NASA-TM-110632

IN-28-TM

P-23

Chapter 1

PROPELLANT INJECTION SYSTEMS AND PROCESSES

Jackson I. Ito
GenCorp Aerojet Propulsion Segment
Sacramento, California U.S.A.

1.1 INTRODUCTION

The Liquid Propellant Combustion Device has always presented design and development risks due to its required harsh operating thermal environment, usually at high pressures, with its secondary goals for small packaging, light weight, high performance efficiency and low cost. The *injector* design has always been recognized as a key component which often controls the success or failure of the combustion device.

When rocketry was in its infancy, the injector design was mainly developed through a time consuming and costly process of trial and error. Once a degree of success was achieved, designers attempted to copy previously successful designs. This approach did not always yield the desired results. Eventually, successful Engineers recognized that it was not copying the *hardware* that assured success, but the proper scaling and control of the *combustion process*. A design that works well for one application may fail in another due to some subtle difference in operational requirement or system constraint. Analytical tools are now available or are being developed to evaluate these critical combustion processes so that candidate designs can be evaluated and optimized conceptually, thus avoiding or minimizing some of the detailed design, manufacturing and test cycles historically required. Even where the models may be incompletely understood or uncertainties exist, it may still be possible to conduct smaller scale, faster and lower cost experiments to validate necessary assumptions or to plan parallel design approaches for a few high risk components to increase subsequent probability of success at lower overall development cost.

Chapter 1 will address the key issues that the designer needs to identify so that they can pick and choose from the technical capabilities provided by the remaining presenters at this **Second International Symposium on Liquid Rocket Propulsion.**

1.2 ROCKET APPLICATION DESIGN REQUIREMENTS

Before one can expect to achieve success in a combustion device design, it is necessary to determine its functional requirements. It is also helpful to understand what types of development risks are most likely to be encountered and what other constraints are imposed by the system within which it will be expected to operate. This allows prioritization of limited technology resources to assure solution of the most troublesome problems before committing an entire system design approach. These requirements can be separated into three major categories.

(NASA-TM-110632) PROPELLANT INJECTION SYSTEMS AND PROCESSES (GenCorp Aerojet) 23 p

N95-26781

Unclas

G3/28 0049665

1.2.1 Thrust Level and Operating Pressure

This requirement determines the size and weight of the combustion device. Figure 1.1 illustrates the range of various combustion devices known within the international propulsion community[1-4].

Boosters are the largest and highest pressure engines. Their high thrust is required to accelerate the entire vehicle's Gross Lift Off Weight including sustainer and/or upper stages as well as payload into orbit. Their high pressure is required because they need to accelerate the nozzle exhaust gases against the atmospheric pressure to high Mach Numbers in order to maximize specific impulse performance. Since their tank volumes are very large, these boosters are pump fed from light weight, low pressure propellant tanks just sufficiently pressurized to suppress pump cavitation.

Sustainers or second stage vehicle propulsion devices effectively operate outside of the earth's atmosphere. They can achieve high performance by merely expanding to very high nozzle exit to throat area ratios. They do not need to operate at as high chamber pressure as boosters and can be either pump fed or operate from pressurized propellant tanks.

Upper Stages are still smaller versions of sustainers. Their propellant mass fractions relative to total stage weight are less than for their lower stages. Thus to save both the weight and costs of a pumping system, they are usually fed from pressurized tanks.

Reaction Control Systems or Satellite Propulsion engines are the smallest rocket thrusters available. These thrusters provide in-flight vehicle guidance or provide in-orbit satellite station keeping functions. They are virtually always pressure fed and operate at low chamber pressures.

1.2.2 Propellant Type

Commonly used propellants can be categorized into three major families which differ in their relative volatilities.

Cryogenics remain in liquid form only if kept sub-cooled below ambient temperature. The most common cryogenic propellant combination is liquid oxygen (LO_2) and liquid hydrogen (LH_2). This pair has the advantage of high specific impulse performance and is environmentally non-polluting. In most cases the hydrogen, which is an excellent coolant, is used to regeneratively cool the combustion chamber and nozzle. Thus it is usually in a gaseous state by the time it is injected into the combustion process.

Liquid Oxygen/Hydrocarbon - The most commonly used hydrocarbon is kerosene due to its ready availability, low propellant cost, ease of storability and moderate bulk density which reduces fuel tank structural weight compared to liquid hydrogen. The LO_2 is highly volatile compared to kerosene. Hence, the combustion chamber length must be designed for the vaporization limited fuel rather than for the oxidizer. This propellant combination offers challenges to balance off the design requirements for thermal cooling, high performance efficiency and combustion stability and could benefit greatly from a systematic combustion analysis approach. Some research test firings have also been conducted with Liquefied Natural Gas (LNG), liquid methane (CH_4) and liquid propane (C_3H_8). All of the latter are typically stored at temperatures approaching that of LO_2 and are sometimes referred to as "Space Storables".

Earth Storables - The oxidizer is usually a nitric acid (HNO_3) mixture or other oxide of nitrogen such as nitrogen tetroxide (N_2O_4). The most common earth storable fuels are amines, a member of the hydrazine (N_2H_4) family or its derivatives. These propellants are liquids at ambient temperature and pressure and are usually hypergolic on contact. Thus, a separate ignition system is not required.

1.2.3 Engine Cycle or Feed System

The Engine Cycle dictates the Propellant Injection System that the Combustion Device and Injector designers must contend with due to its pre-conditioning of propellant states at various component interfaces. A more detailed discussion of liquid rocket engine cycles will be presented in Section 5 (Chapter 21).

Pressurized Propellant Tank provides the simplest feed system. The typical tank pressurant is gaseous helium or nitrogen. Helium is usually used in flight due to its lighter weight; whereas, nitrogen is usually substituted during use in ground test facilities due to its ready availability and low cost. Ground test facilities capable of operating at high pump fed system pressures are usually utilized for initial combustion device development testing to allow parallel development of both combustion devices and turbopumps; but for purposes of this discussion the injectors will be referred to as pump fed designs. To minimize tank structural weight penalty, pressure fed flight tank pressures are kept low and both combustion device operating pressures and feed system pressure drops are also minimized. To further minimize pressurant storage bottle and gas weight, the propellant tank pressure may only be regulated over the initial portion of its mission and permitted to operate in a blowdown mode to its propellant exhaustion. This requires the combustion device to operate in a throttled (reduced thrust) mode late in its mission.

The Gas Generator Cycle is the simplest form of the pump fed engine cycles. A small portion of the main engine propellants are bypassed and burned in a separate combustion device operating at low combustion gas temperature in order to power a turbine which in turn drives the propellant pumps. Since the turbine exhaust gases are dumped overboard at low temperature and low pressure, the lower gas generator exhaust gas performance reduces the overall engine system performance. Because the gas generator mass flowrate fraction must increase linearly with the required turbine horsepower, a tradeoff has to be made between increasing the main combustion device performance with increasing operating pressure against an increasing gas generator engine cycle loss. These systems usually optimize performance at moderate pressures.

The Staged Combustion Cycle flows all of one propellant and a small fraction of the other to keep combustion gas temperature low enough to permit turbine drive then injects the remaining propellant downstream of the turbine to recover maximum engine performance at high gas temperature. The first combustion device referred to as a "Preburner" has similar design criteria as a "Gas Generator" except it is usually larger in size since it has to accommodate a higher mass flowrate and operates at considerably higher pressure since the turbine pressure ratio is in series with the main combustion device rather than being in parallel as in the gas generator cycle. The turbine mass flowrate available in a staged combustion cycle engine greatly exceeds the mass flowrate available in a gas generator cycle engine. Hence, it can thermodynamically optimize performance at significantly higher operating pressures than a gas generator cycle engine. In actual practice, the staged combustion operating pressure is limited from an engine reliability standpoint to a thermal limit to which the combustion device can be cooled. The main combustor in a staged combustion cycle is a gas/liquid injection system since one propellant circuit has already been pre-vaporized before entering the turbine.

An Expander Cycle is somewhat similar to a staged combustion cycle in that no turbine drive gases are exhausted overboard. It has the further simplification that it does not require either a preburner or a gas generator. The turbine drive gases are heated while regeneratively cooling the main combustion chamber and nozzle. In practice, the expander cycle has only been developed for the oxygen/hydrogen propellant combination. Only hydrogen can provide adequate cooling to the regenerative main combustion chamber and still be heated sufficiently to drive the turbine. While hydrogen is an excellent combustion chamber coolant and delivers high combustion performance, it presents a serious challenge for the fuel turbopump designer. Its low density requires high pump speeds and/or multiple pump stages in order to raise its hydrogen pump discharge pressure. This difficult to achieve hydrogen pressure in turn is subject to chamber and nozzle coolant pressure losses, it must supply the required turbine pressure ratio, then still have sufficient pressure remaining to meter the flow into the injector and provide chamber pressure. Expander cycle engines therefore operate at much lower pressures than either staged combustion or gas generator cycle engines. This is not a disadvantage for an upper stage engine operating in space, but it is a serious limitation for a booster engine. Expander cycles also optimize for lower thrust level engines which have a more favorable exposed heating surface area to engine flowrate ratio which also makes it ideal for an upper stage application.

1.3 COMMON COMBUSTION DEVICE DEVELOPMENT RISKS

The different types of combustion device applications discussed in Section 1.2 have different degrees of technical risks. All combustion devices are potentially susceptible, however, to the following primary development problems.

1.3.1 Combustion Instability

Combustion Instability has been the single most significant combustion device development problem since the beginning of Liquid Propellant Rockets. An early recognition of the importance and magnitude of the technical concern in the U.S.A. is indicated by the broad range of investigators contributing to a systematic sharing of viewpoints compiled in 1965[5]. Likewise, demonstrating that Combustion Instability is still a major development risk within the propulsion community, the entire *First International Symposium on Liquid Rocket Propulsion* was devoted to this single subject[6]. The First Symposium was held at the Propulsion Engineering Research Center at the Pennsylvania State University, University Park, Pennsylvania U.S.A. from 18-20 January, 1993.

The deadliest form of combustion instability is usually referred to as "**High Frequency Combustion Instability**" which is characterized by a coupling between the propellant burning rate with one or more of the transverse combustion chamber acoustic modes. This causes a substantial increase in the forward combustion zone heat flux and the usual result of a high frequency combustion instability encounter is immediate catastrophic failure due to a burnout of the combustion chamber and/or injector. Hence, the understandable concern for the phenomenon and its solution(s). This problem is most serious for large booster engines and decreases in severity with diminishing engine size. The problem is most common for liquid oxidizer/liquid fuel injectors utilizing the LO_2/Hydrocarbon or earth storable propellant combinations. It is a lesser problem for the LO_2/H_2 propellant combination, gas/liquid injectors and in general for small thrusters. Acoustic coupling also occurs with the combustion chamber longitudinal modes between the injector face and nozzle throat plane called "longitudinal combustion instability", but these modes are generally less damaging.

"**Low Frequency Combustion Instability**", also called "chugging", is characterized by a coupling of the propellant burning rate with the hydraulics of the propellant feed system. This problem is aggravated by low injector pressure drop and selection of injection elements with long atomization and/or vaporization combustion time lags. The combustion device may not be at risk of catastrophic failure as a result of low frequency combustion instability; but, sensitive payloads may incur structural failure-particularly if they possess natural frequencies which could resonate with the chug frequency.

Gas/liquid injection systems could be susceptible to an additional risk from either sufficiently high amplitude feed system coupled or longitudinal acoustic mode combustion instabilities. The rising pressure at the injector face could cause compressibility of the gaseous propellants to result in flow reversal of the combustion gases into the injector manifolds. If the backflowing combustion gases also entrain unvaporized liquid droplets into the opposite gas manifold, they could result in manifold detonation and its structural failure. There have also been occasions when supposedly "non-damaging" forms of combustion instabilities such as longitudinal or chug mode combustion instability perturbations have triggered the fatal transverse acoustic mode. Thus, all forms of combustion instabilities should be avoided even if they appear to be doing no harm at first assessment.

1.3.2 Combustion Chamber Overheating/Burnout

Considerable progress has been made in combustion chamber heat flux and wall cooling predictive technology. This topic will be covered in more detail in Section 3 (Chapter 16). To seek higher performance (which is always a goal), especially for low altitude booster engine applications, the first reaction is to increase chamber pressure to expand the exhaust gases to a higher nozzle exit area ratio to achieve higher Mach Number. Higher heat flux accompanies higher operating pressures.

For a given *regeneratively cooled combustion chamber* material and wall thickness, a higher heat flux increases the wall temperature differential between the inner coolant wall and outer hot gas wall. The wall thickness can be reduced to limit the maximum hot gas wall temperature. However, the walls must also be designed to withstand a maximum design wall pressure differential, which sometimes occur during transients. This might be achieved by reducing the cooled wall span, which in turn reduces the coolant passage hydraulic diameter and increases the coolant pressure drop.

From the injector design standpoint, one desirable solution would be to reduce the combustion gas temperature immediately adjacent to the walls by incorporating lower mixture ratio injection elements or pure fuel film cooling injection orifices. Making the cool zone wider than absolutely necessary will reduce engine performance inversely with the engine throat diameter. Combustion chamber thermal design margin is determined by the hottest local streak temperature irrespective of the average gas temperature. Regenerative coolant passage burnout resulting in internal leakage is usually self limiting and seldom results in immediate catastrophic failure. It will, however, cause a loss of engine performance and may cause off design engine mixture ratio operation to deplete the fuel tank before all of the oxidizer is consumed compromising mission payload objectives.

An easier solution for sustainer and upper stage engines is to simply operate at lower chamber pressure to reduce heat flux. The only performance penalty of low chamber pressure for an engine in vacuum is possibly a minor increase in the nozzle boundary layer and recombination kinetics performance losses.

Reaction Control Systems and Satellite Propulsion devices have insufficient propellant consumption rate to regeneratively cool their combustion chambers. Furthermore, these engines are frequently required to fire short repeated pulses and require a rapid response. Thus this class of thruster typically use earth storable propellants and depend upon *fuel film cooling* to provide the required thermal margin.

Fuel film cooling thermal and performance characteristics vary widely depending upon the type of fuel being utilized. Hydrogen fuel film cooling is primarily effective when it reduces the local recovery temperature near the design wall temperature. Its high heat capacity still results in relatively high heat flux at moderate temperature ranges. On the other hand, its low molecular weight results in high film coolant specific impulse and low cooling performance losses. The amine fuels undergo monopropellant decomposition and provide relatively stable and predictable heat flux reduction and performance reduction. Hydrocarbon fuels provide very great cooling capacity due to its highly endothermic decomposition, but its performance degradation is also high. Its best thermal to performance trade occurs when its nozzle throat plane recovery temperature is approximately half the stoichiometric temperature. Some trial and error is still required to establish the optimum percentage of fuel film cooling.

1.3.3 Injector Face Erosion

Injector face erosion is a potentially mission compromising failure mode for high pressure engines if it results in burnthrough into the injector manifold. Such an occurrence will result in loss of engine performance, off mixture ratio operation and premature depletion of one propellant tank before the other resulting in possible significant reduction in payload terminal velocity.

Face erosion in low pressure engines is usually limited to superficial erosion which stabilizes after some reduction in local faceplate thickness. This statement precludes the occurrence of combustion instability.

Injector face heat flux models are relatively immature compared to combustion chamber and nozzle thermal models. What has been observed is that high injection velocities tend to aggravate the face heat flux by increasing the recirculation strength. Injector face erosion can be particularly troublesome for the oxygen/hydrogen propellant combination or gas/liquid injection systems if raw oxidizer rich sprays are allowed to recirculate back to the injector face.

1.3.4 Low Thrust Chamber Assembly Performance.

Everyone recognizes the importance of high performance. It is an emotional issue. Low combustion device performance is readily measurable and highly apparent to everyone. Thrust based Specific Impulse (I_{sp}) measurements are most accurate. Chamber pressure based Characteristic Exhaust Velocity (C^*) measurements, although less accurate, can be measured with less sophisticated and cheaper test facilities.

The typical reaction to a low performing combustion device by the novice injector designer is to replace the injector with another having more smaller injection orifices in the belief that more complete combustion will yield higher performance. More often than not, however, the modification can result in combustion instability.

The knowledgeable injector designer understands that low performance can be attributable to any one or more of the following three causes: (1) Non-uniform oxidizer to fuel injection distribution across the injector face, (2) Inadequate (too large) atomization resulting in incomplete droplet vaporization, or (3) Incomplete mixing of fully vaporized combustion products.

1.3.5 Unsafe Transients

Relative to the total range of possible combustion device failure modes, too much time and resources are spent studying the steady state design point and too little recognition is paid to the possible transient operational risks.

Propellant Type transient risks are as follows. Liquid / liquid earth storable engines have the simplest start transients. Liquid oxygen/Hydrocarbons are of intermediate risk. For example, if hydrocarbon contamination were to occur within the LO_2 manifold during an engine shutdown transient from a previous test, the subsequent test start transient could be at risk of having a LO_2 manifold detonation. Cryogenic engines have the most complex start transients because they have severe thermal childown constraints in addition to the usual pressure variation considerations.

Engine Cycle transient risks are rated as follows. Pressurized tank feed systems are easiest to operate. The gas generator cycle has the simplest transient among the pump fed systems. The staged combustion cycle is considerably more complex. The expander cycle is most difficult to start due to its low turbine power margin and deep throttling with a low pressure drop feed system.

Too often, excessive importance is placed upon rapidly achieving steady state pressures and too little attention is paid to understanding the physics of the slow temperature transient. This is especially true for cryogenic propellants and gas/liquid systems.

Possible negative effects attributable to improper transients are: (1) more flight failures have resulted from non-ignition or non-restart of cryogenic upper stages than from any other failure mode, (2) delayed ignition, (3) hard starts, (4) combustion gas reversal causing fire within injector manifolds, (5) engine vibration due to feed system coupling during deep throttle operation, (6) gas generator or preburner temperature spikes to turbine blades, or (7) rapid, cold cryogenic hydrogen quenching of hot turbine blades and/or hot combustion chamber wall. Some failures can result in immediate flight termination and mission loss while others prematurely limit component cycle life.

1.4 INJECTION SYSTEM DESIGN CONSIDERATIONS

To simplify this discussion it will be assumed that the Vehicle and System level trades have already been completed. Assume that the combustion device and injector designers have been given the following design requirements: (1) propellant combination, (2) engine thrust, (3) mixture ratio, (4) pressurized tank or pump discharge pressures, and (5) combustion device length and diameter envelopes.

The following important design parameters must be taken into consideration and preliminary baseline values (subject to continuing review) should be established.

Engine Pressure Schedule - The total pressure available must be allocated between (1) combustion chamber pressure-important to maximize for booster applications to obtain high performance, (2) regenerative coolant pressure drop, if applicable-must be adequate for thermal margin, (3) injector element pressure drop-must be chug stable at lowest expected flowrates, and (4) propellant distribution system-including propellant lines, valves, and injector manifolding.

Nozzle Expansion Ratio - Generally maximize to fill the length envelope for an upper stage combustion device. Check to verify that payload performance advantage over a lower area ratio, shorter length nozzle merits the weight increase and added complexity. For a booster nozzle, optimize flight trajectory performance from liftoff to second stage separation. However, must also beware of asymmetric separation induced side loads at sea level firing and during pump fed start transient.

Contraction Ratio (Area of subsonic combustion chamber to nozzle throat) - Most combustion device contraction ratios are in the two to four range. Liquid/liquid boosters are usually within the lower range; staged combustion cycle main injectors and gas / liquid injectors are in the upper range. Fuel film cooled reaction control systems and satellite propulsion engines typically have high contraction ratios. Rayleigh stagnation pressure loss due to heat addition at finite Mach Number increases rapidly at contraction ratios less than two.

Chamber Length (L') From Injector Face to Nozzle Throat Plane This length needs to be selected together with consideration for probable atomized injector drop size to achieve high (not necessarily complete) propellant droplet vaporization above the nozzle throat.

Injection Element Type and Injector Pattern Selection will be discussed separately in Section 1.6.

1.5 CRITICAL COMBUSTION PROCESSES

Sections 1.2 and 1.3 described various liquid propellant rocket engine applications and their combustion device development problems. This section will describe primary physical mechanisms through which the injector designer can establish control to solve these development problems. A schematic showing some of these combustion processes are in Figure 1.2.

1.5.1 Injector Manifold Distribution

The starting point of any injector design is proper distribution of the fuel and oxidizer across the injector face where you want it! This requirement is so basic that it should be obvious, but its achievement is often taken for granted and its importance is often overlooked. Uniform mixture ratio distribution across the injector core elements will maximize performance. On the other hand, a uniform mixture ratio at the combustion chamber wall may result in excessive heat flux which could cause thermal failure or require excessive regenerative coolant circuit pressure drop in a high pressure engine. In that case, either *fuel film cooling* or a *barrier mixture ratio bias* may be helpful to reduce wall heat flux without reducing chamber pressure. A mass weighted streamtube analysis can provide a way of quantitatively estimating the effect of mixture ratio maldistribution upon performance penalty. It can account for both intentional cooling bias and unintentional maldistribution performance losses.

Compared to the cost of injector re-design and re-testing necessitated by either chamber thermal failure or a disappointingly low injector performance due to injection maldistribution, it would seem prudent to perform simple cold flow hydraulic distribution testing of the injector manifold design prior to committing the injector design to a specific injector pattern. Injector manifold distribution represents a "Necessary But Not Sufficient" criterion for design success. That is, a non-uniform injection manifold distribution can present later development problems, but a uniform manifold distribution is only one of the many design requirements for success.

1.5.2 Injector Spray Atomization

Many liquid rocket propulsion engineers only think that *"atomization"* refers to *droplet diameter* as it affects subsequent propellant vaporization and *performance*. That barely scratches the surface of its importance. In fact, within that context, it is only the *largest droplets* which may exhaust through the nozzle throat without being vaporized that degrades vaporization performance. These maximum diameter droplets only represent the largest 10% to 20% of the total mass distribution.

Everyone acknowledges the critical importance of *High Frequency Combustion Instability*. The *sensitive time lag* is usually approximated by combustion stability analysts with the volume number mean (D_{30}) diameter which typically defines the smallest 20% of the cumulative droplet mass distribution. Other drop sizes typically mentioned in the atomization literature refer to the Sauter mean diameter (D_{32}) and mass median diameter at which half of the mass is below and half is above. It is of less importance to the injector designer to force fit a single *mean diameter* and droplet *distribution function* to describe the entire spray than it is to understand the mass distributions within the range of small, intermediate and large drop sizes required by the various combustion process analysis models.

Another critically important atomization area which few atomization investigators have recognized is the systematic study of the spatial spray atomization distribution from the injector face or from the point of jet impingement. The reason this parameter is so important to the injector designer is that this break up distance divided by the injection velocity represents a significant fraction of the *combustion dead time*. This time lag is needed by the combustion stability analyst to predict the low frequency feed system or chug stability margin that either a pressure fed thruster may be required to operate at the end of its tank pressurization blowdown cycle or the intermediate operating point that all pump fed engines must endure during its start transient before it bootstraps up to full throttle.

Another atomization figure of merit which is critical to the successful injector designer and thermal analyst is an accurate determination of the relative breakup distances from the injector face between the oxidizer and fuel spray fans in a liquid/liquid earth storable or LO_2 / Hydrocarbon injector. This is especially important for injection elements aligned adjacent to the combustion chamber wall. The atomization distance differential represents whether the fuel or oxidizer spray has a head start and the relative propellant volatilities determine whether the real vaporized mixture ratio is more fuel rich or more oxidizer rich than the injection mixture ratio at the injector face. The local axial distribution of vaporized wall mixture ratio strongly influences the chamber heat flux and its cooling margin.

Atomization can be approached in a number of different ways depending upon the resources and preferences of the investigators. They can be measured experimentally and correlated empirically during either cold flow or hot fire testing as will be described further in chapter 6. They can also be modelled analytically based on first principle theories or inferred from previous experience with similar designs.

To fully reap the benefits of atomization, not only for performance prediction, but also for both high frequency and low frequency combustion stability analyses as well as for combustion chamber wall and injector face recirculation thermal analyses, a determination of spatial atomization breakup distances is required as well as a knowledge of drop size distributions.

1.5.3 Propellant Droplet Vaporization

As early as the Mid-1950's, R.J.Priem and M.F.Heidmann of the NASA/Lewis Research Center had concluded that droplet vaporization could be the rate controlling mechanism in the liquid propellant combustion process[7]. Numerous vaporization and spray combustion models are available[5,8] which will be deferred to Section 2 (Chapters 7 through 13).

1.5.4 Bi-Propellant Mixing

Uniform mixing is essential to achieve maximum specific impulse performance. It is also required in Gas Generators and Preburners to achieve uniform turbine inlet gas temperatures which are free from hot streaks which limit turbine life. On the other hand, to maximize combustion chamber and nozzle cooling with minimum cooling performance loss, it is desirable to minimize mixing.

Low molecular weight propellant species such as hydrogen have high diffusivity and mix readily. conversely, high molecular weight propellants such as heavy hydrocarbons mix very slowly. Heavy hydrocarbons have the further disadvantage that they can build up a sufficient insulating layer of cooler fuel vapors surrounding the droplet that they can retard further droplet vaporization as well.

Hypergolic propellants which spontaneously react on contact can undergo Reactive Stream Separation also sometimes called Blow Apart which retards unlike liquid/liquid propellant mixing. Likewise, Gas/Gas injectors are notorious for their low mixing efficiencies due to rapid combustion on their mixing interface. Gas/liquid injectors mix not much differently than liquid/liquid systems.

J.H.Rupe of the Jet Propulsion Laboratory was one of the earliest investigators to recognize the importance of uniform liquid phase mixing as it related to injection element design parameters, propellant properties and injection operating conditions[9]. In essence he reported that optimum unlike mixing could be approached when the propellant jet diameters and injection momentum ratio approached unity.

1.6 CANDIDATE INJECTORS FOR LIQUID ROCKET APPLICATIONS

References[2,10] describe various injection element types which could have beneficial applications to liquid rocket injector designs. Their spray characteristics are depicted schematically in Figure 1.3. A cursory discussion of some significant characteristics and some examples of their possible advantageous application or disadvantages follow.

1.6.1 Co-Axial Jet Injectors

This is the single most common element type used for oxygen/hydrogen injectors. They come in two varieties, the shear co-ax and swirl co-ax. Both usually position the hydrogen in the outer annulus and inject the oxygen in the central jet. Since most oxygen/hydrogen thrust chambers operate in the 5 to 7 mass mixture ratio range, the shear co-ax requires a proportionately higher fuel injection velocity ratio in order to have sufficient injection momentum to adequately atomize and mix the LO_2 jet.

When there is less hydrogen injection momentum available to adequately shear the LO_2, an oxidizer swirl pattern which can either be induced by inserting a mechanical swirl device to impart rotation or by tangential injection can help self-atomize the LO_2 spray fan either with or without the added assistance of the hydrogen jet. The H_2 is usually pre-gassified by regenerative heating in the combustion chamber in a gas generator or expander engine cycle or pre-combusted within the preburner of a staged combustion cycle engine. Thus, the local vaporized mixture ratio asymptotically approaches the design mixture ratio from the thermally benign fuel rich side which benefits both injector face and combustion chamber thermal compatibility. Careful attention must be paid if swirl co-axial injection elements are positioned too close to the chamber wall. Liquid oxygen droplet wall impingement can cause local overheating on the forward chamber wall.

Shear co-axial elements, on the other hand, provide a thermally benign environment on the forward chamber wall. However, shear co-ax's can cause thermally adverse conditions upon the nozzle convergent section if the LO_2 droplets are not completely vaporized by the end of the cylindrical chamber and impinge, shatter and combust on the convergent throat section. In general, a row of finer elements adjacent to the chamber wall provide better compatibility and higher performance potential. A more detailed discussion of Co-Axial Jet Injector atomization will follow in chapters 2 and 4.

1.6.2 Impinging Jet Injectors

Many variations of impinging jet injectors shown in Figure 1.3 are utilized for liquid rocket combustion devices. Some major classifications follow.

The Like on Like Doublet was one of the earliest injection element concepts utilized for liquid rocket injectors. Its popularity was generally attributable to its stable combustion characteristics while delivering moderate performance. The like on like doublet is comprised of both self impinging fuel doublets and self impinging oxidizer doublets. The quantities of fuel pairs and oxidizer pairs need not be equal. A functional advantage can be gained by designing more impinging pairs of the less volatile propellant.

Quadlet elements are like doublet pairs which have been canted toward each other to induce improved unlike propellant mixing. For the same number of impinging pairs and comparable atomization and vaporization efficiencies as like on like doublets, quadlet injectors tend to deliver higher performance in mixing limited Injectors.

Unlike Doublets impinge a single fuel jet upon a single oxidizer jet. This injection element type works best for propellant combinations which have nearly equal fuel and oxidizer injection orifice areas and which also have nearly equal injection momentum ratios.

Unlike Triplets impinge two jets of one propellant upon a single jet of the other. Two opposing fuel jets impinging upon an oxidizer is called a F-O-F Triplet; whereas, two oxidizers impinging upon a single fuel is called an O-F-O Triplet. Most liquid/liquid propellant combinations other than oxygen/hydrogen require finer atomization of the less volatile fuel. The F-O-F Triplet tends to produce finer fuel droplet atomization for a given total injector element quantity. However, since most propellant injection combinations have higher oxidizer injection momentum ratios, the O-F-O Triplet produces better unlike propellant mixing uniformity. The choice between these two triplet orientations depend upon whether the propellant combination is more likely to be fuel vaporization limited or mixing performance limited. Special provisions for wall thermal compatibility may be required if the O-F-O Triplet is the core element of choice. The *Unlike Pentad* is a variation of the triplet elements except that it impinges 4 on 1 instead of 2 on 1.

Unlike impinging elements tend to produce finer atomization than like impinging elements of similar orifice diameter and pressure drop. They are generally higher performing, but also less combustion stable. A coarser unlike impinging element pattern will exist that produces comparable performance efficiency and combustion stability characteristics as a finer like impinging injector. A coarser pattern will probably be cheaper to fabricate, but will also provide wider thermal streaks. A further discussion of impinging jet injector atomization will follow in chapter 3. Experimental techniques for atomization measurements will be covered in chapter 6.

1.6.3 Parallel Jet (Showerhead) Injectors.

The showerhead injection element is seldom used as a thrust producing injector due to its poor atomization and mixing characteristics; however, for these very reasons, it is often used as a barrier fuel film cooling element. It can be advantageously used when the forward chamber can be adequately regeneratively cooled, but when the throat heat flux is excessive for thermal reliability margin or would otherwise require excessive coolant pressure drop. The coolant jet can be either injected axially parallel to the chamber wall, with a slight impingement angle upon the wall or with a tangential swirl component for more uniform front end coverage.

1.6.4 Injector Design Synthesis

Historically, the selection criteria for picking a particular injection element to design and develope has been subjective. Either injector designers or liquid rocket companies have favored certain element types and have used them for all applications disregarding the *Application Design Requirements* discussed in Section 1.2 or the *Development Risk Considerations* in Section 1.3. These choices may either have been based on previously successful design experiences, prior design familiarity or other subjective design considerations.

Aerojet's analytical design approach since 1966 has been based on the design considerations described in Sections 1.2 and 1.3. Atomization breakup distances from the injector face are selected as a design requirement together with a nominal design point pressure drop and injection velocity to determine allowable "combustion dead time" ranges to satisfy feed system combustion stability for transients and required throttle ranges, if applicable.

Characteristic drop sizes for the volume number mean (D_{30}) can be used to predict allowable "sensitive time lags" or characteristic high frequency combustion stability gain relative to the combustion chamber transverse resonance frequencies and combustion damping device margins. Spatial combustion profiles are evaluated or modified to assure thermal heat flux compatibility at hardware surfaces compared to regenerative cooling flux and wall thermal conductivities. The maximum high end droplet diameters are analyzed parametrically to assess acceptable performance losses due to unvaporized droplets exhausting through the nozzle throat plane for given chamber lengths. The droplet mass fractions and species (fuel or oxidizer) impinging upon the convergent throat are used to refine the throat heat flux prediction. Note that the "average" drop size which is the primary focal point of most atomization emphasis was not explicitly mentioned in these functional injector development process models.

The liquid phase or gas/liquid (Rupe) mixing efficiency (E_m) parameter can be used if known to estimate streamtube mixing performance based on distributed mass and mixture ratio distributions.

None of the foregoing Aerojet design criteria have made any reference thus far to a particular element type. Only <u>after</u> the design requirements have been quantitatively defined, does the injector designer attempt to evaluate the repertory of available injection element types, orifice diameters, injection velocities, impingement angles and other design variables to synthesize the injector design which has the highest probability of fulfilling the aforementioned design objectives.

1.7 CONCLUSIONS AND RECOMMENDATIONS

The previous *Art of Injector Design* is maturing and merging with the more systematic *Science of Combustion Device Analysis*. This technology can be based upon observation, correlation, experimentation and ultimately analytical modelling based upon basic engineering principles. This methodology is more systematic and far superior to the historical injector design process of *Trial and Error* or blindly *Copying Past Successes*.

The benefit of such an approach is to be able to rank candidate design concepts for relative probability of success or technical risk in all the important combustion device design requirements and combustion process development risk categories before committing to an engine development program. Even if a single analytical design concept cannot be developed to predict satisfying all requirements simultaneously, a series of risk mitigation key enabling technologies can be identified for early resolution. Lower cost subscale or laboratory experimentation to demonstrate proof of principle, critical instrumentation requirements, and design discriminating test plans can be developed based on the physical insight provided by these analyses.

The reason this overall procedure may appear intimidating at first is because the development of a large, high pressure, liquid propellant combustion device itself is a formidable task with many inherent risks. Injector design is a multiple jeopardy problem. There are many individual reasons that any design may become unacceptable; there are considerably fewer combinations of injector designs that satisfy the many demanding design requirements and often contradictory design trades that must be made. *However, the successful seeker will be richly rewarded by its long term cost and schedule benefits.*

REFERENCES

1. "Technology Week", 10th Annual World Missile and Space Encyclopedia Issue, Volume 19, Number 14, July 25, 1966

2. "Liquid Rocket Engine Injectors", NASA Space Vehicle Design Criteria Monograph SP-8089, 1976

3. Rowe, J. R., "Liquid Rocket Engines", (A Potential Liquid Rocket Users' Guide), Aerojet Liquid Rocket Company, March 1975

4. Isakowitz, S. J., "International Reference Guide to Space Launch Systems", 1991 Edition, AIAA

5. Harrje, D. T. and Reardon, F. H. (Editors), "Liquid Propellant Rocket Combustion Instability", NASA SP-194, 1972

6. "Combustion Instability ?", First International Symposium on Liquid Rocket Propulsion, Pennsylvania State University, U.S.A. 18-20 January 1993 AIAA Progress in Aeronautics and Astronautics, (To Be Published Sept. 95).

7. Priem, R. J. and Heidmann, M. F., "Propellant Vaporization as a Design Criterion for Rocket Engine Combustion Chambers", NASA -TR R-67, 1960

8. Cramer, F. B. and Baker, P. D. (Editors), "Combustion Processes in a Bi-Propellant Liquid Rocket Engine (A Critical Review)", Jet Propulsion Laboratory Report 900-2, Contract NAS7-100, 1967

9. Rupe, J. H., "Correlation Between the Dynamic Properties of a Pair of Impinging Streams and the Uniformity of Mixture Ratio Distribution in the Resulting Spray", Progress Report No. 20-209, Jet Propulsion Laboratory, 1956

10. Penner, S. S., "Chemistry Problems in Jet Propulsion", International Series of Monographs on Aeronautical Sciences and Space Flight, Pergammon Press, New York, London, Paris, Los Angeles, 1957, Pages 360-362

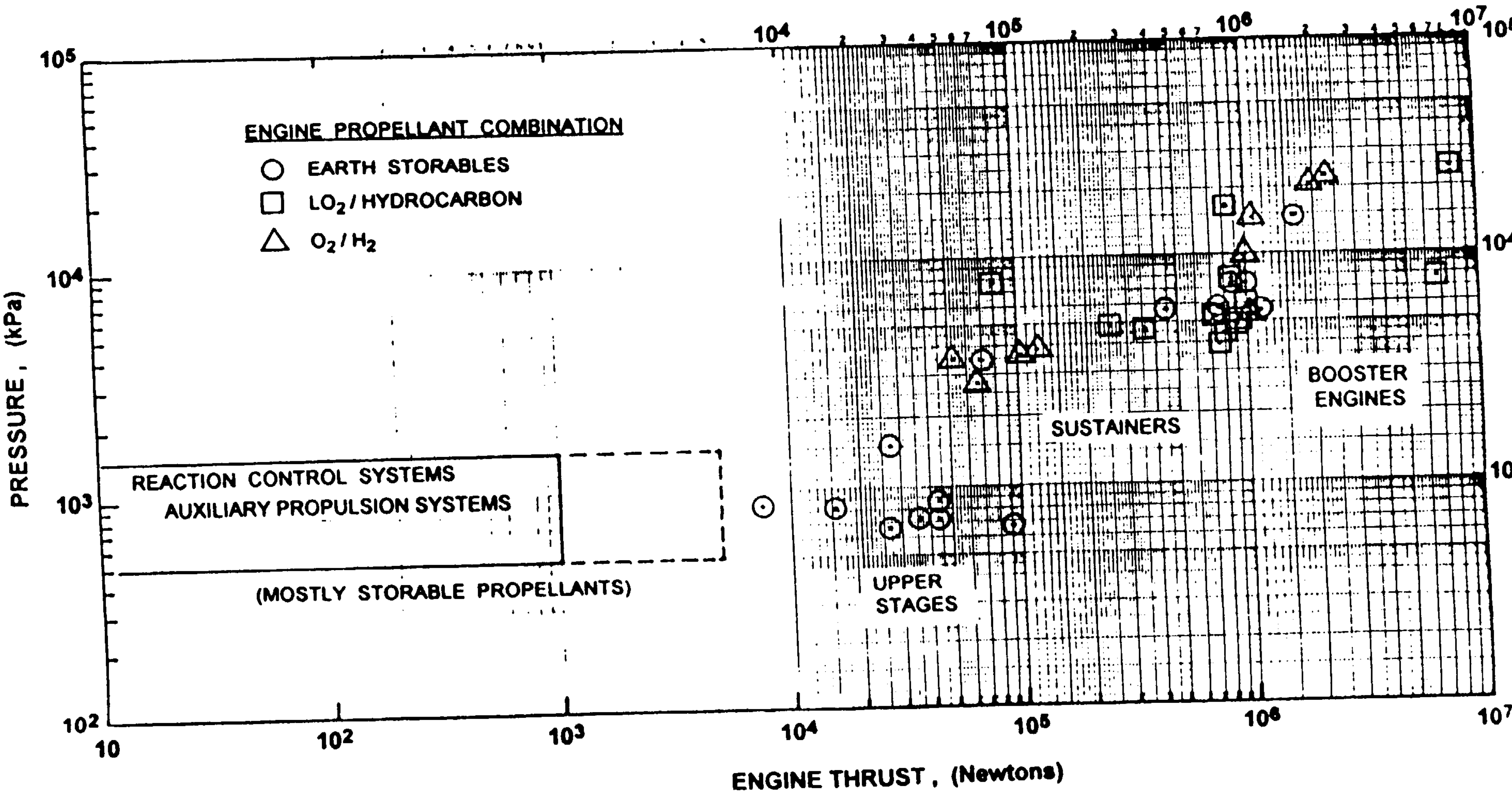

Figure 1.1 Liquid Rocket Engine Design Applications Vary Widely

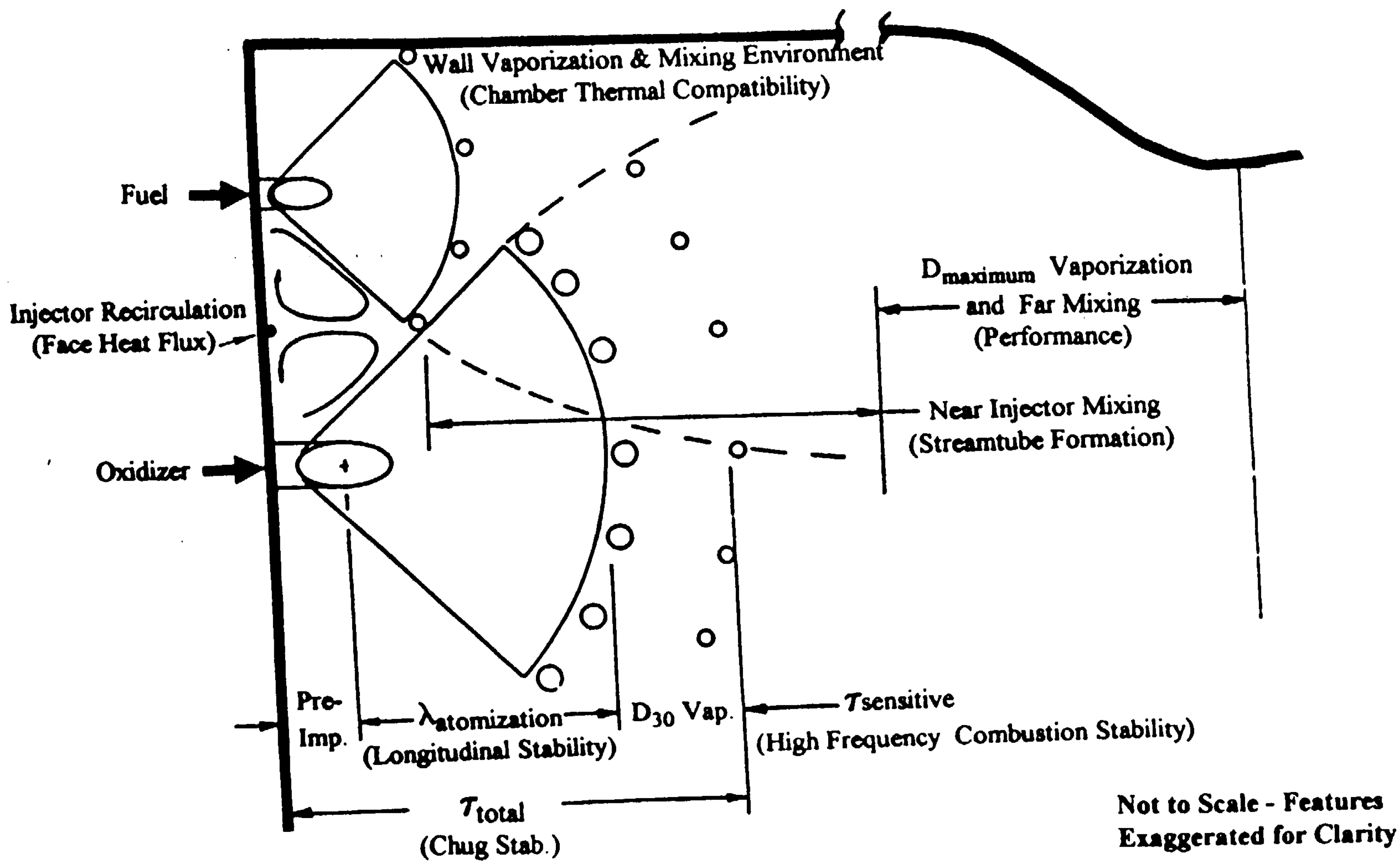

Figure 1.2 Combustion Processes are Mechanistic

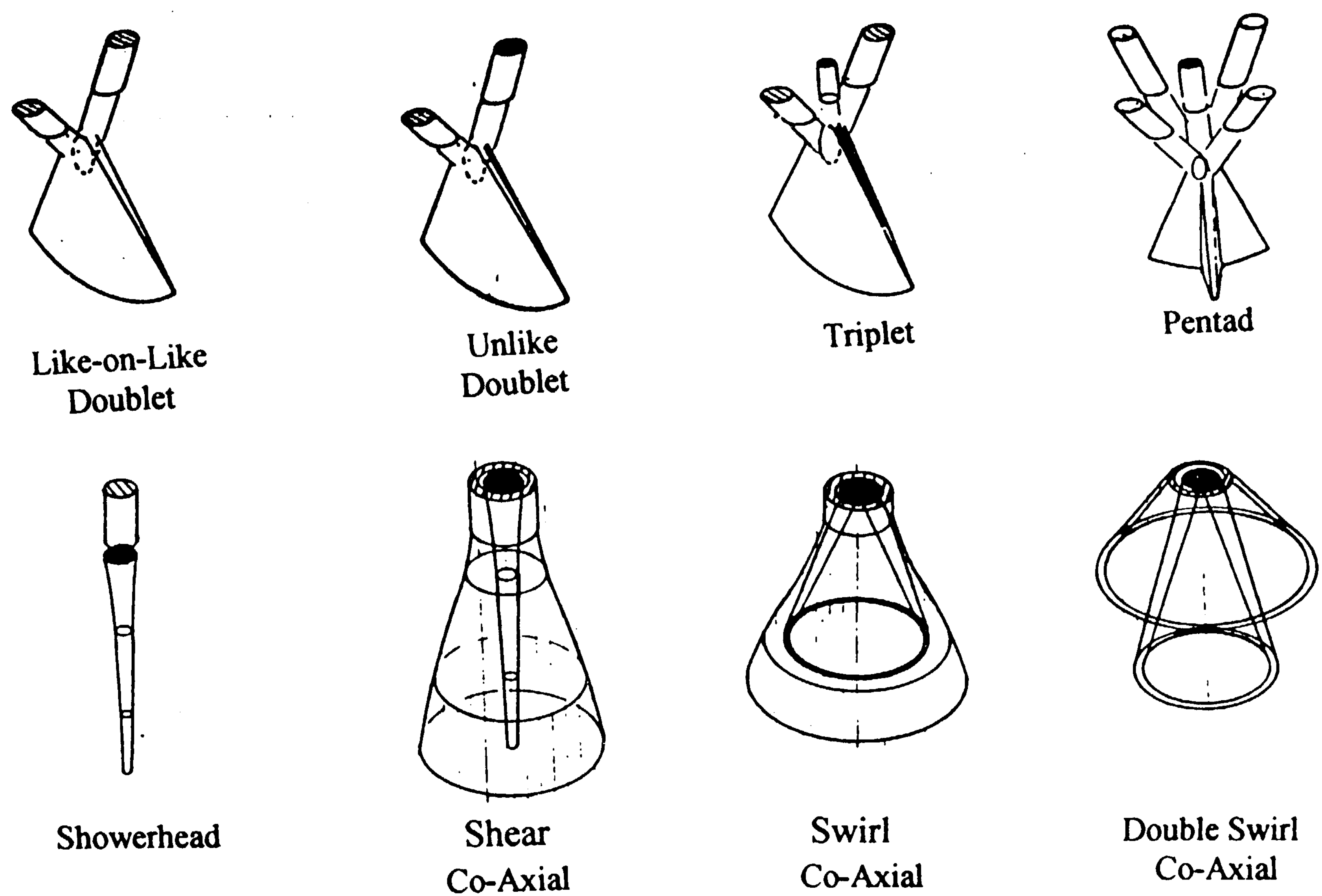

Figure 1.3 Injection Element Spray Pattern Schematics

Second International Symposium on Liquid Rocket Propulsion
ONERA, France, June 19-21, 1995

LIQUID ROCKET COMBUSTION DEVICES
Aspects of Modeling, Analysis, and Design

Section 1. Injection and Atomization Processes

1.1 Propellant Injection Systems and Processes
Principal authors: J. Ito (Aerojet, USA)

1.2 Atomization of Coaxial-Jet Injectors
Principal author: M. Ledoux (CORIA, France)
Co-authors: L. Vingert (ONERA, France), Z. Farago (DLR, Germany), M. Micci (Penn State, USA), Russians

1.3 Atomization of Impinging-Jet Injectors
Principal author: W. Anderson (Penn State, USA)
Co-authors: R. Lecourt (ONERA, France), P. Berthoumieu (ONERA/CERT, France), N. Zhu and C. Huang (China)

1.4 Correlation of Droplet Sizes for Coaxial, Tangential-Entry Liquid-Rocket Injector Elements
Principal author: J. Clark (Pratt & Whitney, USA)

1.5 Dynamics of Liquid Rocket Injectors
Principal authors: V. Bazarov (Moscow Aviation Institute, Russia)

1.6 Experimental Diagnostics and Statistical Procedures for Atomization and Spray Pattern Analysis
Principal authors: C. Tropea (LSTM, Germany)
Co-authors: R. Ingebo (NASA LeRC, USA), French

Section 2. Droplet Vaporization and Spray Combustion

2.1 Modeling of Liquid-Propellant Spray Combustion in Rocket Engine Combustors (overview paper, to identify state of the art, merits, shortcomings, critical issues, research needs, etc.)
Principal author: R. Borghi (CORIA, France)
Co-authors: F. Lacas (ECP-EM2C, France)

2.2 Liquid-Propellant Droplet Vaporization and Combustion
Principal author: V. Yang (Penn State, USA)
Co-authors: P. Lafon and M. Habiballah (ONERA, France), F. Zhuang (China)

2.3 Droplet Cluster Behavior in Dense and Dilute Sprays
Principal author: J. Bellan (NASA JPL, USA)
Co-authors: M. Sommerfeld (LMU,Germany)

2.4 Modeling of Turbulent Mixing in Liquid-Propellant Sprays
Principal author: C. Chen (UAH, USA)
Co-authors: S. Jeng (UC, USA)

2.5 Spray Combustion in Storable-Propellant Combustion Chambers
Principal author: D. Preclik (DASA, Germany)
Co-authors: D. Wennerberg (Battelle, Germany), D. Estublier (ESTEC, Netherlands)

2.6 Experimental Investigation of Liquid-Propellant Spray Combustion
Principal author: O. Haidn (DLR, Germany)
Co-authors: N. Yatsuyanagi (NAL, Japan), E. Gokalp (CNRS, France)

2.7 Propellant Ignition and Flame Propagation
Principal author: E. Hurlbert (NASA JSFC, USA)
Co-authors: R. Moreland (NASA JSFC, USA), L. Liou (LRC, USA)

Section 3. Thrust Chamber Performance and Heat Transfer

3.1 Assessment of Thrust Chamber Performance
Principal author: D. Coats (SEA, USA)
Co-authors: N. Girard (CNES), K. Denisov (NIIHIMMASH, RUSSIA)

3.2 Numerical Analysis of Combustor and Nozzle Flows
Principal author: C. Merkle (Penn State, USA)
Co-authors: P. Liang (Rocketdyne, USA)

3.3. Thrust Chamber Cooling and Heat Transfer
Principal author: M. Popp (DASA, Germany)
Co-authors: D. Quentmeyer (NASA LeRC, USA), S. Fisher (Rocketdyne, USA)

Section 4. Experimental Diagnostics and Testing

4.1 Technology Test Bed for Engine Development
Principal author: H. McConnaughey (NASA MSFC, USA)
Co-authors: Russians, Europeans, Chinese

4.2 Planar Laser Diagnostics of Liquid Propellant Jets in Dense Spray Regions
Principal author: D. Stepowski (CORIA, France)
Co-authors: R. Santoro (Penn State, USA)

4.3 Laser Diagnostics for Cryogenic Propellant Combustion Studies
Principal author: M. Oschwald (DLR, Germany)
Co-authors: S. Candel (ECP-EM2C, France), M. Péalat (ONERA, France)
L. Vingert (ONERAR, France)

4.4 Data Analysis and Scaling Techniques for Combustion Devices Testing
Principal authors: C. Dexter (NASA MSFC, USA), J. Hulka (Aerojet, USA)
Co-authors: K. Denisov (NIIHIMMASH, RUSSIA), J. Hutt (NASA MSFC, USA)

Section 5. Design and Development

5.1 Thermodynamic Power Cycles of Liquid Rocket Engines
Principal author: C. Erickson (Rocketdyne, USA)
Co-author: B. Zhang (China), P.Pempie (CNES, France)

5.2 Combustion Devices Design and Optimization
Principal author: D. Herbeaux (SEP, France)
Co-authors: D. Sion (SEP, France), DASA (Germany)

5.3 Advanced Nozzle Technology for Cryogenic Engines
Principal author: P.Vuillermoz (CNES, France)
Co-authors: H. Grosdemange (SEP, France), Weiland (DASA, Germany),
M. Bigert (Volvo, Sweden), B. Aupoix (ONERA/CERT, France)

5.4 Current Status of Tripropellant Combustion Technology
Principal author: L. Tanner (Pratt & Whitney, USA),
Co-authors: F. Chelkis (NPO Energomash, Russia)

5.5. Oxidizer-Rich Preburner Technology for Full Flow Cycle Applications
Principal authors: R. Jensen (Rocketdyne, USA)

TABLE 1.1A BI-PROPELLANT LIQUID ROCKET ENGINES (USA)

VEHICLE	ENGINE	PROPELLANT OXID	PROPELLANT FUEL	THRUS (LBF)	PC (PSIA)	ENG CYC	STG	THRUST (NEWTON)	PC (kPa)	COUNTRY	REF SOURCE	REF PAGE
TITAN III	LR87AJ-11	N2O4	A-50	260000	809	GG	1	1156532	5578	USA	3	28
TITAN III	LR91AJ11	N2O4	A-50	100850	827	GG	2	448601	5702	USA	3	14
APOLLO	SPS	N2O4	A-50	20500	97	P	4	91188	669	USA	3	70
DELTA	AJ10-118	N2O4	A-50	9763	125	P	2	43428	862	USA	3	66
TRANSTAG	ITIP	N2O4	A-50	8000	105	P	3	35586	724	USA	3	54
SHUTTLE	OMS	N2O4	A-50	6000	125	P	3	26689	862	USA	3	50
APOLLO	LEM-D	N2O4	A-50	9850	104	P	5	43815	717	USA	2	5
APOLLO	LEM-A	N2O4	A-50	3500	120	P	6	15569	827	USA	2	5
APOLLO	RCS	N2O4	MMH	93		P		414	0	USA	1	70
GEMINI	OAMS	N2O4	MMH	100		P		445	0	USA	1	70
GEMINI	OAMS	N2O4	MMH	85		P		378	0	USA	1	70
GEMINI	OAMS	N2O4	MMH	25		P		111	0	USA	1	70
AGENA	LR81BA11	RFNA	UDM	15800	506	GG	2	70282	3489	USA	2	3
AGENA	RCS	RFNA	UDM	200		P		890	0	USA	1	70
AGENA	RCS	RFNA	UDM	16		P		71	0	USA	1	70
SATURN 1C	F-1	LO2	RP-1	1522000	1128	GG	1	6770160	7777	USA	2	3
SATURN 1B	H-1	LO2	RP-1	204300	705	GG	1	908767	4861	USA	2	3
TITAN I	LR87AJ-3	LO2	RP-1	180000	637	GG	1	800676	4392	USA	2	3
THOR	MB-3	LO2	RP-1	170000	588	GG	1	756194	4054	USA	2	3
ATLAS	MA-5	LO2	RP-1	165000	577	GG	1	733953	3978	USA	2	3
TITAN I	LR91AJ-3	LO2	RP-1	80000	682	GG	2	355856	4702	USA	2	3
ATLAS	MA-5(SUS	LO2	RP-1	57000	706	GG	2	253547	4868	USA	2	3
SHUTTLE	SSME	LO2	H2	509000	3250	SC	1	2264134	*****	USA	2	5
SATURN IV	J2	LO2	H2	230000	780	GG	2	1023086	5378	USA	2	5
CENTAUR	RL-10	LO2	H2	15000	400	EXP	3	66723	2758	USA	2	5

TABLE 1.1B BI-PROPELLANT LIQUID ROCKET ENGINES (FOREIGN)

VEHICLE	ENGINE	PROPELLANT OXID	PROPELLANT FUEL	THRUS (LBF)	PC (PSIA)	ENG CYC	STG	THRUST (NEWTON)	PC (kPa)	COUNTRY	REF SOURCE	PAGE
ARIANE 5	L7	N2O4	MMH	6140	218	P	2	27312	1503	EUROPE	4	37
PSLV		N2O4	MMH	1700	123	P	4	7562	848	INDIA	4	57
TSYKLON	RD-712	N2O4	UDMH	667000			1	2966949	0	FSU	4	151
PROTON	RD-253	N2O4	UDMH	368000	2130	SC	1	1636938	14686	FSU	4	137
GSLV	VIKAS	N2O4	UDMH	165000	763	GG	0	733953	5261	INDIA	4	58
LM-2	YF-20	N2O4	UDMH	165000		GG	1	733953	0	CHINA	4	14
LM-2	YF-22	N2O4	UDMH	165000		GG	2	733953	0	CHINA	4	15
PROTON		N2O4	UDMH	135000		SC	2/3	600507	0	FSU	4	137
TSYKLON	RD-219	N2O4	UDMH	223000	1066	GG?	2	991949	7350	FSU	4	151
KOSMOS	RD-216	N2O4	UDMH	194000	1066	GG	1	862951	7350	FSU	4	125
LM-1	YF-2A	N2O4	UDMH	62000		GG	1	275788	0	CHINA	4	14
LM-1	YF-3	N2O4	UDMH	33050		GG	2	147013	0	CHINA	4	15
TSYKLON		N2O4	UDMH	17500		GG?	3	77844	0	FSU	4	151
ARIANE 4	VIKING	N2O4	UH25	170000	848	GG	1	756194	5847	EUROPE	4	36
ENERGIA	RD-170	LO2	RP-1	1777000	3556	SC	1	7904451	24518	FSU	4	110
VOSTOK	RD-107	LO2	RP-1	225000	848	GG	1	1000845	5847	FSU	4	163
VOSTOK	RD-108	LO2	RP-1	211000	740	GG	1.5	938570	5102	FSU	4	163
ZENIT	RD-120	LO2	RP-1	187000	2364	SC	2	831813	16299	FSU	4	174
SOYUZ	RD-461	LO2	RP-1	67000		GG	2	298029	0	FSU	4	163
ZENIT		LO2	RP-1	19400	1124	SC	3	86295	7750	FSU	4	174
MOLNIYA		LO2	RP-1	15000		GG	3	66723	0	FSU	4	163
VOSTOK	RD-448	LO2	RP-1	12000		GG	2	53378	0	FSU	4	163
ZEN/vernier	RD-8	LO2	RP-1	18000	1117		RCS	80068	7701	FSU	4	174
N-2	MB-3	LO2	RJ-1	170000	569	GG	1	756194	3923	JAPAN	4	77
H-2	LE-7	LO2	H2	243000	2090	SC	1	1080913	14410	JAPAN	4	78
ARIANE 5	VULCAIN	LO2	H2	225000	1450	GG	1	1000845	9997	EUROPE	4	37
ENERGIA	RD-120	LO2	H2	441000	3000	SC	2	1961656	20684	FSU	4	110
H-2	LE-5A	LO2	H2	27300	555	EXP	2	121436	3827	JAPAN	4	77
H-1	LE-5	LO2	H2	23150	526	GG	2	102976	3627	JAPAN	4	77
LM-3	YF-75	LO2	H2	17600		GG	3	78288	0	CHINA	4	16
ARIANE 4	HM7B	LO2	H2	14100	508	GG	3	62720	3503	EUROPE	4	36
LM-3	YF-73	LO2	H2	2475		GG	3	11009	0	CHINA	4	16