Design of Liquid-Propellant Rocket Engines

Nikola Samardzic

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1 Introduction

1. Generation of Thrust

   Rocket engines generate thrust through exchange of momentum. They eject relatively low quantities of mass out of the nozzle at relatively high speeds.

   Say the rocket has mass m and is moving upwards with velocity u, and the propellant is exiting the nozzle at velocity v_e relative to the rocket. Let the amount of propellant exiting the nozzle per unit time be denoted as Δm. Then the net momentum of the system is

   $$M(t_0) = (m_0 - \Delta m t_0)u(t_0) + \int_0^{t_0} (u(t) - v_e)\Delta m \, dt \tag{1}$$

   Assuming no outside forces and that v_e (the velocity of the propellant relative to the rocket) and Δm (the rate at which the propellant is exiting the nozzle) are constant, we can differentiate (1) with respect to t_0 to get

   $$0 = -\Delta m u(t_0) + (m_0 - \Delta m t_0)u'(t_0) + (u(t_0) - v_e)\Delta m \tag{2}$$

   Notice that the force acting on the rocket at time t_0 is $F(t_0) = (m_0 - \Delta m t_0)u'(t_0)$. We call this the net force on the rocket (N_f), and write

   $$N_f = \Delta m u(t_0) + (v_e - u(t_0))\Delta m = \frac{dm}{dt} v_e. \tag{3}$$

   This is called the *net force of the engine in a vacuum.*

   The pressure of the envirionment – air ambient pressure on a rocket in flight – has an influence on the magnitude of the net force. At the exit plane of the nozzle there is a difference in the pressure of the hot gas comming from the engine and the outside atmospheric pressure. For a rocket engine with pressure at the exit plane of the nozzle P_e and exit area A_e flying in an athomsphere with pressure P_a the net force of the engine is going to be $F = N_f + A_e(P_e - P_a)$.

2. Gas-flow Processes in the Combustion Chamber and Nozzle

   Usually, gas-flow calculations concerning rocket engines assume the following conditions:

   - The gas in the nozzle is ideal, homogeneous
   - No heat transfer through the motor walls in either direction (i.e. adiabatic process)
   - No friction
   - Steady flow rate
   - One-dimensional flow (all gas flows on parralel lines)
   - All particles on a specific cross-section of the nozzle have the same velocity (radial symmetry)
   - Chemical equilibrium within the combustion chamber

   In the actual design process, empirically established corrections will be added to the equiations obtained bellow.

   In an adiabatic process, the increase in kinetic energy of the flowing gases between any two points equals the decrease in enthalpy (Like gas comming out of the spray can). Looking at the nozzle, this fact is expressed by

   $$\frac{1}{2gJ}(v_x^2 - v_i^2) = C_p(T_i - T_x) \tag{4}$$

   Also, for isentropic flow we know that pV^{gamma} is constant.

A liquid-rocket combustion chamber converts propellants into high-temperature, high-pressure gas through combustion, which releases the chemical energy of the propellant, resulting in an increase in internal energy of the gas. The liquid propellants are injected at the injection plane with a small axial velocity which is assumed to be zero in gas-flow calculations. The combustion process proceeds throughout the length of the chamber and is expected to be completed at the nozzle entrance. Heat liberated between the injection plane and nozzle inlet increases the specific volume of the gas. To sastisfy conditions of constant mass flow, the gas must be accelerated toward the nozzle inlet with some drop of pressure.

The gas-flow process within the combustion chamber, that is, within the volume upstream of the nozzle entrance, is not entirely isentropic but rather is a partly irreversible, adiabatic expansion. Although the stagnation temperature remains constant, the stagnation pressure will decrease. This causes permanent energy losses. These losses are a function of the *contraction area ratio* $\epsilon_C = A_C/A_t$. The greater the contribution of the nozzle to overall thrust, the more efficient will the gas acceleration be. Conversly, with no nozzle attached the losses are maximized.

3. Gas flow through rocket nozzles

   The prime function of a rocket nozzle is to convert efficiently the enthalpy of the comustion gases into kinetic energy and thus create high exhaust velocity of the gas. It is assumed that the gas flow through the nozzle will be an isentropic expansion, and that the total pressure and temperature remain the same throughout the nozzle.

   The *critical pressure ratio* is the ratio of the pressures at the throat of the nozzle and the combustion chamber. It is a function of only the specific heat ratio.

$$\frac{p_t}{(p_c)_{ns}} = \left[\frac{2}{\gamma} + 1\right]^{\gamma/(\gamma-1)} \tag{5}$$

   The static pressure at the nozzle throat (p_t), where the maximum flow per unit area occurs, is defined as *critical pressure.*

   Useful derivations for ideal gas liquid propellants follow.

   - The theoretical exit velocity

$$v_e = \sqrt{\frac{2g\gamma}{\gamma - 1} RT_i \left[1 - \left(\frac{p_e}{p_i}\right)^{\frac{\gamma-1}{\gamma}}\right] + v_i^2} \tag{6}$$

   - The theoretical gas weight flow rate

$$W = A_t(p_C)_{ns} \sqrt{\frac{g\gamma\left[2/(\gamma+1)\right]^{\frac{\gamma+1}{\gamma-1}}}{R(T_C)_{ns}}} \tag{7}$$

   - The theoretical nozzle expansion ratio

$$\epsilon = \frac{A_e}{A_t}$$

   - Pressure and velocity at throat

$$p_t = (p_C)_{ns}\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma}{\gamma-1}}$$
$$v_t = \sqrt{\frac{2g\gamma}{\gamma+1} R(T_C)_{ns}}$$

4. Performance Parameters of Liquid-Propellent Rocket Engines *Specific impulse* (I_s) equal to F/W.

5. Liquid Propellants The term *liquid propellant* covers both *liquid oxidizers* (liquid oxygen, liquid fluorine, nitric acid, etc.) and *liquid fuels* (RP-1, alcohol, liquid hydrogen, etc.).

   Bipropellant systems employ two different propellants, usually an oxidizer and a fuel. Separate tanks hold oxidizers and fuel, which are not mixed until they reach the combustion chamber. Present day liquid-propellant rocket engines use bipropellants almost exclusively because they offer higher performance and safer operation.

   Ignition devices used to produce chemical reaction include chemical pyro-technic ingiters, electric spark plugs, injection of a spontaneously ignitable liquid fuel or oxidizer (*pyrophoric fluid*) ahead of the propellant.

Other bipropellant combinations, called *hypergolics*, ignite spontaneously upon mixing and thus permit greatly simplified ignition, but also pose certain hazards.

Some liquid propellants are liquefied gases of very low boiling point at ambient pressure and low critical temperature. These propellants are defined as *cryogenics*. The most common cryogenic propellants for rocket applications are liquid oxygen (O_2), liquid hydrogen (H_2), liquid flourine (F_2), oxygene diflouride (OF_2), or a mixture of these. The biggest challenge with cryogenics is that their pathways within the rocket must be addequately heat-isolated to prevent boil-off.

Density impulse of an engine is the measure of impulse delivered to the rocket through the engine per unit volume of the propellant.

6. Selection of Liquid Propellant

   The factors that are relevant to consider when selecting a liquid propellant are

   - Amount of energy produced per unit mass of propellant combined with a low molecular weight of the combustion gases.
   - Ease of ignition
   - Stable combustion
   - High density
   - Ability to serve as an effective coolant for the thrust chamber (optimally has high specific heat, high thermal conductivity, and high critical temperature)
   - Reasonably low vapor pressure at 160° F for low tank weight
   - Low freezing point (preferably less that -65° F)
   - High thermal and shock stability to minimize explosion and fire hazard.
   - Low toxicity of propellants, their fumes, and combustion products.
   - Low cost
   - Availability

2 Design of Thrust Chambers and Other Combustion Devices

1. Basic Thrust-chamber Elements

   In a liquid-bipropellant rocket engine, the following basic steps characterize the conversion of the energy of propellants into thrust:

   (a) The liquid propellants, at proper oxidizer/fuel mixture ration (O/F), are injected into the combustion chamber and atomized into droplets.

   (b) The droplets are subsequently vaporized by heat transfer from the surrounding gas.

   (c) The vaporized propellants are mixed rapidly, further heated, and promptly react, thus continuously increasing gaseous mass flowrate within the combustion chamber. This gas-phase reaction is aided by high-speed diffusion of active molecules and atoms. Combustion is considered complete when it reaches the upstream of the chamber throat, when all liquid droplets are vaporized.

   (d) As the gaseous products of the combustion process pass toward and through the throat, they are accelerated to sonic, and then to supersonic, velocity within the diverging nozzle section, and are finally ejected.

2. Combustion-Chamber Volume

   The combustion chamber serves as an envelope to retain the propellants for a sufficient period (*stay time*) to ensure complete mixing and combustion. The stay time is theoretically independent of combustion-chamber geometry. However, a long chamber with a small cross section entails high nonisentropic pressure losses. With a short chamber of large cross section, the propellant atomization and vaporization zone occupies a significant portion of the chamber volume.

   There are 3 shapes that are generaly used in combustion-chamber design. While spherical and near-spherical chambers were used in early European designs, the cylindrical chamber has been employed most frequently in the US. Compared to a cylindrical chamber of the same volume, a spherical or near-spherical chamber offers the advantage of less cooling surface and weight. A sphere has the best surface-to-volume ratio and the minimum wall thickness required for sustaining the pressure inside (about half that of the cylinder). However, it is harder to manufacture.

3. Nozzle Shape Most rocket nozzles are of the converging-diverging De Laval type. The selection of an optimum nozzle shape for a given expansion area ration is generally influenced by the following design considerations and goals:

- Uniform, parallel, axial gas flow at the nozzle exit for maximum momentum vector
- Minimum separation and turbulence losses within the nozzle
- Shortest possible nozzle length for minimum space envelope, weight, wall friction losses, and cooling requirments
- Ease of manufacturing

Any abrupt change or discontinuity in the nozzle wall contour should be avoided to prevent the possibility of shock waves or turbulence losses. Theoretically, the nozzle throat is simply the unique plane of minimum cross-sectional area. In practice, a well rounded throat section is employed.

Conical nozzle. In early rocket-engine applications, the conical nozzle, which proved satisfactory in most respects, was used almost exclusively. Most importantly it was easy to produce. Usual tilt of the divergent portion was 15^o. Certain performance losses occur in a conical nozzle due to the non-axial components of the exhaust gas velocity, a correction factor, λ, is applied in the calculation of the exit-gas momentum ($\lambda = 1/2(1 + \cos \alpha)$). For $\alpha = 15^o$ we get $\lambda = 0.983$.

Bell nozzle. To gain higher performance and shorter length, engineers developed the bell nozzle. It employs a fast-expansion section in the initial divergent region, which leads to a uniform, axially directed flow at the nozzle exit. The wall of the contour is changed gradually enough to prevent oblique shocks.

Clustered-Nozzle Concepts. Two dimensional, or cluster-type, nozzles are special cases related to the annular-nozzle family.

4. Thrust-Chamber Cooling

Due to high combustion temperatures (4000 to $> 6000^oF$) and high heat-transfer reates from the hot gases to the chamber wall, thrust-chamber cooling is a major design consideration. For short-term operation (up to a few seconds) uncooled chambers may sometimes be used, but for other operations a steady state chamber cooling system must be employed. One or a combination of the following cooling techniques might be used:

(a) *Regenerative cooling.* The most widely applied method; utilizes one or possibly both of the propellants fed through passages in the thrust-chamber wall for cooling, before being injected into the combustion chamber.

(b) *Dump cooling.* With this principle, a small percentage of the propellant, such as the hydrogen in a LO_2/LH_2 engine, is fed through passages in the thrust chamber wall for cooling and is subsequently dumped overboard through openings at the rear end of the nozzle skirt. Inherently, this method implies performance losses and is therefore rarely used.

(c) *Film cooling.* Here, exposed chamber-wall surfaces are protexted from excessive heat by a thin film of coolant or propellant introduced through orifices around the injector and sometimes in several more planes towards the throat. Often used together with regenerative cooling.

(d) *Transpiration cooling.* Introduces a coolant (gaseous or liquid propellant) through porous chamber walls at a rate sufficient to maintain the desired temperature of the combustion-gas-side chamber wall. Essentally, a special type of film cooling.

(e) *Ablative cooling.* In this process, combustion-gas-side wall material is sacrificed by melting, vaporization, and chemical changes to dissipate heat. As a result, relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process. In addition, the ablative material is often a good heat insulator, keeping to a minimum the heat transmitted to the outer structure.

(f) *Radiation cooling.* With this method, heat is radiated away from the surface of the outer thrust-chamber wall. It has been successfully applied to very small, high-temperature materail combustion chambers and to low-heat-flux regions, such as nozzle extensions.

The main factors that influence the selected cooling design approach are:

(a) *Propellants.* The properties of the combustion products, such as temperature, specific heat, specific weight, viscosity, etc. have a direct bearing on the heat-transfer rate and this affect chamber cooling requirments and methods.

(b) *Chamber pressure.* Higher chamber pressures are linked with higher combustion-gas mass flowrates per unit area of chamber cross section and therefore with higher heat-transfer rates. Regenerative and film cooling are ussually combined to meet the stringent requirments of high-chamber-pressure applications.

(c) *Propellant-feed system.* In a turbopump-fed engine, more pressure drop is usually available for chamber cooling. The availability of this drop permits the use of regenerative cooling, which requires sufficient pressure to force the coolant through the cooling passages before entering the injector. A pressure-fed engine usually has more stringent pressure limitations and operates at relatively lower chamber pressures. This implies the use of film, ablative, or radiation cooling.

(d) *Thrust-chamber configuration.* The geometric shape of the chamber affects local combustion-gas mass flowrates and wall surface area to be cooled.

(e) *Thrust-chamber construction material.* Strength at elevated temperature and thermal conductivity will determine the suitability of a given material of regeneratibe cooling.

Chamber cooling design ties in tightly to other considerations in engine design. For example, optimization of the chamber pressure of a high-performance engine may be largely limited by the capacity and efficiency of the chamber-cooling system. In turn, chamber pressure affects other parameters, such as the nozzle expansion ratio, propellant feed pressure, and weight.

(a) Gas Side Heat Transfer

A primary step in the design of a thrust-chamber cooling system analizes heat transfer from the combustion gases to the chamber walls (gas-side heat transfer), which occurs by forced convection. Before the gases can transfer heat to the wall, the heat energy must pass through a layer of stagnant gas along the wall, the boundary layer. This is done via Newton's law of cooling:

$$q = h_g(T_{aw} - T_{wg})$$

where q is the heat flux (heat transferred across the stagnant gas film per unit surface area per unit time), h_g is the gas-side-heat transfer coefficient, T_{aw} is the adiabatic wall temperature of the gas, and T_{wg} is the hot-gas-side local chamber-wall temperature.

(b) Regenerative Cooling

Advantages of regenerative cooling include no performance loss (thermal energy absorbed by the coolant is returned to the injector), essentially no change in wall contour as a function of time, indefinite firing duration, and relatively light-weight construction.

Disadvantages include limited throttling with most coolants, reduced reliability with some (e.g. hydrazine), high pressure drops required at high-heat-flux levels, and thrust levels, mixture ratios, or nozzle area ratios possibly limited by maximum allowable coolant-temperature.

The bulk temperature of the coolant increases from the point of entry of entry until it leaves the cooling passages, as a function of the heat absorbed and the coolant flowrate. The differences between combustion-gas temperature and wall temperature range from 2500 to 6000^oF.

5. Injector Design

The injector injects the propellants into the combustion chamber in the right proportions and the right conditions to yield an efficient, stable combustion process. The injector also performs the structural task of closing off the top of the combustion chamber against the high pressure and temperature it contains. It is comparable to a carburator of a car engine.

No other component of a rocket engine has as great an impact upon engine performance as the injector. The measure of delivered performance (specific impulse) is the number of pounds of thrust provided per pound of propellant consumed per second. Each loss in injector combustion efficiency means a loss of the same magnitude in overall specific impulse. The SSME engine's injector has an efficiency of 99.7%. Note that performance considerations do become secondary if the injector is easily triggered into destructive instability.

(a) Injector Design Issues

The optimal injector entails the following characteristics: light weight, high performance, low cost, reliability, etc.

The SSME requirments called for a highly sophisticated, high-performance booster/core engine using liquid-oxygen/liquid-hydrogen propellants. It was to be reused in multiple missions in manned spacecraft, a use that dictates a higher reliability than any other application. This implied an expensive and heavy injector. The SSME actually has 3 injectors: the usual main-engine injector and one injector each for the fuel and oxidizer turbine-drive preburners, each specificaly developed for its task.

At the other end of the spectrum, a "divert" engine for a small kinetic-energy weapon ("smart bullet"), called for light weight, small size, and rapid response. This engine had a firing life of only a few seconds, but had to reach thrust in only a couple of milliseconds. The injector has a unlike-impinging, 36-element pattern supplied by a manifold with short direct passages from a valve integrated into the injector body. The injector face had a diameter of only 1 inch. The engine thrust-to-weight ratio set a new record, and steady state performance proved very good, considering the rapid response and small combustion chamber.

For liquid oxygen and the fuel a lightweight gas (i.e. hydrogen), a coaxial element will be the primary candidate. For hypergolic liquids and rapis response in a small chamber, an unlike-impinging pattern will be the logical starting point.

For a large booster with LOX/RP-1 propellants, combustion stability becomes a major issue and the density of liquid-phase injection generally rules out the use of a coaxial pattern. The element of choice here would probably be a like-impinging type, such as a like-doublet. The possibly better mixing from an unlike-impinging element would be outweighed by the probability of sensitivity to instabilities.

With combustion chambers made out of metals (copper, nickel, steel) that are fuels, it is important to avoid scrubbing of the chamber wall by high-temperature oxidizing streams. Most injection patterns are therefore designed to provide an excess of fuel in these areas (adjecent to the walls). This implies performance penalties, as the protective fuel on the walls is mostly unsuitable for combustions.

(b) Combustion Stability

All systems which release large amounts of energy have the potential for destructive oscillations, particularly if there is regenerative feedback between the combustion and the rate of energy release. This is particularly true of the combustion process, since temperature and pressure variations can directly impact the rates of vaporization and reaction. Stable operation can be achieved by either damping or detunning these processes. The injection flow resistence provides isolation between propellant flows and chamber disturbances. In general, unlike-impinging elements are the most easily excited to instability, and nonimpinging elements the most stable.

Manifolding, as well as other elements of the propellant-feed systems, may have frequency characteristics that can couple with chamber acoustic modes. These must be addressed in the injector design. These frequencies must be damped where appropriate. Injectors often have physical devices to aid in damping or to otherwise change frequency response (typically acoustic cavities and baffles). These devices increase the resonant frequency of the object they are attached to, and higher frequencies are easier to damp. In rocket engines, they are most effective when located between the outer perimeter of the injector face and the start of the combustor wall.

(c) Manifolds

Selecting the proper injector is tightly intervowen with the design of a feed-system manifold – interconnecting multiple feed paths to a single supply source. The manifold myst first provide passages to feed the often complex, interrelated injector orifices in the desired pattern. The volume of the manifold determines the time required to prime the system and the length of time required to prime the system and the length of time during which the vapor pressure maintains "dribbling" flow after the inlet valves have been closed. This dribble volume very adversely affects repeatability and consistency of propellant use until cutoff, and incure a loss in efficiency. On the other hand, greater volume of the manifold provides the free-flow of propellant and avoids nonuniform distribution of mass and mixture ratio across the injector face. A rule of thumb compromise gives each manifold run four times the flow area of the total group of injection orifices that are fed by it.

(d) Nonimpinging Injection Elements

Coaxial. The most common type of non-impinging element, the coaxial configuration, characterizes the SSME injector and other oxygen/hydrogen engines. The coaxial, or concentric, injection element usually has a slow-moving central stream of liquid oxidizer surrounded by a high-velocity concentric sheet of gaseous fuel. Initially developed by NASA. Mixing, atomization of the liquid, and mass distribution are provided by the shearing action of the high-velocity gaseous fuel on the surface of the liquid. Typically, the liquid velocities are less than 100 ft/s and gaseous fuel velocities are over 1000 ft/s. The fuel surrounding the oxidizer tends to shield the combustion process, which enjoys a favorable combustion-wall heating envirionment and also benefits combustion stability. This method is less suited for liquid or high-pressure gas fuels, since the velocity relationships required to make it work well are difficult to obtain.

(e) Unlike-Impinging Injection Elements

Unlike doublets. A straighforward way of mixing two different fluid streams directs one against the other; this is the basis of an unlike-impinging doublet. The impact produces a fan-shaped spray made up of a mixture of the two impinging fluids. Mixing here is not perfectly distributed as it is adversely affected by any momentum and/or stream-diameter mismatch of the impinging fluids. other effects might also cause problems. One example is that affects the performance of highly-reactive, unlike-impinging reactants is *"blow appart"* or reactive-stream separation, or reactive demixing, which occurs because hypergolic propellants usually have extremely short ignition delay and this start generating gases before completion of the mechanical impact of the two streams. These gases add forces to the system which tend to separated the surfaces of the reactants. At high chamber pressures, there is evidence that liquid propellants not normally considered hypergolic (such as LOX or kerosene) may exhibit this phenomenon as well. Also turbulence from the engine might affect the mixing.

Unlike triplets. A mismatch in stream size and momentum between the oxidizer and the fuel in unlike doublet elements will force the spray away from the desired axial direction and distort the fan, resulting in poorer mixing. This may be avoided by using a symmetrical, unlike-injection element consisting of an axial central stream of one propellant and two symmetrically-impinging outer streams of the outer propellant. In most propellant combinations, the total oxidizer flow area will be greater than that of the fuel so an oxidizer-fuel-oxidizer layout is used.

This approach tends to be sensitive to stability problems, due to the fact that they might produce oxidizer rich streaking on the perimeter of the combustion chamber in the oxidizer-fuel-oxidizer layout.

Obviosly this idea could be extended to any number of streams. A greater number of streams might be suitable for very fuel-rich mixtures (like with gas generators). However, this approach is senstive to combustion instability and is probably worst than the triplet element.

(f) Like-Impinging Elements

Like doublets. Like-impinging (or self-impinging) elements impinge the injected streams (liquid or gas) directly on other streams of the same propellant. The most common of these, a doublet configuration, has two like-fluid streams angled together to an impact point, producing in a fan-shaped spray of droplets similar to that of an unlike doublet. However, there is no mixing within this fan, since only one reactant is present in each. Energy dissipated by the impingement atomizes the liquids. Orientation of the initial fans for secondary impingement and overlapping of the sprays mixes the two propellants. Provision of this interaction between doublet sprays will be the key to a succesful like-impinging doublet design. This is frequently used in liquid-liquid propellant systems in which reacton or heat transfer between unlike-impinging streams is undesirable.

Like-impinging triplets. The like-impinging triplet usually produces narrower spray fans and larger drops thn an equivalent doublet, resulting in an overall net loss rather than gain.

6. Throttling

Many ricjet engines have a requirment for variable thrust over a wide range of operating conditions. This is an additional concern for injector designers. Most well-designed injector systems with reasonable pressure isolation can be throttled through a modest range of operating levels by upstream flow restriction. However, the operating range will be limited in liquid injection systems because of pressure drop/flow relationships and the need to provide a significant pressure drop across the injection elements. Namely, liquid pressure-drop changes as the square of flowrate while chamber pressure is linear with flow. This implies that significant throttling would lower the injection-pressure to dangerously low levels. The Saturn/Apollo Lunar Module descent-engine required a 10-to-1 throttling range, and new approaches had to be proposed to meet this demand.

7. Experimental Evaluation of Injector Designs

The design of an injector can be improved by proper application of experimental results obtained in nonreactive (cold-flow) and hot-fire testing.

Cold-flow calibration tests. The most basic cold-flow tests calibrate flow resistence and visual assessment of the injected streams. These are done before any hot-fire testing. Flow-resistence data is required for setting inlet pressures.

Cold-flow mixing tests. Liquid-liquid patterns can be evaluated fairly well by using two nonsoluble liquids as propellant simulants and collecting the spray in a grid system. The grid is then examined for mass ratio. Gas-liquid nonreactive mixing tests are much more difficult to do and are subject to greater unknowns.

Atomization testing. Attempts to examine the ability of the injector to atomize the propellants. This has proven to be very hard to do without hot-fire testing.

Hot-fire tests. Usually first applied to subscale simulations of planned larger assemblies. Stability characteristics are often assessed by introducing chamber-pressure disturbances such as *bombs*, *pulse guns*, or rapid perturbations of propellant flowrates. The time required to damp out any ringing resulting from such disturbances helps indicate stability margin in the system. Other elements of the system are also taken to their extremes. Due to the necessity of using specific measurment tools during testing, appropriate ports for pressure, temperature and vibration measurements must be incorporated into the design of the injector.

8. Gas Generators

In liquid-propellant engine systems, gases are required to power the propellant feed systems and other subsystems. Gases in the range from 1200 to 1700^oF have been used to drive gas turbines for pump-fed systems.

The following are the design objectives for operational gas generators:

- Ability to produce gases safely with required properties, in a compact unit at the required flowrate.
- Ability to start and stop smoothly without abrupt temperature surges, pressure oscillations, or overflow of unburned propellants.
- Ability to operate over a wide range of propellant flow rates and (in case of bipropellants) mixture ratios and to respond closely to the control system.
- Ability to maintain safe shutdown without complicated purging and draining systems.
- Ability ot restart safely (for restartable engine systems only).

(a) Solid-Propellant Gas Generators

Solid-propellant gas generators are used in liquid-propellant propulsion systems for limited-duration applications, such as providing a supply of pressurized gas to power turbines for engine start or to provide gas pressurant for short-duration, pressure-fed systems. The temperature of gases generated by solid propellants generally exceed 2000^oF, which is too high for uncooled components over extended duration.

(b) Liquid-Monopropellant Gas Generators

Monopropellants such as hydrogen peroxide (H_2O_2) and hydrazine (N_2H_4) have been used as gas generants in many applications. These systems are relatively easy to control and the gases ae generated at predictable temperatures. The generator system introduces a third propellant, often requiring special handling and tankage.

(c) Liquid-Bipropellant Gas Generators

Bipropellant gas-generation systems are used in most current large engines because they utilize the same propellants and generally provide the most flexible high-performance systems. Primarily these components provide the motive gases for turbine-driven machinery in the engine (mainly the propellant pumps). This application requires a discharge temperature that the turbine blades can accomodate. Generally, delivered temperatures can be held bellow 1800^oF, and current practice is closer to 1140^oF. Modern rockets in the US (which use $LO_2/RP-1$) use gas generators to drive the turbine of a pump-fed system. The control system consists of two normally-closed linked poppet valves which control the flow of propellants to the gas-generator injector. The valve assembly is actuated by gas pressure, which forces the piston down on the fuel side, to open the fuel poppet. A yoke integral with the piston actuates the oxidizer poppet. The valve design, through a combination of manifolds and LOX-poppet adjustments, effects a slight oxidizer lead to prevent detonations and a fuel-rich cutoff to avoid possible turbine burining. The propellants flow through the poppets to the injector and into the gas-generator combustor and are mixed and reacted within the inner chamber and combustor body. Two pyrotechnic igniters start the propellants. A gas duct with two opposing flanges is located at the end of the combustor body. These flanges connect with the solid-propellant gas generator's turbine spinner and the turbine-inlet duct.

The basic design parameters for bipropellant gas generators are virtually the same as for thrust-chamber-injector/combustor designs. In general, the effective L^* of a gas generator (the characteristic length of the combustion chamber) will be about twice that of an equivalent thrust chamber and the effective contraction ratio will also be greater. This is due to the lower vaporization rates of liquid propellants at lower temperatures and the sensitivity of the turbine structure to relatively small levels of temperature striations ("streaks").

Because of the relatively low product-gas temperatures, gas generator chamber designs do not usually employ cooling provisions.

(d) Thrust-Chamber Gas-Tapoff Systems

In these systems, combustion-product gases are bled from the main thrust chamber and ducted to the turbine, where they are used as the working fluid. In a tapoff system, the bulk of the extremely hot gases of the main chamber would not be suitable as turbine-drive fluid because of the limitations of the turbine construction materials. It has been successfuly demonstrated, however, that by withdrawing chamber gases from the boundary zones only, and by proper shapping and location of the bleed ports, the desired turbine inlet temperature (usually $< 1700^oF$) can be reliably and repeatably produced. Some other considerations need to be taken into account if the engine requires throttling.

9. Ignition Devices

Release of chemical energy stored in liquid propellants can be initiated by a number of methods; and they have been used for both thrust chambers and gas generators. Selection of an ignition system depends on the nature and phase of the propellants (bi- or mono-propellants), need for altitude start, need for restart, system safety, compatibility with overall engine design, weight and space considerations.

All ignition methods, particularly those for bipropellant systems, have one overriding requirment in common: rapid, reliable ignition of incoming propellants before accumulation of reactive material. Propellants entering the combustion chamber not promptly ignited can cause explosive mixtures to form and detonate.

(a) Igniters

Igniters are defined as devices that release heat and thereby initiate reaction of the main propellants. Igniters derive power from an outside source or from a limited quantity of internally stored energy. Once ignited, main propellants remain ignited, so it is not necessary to maintain an ignition source, although some systems do.

Pyrotechnic igniters. Pyrotechnic igniters are electrically initiated slow-burning pyrotechnic torches. They are designed with solid-propellant technology and typically incorporate one or more solid-propellant charges. Burn duration ranges from 2 to 10 seconds. Pyrotechnic igniters can be mounted to the thrust-chamber injector at the injector face. They are initiated by electric squibs that light the igniter propellant. High-voltage currents are used to prevent spontaneous ignition (500 V AC-current). For reliability, redundant igniters are often used.

The reason pyrotechnic igniters are generally avoided today are:

- Electro-explosive interfaces and electrical components necessutate redundency.
- Many large injectors are compartmented, requiring separate ignition sources for each compartment.
- Pyrotechnic devices are encumbered by stringent safety requirments that raise hardware costs.
- Field checkout and installation of components are frequently required.
- The igniter must be installed for each engine test and for launch.

Hypergolic igniters. The term *hypergolic* refers to a bipropellant combination that ignites spontaneously when the two components meet. Such a system was used as an ignition source for the German V-2 engine, which used hydrazine-hydrate($N_2H_4 \times H_2O$) and 80% hydrogen peroxide(H_2O_2). Upon an ignition signal, a ground-mounted supply unit, including remotely operated valves, fed the two components to the injection region, where they burned with a spontaneously igniting hot flame. This method has a disadvantage of frequent clogging of feed lines, and the need to eject a considerable amount of inert solid material made it undesirable. Also, adapting the method for repeated starts would be difficult.

An improved method to the above uses a *hypergolic slug.* In this design, a small amount of fluid, hypergolic with one of the main propellants, is stored in a cylindrical cartridge that has burst diaphragms at both ends. The cartridge, in turn, is loaded into a housing forming part of a bypass line paralleling a high-pressure main-propellant-feed line. A fluid chosen to be hypergolic with the oxidizer but neutral to the fuel would be installed in the fuel system, and vice versa. The former case is more common and we will focus on it.

When the turbopump starts and outlet pressures rise, the oxidizer valve is opened. As pressures rise further, burst diaphragms in the hypergolic-fluid cartridge rupture and the fluid meets with oxidizer in the chamber, igniting spontaneously. The fuel following the slug sustains the ignition flame. The main fuel valve is opened now and all parameters reach mainstage level.

Triethylaluminium(TEA), a room-temperature storable liquid, hypergolic with liquid oxygen, offers excellent ignition-delay characterstics. However, a major combustion product is aluminium oxide, a tenacious solid that can clog small feed passages and orifices.

Triethylboron(TEB), another room-temperature-storable liquid hypergolic with oxygen, produces a softer, less-tenacious residue than triethylaluminium. However, TEB has demonstrated long ignition delays when injected into very cold oxygen. Mixtures of 10-15% by wieght of TEA in TEB provide satisfactory ingition-delay characteristics with liquid oxygen, while producing an acceptable residue. Just a fraction of a cubic inch of hypergol will cause ignition at each location.

The hypergol-slug system has not found application for gas-generator ignition, because it generates solid particles and requires an oxidizer lead; these conditions are not compatible with most turbine-drive designs.

Most, if not all, hypergols are toxic.

Spark plugs. Spark-ignition systems are well-suited for applications requiring repeat starts witout servicing between starts. The plugs can be used in direct spark applications or to ignite propellants in a precombustor or torch device. Direct-spark systems, in which sparkplugs directly ignite gas generators or main-injector propellants, are not usable for all applications.

Three types of spark-ignition systems and three types of sparkplugs have been used for rocket-engine ignition. High-tension and low-tension capacitive-discharge and inductive discharge ignition-exciters have been designed, fabricated, and tested. Sparkplugs have been designed with air, surface, and shunted-surface gaps. The most robust, flexible electronic systems incorporate high-tension, capacitive discharge exciters mated to recessed surface-gap plugs. The designer should provide dual, redundant plugs in all high-reliability spark applications to preclude system failure.

Spark-torch igniters. Limitations of direct-spark systems in providing a widespread ignition source, in withstanding spark-quenching GG pressure envirionments, and in surviving main combustor operating conditions led to development of spark-torch ignition systems, or *augmented-spark-ignition(ASI)* systems, in which small amounts of engine propellant are fed into an igniter combustor and ignited by electric sparks. THe flame output is ducted to the required area and, in turn, ignites main the main propellants.

There are other igniters mentioned in the book that I did not go over.

(b) Ignition Detection

Ignition detection methods include:

- *Visual detection.* This method was used exclusively for the German V-2 engine, but has become increasingly difficult with mother rockets.
- *Optical detection.* Ground-mounted optical devices can be moved close to the chamber exit.
- *Fusible-wire links.* A wire is strung across the chamber exit, wich, when fused by the ignition flame, interrupts a circuit and signals that ignition has occured. For pyrotechnic igniters, the wire can be broken by inert particles or even by a dud igniter coming out of the chamber, giving a false positive. This is overcome by providing redundency with multiple parallel wires.

- *Pressure-sensing devices.* The idea is to try and measure the pressure increase inside of the combustion chamber resulting from the burning igniter flame. Good for multi-start engines.

Other methods are discussed in the book.

10. Combustion Instability

Combustion instability is defined in terms of the nature of pressure fluctuations in the combustion chamber. In unstable combustion, large concentrations of vibratory energy appear at one or more frequencies in the spectrum and can easily be recognized against the normal random-noise background. High-frequency combustion instabilities carry the potential for serious damage and catastrophic engine failure. Low-frequency instabilities can interfere with vehicle operation or damage instrumentation.

Guidelines for Combustion Stability Specifications and Verification Procedures for Liquid-Propellant Rocket Engines suggests that sustained oscillations with peak-to-peak amplitudes greater than 10% of the steady-state chamber pressure are generally considered as constituting combustion instability.

3 Design of Gas-Pressurized Propellant Feed Systems

4 Literature

- Dieter K. Huzel, David H. Huang, *Modern Engineering for Design of Liquid-Propellant Rocket Engines*