chemical reaction. It is assumed in case B that the chemical reaction begins while the liquid phase is still present (hypergolic propellants) and continues directly into the gaseous phase. Accordingly, the change in specific volume beings immediately. It follows from the presentation that the actual stay time depends heavily on the type of propellant. Feodosjew and Siniarjew indicate stay times of 0.003-0.008 sec. Calculation of the combustion chamber volume on the basis of (1) is highly uncertain. Furthermore, according to (1) and (3), the stay time is independent of combustion-chamber shape — something with which we definitely cahnot concur. For this reason, attempts have been made to establish more accurate relationships for the stay time or to dispense with the conception altogether.

The characteristic length can be used in place of the stay time. By the characteristic length L*, we mean the quotient

$$L^* V_0/F_s = \frac{\text{Combustion-chamber volume}}{\text{Area of narrowest nozzle cross section}}. \qquad (4)$$

Experimental studies have indicated that the ratio V_0/F_s must exceed a certain minimum value for adequate combustion to take place.

We shall restrict our further discussion to cylindrical combustion chambers. The classical method consists in static testing of a series of small model "furnaces" with various lengths and measuring the characteristic velocity. For the latter, we have from (4) in 3.1.2.

$$c^* = \frac{F_s p_0}{\dot{m}}. \qquad (5)$$

The quantities p_0 and $\dot{m}$ that appear here are easily measured, and F_s is known. If the characteristic velocity c* is plotted against the length of the model combustion chamber, we obtain a curve of the form shown in Fig. 3.38. From a certain combustion-chamber length on, c* no longer increases. Thus the minimum length and hence the minimum volume of the combustion chamber are given. Now L* is also known. This experimentally determined value of L* can now be extended to a larger combustion chamber. Here however, it must be remembered that the curve determined applies to a given propellant and a given injection system.

It has been found for a constant cross-section ratio of

$$\varepsilon_0 = F_0/F_s = \frac{\text{Combustion chamber cross section}}{\text{Narrowest nozzle cross section}}$$

and for a constant mixture ratio. A number of trials are required for determination of optimum dimensions.

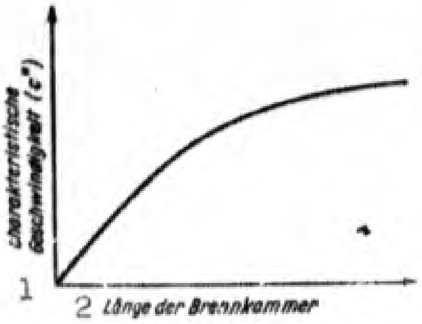


Fig. 3.38. Characteristic velocity c* as a function of the length of a cylindrical combustion chamber. 1) Characteristic velocity (c*); 2) combustion chamber length.

The following characteristic lengths have been obtained on the basis of experiments:

nitric acid-hydrocarbon	$2 \text{ m} < L^* < 3 \text{ m}$
liquid oxygen-ethyl alcohol	$2.5 \text{ m} < L^* < 3 \text{ m}$
liquid oxygen-kerosene	$1.5 \text{ m} < L^* < 2.5 \text{ m}$
fluorine-ammonia	$1 \text{ m} < L^* < 1.5 \text{ m}$

The optimum cross-section ratio ε_0 depends primarily on the size of the engine. It becomes smaller as thrust increases. The earliest engines had ε_0-values between 4 and 15. In the USA, preference is given to more slender combustion chambers with $1.2 < \varepsilon_0 < 4$, and specifically $\varepsilon_0 = 3\text{-}4$ for a thrust S = 1000 kgf, $\varepsilon_0 = 2\text{-}3$ for S = 10,000 kgf and $\varepsilon_0 = 1.2\text{-}2$ for S = 100,000 kgf.

Example of calculation for cylindrical combustion chamber

Suppose that a cylindrical combustion chamber is to be designed for a thrust of S = 60,000 kgf. The propellants are liquid oxygen and ethyl alcohol ($\gamma = 1.27$), the combustion chamber pressure is to be 25 atmospheres, and the nozzle is to expand to 1 atmosphere. First we compute the narrowest nozzle cross section F_s. For this we have according to (23) in 3.1.1.

$$F_s = \frac{S}{\zeta_0 p_0}.$$

For ζ_0 we obtain from (22) of 3.1.1.: $\zeta_0 = 1.429$, so that

$$F_s = \frac{60000 \text{ kp}}{1{,}429 \cdot 25 \cdot 10^4 \text{ kp/m}^2} = 0{,}1680 \text{ m}^2.$$

On the basis of (4), we obtain for the combustion chamber volume

$$V_0 = L^* F_s.$$

Since for the liquid oxygen-ethyl alcohol propellant combination L* lies between 2.5 m and 3 m, we obtain for V_0 the values

$$0{,}4200 \text{ m}^3 < V_0 < 0{,}5040 \text{ m}^3.$$

If, considering the magnitude of the thrust, we assume a cross-section ratio $\varepsilon_0 = 1.5$, i.e.,

$$F_0 = 1{,}5 \cdot 0{,}1680 \text{ m}^2 = 0{,}2520 \text{ m}^2,$$

then we obtain for the length ℓ of the cylindrical combustion chamber

$$\frac{0{,}4200 \text{ m}^3}{0{,}2520 \text{ m}^2} < l < \frac{0{,}5040 \text{ m}^3}{0{,}2520 \text{ m}^2},$$

i.e.,

$$1{,}67 \text{ m} < l < 2{,}00 \text{ m}.$$

From (24) in 3.1.1., we obtain for the nozzle expansion ratio

$$\frac{F_e}{F_s} = 3{,}859,$$

from which $F_e = 3.859 \cdot 0.1680 \text{ m}^2 = 0.6483 \text{ m}^2$ and for the diameter

$$d_e = 0{,}91 \text{ m},$$
$$d_s = 0{,}46 \text{ m}.$$

The length ℓ_d of the nozzle is, according to (16) in 3.1.1,,

$$l_d \leqq \frac{0{,}91 \text{ m} - 0{,}46 \text{ m}}{0{,}536} \leqq 0{,}84 \text{ m}.$$

In general, it can be affirmed that with increasing thrust, the volume represented by the combustion chamber out of the total volume (combustion chamber + nozzle) becomes smaller, while the volume proportion of the nozzle becomes larger.

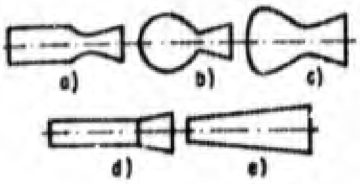


Fig. 3.39. Combustion chamber shapes. a) Cylindrical combustion chamber; b) spherical combustion chamber; c) pear-shaped combustion chamber; d) combustion chamber and nozzle without narrowest cross section; e) divergent combustion chamber.

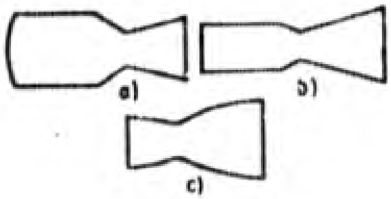


Fig. 3.40. Early and more recent rocket-engine types. a) Earlier type (large characteristic length); b) more recent type with increased nozzle divergence ratio; c) combustion chamber with convergent shape; nozzle no longer conical.

As regards the shape of the combustion chamber, we can distinguish three basic types. The cylindrical combustion chamber, whose production is particularly simple, has been used most extensively in the past (Fig. 3.39a). The injection head can be built as a separate omponent. Spherical combustion chambers are particularly strong (Fig. 3.39b). They have the largest volume together with the smallest surface area, something that is of particular importance for cooling. To obtain large thrusts, particularly in the early days, pear-shaped combustion chambers (Fig. 3.39c) were used — for example, in the V2. There is also the possiblity of using combustion chambers without a narrowest cross section (Fig. 3.39d). The heat liberated in the cylindrical part accelerates the gases, and, with sufficient length, the velocity of sound is attained at the end of this segment. In the divergent segment that follows, the gases are further accelerated. Such engines are known as flared-tube engines. Combustion chambers with only the divergent shape and no nozzle are also possible (Fig. 3.39e). They require propellants with high reactivity. Experiments have already been conducted with such combustion chambers. It has been possible to establish by pressure measurements that combustion is particularly intense in the vicinity of the injectors of such combustion chambers.

Until a few years ago, it was necessary, in view of the injection system design and the nature of the propellant, to use large characteristic lengths in order to ensure adequate combustion. The shape of the engine corresponded approximately to type a in Fig. 3.40, which is characterized by large volume, a large cross-section ratio ε_0, and a short nozzle with a small divergence ratio. The object of using the large cross section ratio ε_0 was to avoid excessive heat throughput.

Advances in the design of the injection and cooling systems, and primarily in the development of new materials for combustion chambers and new propellants, made it possible to reduce L* and ε_0. The nozzle divergence ratio, on the other hand, was increased; Fig. 3.40b shows a modern combustion chamber with its nozzle. More recently, shape c of Fig. 3.40 has been used more and more frequently; here, the cylindrical combustion chamber is replaced by a convergent shape. The divergent segment is no longer conical and has a large divergence ratio.

The design of an adequate cooling system is one of the most difficult problems in the design of a propulsion system. An effort is made to make the engine as light as possible. Here, however, we run up against the enormous quantities of heat that flow through the walls and necessitate efficient cooling arrangements. This heat transfer is more than eight times that of a heavy-duty industrial smelting oven. A rocket engine with a thrust of 2300 kgf, using a propellant combination with a combustion heat of 1100 kcal/kg and a combustion temperature of 3060°K has a heat transfer rate of 756,000 kcal/min for a combustion volume of 0.028 m^3. It is nonetheless possible to use cooling systems to obtain efficient propulsion systems whose heat losses amount only to 3 to 4%.

No special cooling devices are necessary at short burn times, up to about 30 sec, such as are encountered in solid rockets. It is enough to make the walls of the combustion chamber and nozzle sufficiently thick, since these are heated capacitively. With longer burning times, auxiliary cooling is necessary, since the combustion chamber temperature is far above the melting point of the combustion chamber wall materials. The most important type of cooling is liquid cooling or regenerative cooling. In this form of cooling, one of the propellant components is ducted through a cooling jacket surrounding the combustion chamber and nozzle before being sprayed into the combustion chamber. The result is a steady state in which a constant quantity of heat flows through the combustion chamber wall per unit of time, heating it only up to a constant temperature below the melting point of the combustion chamber wall material. Regenerative cooling has the advantage that it returns most of the heat flowing through the combustion chamber wall back into the combustion chamber. Figure 3.34 shows a rocket engine surrounded by a cooling jacket. Regenerative cooling has, on the other hand, a disadvantage in that it remains restricted to propellant-component types that have the required physical and chemical properties, namely, high specific heat and high boiling point, as well as the appropriate chemical stability at high temperature.

Figure 3.41 shows the temperature curve in the vicinity of the combustion chamber wall. T_g is the gas temperature in the combustion chamber, v_g is its velocity. T_i is the temperature of the inner surface and T_a the temperature of the outer surface of the combustion chamber wall. T_f is the temperature of the coolant fluid, v_f is its flow rate. q_g is quanity of heat liberated per unit time by the gases

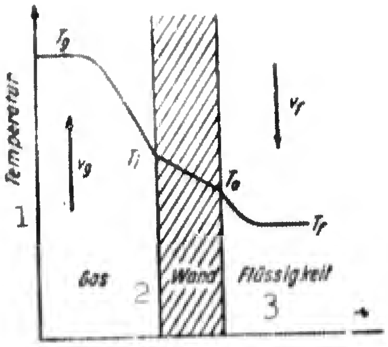


Fig. 3.41. Temperature curve in the vicinity of the combustion chamber wall. v_g) Combustion gas velocity; v_f) coolant flow rate; T_g) gas temperature; T_i) temperature of inner surface; T_a) temperature of outer surface; T_f) temperature of coolant. 1) Temperature; 2) wall; 3) liquid.

at the surface F of the combustion chamber inner wall. The following relationship applies for the surface unit:

$$\frac{q_g}{F} = \alpha_g(T_g - T_i). \tag{6}$$

α_g is known as the heat-transfer coefficient and depends on the specific heat and conductivity of the gases, but primarily on the velocity v_g and, furthermore, on the diameter of the combustion chamber and nozzle.

A corresponding relation applies for the heat transfer to the coolant:

$$\frac{q_f}{F} = \alpha_f(T_a - T_f). \tag{7}$$

where α_f is again the heat-transfer coefficient.

During a time t, a quantity of heat

$$Q = \lambda F t \, \frac{T_i - T_a}{\delta} \tag{8}$$

flows through the surface F of a combustion-chamber wall with a thickness δ.

The quantity of heat is proportional to the temperature gradient $T_i - T_a/\delta$ and to a factor λ, which is known as the thermal conductivity.

λ depends essentially only on the nature of the substance and is therefore a specific quantity for a given substance. A few values of λ follow:

Copper	99.9 %	$\lambda = 0.092 \frac{\text{kcal}}{\text{m-sec-deg}}$
Aluminum	99.75% Al 99 % Al	0.055 0.050
Steel V2A		0.0036
Steel with 20% Ni		0.0044
Tungsten		0.0400
Tantalum		0.0130

According to (8), a quantity of heat

$$\frac{q_w}{F} = \frac{\lambda}{\delta}(T_i - T_a) \qquad (9)$$

flows through the wall per unit time if

$$q_w = \frac{Q}{t}.$$

Since we are concerned with a steady-state process,

$$\frac{q_g}{F} = \frac{q_w}{F} = \frac{q_f}{F} = \frac{q}{F}. \qquad (10)$$

Accordingly, on the basis of (6), (7) and (9),

$$\frac{q}{F} = \alpha_g(T_g - T_i) = \frac{\lambda}{\delta}(T_i - T_a) = \alpha_f(T_a - T_f). \qquad (11)$$

If we eliminate the two quantities T_a and T_i from the above equations, we obtain

$$\frac{q}{F} = \frac{T_g - T_f}{\frac{1}{\alpha_g} + \frac{\delta}{\lambda} + \frac{1}{\alpha_f}}. \qquad (12)$$

Substituting this relation in (6), we obtain

$$T_i = T_g - \frac{T_g - T_f}{1 + \alpha_g\left(\frac{\delta}{\lambda} + \frac{1}{\alpha_f}\right)}. \qquad (13)$$

Now we must make an effort to keep T_i as low as possible, and in any event considerably below the melting point of the material of which the wall consists. T_g and T_f are given quantities. The denominator of the second term on the right side, $1 + \alpha_g \left(\frac{\delta}{\lambda} + \frac{1}{\alpha_f}\right)$, must be as small as possible if T_i is to be small. This can be obtained by making δ small, i.e., by using a thin combustion-chamber wall, selecting a material with a high thermal conductivity, or both. For strength considerations, however, δ cannot be made arbitrarily small.

TABLE 3.11

Comparison of Certain Combustion Chamber Materials

1 Werkstoff	2 Schmelzpunkt T_s[°K]	3 Wärmeleitzahl $\lambda \left[\frac{\text{kcal}}{\text{ms grd}}\right]$	4 Temperatur der Innenwand T_i[°K]	$T_s - T_i$ [°K]	Temperaturgefälle in der Brennkammerwand [grd/mm] 5
6 Aluminium	931	0,0550	520	411	31
7 V 2 A-Stahl	1673	0,0036	1110	563	349
8 Wolfram	3653	0,0400	540	3113	42

1) Material; 2) melting point T_s [°K]; 3) thermal conductivity; λ [kcal/m-sec-deg]; 4) inside wall temperature; 5) temperature gradient in combustion chamber wall [deg/mm]; 6) aluminum; 7) steel V2A; 8) tungsten.

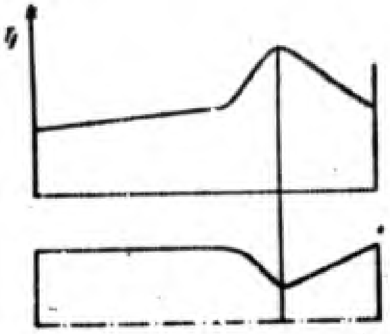


Fig. 3.42. Temperature T_i of inner surface of combustion chamber and nozzle wall at various distances from combustion-chamber base.

Metals that have the highest possible melting points and the largest possible thermal conductivities are particularly suitable as materials for the combustion-chamber walls of powerplants using liquid propellants. Unfortunately, the metals with high melting points usually have low thermal conductivities. Corrosion resistance with respect to the oxidizers and combustion gases is also required. This eliminates copper completely. Aluminum and stainless steels can be used to advantage. Tantalum and tungsten are highly qualified by their high melting points, but are very expensive. In Table 3.11, some of H.G. Mebus's values [12] for a combustion chamber with 1000 kgf of thrust and a combustion chamber pressure of 40 atmospheres are assembled. These figures assume a nitric-acid-based propellant combination and cooling by the oxidizer. The combustion chamber wall is 2 mm thick.

On the basis of the table, tungsten would be particularly attractive, since the difference $T_s - T_1$ is very large. A large temperature gradient in the combustion chamber wall (Column 6) is undesirable, since the chamber wall is heavily stressed by widely different thermal expansions. The temperature stresses are further compounded by the mechanical loads.

We had disregarded the influence of the heat transfer coefficient α_g on T_1. It can be seen from (13) that an increase in α_g causes an increase in the denominator of the 2nd term of the right member and thus an increase in T_1. α_g depends particularly strongly on the gas flow velocity. A 10-fold increase in the velocity increases α_g by a factor of 5.6. A decrease in diameter also increases α_g. A consequence is that the temperature of the inner wall, T_1, and with it the thermal stressing vary widely. The heaviest stressing occurs in the narrowest cross section. Figure [illegible] shows the curve of T_1 as a function of distance from the base of the combustion chamber.

In recent years, the prime concern has been to reduce the structure weight of powerplants. Since the cooling jacket is relatively heavy, a completely new construction has been coming into favor. The heavy double-jacket system is replaced by several hundred axial tubes.

At first, 0.5-mm-thick nickel tubes with rectangular cross sections were employed and given the shape of the combustion chamber and nozzle. They were joined to one another with silver solder. The internal pressure was first taken up by steel straps for the nozzle and a secondary steel-plate jacket for the combustion chamber. Later, these reinforcing members were replaced by a fiberglass-reinforced apoxy-resin jacket.

The term mass-to-power ratio refers to the powerplant mass/thrust ratio. In the A4, it was 40 kg/Mgf. In contemporary propulsion systems, it has been reduced to as little as 12 kg/Mgf. It is assumed that 9 to 10 kg/Mgf will be the lowest attainable limit. Counterflow cooling, which is illustrated in Fig. 3.43, is also used for powerplants consisting of clustered chambers.

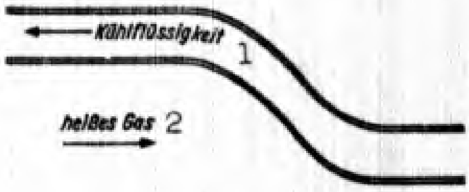


Fig. 3.43. Counterflow cooling by means of tubes. 1) Coolant; 2) hot gas.

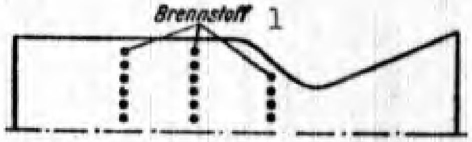


Fig. 3.44. Distribution of orifices for fuel in film cooling. 1) Fuel.

In addition to regenerative cooling, film cooling is often used. Usually, both methods are employed simultaneously. In this type of cooling, a liquid or gaseous film is produced along the combustion chamber and nozzle wall. For this purpose, the fuel is allowed to flow

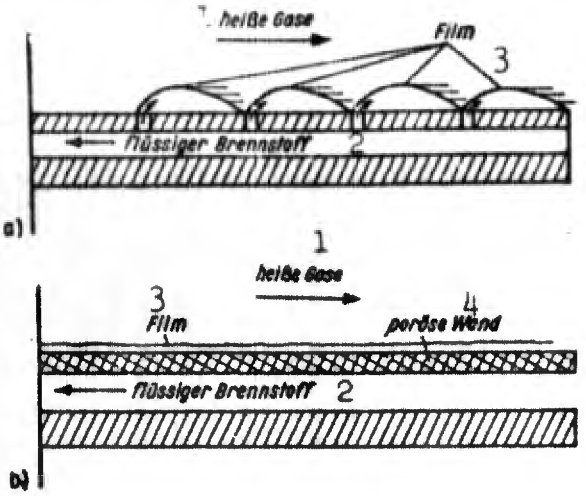


Fig. 3.45. a) Film cooling; b) sweat cooling. 1) Hot gases; 2) liquid fuel; 3) film; 4) porous wall.

out through fine holes in the inner wall. Figure 3.44 shows an example of the manner in which the holes might be distributed. The emerging fuel is entrained by the streaming combustion products. It is at first liquid and evaporates gradually. The film formed in this manner insulates the wall from the hot gases (Fig. 3.45a).

Combustion of the fuel cannot take place immediately, since there has as yet been no mixing with the oxidizer.

Occasionally a porous inner wall that transmits small quantities of fuel is used instead of the performated metallic inner wall. In such cases we speak of sweat cooling (Fig. 3.45b).

Studies have shown that effective film cooling requires a certain quanity of fuel, up to about 10% of the total amount, and that this results in a decrease in specific impulse.

More recently, combustion chambers with fusion cooling for liquid rockets have been tested. The combustion chamber and exhaust nozzle are lined with a material that melts away. This lining is said to render long burn times possible. Fusion-cooled combustion chambers have an advantage in that they require no plumbing system for regenerative cooling and are therefore simpler and cheaper to manufacture. Another benefit is lower weight, so that the thrust/weight ratio is improved. Used combustion chambers can be reconditioned by giving them new linings.

In one test, it was possible to operate a 1-Mgf-thrust fusion-cooled combustion chamber for 4.5 min and, after a pause, for yet another 3 min. Tests of such powerplants in various sizes have been reported.

3.1.4.3. Instability in combustion processes in liquid powerplants

The term instability as applied to combustion processes is not entirely accurate; it would be more correct to speak in general of oscillations or fluctuations in the combustion processes. Such occurrences give rise to the greatest difficulties encountered in the development of liquid-propellant powerplants. Numerous experimental and theoretical investigations of this problem have been published. In view of the scope and purpose of the present book, this field can be treated only briefly. In static tests with liquid rockets, it is occasionally observed that oscillations arise in the combustion chamber during the combustion process. The cause is a periodic variation of the pressure in the combustion chamber. We distinguish between a low-frequency instability at 100-200 cps and a high-frequency instability at about 1000 cps. The latter is announced by loud noise.

Variations in the pressure result in variations in the thrust and hence in acceleration. This is detrimental to the strength of the rockets, which have been built as light as possible, and to the performance of the instrument packages. In some cases, the effect may result in destruction of the rocket.

Thus these oscillations must be avoided to the greatest possible extent. In unfavorable cases, it is at least necessary to achieve considerable reduction of the amplitudes.

In the theoretical investigations, the fundamental premise has been that the propellant does not burn immediately, and that a certain time delay (conversion time) is necessary to permit mixing of the oxidizer and fuel particles and their absorption of the energy required for initiation of combustion. It has been possible to show theoretically that this time τ is of basic significance for the low-frequency combustion oscillations. The studies showed that even true instabilities, i.e., building up of the oscillation amplitudes, might occur and lead to destruction of the powerplant.

We refer now to a simple example to explain the origin of the oscillations. We assume first that the pressure in the feed system is constant. We assume further that the combustion-chamber pressure falls for some accidental reason. The amount Q of fluid passing through a certain nozzle per unit of time depends on the pressure difference between the beginning and end of the nozzle, i.e., on the pressure gradient Δ_p. As Δ_p increases, Q also increases. In the case taken above, Δ_p becomes larger, so that more propellant flows into the combustion chamber. This amplified influx lasts for the above-defined time τ, which elapses before the propellant has been gasified and begun to flow out. At this instant, the pressure in the combustion chamber rises in proportion to the per-second inflow Q and exceeds the normal value. As a result, Δ_p and hence the quantity flowing out per second now becomes smaller. This state prevails for another interval τ. When the time τ has elapsed, the combustion chamber pressure again drops below the normal value.

In this simplified example, we can see that the pressure difference Δ_p and the time τ are of decisive importance. By making Δ_p as large as possible, i.e., by raising the entry pressure at the nozzle, we can make the relative fluctuations of Δ_p smaller and lower the oscillation amplitudes. Experiments have shown that the oscillation frequency is raised in this process. τ is a function of the chemical nature of the propellants and the mixture ratio.

Other factors of influence are the volume of the combustion chamber, i.e., the characteristic length L*, and the average combustion chamber pressure p_0. The frequency of the oscillations diminishes with increasing characteristic length. For a given average combustion chamber pressure p_0 and a given injection pressure, the amplitude drops off slowly when L* increases. The larger combustion space "softens" the oscillation process.

The oscillations in the combustion chamber may have back effects on the feed system, setting up similar oscillations in it. In unfavorable cases, the oscillations may act to amplify one another to the extent that the powerplant is destroyed.

Since the oscillations, their amplitudes and frequencies, depend, as we have seen, on numerous factors, the problem of eliminating the flow-frequency oscillations is extraordinarily complicated. In many cases, it is sufficient just to change the mixture ratio and the combustion-chamber pressure, while other cases require radical alterations, such as, for example, changes in the size and shape of the combustion chambers.

High-frequency oscillations, usually of smaller amplitude are superimposed on the low frequencies. The reasons for their appearance are similar to those for the appearance of the low-frequency oscillations. Nevertheless, there is an essential difference insofar as they depend not on the chamber combustion processes as a whole, i.e., for example, not on the time τ, but rather on the processes that occur during combustion of the heated gases, characterization of which would require studying smaller volumes.

3.1.5. Hybrid Brackets

A rocket in which one propellant is liquid and the other solid is known as a hybrid rocket. Such an arrangement offers the possibility of combining certain advantages of the liquid and solid propulsion systems with one another. They make it possible to shut down the system temporarily, and, within certain limits, also to regulate its thrust.

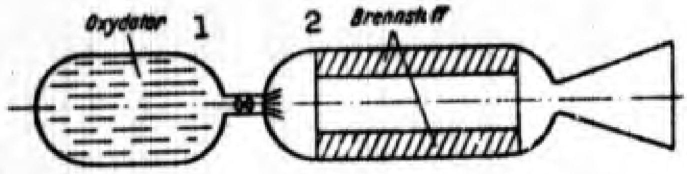


Fig. 3.46. Liquid-solid rocket. 1) Oxidizer; 2) fuel.

Figure 3.46 gives a schematic representation of a liquid-solid rocket. The oxidizer is liquid. It flows into the combustion chamber, which holds the solid fuel. The latter takes the form of a hollow cylinder that protects the combustion-chamber walls from heating.

The solid fuel, for example, polyethylene, a synthetic resin whose heat is about one-third the weight of a normal solid propellant delivering the same impulse is easy to produce. It is homogeneous and has a constant combustion rate. As a consequence, excess combustion-chamber pressures cannot arise. Since the fuel does not react with air, handling and shipping are not hazardous.

The number of possible propellant combinations is very large, since the basic chemical materials coming under consideration can be used in their natural form. Among the numerous fuels, we mention only synthetic rubber and polysulfides, and, among the oxidizers, hydrogen peroxide and nitric acid.

The specific impulse corresponds approximately to that obtainable with two-component liquid propellants.

In March 1962, a series of hybrid-rocket tests were conducted at the French proving grounds at Vernon. Thrusts of 150 and 500 kgf were attained for 100 sec. Other experiments produced thrusts of 1000 kgf and 1500 kgf. Considering the many advantages of the hybrid rockets, it is astonishing that experiments have begun at so late a date.

3.2. FREE-RADICAL PROPULSION

At the present time, this type of propulsion is only of theoretical interest, because of the great difficulties that stand in the way of its application. Under the influence of high temperatures, molecules, such as, for example, the H_2 molecule, are split into two hydrogen atoms in accordance with the endothermal reaction

$$H_2 \rightarrow 2H$$

Large quantities of heat are required here.

TABLE 3.12

Free Radicals and Required Dissociation Energies

Ausgangsmolekül a	Radikale b	erforderliche Dissoziationsenergie [kcal/kg] c
H_2O	OH, H	6550
CH_4	CH_3, H	6300
N_2	N	8000
O_2	O	3660
H_2	H	51 200

a) Initial molecule; b) radicals; c) energy of dissociation required [kcal/kg].

Now if it were possible to store this atomic hydrogen and react it in the combustion chamber to form molecular hydrogen in the exothermal reaction

$$2H \rightarrow H_2$$

these large quantities of heat would again become available and produce higher exhaust velocities than is the case with ordinary exothermal chemical reactions. Exhaust velocities up to about 12,000 m/ /sec would theoretically be possible. In general, electrically neutral atoms or groups of atoms with unstable electron configurations are known as radicals. Table 3.12 lists a few radicals and their starting molecules, as well as the amounts of energy required for dissociation in kcal/kg of the initial molecule.

The reaction $H_2 \rightarrow 2H$ requires by far the largest energy of dissociation. For this reason, particular attention has been directed to atomic hydrogen.

As regards production and storage, however, great difficulties are encountered in attempts at practical application of free radicals. The lifetime of free radicals is very short at normal temperatures. It runs to about 10^{-3} to 10^{-8} sec. Only at temperatures in the vicinity of absolute zero would the lifetime be prolonged adequately.

If and when practical applications materialize, prime consideration will probably be given a mixture of atomic and ordinary hydrogen, i.e., the H-H_2 system. At a concentration of 10% atomic hydrogen, the system would be superior as regards performance to the best chemical propellants. Up to the present time, it has been possible to stabilize only a 1% concentration.

Manuscript Page No.	[List of Foreign Abbreviations]
2	W = w = watt
2	MW = Mw = megawatt
2	s = sec = second
3	kp = kgf = kilogram-force (weight)