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# EXPLORING IN AEROSPACE ROCKETRY
# 7. LIQUID-PROPELLANT ROCKET SYSTEMS

by E. William Conrad
Lewis Research Center
Cleveland, Ohio

Presented to Lewis Aerospace Explorers
Cleveland, Ohio
1966-67



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# 7. LIQUID-PROPELLANT ROCKET SYSTEMS

## E. William Conrad*

Liquid-propellant rockets may be classified as monopropellant, bipropellant, or tripropellant. In a monopropellant rocket, a propellant, such as hydrazine, is passed through a catalyst to promote a reaction which produces heat from the decomposition of the propellant. A bipropellant rocket burns two chemical materials, a fuel and an oxidizer, together. In a tripropellant rocket, three different chemical species, such as hydrogen, oxygen, and beryllium, are mixed in the combustion chamber and are burned together. These tripropellant rockets have great potential, but they are not yet in actual use because they present many developmental problems. The following discussion will be restricted to bipropellant rockets because this type is used for the bulk of our present space activities.

## ROCKET ENGINE

A simple, liquid-propellant rocket engine is shown in figure 7-1. The principal components of this engine are the injector, the combustion chamber, and the exhaust nozzle.

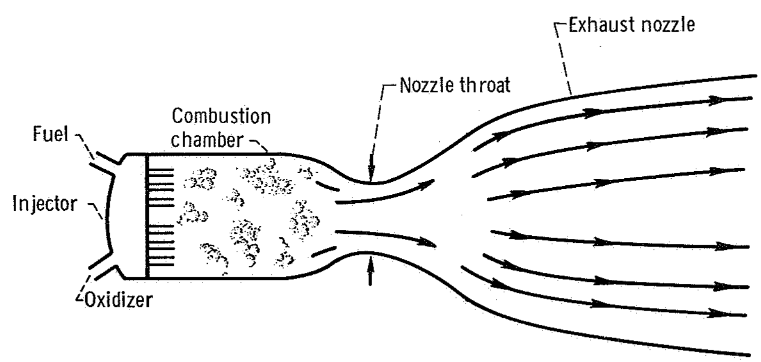


Figure 7-1. - Bipropellant liquid rocket engine.

*Chief, Chemical Rocket Evaluation Branch.

The propellants enter the combustion chamber through the injector. In the combustion chamber the propellants mix and are ignited. Some propellants, such as the oxygen-kerosene combinations used in the Atlas launch vehicle, are ignited by means of a spark plug. Other propellants, such as the nitrogen tetroxide - hydrazine combination used in the advanced Titan launch vehicle, are hypergolic; that is, when the two propellants are mixed, they ignite spontaneously. When the propellants burn, they produce very hot gases. The high temperature, in turn, raises the pressure of these gases in the combustion chamber. The increased pressure causes the gases to be discharged through the exhaust nozzle. As these gases pass through the exhaust nozzle, they are accelerated and expanded. The area reduction at the nozzle accelerates the gas to sonic velocity at the throat. Then, in the diverging portion of the nozzle, the gases are expanded and accelerated to supersonic velocities. (This flow process is discussed in chapter 2.)

## Fuel Injector

The design of the injector is of great importance because the propellants must be introduced into the combustion chamber in such a way that they will mix properly. The objectives of the mixing process are to attain fine atomization of the propellants, rapid evaporation and reaction of the propellants as close as possible to the injector face, and a uniform mixture ratio throughout the combustion chamber. The ultimate goal is to have each molecule of fuel meet an appropriate number of oxidizer molecules and be completely consumed in the combustion process. A detailed discussion of the fundamental processes of combustion is presented in chapter 4.

Injectors of many types are used to accomplish the desired mixing of the propellants. Some of the most commonly used injectors are shown in figure 7-2. The double impinging stream injector, shown in figure 7-2(a), is a relatively common design. Fuel and oxidizer are supplied to the combustion chamber through alternate manifolds, so that each fuel stream impinges on an oxidizer stream. This impingement shatters the streams into ligaments, which, in turn, break up into droplets. Finally, the droplets evaporate and burn. The triple impinging stream injector (fig. 7-2(b)) is also very common and highly efficient. With this design, two streams of one propellant impinge on a stream of the other propellant at a common point. Figure 7-2(c) shows the self-impinging pattern, in which two streams of the same propellant impinge on each other and shatter to produce a fine, fan-shaped, misty spray. Alternate manifolds in the injector produce fans of fuel mist and of oxidizer mist. These fans mix and burn along their intersections. The shower-head stream injector, shown in figure 7-2(d), was very common in the early days of rocketry. With this design, streams of each propellant are simply injected parallel to one another. The efficiency of this system is, in general, relatively poor. Too much of

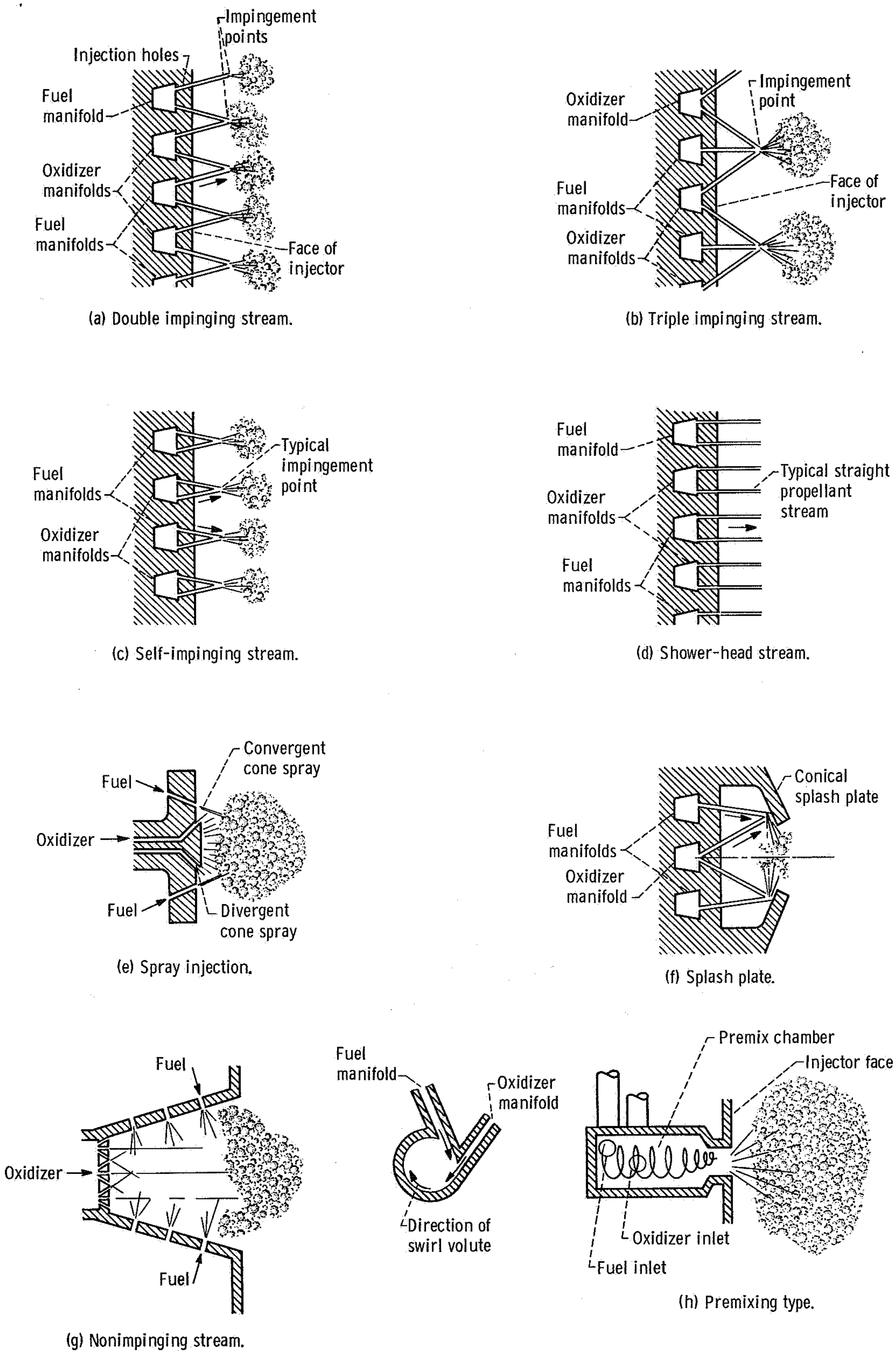


Figure 7-2. - Several injector types.

each propellant goes out the exhaust nozzle without mixing and reacting with the other propellant. Because of this low efficiency, very few current engines use shower-head injectors.

Also shown in figure 7-2 are some rather unusual injector designs. The spray injector (fig. 7-2(e)) produces a cone of oxidizer and impinging streams of fuel. Figure 7-2(f) shows a splash-plate injector, in which streams of fuel and oxidizer impinge at a point on the splash plate. This impingement produces sprays that eddy around the splash plate and promote further mixing of the propellants. The nonimpinging stream injector, shown in figure 7-2(g), has a precombustion chamber in the form of a cup sunk into the injector face. Many streams of both propellants are injected into this precombustion chamber to produce a rather violent mixing. The premixing injector (fig. 7-2(h)) has a premixing chamber into which the two propellants are injected tangentially to produce a swirling mixing action. There is a new and relatively efficient injector which uses a quadruple impinging stream pattern. With this design, two streams of each propellant impinge at a common point. This injector is particularly effective for use with storable propellants.

The concentric tube injector, shown in figure 7-3, is probably the optimum design for hydrogen-oxygen propellant combinations. The oxidizer enters the oxidizer cavity through the center pipe, then flows outward throughout this cavity, and enters the combustion chamber through the hollow oxidizer tubes. The fuel enters the fuel cavity, which is just under the injector face, and thence it flows into the combustion chamber through the annuli which surround the oxidizer tubes.

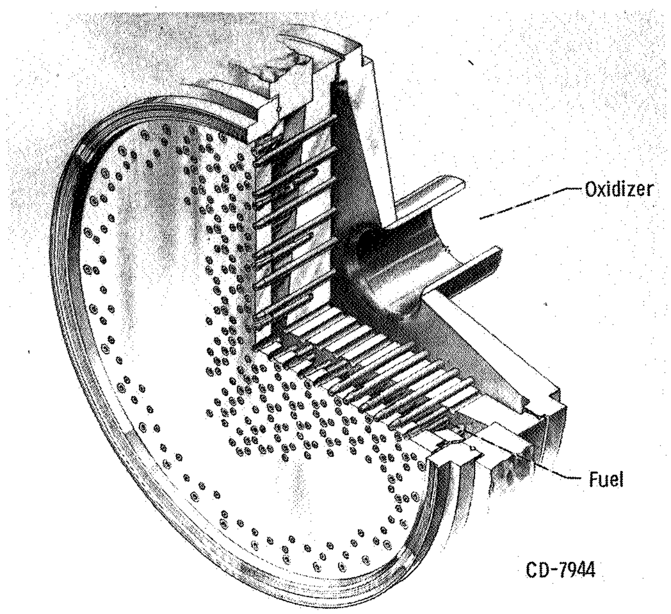


Figure 7-3. - Cross section of concentric-tube injector.

All these injection techniques create a combustion zone which has a great deal of energy contained in it; this concentration of energy can cause severe problems. One great difficulty in developing new rocket engines is combustion instability, particularly the variety at high frequency which we call "screech" or "screaming." This phenomenon has plagued propulsion people since afterburners were developed in the late 1940's and has continued through ramjets and into the rocket field. Screech can increase heat transfer by a factor of as much as 10 and thus is extremely destructive. As an example, in figure 7-4 is shown an injector face which experienced screech for only 0.4 second. Why screech happens is not yet fully understood, but there has been, and there still is, a great effort aimed at trying to solve the combustion instability problem. What apparently happens in the engine is that pressure waves are set up which have various possible modes of oscillation as acoustic systems. The waves may oscillate, or pulse, from the injector face down to the exhaust nozzle throat where there is a sonic plane, and bounce back to the injector face - the longitudinal mode. They do this at the speed of sound,

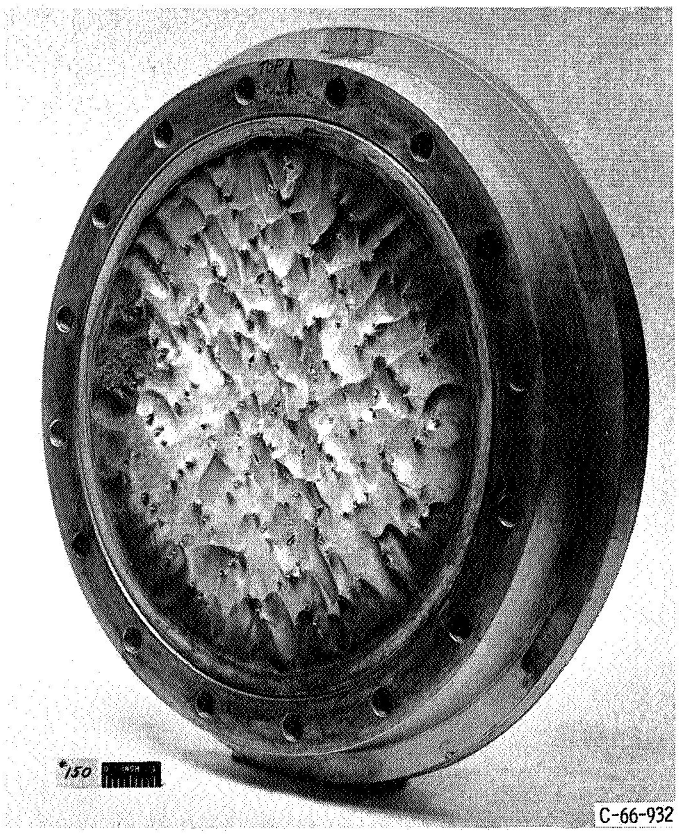


Figure 7-4. - Injector after operation with screech for 0.4 second.

which in that medium is very high so that the frequencies in a large engine would be of the order of 2500 to 3000 cycles per second. It is also possible for the pressure wave to travel from the center of the engine or injector area out radially to the walls of the chamber and back into the center in an expanding-contracting fashion - the radial mode. There is another mode, the transverse, or tangential, where the pressure waves start at the top point and travel around and bump into one another at the bottom and reflect back around to the top. This tangential mode of screech is particularly destructive.

Acoustic liners. - There are numerous ways of combating high frequency instability; one of the most promising techniques is the use of an acoustic liner which works much like the acoustic tile used on ceilings. The liner presents a perforated surface to the combustion zone and although part of each wave hits the solid part of the surface, bounces back, and is not dissipated efficiently, other parts pass through the liner into an acoustic resonator, an example of which is shown in figure 7-5. Helmholtz resonator theory is used to determine the size of the cavity behind the holes so that the sound energy, or pressure wave energy, that passes through the hole is broken up and dissipated. These liners have been quite effective, but they are not a cure-all. They are a very valuable tool, but we have not yet learned how to fully optimize the design of these devices to achieve maximum effectiveness.

Baffles. - Another way of eliminating this instability which is usually successful is by the use of baffles on the injector face. Shown in figure 7-6 is an injector with such baffles from the Air Force Transtage engine. Four baffles are used that extend down into

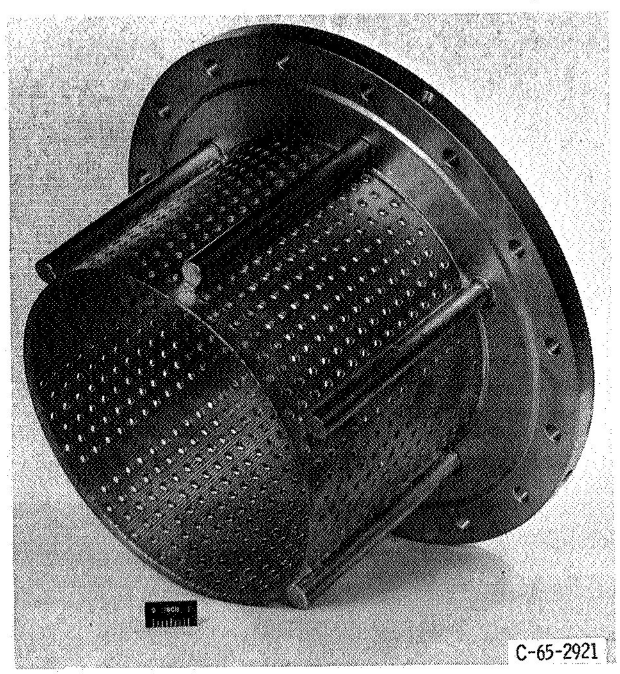


Figure 7-5. - Flight weight acoustic liner for screech suppression.

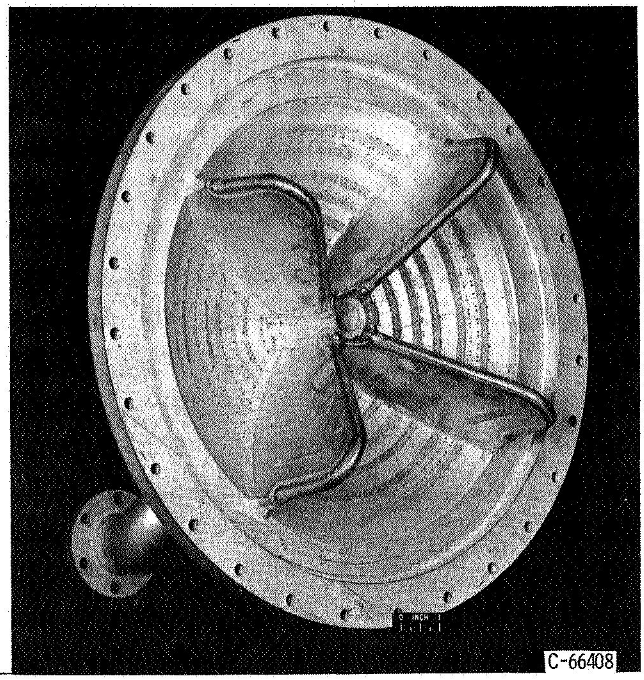


Figure 7-6. - Baffles attached to injector face for screech prevention.

the combustion zone to interrupt the progress and reinforcement of these tangential pressure waves. This method is based on the hypothesis that screech originates much like a detonation wave; once the disturbance is created in the combustion zone, the waves accelerate the combustion process locally, which, in turn, provides energy to reinforce the wave. These waves will propagate and grow as they feed on the chemical energy that is present. The baffles represent an attempt to interrupt the waves and reflect them back into a zone where there is no unburned propellant to supply energy for their continuation. Other hypotheses have been advanced, however, and no theory is yet confirmed.

## Cooling

Having created an inferno, the designer is next faced with the problem of how to contain it. The gas temperatures vary between about $4000^{\circ}$ and $7000^{\circ}$ F, depending upon the particular propellant combination. However, most of the structural materials in common use melt at much lower temperatures. For example, stainless steel or Inconel melt at about $2200^{\circ}$ or $2400^{\circ}$ F, much below the combustion gas temperature. There are refractory alloys such as molybdenum which melts at around $4700^{\circ}$ F. However, the use of molybdenum poses some problems because it oxidizes extremely rapidly; in fact, it will simply sublimate, going directly from the solid phase into the gas phase unless it is protected from oxygen attack. Coated molybdenum is, therefore, one of the materials that are being carefully considered in advanced engines.

Regenerative method. - Since no known material will work unassisted, the designer must employ active cooling techniques. There are many ways to cool engines, none of which are optimum for all propellant combinations or all types of engine. Therefore, several different techniques are used. Regenerative cooling, perhaps the most common system, is used in the Atlas engines, in the F-1, and the J-2 engines used in the Saturn booster stages. Figure 7-7 shows a cross section of a regeneratively cooled engine

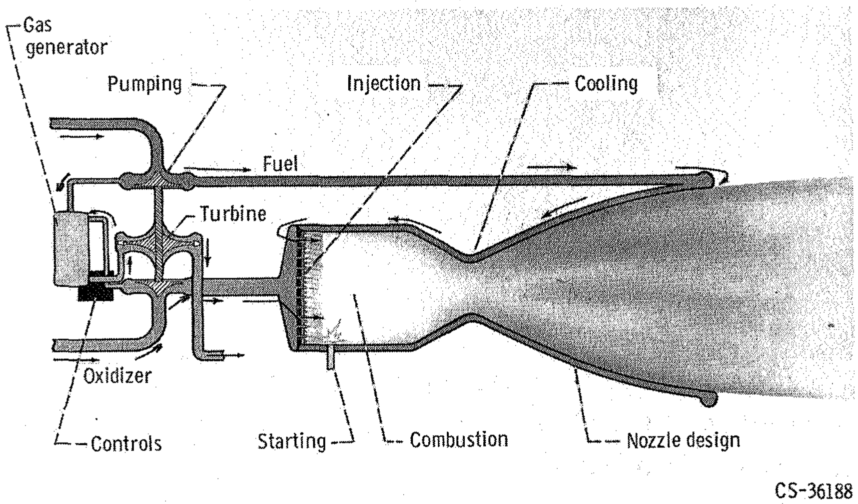


Figure 7-7. - Regeneratively cooled rocket engine.

where one of the two propellants is used as a coolant. The propellant used for cooling is piped into the engine at the downstream end of the exhaust nozzle. There it is divided among as many as perhaps 100 small tubes which extend the entire length of the engine and are brazed together to form the walls of the engine. Each of these tubes is specially formed and tapered to change the local velocity of the coolant as it passes forward to the injector. The local coolant velocity is calculated to match the expected heat distribution from the combustion process. In the case of the hydrogen burning engines, hydrogen is used as the coolant because it is excellent in this regard. It can be heated to almost any temperature and will continue to absorb heat so that the hydrogen itself does not pose a limitation. If, on the other hand, hydrazine is used to cool an engine, like those in the Titan, local velocity must be very high because the hydrazine will detonate if it gets above $210^{\circ}$ or $220^{\circ}$ F.

Ablative cooling. - There are some cases where the regeneratively cooled engine cannot be made to cool properly. One of these cases is where there is a need to throttle the engine to produce a lower thrust. When the engine is throttled, propellant flow is reduced and the velocity of the coolant in each of the coolant tubes is reduced. Accordingly, the heat-transfer capability of the coolant in the tube decreases. Consequently, the metal begins to overheat and burnout will occur. The engines for the Apollo landing vehicle, which require a 10:1 throttling, use ablative cooling. These thrust chambers are made essentially of glass fibers and a plastic. The problem of reduced propellant velocity is avoided because the ablative engine is not adversely affected by operation at reduced thrust. In figure 7-8 is shown an example of an ablative engine after firing.

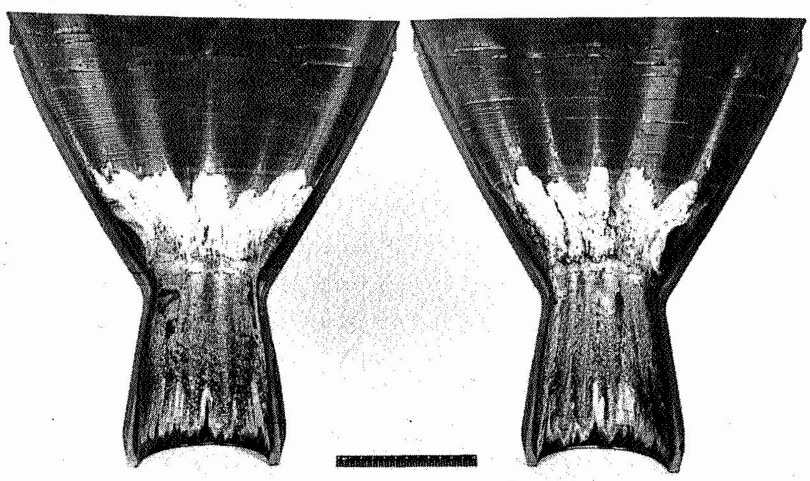

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Figure 7-8. - Sectioned ablative combustion chamber after 151 seconds of operation.

This engine is almost the same size as the engine to be used in making the lunar landings with Apollo. The Apollo engine is made of quartz fibers which are more or less perpendicular to the engine centerline; these are imbedded in a phenolic resin. The principle of operation here is this: the heat vaporizes the resin, each pound of resin absorbing between 2000 and 5000 British thermal units of heat in the process. The gases that evaporate from the resin flow over the hot inside surface of the combustion chamber and nozzle, cooling these surfaces by evaporation. Furthermore, the gases act as a barrier to prevent more heat from entering the ablative wall from the hot combustion gases. Eventually, however, the quartz fibers melt and beaded runoff of quartz occurs. This principle was used on the heat shield of the first reentry body returned from space. It is also the principle that is used on the Mercury capsule heat shield and on the Apollo heat shield as well as the engines.

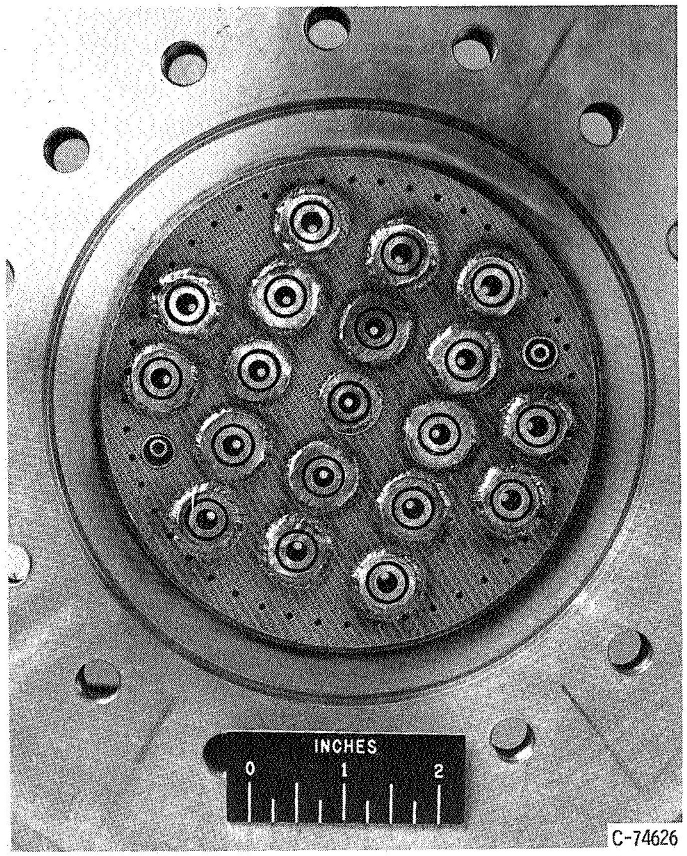

Figure 7-9. - Injector with porous faceplate for transpiration cooling.

Transpiration cooling. - Another technique for cooling rocket engines is transpiration cooling. Transpiration cooling simply consists of using a porous material to fabricate the chamber wall, and forcing through the porous wall a small quantity of one of the propellants. This is practical, particularly with a propellant such as hydrogen which is gaseous by the time it emerges on the hot side of the wall. Figure 7-9 is a view of an injector with a porous faceplate for transpiration cooling. One common way of making a porous wall consists simply of layers of stainless screen that are then pressed and sintered to make a rather rigid structure. The merit of the transpiration cooling system is that it probably can be made to work under the most severe heat-transfer conditions we can imagine. It will probably allow higher chamber pressure (with its consequent higher heat transfer) than will either the regenerative system or the ablation system. There is a penalty, however, in performance in using transpiration cooling, particularly if extremely severe heat-transfer conditions require high coolant flows. Much of the propellant used as a coolant flows through as a boundary layer and does not become fully involved in the combustion process. This means that more propellant is needed with this cooling system than with either of the others, propellant which adds weight but little thrust. Therefore, by comparison, a transpiration cooled engine is slightly less efficient than either a regenerative or an ablative cooled one.

Dump cooling. - An experimental engine using the dump cooling technique is shown in figure 7-10. The dump coolant, hydrogen in this case, enters a concentric shell

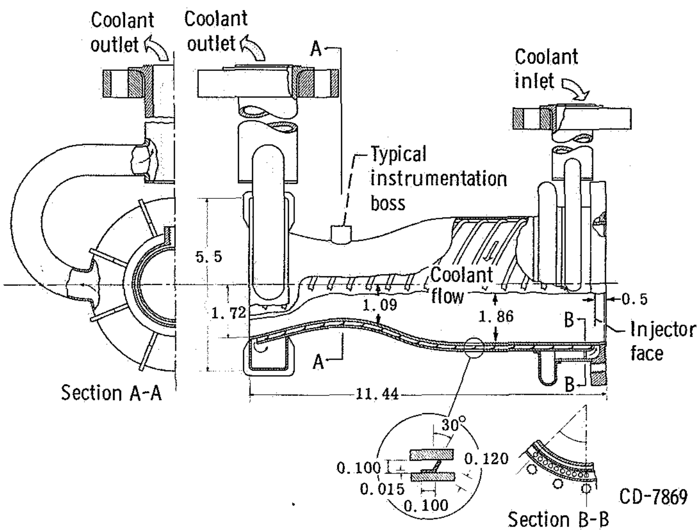


Figure 7-10. - Sketch of experimental engine with "dump" cooling.

around the combustion chamber at the injector end and flows around the chamber through helical passages to the aft end, thus cooling the chamber wall. In a flight engine, the warm hydrogen resulting would flow through an exhaust nozzle independent of the main stream and would produce an impulse almost as high as the main propellant stream. This technique is probably most applicable to engines operating at low chamber pressure.

Radiation cooling. - A final method, which has been used on small engines in particular, is radiation cooling. With this technique, the combustion chamber is made from a coated refractory alloy such as columbium, molybdenum, or tantalum-tungsten which can operate at metal temperatures of the order of 3000° F; such a chamber operates white hot. The inside of the chamber receives heat from the combustion gases which are of the order of 5000° F, while the outside of the chamber radiates the heat so received into space. The heat transfer due to radiation is in accordance with the Stefan-Boltzmann Law, which states mathematically:

$$Q = \epsilon\left(T_m^4 - T_o^4\right)$$

where

$Q$ quantity of heat transferred/unit time

$\epsilon$ emissivity (usually about 0.9), the ratio of radiation emitted by a surface to that emitted by a perfect radiator

$T_m$ thrust chamber temperature, $^oR$

$T_o$ temperature of matter surrounding engine, $^oR$

Inasmuch as the temperature $T_o$ in space is almost absolute zero, the equation becomes

$$Q = \epsilon T_m^4$$

When a radiation-cooled engine is started, the chamber will quickly heat to some very high temperature at which the heat rejected by radiation is exactly equal to the heat received from the combustion in the chamber. In practice, such engines must be carefully located to avoid overheating any portion of the spacecraft which can "see" the hot chamber and, therefore, can receive radiation from it.

## PROPELLANT SUPPLY

Since the combustion chamber and its operation have been examined, it is appropriate to examine next the systems which deliver the propellants to the combustion chamber. Basically, only two system types are used: pressure-fed or pump-fed, although many variations are possible with either system.

Pressure-fed system. - A typical pressure-fed system is shown schematically in figure 7-11. An inert gas, usually helium or nitrogen, is stored under high pressure in the bottle labeled "gas supply." Prior to engine start, the gas is allowed to enter the fuel and oxidant tanks through check valves (one way) and a pressure regulator which will maintain the pressure in the tanks at some preset value higher than the desired thrust chamber pressure. The engine is started by opening the fire valves in a carefully controlled sequence (with the use of an electronic timer) to allow the gas pressure to force the propellants into the thrust chamber. As the propellants are consumed, additional gas is supplied to the tanks by the pressure regulator to maintain constant pressure until the tanks are empty or shutdown is commanded. More sophisticated systems include various methods of heating the pressurant gas to reduce the quantity required. It should be noted that with pressure-fed systems, the propellant tanks must operate at high pressure and must, therefore, be strong and heavy. As a result, this system is competitive with pump-fed systems only for fairly small missile stages.

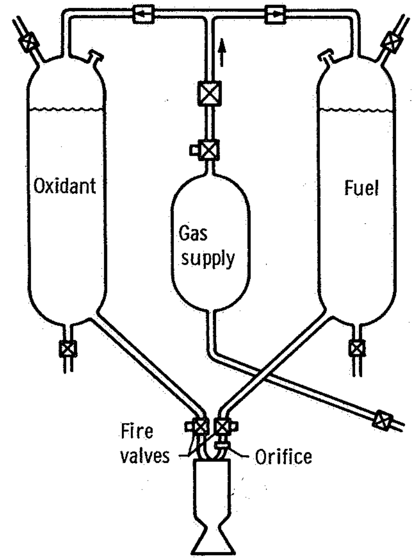


Figure 7-11. - Typical pressure-fed propellant system.

Pump-fed system. - One version of a pump-fed system, used for the RL-10 engines in the Centaur vehicle, is shown schematically in figure 7-12. Boost pumps at each propellant tank, not shown, are driven by catalytic decomposition of hydrogen peroxide in a gas generator to supply propellants at relatively low pressures to the inlet of the pumps shown in the figure. The oxygen is pumped to a pressure of about 500 psi and then passes through the mixture ratio control valve to the injector, which sprays it into the combustion chamber. Hydrogen, on the other hand, is pressurized by a two-stage pump; from there,

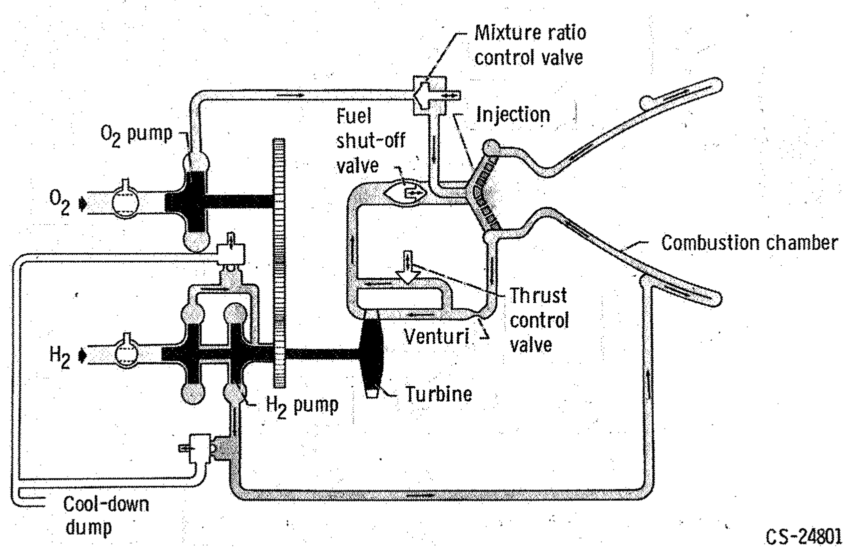


Figure 7-12. - Standard RL-10 engine and propellant supply system.

it enters the engine cooling jacket where, in cooling the chamber its temperature is increased from about $40^o$ to $300^o$ R. The warm gaseous hydrogen leaving the front end of the cooling jacket is then used to drive a turbine which powers both the hydrogen and oxygen pumps. After leaving the turbine, the hydrogen passes through the injector into the combustion chamber. Pump speed is regulated by a controlled bypass of some of the hydrogen around the turbine. The system just described is referred to as a "bootstrap" system. Another common system simply employs a separate combustion chamber or gas generator using engine propellants tapped off the main supply lines to drive the turbine which drives the pumps. The exhaust gas from the turbine is ducted overboard through a separate nozzle.

## Propellants

To round out this brief description of chemical rocket propulsion, it is appropriate to consider the propellants - and the reasons for their selection. In figure 7-13, specific impulse is plotted as a function of the bulk density of the propellant combination for a number of combinations. It should be recalled from a preceding chapter that specific impulse is a figure of merit much like gas mileage of an automobile. It is equal to the pound-seconds of thrust produced for each pound-per-second of propellant flow. It will be

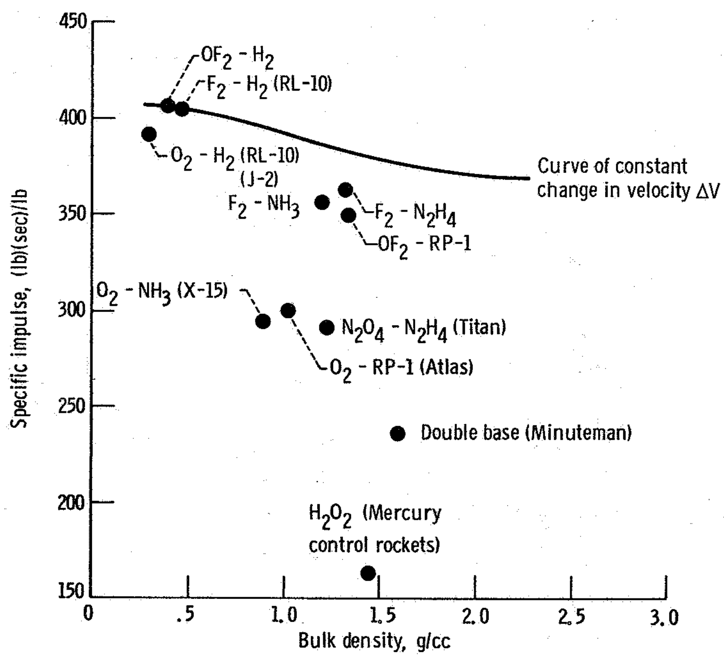


Figure 7-13. - Performance characteristics of typical propellant combinations.

seen that the maximum values of specific impulse are achieved with $OF_2$-$H_2$ and $F_2$-$H_2$, followed by $O_2$-$H_2$. These propellant combinations are, of course, of considerable interest since they give significantly higher values of impulse than the $O_2$-RP-1 used in Atlas or the $N_2O_4$-$N_2H_4$ used currently in Titan.

By itself, however, specific impulse is not a singular criterion of merit - other factors must be considered, such as the bulk density shown by the abscissa. The overall objective of a missile stage is to impart a maximum change in velocity to the stage and its payload. As given in chapter 2, the equation relating these factors in space is:

$$V_f - V_i = \Delta V = I_{sp} g \ln\left(\frac{W_f}{W_e}\right)$$

where

$V_f$ final velocity

$V_i$ initial velocity

$I_{sp}$ specific impulse

$g$ universal gravitational constant (32.217 ft/sec$^2$)

$W_f$ weight of stage loaded with propellant

$W_e$ empty weight (at burnout)

Inspection of this equation shows that the velocity change can be increased by increasing specific impulse, but also, it can be increased by decreasing the vehicle empty weight or structural weight. In this regard then, the use of higher density propellants will result in smaller propellant tanks and, hence, a lower structural weight. The trade-off between specific impulse and bulk density is indicated in figure 7-13 by the curve of constant $\Delta V$. From this, it may be seen that $F_2$-$H_2$ will produce a greater velocity change than will $OF_2$-RP-1 (for an equal velocity change, $OF_2$-RP-1 would have to show a specific impulse of 380 seconds instead of 350). Thus, the $\Delta V$ produced is not simply given by the ratio of the specific impulse values, but is also affected by the bulk density.

## Miscellaneous Considerations

There are other factors that are important in the propellant selection; one of them is hypergolicity. This ability of some propellant combinations to burn spontaneously is a desirable characteristic because it eliminates spark plugs or other ignition devices and thereby improves reliability. Another important consideration is the temperature at which the fuel and the oxidizer are liquid. Ideally, this temperature should be the same for both; otherwise, the colder propellant will freeze the warmer one solid. For example, although hydrogen and nitrogen tetroxide make a good propellant combination, their liquid temperature ranges are so different as to require heavy insulation between the storage tanks. This means extra weight with its consequent interference with performance unless the storage tanks are placed far apart - a design difficult to achieve in most rockets, and one which frequently adds weight in the form of extra piping and bracing. Toxicity is another factor. Fluorine is extremely toxic; nitrogen tetroxide is somewhat more toxic than phosgene. These toxic propellants have to be treated with respect and involve costly safety equipment and procedures which the user would like to avoid. The designer must consider propellants with an adequate supply and reasonable cost. These have generally not been factors in the programs to date; however, they are important in selecting propellants for advance missions, inasmuch as it is possible to specify something that is beyond our ability to supply. For example, if the tripropellant combination involving finely powdered beryllium were suddenly the only way of meeting a very energetic mission requirement, there would be enormous problems of finding supplies and suppliers. Storage and insulation are important; for example, if the mission were to fly close to the Sun, say within 1/10 astronomical unit, and return, the mission time would be about 220 days. To contain liquid hydrogen for long time periods such as this without excessive boiloff requires exceptional insulation, and the designer must pay the penalty in terms of weight. Other propellant combinations then, in this particular type of mission, are competitive, because, even though their impulse may be lower, the designer can avoid all this insulation weight. Finally, there is the consideration of the cooling capacity of the propellant; for example, hydrazine detonates at about $210^{\circ}$ F and, therefore, is limited in the amount of heat it can absorb, but hydrogen can absorb any quantity of heat without limitations except those imposed by the materials to contain the hot hydrogen.

This discussion has covered the more important factors of the many that govern the behavior of liquid propellant rockets and which the designer must consider. A detailed treatment of the subject is contained in Rocket Propulsion Elements by George P. Sutton, Third ed., John Wiley and Sons, Inc., 1963.

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