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LIQUID ROCKET TECHNOLOGY FOR THE CHEMICAL ENGINEER

by Richard J. Priem
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TECHNICAL PAPER proposed for presentation at
62nd Annual Meeting of the American
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABSTRACT

Recent rocket technological developments of interest to the chemical engineer are presented. The path of liquid propellants through a propulsion system is followed to illustrate the problems encountered in developing a lightweight and reliable system. As the problems are defined, the technical advancements made to overcome them are presented. Improved techniques for insulating tanks and providing for flexible lines are given. Data are presented to show how various parameters influence propellant sloshing and consumption of pressurizing gas when used with cryogenic propellants. Recent advances are presented for using combustion products to pressurize tanks. New methods for pumping cryogenic propellants that are near their boiling point are discussed. Methods for predicting the very high heat transfer rates encountered with accelerating gas streams in rocket engine combustors are presented. A new injector concept is discussed to illustrate the technology developed for obtaining

TM X-52695

high performance in a combustor with high volume heat release rates. Computer programs for calculating equilibrium combustion products and combustion rates for various propellants and combustor operating conditions are discussed.

INTRODUCTION

The development of liquid rockets for space travel has been accompanied by numerous research and technology programs. The objective of these programs has been to provide new ideas, information, and data to aid the rocket development engineer in obtaining a reliable, efficient, and light-weight system. While these studies have been directed toward helping the liquid rocket engineer, the information and data can also be used for other applications. This paper is a review of some of the results from liquid rocket research programs that could be of interest to a chemical engineer.

Because it is impossible to review all the programs and material that would be of interest to a chemical engineer, a format has been developed for selection of the material to be covered. The approach was based on following the liquid propellant throughout its lifetime in a rocket vehicle, and describing various problems and methods by which the problems were overcome in developing liquid rockets.

A schematic of a liquid rocket vehicle is shown in figure 1 which illustrates how the propellants flow through a vehicle and the technology areas that will be covered herein. The four main components of a vehicle are: tanks, lines and valves, turbopump, and combustion chamber. In the area of the tanks there are three sub-areas of interest as follows: insulation for cryogenic tanks, motion of the propellants in the tank, and pressurization of the tank to expel the propellants. Likewise the combustion chamber can be subdivided into technologies concerned with the injector which introduces the propellants into the combustor so that high combustion efficiency is obtained, heat transfer between the very hot combustion gases and the chamber walls, and heterogeneous combustion.

By selecting the format discussed above it is hoped that the various areas in which useful information exists will become apparent, and that the solutions developed for liquid rockets can be of value in other applications. As the problems in each area are described, the most recent and pertinent results will be reviewed to illustrate the available technology. The reader will be able to find additional information in the material referenced herein.

PROPELLANT TANKS

The largest single components in liquid rocket propulsion systems are the tanks which contain the propellants, i.e., the fuels and the oxidizers. Nearly 90 percent of the volume of a space vehicle is required for propellant storage. In addition to performing the obvious function of confining the propellants, rocket vehicle tanks and related accessories must fulfill a variety of heretofore unusual requirements. Chief among these are (1) the prevention of propellant boiloff by using new materials of insulation, (2) the provision of stability against fluid sloshing, and (3) a means of driving the fluid from the tank by pressurizing it.

Cryogenic Tank Insulation

Many of the rocket propellants are cryogenic liquids (liquified gases); therefore, the tanks must be provided with thermal insulation to minimize boiling of the liquid. Advances in the effectiveness of thermal insulations can be directly attributed to concentrated development efforts in support of liquid rockets. The starting point was the double-wall, vacuum insulated glass Dewar flask. The Dewar flask was widely used to store cryogenic fluids until the demand for such fluids, particularly liquid oxygen and liquid hydrogen, became one of large scale.

The thermal insulations available today fall into two major categories: insulations which are gas filled, and insulations in which the gas has been evacuated to achieve a required low pressure. Figure 2 shows the effectiveness of different classes of insulating materials (foam, powder, fiber, and multi-layer insulations) which span a range of more than 3 orders of magnitude of thermal conductivity. Multilayer insulation has the most impressive thermal conductivity and will be discussed further herein, although the other insulating materials have other desirable features. An excellent survey of insulating materials covering all types of insulations can be found in reference 1.

The insulated structure concept seen in figure 3 is composed of a superalloy heat shield for temperatures up to 1800° F, fibrous high temperature insulation, a primary structure, cryogenic insulation, and a fuel tank structure. The temperature of the primary structure is partly dependent on the thickness of the cryogenic and fibrous insulations. This type of construction requires essentially three leaktight shells: (1) the internal hydrogen tank, (2) the primary structure which precludes liquefaction of air that enters the cryogenic insulation area and (3) the heat shield which prevents trapping and freezing of moisture within the fibrous insulation area.

Compressive loads, either those caused by atmospheric pressure, when a flexible outer skin is used, or those developed during application of the insulation will reduce the overall insulating effectiveness. Even if the compression by the weight of the upper layers on the lower layers is disregarded, external forces (e.g., tension applied during wrapping of a multilayer insulation around a cylindrical object, thermal expansion or contraction of the insulation components with respect to the object, and localized loads in the vicinity of the object's supports) can compress the insulation. These compressive loads may be in the range from 0.01 to 1 psi.

Figure 4 shows the effects of compression on the heat flux through 16 different multilayer insulations. When compression up to 2 psi is applied, the heat flux for the majority of the insulations is about 200 times greater than at the no-load condition. Figure 4 shows that the heat flux is proportional to the compressive load to the 0.5 to 0.67 power.

The performance of any given insulation in an actual installation will be greatly affected by the following variables:

(1) Applied compressive load.

(2) Lines and supports connected to the tank.

(3) Number of shields used in the sample.

(4) Kind of gas filling the insulation and its pressure.

(5) Size and number of perforations in the insulation to permit out-gassing.

(6) Temperatures of the warm and cold boundaries.

Data and equations to evaluate these variables can be found in reference 1 or the references found therein.

Fluid Motion

Most of the initial mass of launch vehicles is comprised of the liquid propellants. The dynamic forces in flight resulting from the motion, or slosh, of these large liquid masses can be very substantial, even beyond the capabilities of the vehicle attitude control system to counteract them, or the structure to resist them. To prevent interaction between the various parts of the vehicle, it is desirable to have the natural frequencies of the various components widely separated. Table I gives data for several representative vehicles, whose sizes are representative of the relative sizes of the tanks involved, and the various natural frequencies.

TABLE I. - Characteristics of Some Representative Launch Vehicles

Vehicle	Length, m	Diameter, m	Control frequency, Hz	Fundamental slosh frequency at liftoff, Hz	Fundamental bending frequency at liftoff, Hz
Redstone	21	1.78	0.5	0.8	10-12
Redstone-Mercury	25	1.78	.5	.8	10
Jupiter	20	2.65	.4	.6	9
Juno II	25	2.65	.4	.6	8
Saturn I	60	6.5	.3	.45	2
Saturn V	130	10	.16	0.3-0.4	1

From Table I, one can see that the various frequencies are indeed not always widely separated. (Consideration of harmonic frequencies would show additional frequencies very close to each other.)

If the dominant frequency of the propellant motion, called propellant slosh frequency, is close to any of the control system frequencies, an instability in the flight characteristics can result. When the slosh frequencies are close to the elastic body bending frequencies, a large amplitude dynamic response problem may arise. In any case, the governing design problem is that of stability and control, so that one must also consider the location and configuration of the tanks.

Numerous theoretical and experimental studies have been conducted to determine the fluid motion in tanks with various shapes, damping devices, and fluids. Excellent reviews of the dynamics of liquids in tanks are given in references 2 and 3. The theories predict the frequencies for various modes of oscillation and tank parameters.

The theoretical and experimental results show that numerous frequencies exist for natural oscillations in a tank. Therefore, it is very difficult to design tanks that do not have frequencies close to elastic body frequencies or control system frequencies. To overcome this difficulty of matching frequencies it has been found necessary to use baffles in the tank to increase the damping of the oscillations in an attempt to eliminate the coupling between various system components. Baffles, however, also change the natural frequencies of the tank, as shown in figure 5. Maximum frequency is obtained with the baffle at the free surface (baffle depth = 0). As the baffle is lowered, the frequency decreases to a minimum value. As the baffle is further lowered, the frequency increases to the free surface or unbaffled frequency.

An example of the level of damping produced by a simple baffle is shown in figure 6. This figure also indicates the extent of agreement between theoretical and experimental results. Theoretical studies indicate maximum damping is obtained with a baffle at the liquid surface, while the experimental results give maximum damping with a baffle slightly below the liquid surface. With baffles well submerged in the liquid, the agreement between theory and experiment is excellent.

Tank Pressurization

Discharge of propellants from the tanks of a liquid rocket is accomplished by providing a positive tank pressure. With a minimum weight system, the pressurizing gas is usually at a temperature considerably higher than that of the cryogen. The resulting energy exchanges involve complex interactions which are influenced by many physical parameters of the system. Such interactions result in losses in the overall effectiveness of the pressurizing system, and thus directly influence the total pressurizing mass requirements.

The magnitude of these losses, when liquid hydrogen is pressurized with hot hydrogen, is given in reference 4. A typical curve showing how tank size and expulsion time influence the pressurant mass requirements to expel liquid hydrogen with gaseous hydrogen at 500° R is shown in figure 7. The actual mass requirements are two to three times greater than the quantity that would be required if the pressurant gas in the ullage had remained at a constant high temperature. For smaller tanks with increased surface-to-volume ratios and longer expulsion times, an even greater increase in the relative amount of pressurant gas is required. Measurements of tank wall, propellant, and ullage gas temperatures to determine where the energy is lost gave the results shown in figure 8. The energy balance shows that 20 percent of the energy added by the pressurizing gas was absorbed in the liquid and between 40 to 50 percent was absorbed by the light weight (thin) walls of the tank.

The effects of liquid propellant (hydrogen) sloshing and baffles were also investigated in reference 4 and the results are shown in figure 9. As the tank oscillation amplitude is increased the fluid motion changes from a planar wave to a splashing wave. The splashing wave, obtained in the 13-foot diameter tank at a tank oscillation amplitude of ±1.10 in. at the first mode frequency, increased the pressurant requirement approximately 30 percent from that obtained with no oscillation present. When antislosh baffles were added, the pressurant requirements increased 100 percent over that obtained with no liquid sloshing. Although antislosh baffles decrease the maximum wave height, they also cause more liquid splashing which cools the gas in the ullage. Other parameters that were investigated in reference 4 were pressurant gas temperature, pressurant injection techniques, pressure level, and tank wall thickness. All parameters were found to have an influence on the pressurant requirements to expel cryogenic propellants from a tank.

For liquid propellants (fuel and oxidizer) of the type which ignite by spontaneous reaction upon contact, another pressurization scheme has been considered. This consists of introducing a limited amount of one propellant into the bulk of the second propellant so that the resultant reaction and heat release will create hot gases to pressurize the second propellant tank. A recent study (reported in reference 5) investigated the feasibility, limitations, and operating characteristics of a propellant tank pressurization system in which fluorine is injected into a liquid hydrogen tank to generate pressurizing gas by vaporizing hydrogen from the heat produced in the high temperature reaction. A summary of the experimental results of the study is shown in figure 10. Pressurization efficiency (compared to analytical model results) is shown for various injectors and ullages. Three injection schemes were used: (1) simple injection into the ullage volume (US), (2) submerged injection with an aspirator (SA), and (3) simple submerged injection (SS). Reliable ignition and re-ignition was demonstrated with all injections. With submerged injection, freezing, and detonations were an occasional problem. Injection into the ullage volume demonstrated the most efficient and rapid tank pressurization without detonations.

PROPELLANT LINES AND VALVES

As the propellant leaves the tank it goes into lines (ducts) which are used to transfer the fluid to the engine or its pump. The lines must meet various requirements depending on the specific application. Generally, they must be flexible to allow for the motion between various components, and they must be light in weight. With cryogenic propellants the lines must be insulated and yet retain sufficient flexibility to allow for thermal expansion as well as relative motion between parts. The large demand for insulated flexible lines for rockets using liquid oxygen (LOX) or liquid hydrogen has resulted in the development of many new companies that produce them. Currently it is possible to order flexible insulated lines from catalogs to meet almost any demand. These lines vary from non-metallic laminated tubes to vacuum jacketed bellows. It would be impossible to review the various developments in this area; therefore, the reader is referred to the catalogs of various manufacturers to determine the latest advances in this area.

Flexibility in piping arrangements and easy installation frequently require joints that can be readily connected and disconnected. Figure 11 is a schematic illustration of the construction of a bayonet type joint for a vacuum-jacketed line. The vacuum insulation overlaps on both the male and female halves of the joint, and the radial clearance is kept small to allow a quiescent gas film to develop. The gas film blocks the liquid and keeps it from flowing out to the warm region of the seal, where it would boil and cause a high heat leak. Another significant design feature is that the parts are made long and thin to minimize the heat-conduction path from the warm region to the cold region. The bayonet concept can be used to develop complete piping systems. Piping components are now manufactured in sizes from a fraction of an inch to over 1-foot in diameter.

To provide a seal for the propellant lines, gaskets are frequently used as shown in figure 11. Adequate information is available for gaskets used at conventional temperatures (see reference 6 for an excellent bibliography). Additional technology was developed, however, for gasket materials for use in contact with LOX or liquid fluorine. In addition to being able to withstand cryogenic temperatures the gasket material had to be inert to LOX or fluorine. This requirement limited the choice of polymeric materials to fluorocarbon polymers. The fluorocarbon polymers presented a special problem, however, because all known fluorocarbons creep (deform) excessively under load, even at the temperature of LOX (ref. 7).

Two types of gasket structures have been investigated to determine if a suitable laminate of glass fabric and fluorocarbon polymer could be used to improve the low-temperature performance characteristics of gaskets (see ref. 8 for details). In one type, alternate layers of glass fabric and sheets of tetrafluoroethylene polymer were laid together, with the fabrics arranged so that the yarns of the fabric in each layer were at an angle to those in the nearest layers. Since the glass fibers in these laminates were not completely surrounded by the polymer, the edges of sections cut from the laminate were not sealed and liquid could soak into the gasket material. To prevent this, gaskets cut from the laminates were either encapsulated in fluorinated ethylene-propylene polymer, or the cut surfaces were coated with this polymer.

These gaskets were evaluated using a conventional test machine recording the deflection vs. load for several load, unload cycles. The area under the load deflection curves were calculated and taken as measures of the energy absorbed by the gasket when compressed. Materials showing the highest energy-absorption values and minimum fall-off in energy absorption with cycling were considered the best materials. Figure 12 shows the energy-absorption data for three gasket materials. The materials evaluated were a good-quality glass reinforced laminate, a glass-reinforced laminate of unacceptable quality, and a gasket made of styrene-butadiene rubber reinforced with asbestos fiber and saturated with a liquid fluoroethylene polymer. The optimum glass laminate was a much superior gasket than the asbestos filled rubber that was commonly used for LOX service prior to the development of better materials.

Fluorocarbons have been used for gaskets with fluorine when the gasket is not in direct contact with liquid fluorine. When the gasket is in direct contact with flowing fluorine, metallic confined ring gaskets (O-ring, U-ring, Crunch gasket, etc.) must be used. An excellent review of techniques for handling and using fluorine and fluorine-oxygen mixtures can be found in reference 9.

Valves for liquid rocket systems must meet very stringent requirements regarding reliability and leakage. A valve that leaks can produce severe problems at the launch site because many of the propellants are toxic and all of the fuels represent a potential fire or explosion hazard if they are allowed to leak out of the tanks. For deep space missions where the tanks must be sealed for long periods (1200 days for a Jupiter orbitor mission) the valve requirements are very stringent. Several studies have been made to determine what design and fabrication features produce low leakage valves. An excellent summary of valve and seat design data for aerospace valves is given in reference 10 where information is presented to calculate leakage flow rates in various regimes of valve flow, from nozzle flow down to molecular flow. Influence of valve seat surface irregularities such as surface finish inadequacies, scratches, and surface deformation on leakage rate have been thoroughly examined for various valve designs and are also presented.

An example of a conical valve design that incorporates many desirable features of a low leakage valve for fluorine is shown in figure 13. With this valve, the valve plug has a conical shape and the seat is spherical. Any desired mechanical advantage can be obtained for plugstem force by selecting various cone angles from 0^{o} to 90^{o}. Since the cone length of the plug and radius of the spherical seat are the same, the sealing force is applied normal to the valve seat. This eliminates the sliding contact found in most valve plug-seat combinations. In general, a minimum lip width is desirable to avoid stress concentration on sharp corners, with resultant strain-hardening of the material (inset B, figure 13). If thicker cross-sections are used, the cone lip should be ground and lapped to provide a spherical surface to mate with the spherical valve seat (inset A, figure 13).

TURBOPUMPS

After the propellant flows from the tanks and into the lines, it goes to the turbopump. The purpose of the turbopump is to increase the pressure so that the propellant will flow from a low-pressure tank to a high-pressure combustion chamber. To prevent propellant boiling in the pump, the tank pressure must be greater than the vapor pressure of the propellant. Yet, since the largest structural weight of a liquid rocket vehicle is in propellant tanks, it is necessary to minimize tank pressures to minimize structural weight. Tank pressures are, therefore, only a few pounds per square inch greater than the vapor pressure for cryogenic propellants. Under these conditions the propellant is in a near-boiling state.

With a near-boiling liquid in the tanks any static pressure drop experienced by the liquid as it flows from the tank to the pump and into the pump passages causes a portion of the liquid to flash to vapor (boil) within the pump passages. This vapor formation, called cavitation, can cause a loss in pump performance. To illustrate the effects of cavitation on flow conditions about pump blades, blade pressure distributions with and without cavitation are compared in figure 14. The horizontal dashed line represents the vapor pressure of the incoming fluid. If the inlet pressure is sufficiently high, the blade surface pressures are everywhere greater than the liquid vapor pressure. However, if inlet pressure is reduced sufficiently, local pressures on the suction surface of the blade are reduced to the fluid vapor pressure, and cavitation occurs. Further reduction in inlet pressure leads to rapid deterioration of pump performance because of extensive cavitation and accompanying flow separation from the suction surface of the blade.

The problem of pumping near-boiling fluids has been overcome by a prepumping stage, which operates satisfactorily even under conditions of extensive cavitation (references 11 and 12). Such a prepumping stage is called an inducer. A typical inducer is presented in figure 15. The three exceptionally long blades are mounted on the hub in a manner approximating a helix. At low inlet pressures, cavitation and flow separation occur on the forward portion of the blade; but, because of the length and hydrodynamic design of the blades, the separated flow re-attaches to the blade surface behind the cavity to reestablish the desired force. Although the pressure forces on the forward portion of the blade have decreased, the forces exerted by the rear portion of the blade have increased. This compensating effect results in extended high performance despite extensive cavitation.

Another and perhaps more familiar aspect of cavitation is the structural failure that it can cause. An example of pump blade cavitation damage is given in figure 16. The cavitation damage occurs on a surface where adjacent vapor cavities rapidly collapse. Although various studies have shown that pure vaporous cavities can collapse, the basic mechanics of damage by imploding vapor bubbles is not clearly established. Several programs have been conducted however to determine materials that resist cavitation damage, i.e., those reported in references 13 and 14. High values of yield strength, ultimate tensile strength, hardness, and ductility are beneficial in providing resistance to cavitation damage.

COMBUSTION CHAMBER

The last place of residence of the propellants before they leave the liquid rocket vehicle is the combustion chamber of the engine. Besides providing the appropriate conditions for the propellants to react, the combustion chamber must provide other features to obtain a reliable lightweight vehicle. Because the combustion gases are extremely hot and reactive, some protection must be provided to control the heat transfer to the walls of the combustion chamber. This protection can be in the form of cooling the walls with one of the propellants (called regenerative cooling), or coating the walls with high temperature materials that reduce the heat load and provide a surface that will withstand the gas environment. The combustion chamber also includes the device through which the propellants are introduced into the chamber. This device is called the injector. The injector plays an important role in determining the rate and efficiency of the combustion process. The discussion on combustion chambers will, therefore, be divided into the topics of heat transfer, injectors, and combustion.

Heat Transfer

A simple method of predicting the heat transfer to a combustion chamber is to treat the chamber as if it were a variable diameter pipe. Correlations for turbulent heat transfer in pipes are well known and involve a relation between Nusselt and Reynolds numbers. The heat flux values that are predicted from this type of calculation technique, however, exceed the heat flux actually experienced at the nozzle throat by a factor of 2. Numerous studies have been made, therefore, to delineate this discrepancy in predicted heat transfer rates (references 15, 17, and 18). Measurements of the boundary layer in a nozzle have shown that with accelerating flow the velocity profile is flatter than predicted in the portion away from the wall and steeper in the portion of the boundary layer near the wall. With the steep velocity profile near the wall, it would be expected that the heat transfer with accelerating flow would be even higher than predicted, rather than lower.

Analyzing the turbulent boundary layer with an integral equation approach (reference 18) shows that there is an obvious similitude between the energy and momentum equations when no acceleration is present. Applying the integral equations approach and including terms involving gas acceleration has decreased the predicted heat transfer rates in the nozzle throat area and obtained reasonable agreement with measured results.

Injectors

The role of the liquid propellant injector is to control the flow of propellants into the combustion chamber so that rapid and controlled combustion is accomplished. Because the combustion occurs at high pressures and in a very small volume, the flow rate for a unit injector face area is very high. This has required developing novel injection techniques to obtain the required high performance. Most of the technology developed for injection techniques has been obtained in the form of understanding how different design variables influence combustion rate and performance. A review of this technology is given in reference 19.

Recent injector development programs (ref. 20 and 21) illustrate some of the new design features incorporated in liquid rocket injectors. The injector concept employs a large number of thin metal platelets, with integral propellant flow passages, stacked and bonded together. The platelets have patterned flow channels on the surface to form the flow passages through the stack and provide the means for bringing the propellants from their manifolds to the injector face. The injector face itself resembles a porous surface, containing many hundreds of orifices per square inch of face area. The orifices and flow passages have hydraulic diameters in the range of a few thousandths of an inch, so that laminar flow is maintained. The small streams and uniformity of pattern provide efficient combustion and a high level of homogeneity in the combustion products within a short distance of the injector face. Face cooling problems are minimized since the injector face is effectively a transpiration cooled surface, with the entire propellant flow being used as coolant.

With laminar flow of propellants, a linear flow versus pressure drop relationship is obtained. Because of the laminar flow, the relative injector hardness (as measured by the ratio of injector pressure drop to chamber pressure) would remain constant over a wide throttling or thrust range. Thus, in theory, the concept provides good protection against flow oscillations induced by chamber pressure oscillations.

A problem with this injector concept (and many others) stems from interpropellant heat transfer. Due to the small size and large number of propellant feed passages, the injectors are excellent heat exchangers. Designing injectors to provide laminar flow characteristics for large variations in properties and changes of phase that are encountered with interpropellant heat transfer is very difficult.

Combustion

The simplest idealization of the combustion process in a liquid rocket engine is to regard it as a chemical reactor in which a fuel reacts with an oxidant. This reactor differs from those used in chemical-process industries in two important ways: (1) the process is adiabatic rather than isothermal, (2) thrust is wanted and not chemicals. Therefore, the reaction products are exhausted through a nozzle to convert the random thermal energy of the high temperature gas into directed kinetic energy, or thrust. The optimum behavior of a liquid rocket can be estimated by assuming that the fuel and oxidant mix perfectly and react instantaneously to give an equilibrium mixture of reaction products. With a hydrocarbon fuel and oxygen as the oxidizer, the products are not merely carbon dioxide and water. At the high combustion temperatures and pressures of a liquid rocket reactor a considerable amount of dissociation takes place.

To simulate an ideal liquid rocket thermodynamically, it is assumed that all of the fuel and oxidant react at a constant pressure and energy and then the hot combustion gases expand adiabatically through the nozzle. Thermodynamically the combustion process can then be illustrated by considering the combustion as it occurs on a kitchen range when a mixture of air and methane react to form a high temperature flame containing the reaction products (figure 17). The system of equations that describes this process contains (1) equations for conservation of mass for each of the four elements (carbon, hydrogen, oxygen, and nitrogen in this case), (2) equation for conservation of energy, (3) equation specifying the pressure at which the combustion takes place, and (4) sixteen equilibrium constant equations for a total of 22 simultaneous equations. These simultaneous equations are not linear equations and cannot be solved analytically, but require an iterative solution on a computer.

Any chemical system other than methane and air requires both different equations and a different number of equations to be solved. A new computer program would appear to be required for each chemical system encountered. General programs have been compiled, however, that can handle many elements and species. References 22 and 23 describe a computer program and data to calculate thermodynamic properties and equilibrium combustion products for 210 substances involving the first 18 elements of the periodic chart, that is, elements such as hydrogen, boron, carbon, nitrogen, up to and including aluminum, silicon, phosphor, sulfur, chlorine, and argon. The program and data are on a single reel of magnetic tape and the program automatically selects the data and computation procedure required for any combustion system. The program adapts to several computers and copies of the magnetic tape and instructions have been sent to more than 70 universities and industrial laboratories.

Expansion of programs like that described above have been made to calculate the thermodynamics of reaction products flowing through a nozzle. These programs include changing composition due to a change in the temperature and pressure with infinite reaction rates and also with finite rates. Five standardized computer programs have been developed under the authorization of the

Interagency Chemical Rocket Propulsion Group, (ICRPG). These five reference programs are:

- ODE One-Dimensional Equilibrium Nozzle Analysis Computer program, developed by NASA-Lewis Research Center
- ODK One-Dimensional Kinetic Nozzle Analysis Computer Program developed by Dynamic Science Corporation
- TBL Turbulent Boundary Layer Nozzle Analysis Computer Program, developed by Pratt and Whitney Aircraft Corporation
- TDE Two-Dimensional Equilibrium Nozzle Analysis Computer Program, developed by Pratt and Whitney Aircraft
- TDK Two-Dimensional Kinetic Nozzle Analysis Computer Program, developed by Dynamic Science.

Copies of the computer programs and documentation for use of the programs are available from the ICRPG Performance Standarization Working Group through Dynamic Science, Irvine, California.

In an actual liquid rocket combustor the propellants are introduced as liquids to form a heterogeneous mixture of liquid propellant and gaseous combustion products instead of the assumed perfect gas mixture for the idealized rocket discussed above. The physical processes of atomizing the liquid into drops, vaporizing the drops, and mixing the vapors of the fuel and oxidizer all require a finite distance and time. Therefore, numerous studies have been conducted to aid in determining the rates of the various physical processes involved in converting two liquid propellants to mixed gaseous proepllants. A review of the more recent work can be found in reference 19.

Because the chamber pressure of liquid rocket combustors is fairly high (between 100 and 1500 psi) and the temperature of the propellant vapors after they are mixed with combustion products is very high, the chemical reaction rate is very rapid, see reference 24. The slower step, therefore, in the complete combustion process is generally the vaporizing of the liquid propellant.

To determine the vaporization rate, or time, several computer programs have been developed to calculate drop histories (references 25 and 26). These programs use standard equations for heat, mass, and momentum transfer. A typical result of such a calculation is shown in figure 18. The drop heats rapidly to its wet bulb temperature, in this case 845° F. The heating period requires about 20 percent of the total distance required to vaporize the drop. Meanwhile the radius of the drop initially increases slightly by thermal expansion and then decreases rapidly as vaporization progresses. The velocity of the gas in the combustion chamber increases from zero at the injector end to 790 feet per second near the nozzle as drops vaporize and the vapors burn. As the gas accelerates the drops experience a drag force, causing them to accelerate until they approach the gas velocity. The difference between the velocities of the surrounding gas and the drop is important in determining drag and vaporization rates.

Calculations like those described above have been carried out for a variety of conditions to show that under many conditions the performance of a rocket combustor can be predicted by assuming that the vaporization process controls performance. When the chamber pressure is above the critical pressure of the propellant, the calculations described above show that the propellant does not reach a steady "wet bulb" temperature before reaching the critical temperature. Therefore, several studies have been conducted to determine combustion rates at pressures above the critical pressure of the propellant (references 27 and 28). These studies have provided techniques for predicting combustion rates in high pressure environments.

More recent computer programs have incorporated most of the steps that occur in the combustion process (reference 19). This includes atomization, vaporization, mixing and chemical reaction. However, these programs are specialized for specific types of injection processes and, therefore, are limited to these systems.

SUMMARY

Recent liquid rocket technological developments of interest to the chemical engineer are presented. These include the following:

recent advances in insulating propellant tanks; methods of predicting and controlling the motion of liquid propellants in a tank; advances in pressurizing propellant tanks including using combustion products as the pressurant; techniques for making lightweight flexible lines and seals that can be easily disconnected; the pumping of a fluid near its boiling point, and the concepts of an impeller to eliminate pump cavitation; methods for predicting the very high heat transfer rates encountered in rocket engine combustors with accelerating gas flow; a new injector concept to obtain high performance in a combustor with high volume heat release rates; computer programs for calculating equilibrium combustion conditions and combustion rates for various propellants and combustor operating conditions.

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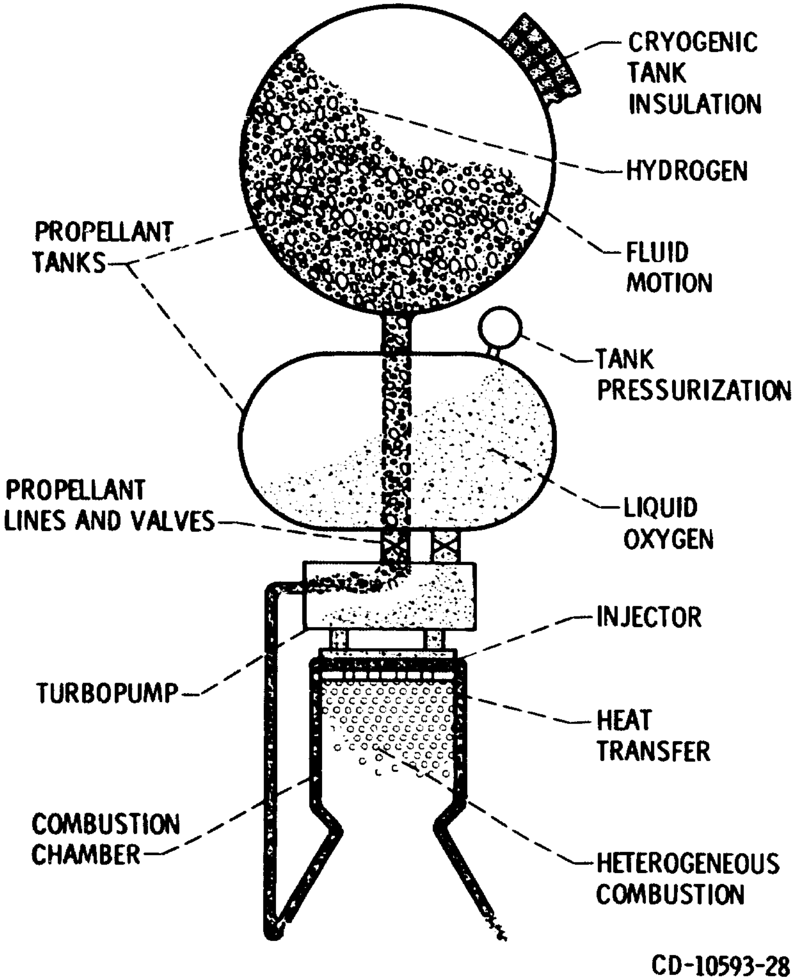

Figure 1. - Schematic of liquid rocket vehicle.

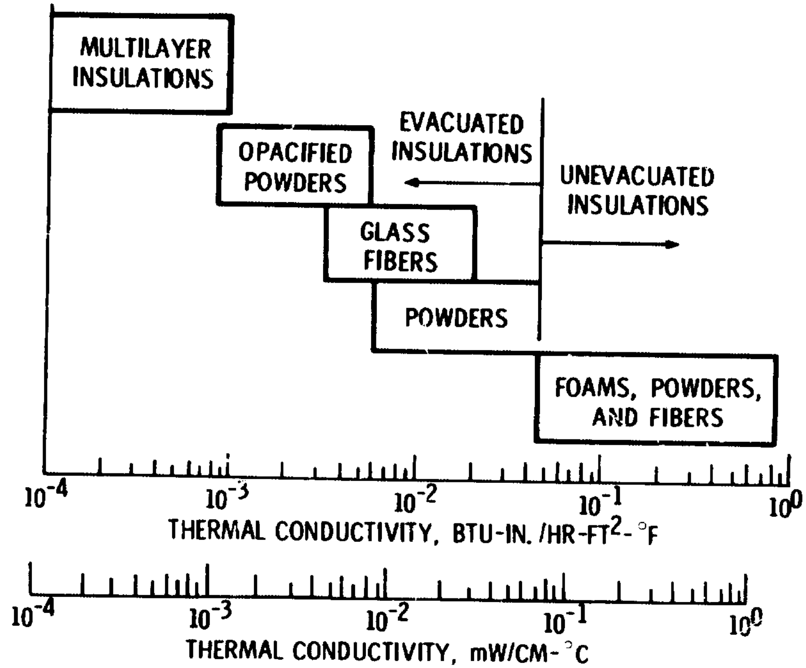

Figure 2. - Thermal conductivity of thermal insulation materials. (Temperature range, 60° to -320° F (238° to 77° K); density range, 2 to 5 lb/ft^3 (0.03 to 0.08 g/cm^3.))

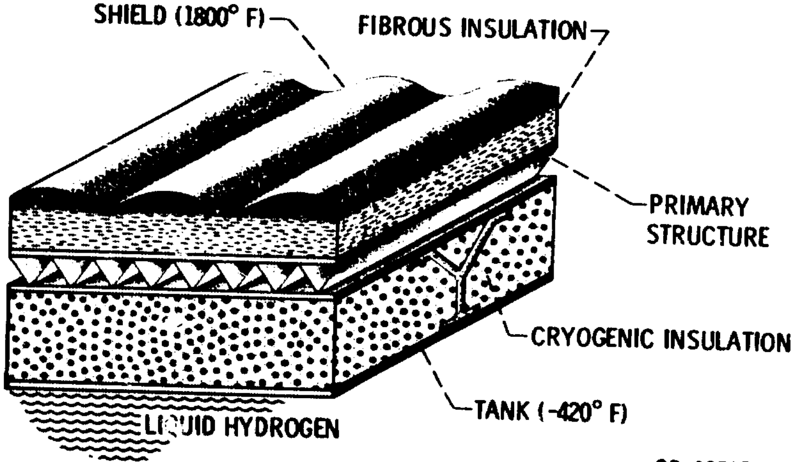

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Figure 3. - Cryogenic tank.

	Run number	Number of layers	Material
- - - - -	3037	10	1145 - H19 tempered aluminum
		11	Nylon netting
——— - -	3041	10	Aluminized (both sides) polyester
		22	Glass fabric
— — —	3042	10	Aluminized (both sides) polyester
		33	Silk netting
———	3043	10	Aluminized (both sides) polyester
		11	2 lb/ft^3 polyurethane foam
—— - -	3044	10	Aluminized (both sides) polyester
		11	Silk netting with 0.004-in. by 0.5-in. strips of glass mat
——— -	3046	10	Aluminized (both sides) polyester
		11	Silk netting with 0.008-in. by 0.25-in. strips of glass mat

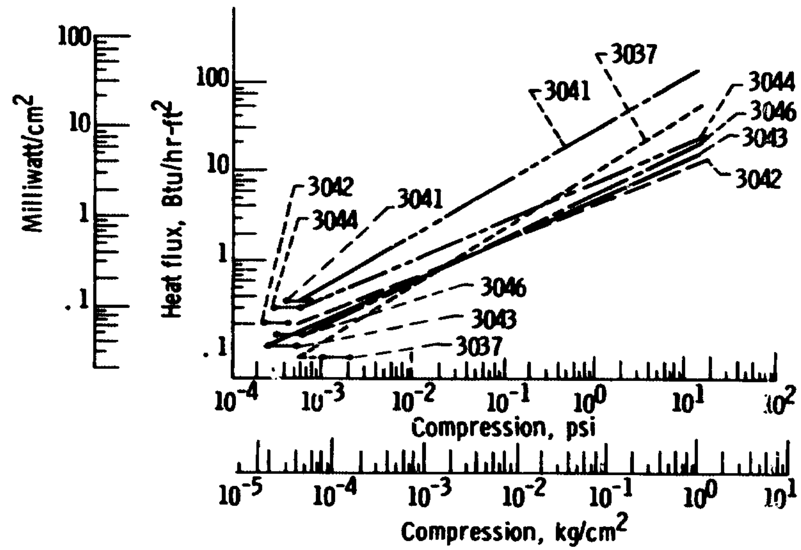


Figure 4. - Effect of external compression on the heat flux through multilayer insulations (ref. 1).

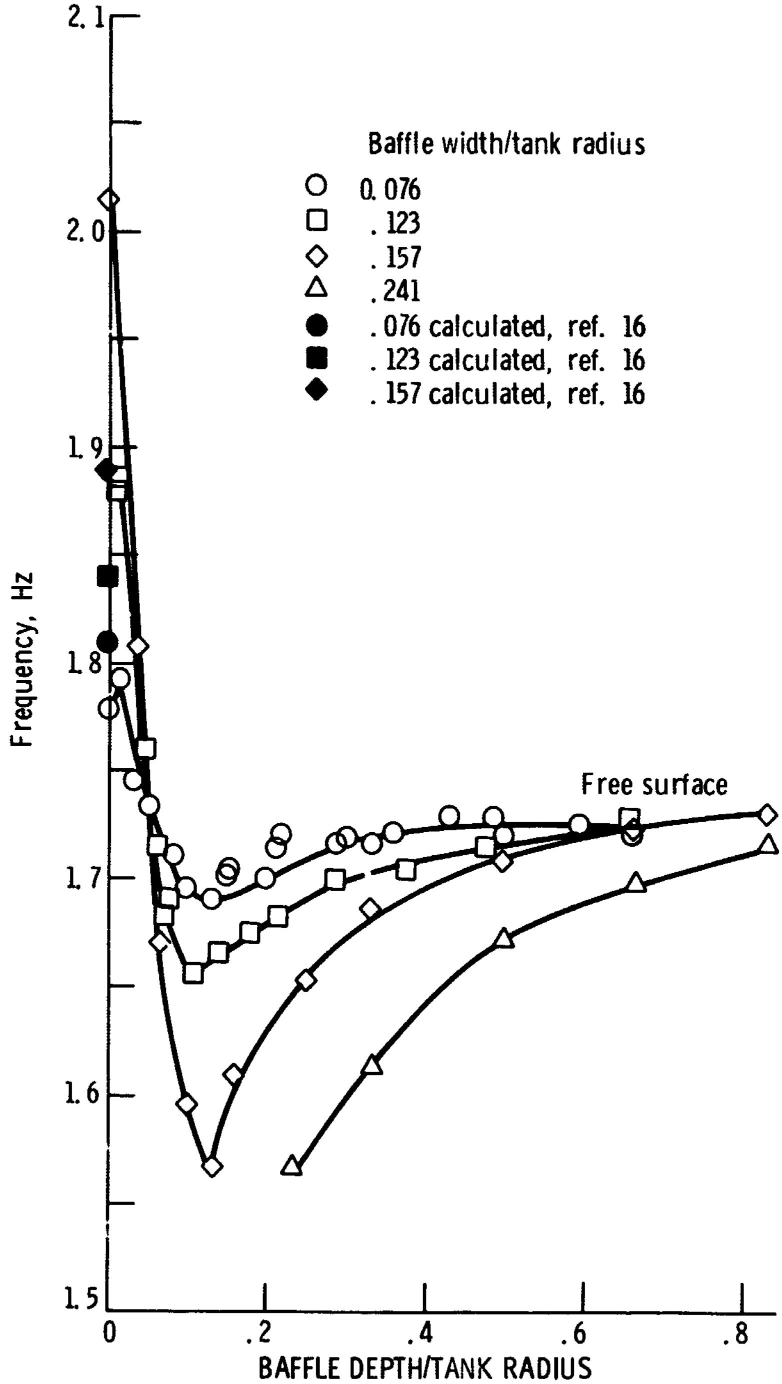


Figure 5. - Variation of fundamental slosh frequency with baffle location for single fixed-ring baffle in a 12-inch diameter cylindrical tank (ref. 2).

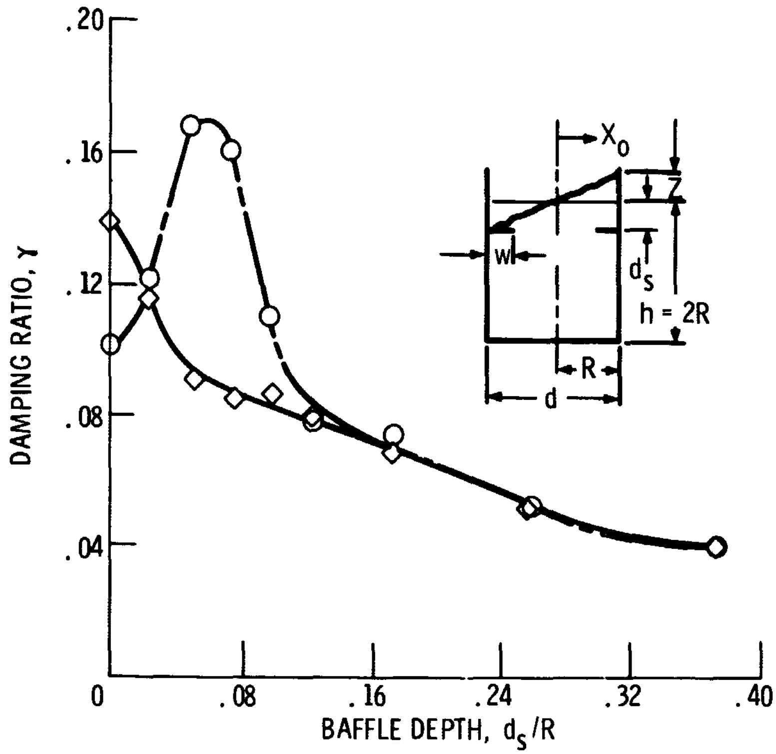


Figure 6. - Comparison of theory and experiment for damping provided by a flat solid-ring baffle as a function of baffle depth (ref. 3).

P_T = 50 PSIA, $T_{INLET} \approx 500°$ R, GH_2 PRESSURANT, HEMISPHERE INJECTOR

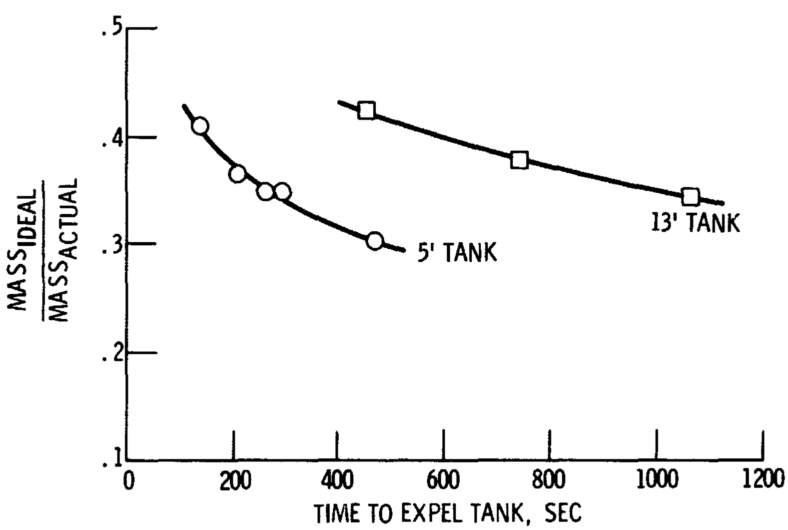


Figure 7. - Pressurant mass ratio as f (expulsion time) for two tank sizes (ref. 4).

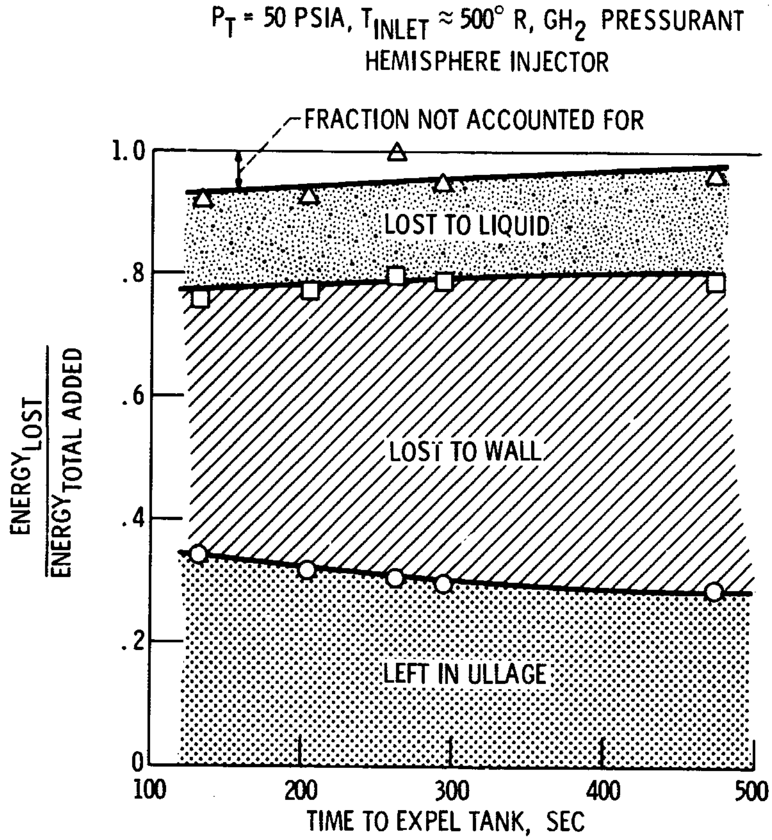


Figure 8. - Energy distribution in 5-foot tank at the end of expulsion (ref. 4.)

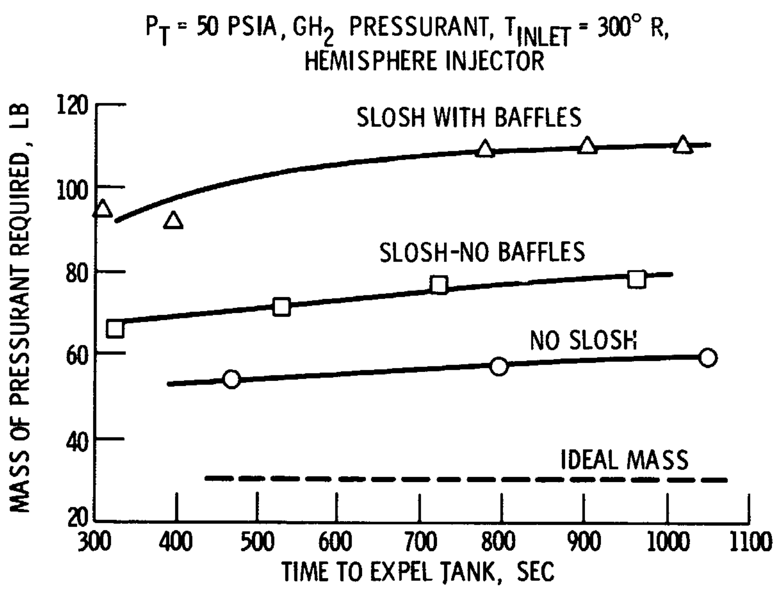


Figure 9. - 13-Foot tank pressurant gas requirements under slosh conditions (ref. 4).

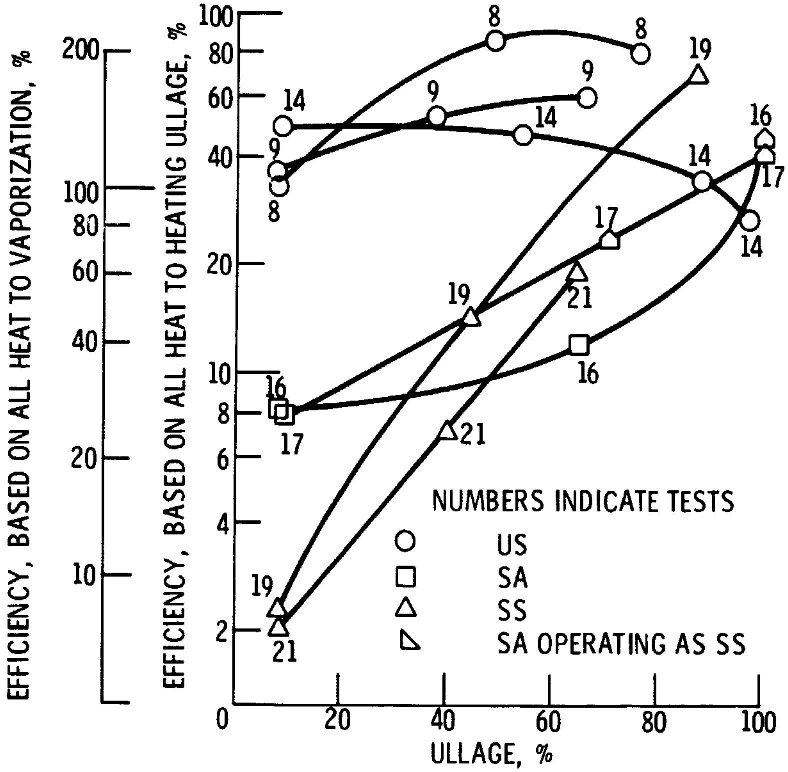


Figure 10. - Prepressurization efficiency (ref. 5).

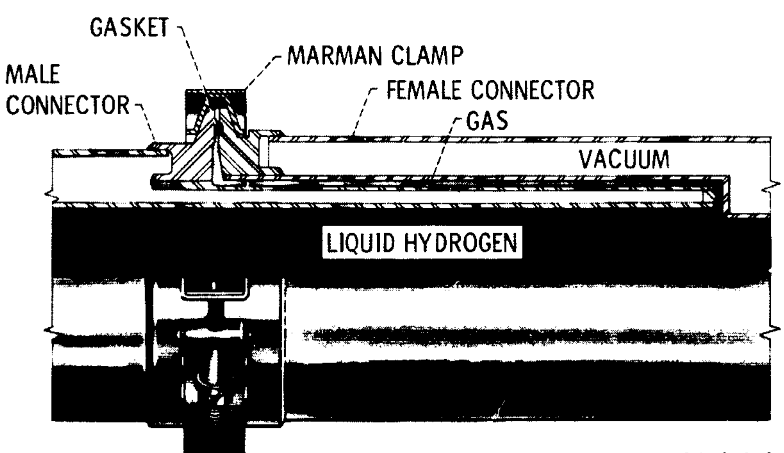


Figure 11. - Liquid-hydrogen bayonet-type low-temperature joint.

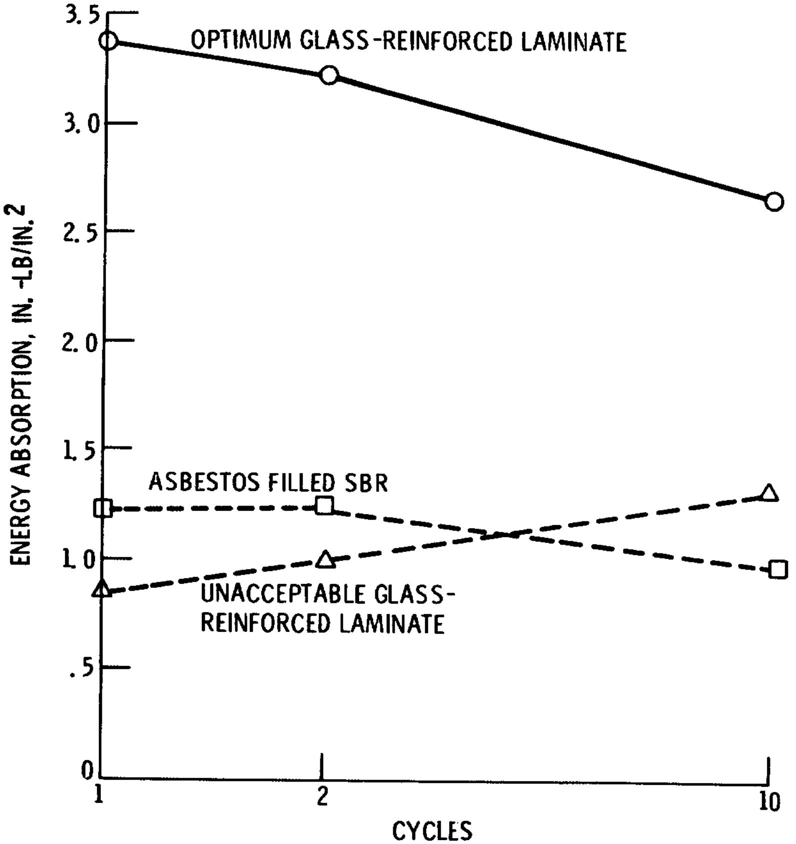


Figure 12. - Energy absorption of gaskets.

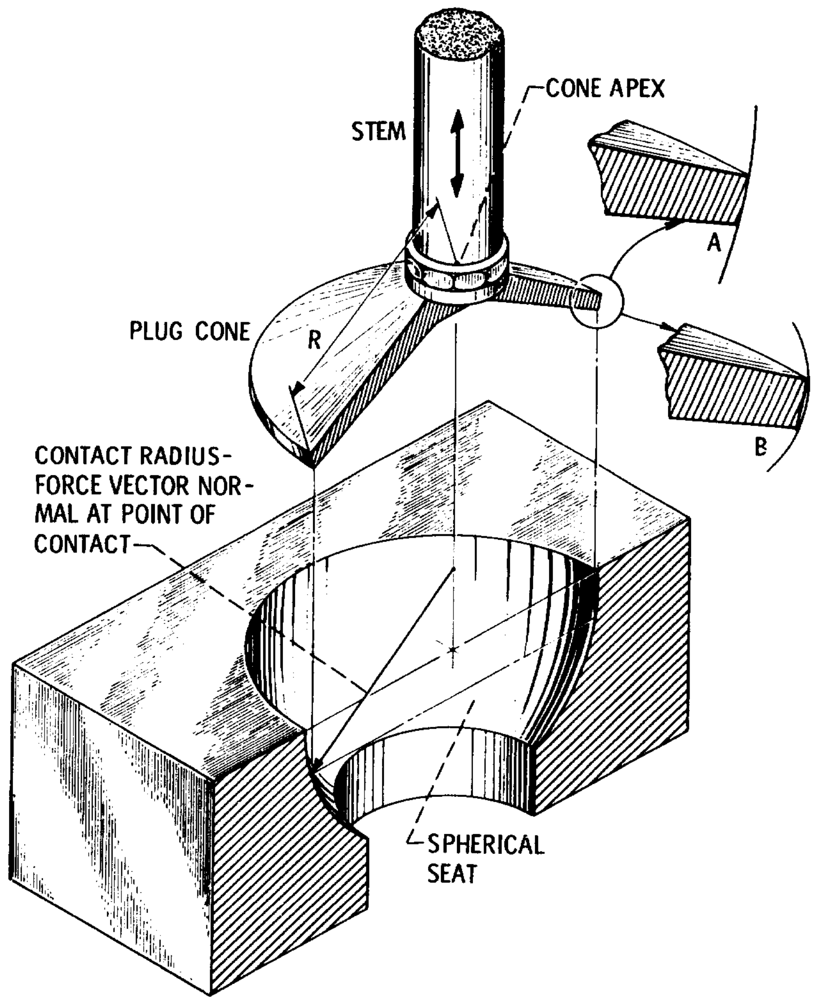
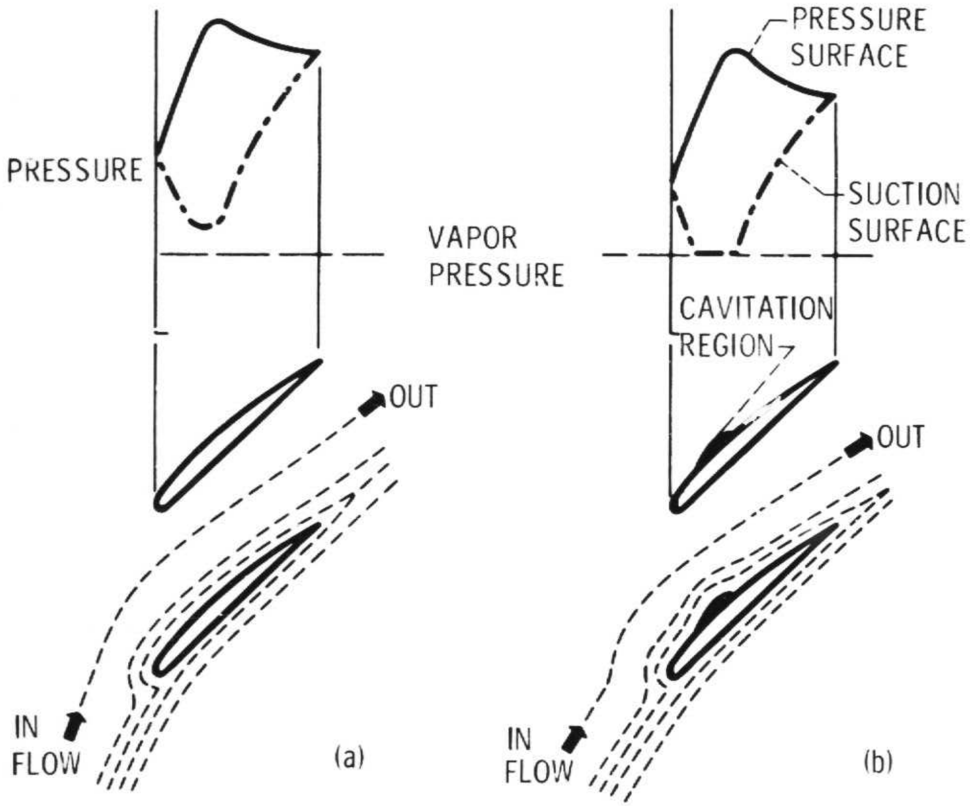


Figure 13. - Schematic drawing of conical valve design and operation.



(a) Noncavitating.

(b) Limited cavitation.

Figure 14. - Blade pressure distributions with and without cavitation.

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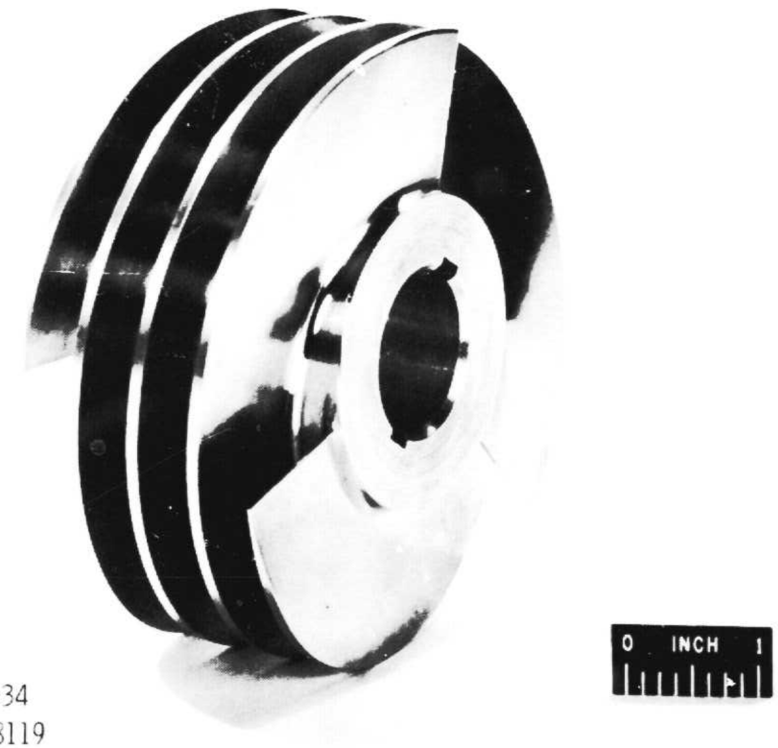

Figure 15. - Inducer.

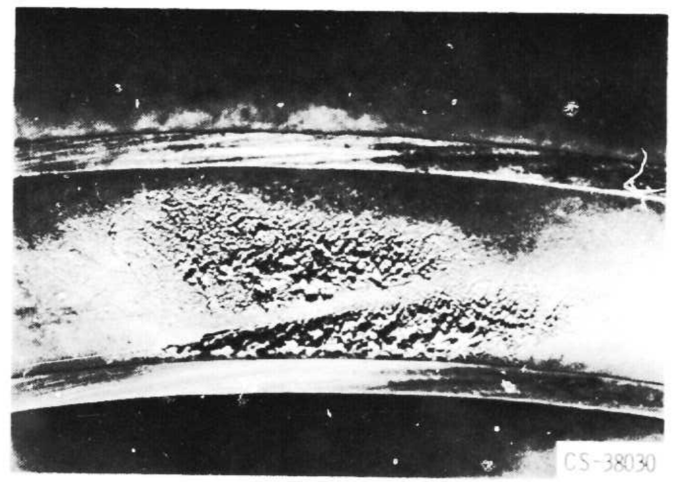

Figure 16. - Pump blade cavitation damage.

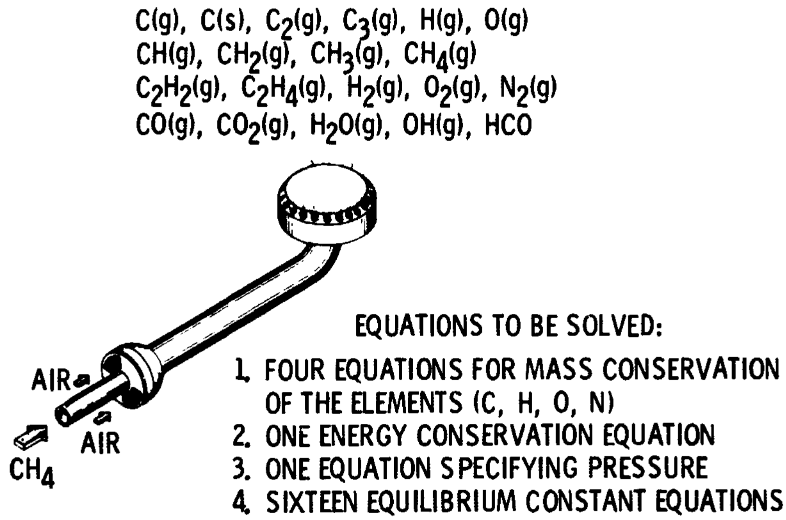

Figure 17. - Combustion process.

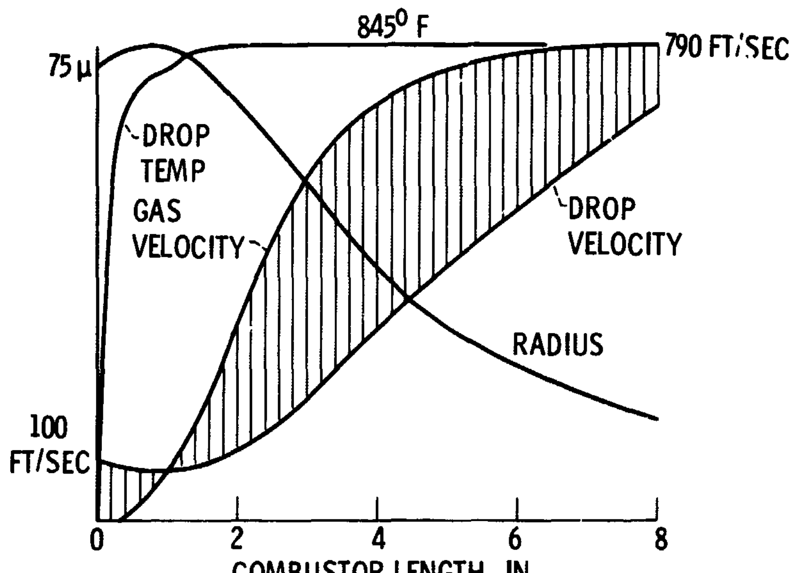

Figure 18. - Calculated vaporization process for uniform heptane drops.