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History of Liquid Propellant Rocket Engines in the United States

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I. Introduction

The liquid propellant rocket engine (LPRE) is a proven means of propulsion. It was conceived over 100 years ago, but its first actual construction in the United States (and in the world) was accomplished by an American, namely, Robert H. Goddard, in 1921 (82 years ago). His first static hot-firing test was in 1923 and the historic first flight with a LPRE occurred in 1926. Today this technology is sufficiently well developed and proven that we can design, build, and fly with confidence any kind of LPRE. In 1940, there were only a few outstanding individuals and groups that were struggling with early research and development efforts. The LPRE capability has proliferated and grown, and today there are several active U.S. companies and several government laboratories that have a mature broad LPRE technical base.

The reader should have an understanding of LPREs, a background or exposure to some aspect of the subject. For basic information refer to the general Refs. 1–8 and, for more detail, to a future book by the author on a world wide history of LPREs, scheduled to be published by the AIAA in late 2004.

There is no single LPRE concept or type, but rather several that are related and tailored to specific applications. All have one or more thrust chambers (TCs), a feed system for providing the propellants under pressure to the thrust chamber(s), and a control system. There are significant differences between LPREs with high thrust and low thrust, cryogenic vs storable propellants, monopropellants or bipropellants, single use or reusable, one run per flight vs multiple restarts during flight, random variable thrust or nearly constant thrust, and those with pumps or gas pressure expulsion of propellants in their feed systems. The history of each of these types will be discussed.

In this paper a "successful LPRE" is defined as one that 1) has been put into production and/or 2) has flown its mission satisfactorily more than once. After all, the ultimate objective is to propel a vehicle. There have been many LPREs, engine components, and propellants, but for various reasons they were never successful, and most fell by the wayside. Yet we learned some important lessons from them. We will concentrate on some of the successful LPREs, but we will also discuss some others that have interesting technology or historic significance.

An estimated 300–350 different LPREs have been designed, built, and static tested in the United States. Because of space limitations, only a few of them have been selected for this summary paper. For each of those, only a few pieces of data or a figure will be given here. If a significant LPRE or outstanding accomplishment was omitted, it was not by intent, but by the lack of information available to the author and/or the space limitation for this paper. Although some of the flight vehicles driven by a LPRE (airplanes, missiles, or spacecraft) are mentioned or identified here briefly, the emphasis in this work is on the rocket engines and not on the rocket vehicles themselves. Gaseous propellant engine systems are included because they are usually grouped with the LPREs. We will not cover solid propellant rocket motors, electrical propulsion, hybrid propulsion, and combination rocket-airbreathing engines.

II. Need for LPREs

Why were LPREs used? Because they propelled certain military and space vehicles better than any other type of chemical propulsion and because they provided some operating characteristics that could not be duplicated at the time by any other means of propulsion.[2] LPREs made it possible to build sounding rockets (1926–1960); they propelled military aircraft and assisted with their takeoff (1942–1970). They went into production for several early tactical missiles (1951–1973) because solid propellant rocket motors could not meet the operating temperature limit requirements during the 1940s and 1950s. LPREs were selected for all the initial ballistic missiles, helping to build up the military missile inventory needed urgently by the U.S. Government in the 1950s–1970s. Since 1960, LPREs propelled all of the large space launch vehicles and just about all the U.S. spacecraft and satellites. They constitute the propulsion machinery that drove us into the space age.

The features and performance characteristics of LPREs that allowed their selection for the mentioned missions, were unique and are briefly reviewed next.[2] Liquid bipropellants generally give a higher specific impulse than other chemical propulsion means, such as monopropellants or those using solid or hybrid propellants. Cryogenic propellants give the highest specific impulse. LPREs can be designed over a very wide range of thrust values to fit specific applications (by a factor of 10^8). They are the only form of chemical

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propulsion that can be designed for quick restart, fast pulsing, and ready reuse. They can be designed for a random thrust variation on command. They have been uniquely suitable for controlling quick attitude (pitch, yaw, or roll) changes and minor velocity changes of individual stages of missiles, space launch vehicles, spacecraft, and satellites. A precise repeatable thrust termination permits an accurate terminal flight velocity. LPREs can be functionally checked out and even fully tested before they are used. An engine-out capability can be designed into engine clusters. A remarkably high reliability has been achieved in production LPREs. Lightweight LPREs have allowed flight vehicles to achieve a high propellant fraction and a high vehicle mass ratio. Instant readiness has been achieved with storable propellants. These propellants have been stored for 20 years in a vehicle.

All common propellants used today can discharge a very clean transparent exhaust gas without smoke. Gas from certain of the storable propellants can give a trace of smoke, but their particulates do not usually form a noticeable deposit on sensitive vehicle surfaces, such as windows. The exhaust gas of today's LPREs is not toxic and environmentally friendly.

III. Technology Trends and Changes

Throughout this LPRE history, one can discern some technical trends, growth patterns, or directions for improvement. Some are briefly listed later. There are several other trends, but their write-up had to be omitted from this paper. They were thrust vector control, injectors, gas generators or preburners, major reductions of inert engine mass, extending or uprating a family of existing engines, and reducing lifetime costs. They are planned to be included in the upcoming book version.

A. Expanding the Range of the Thrust, 0.01–1,800,000 Pounds Force (Refs. 1–4)

The thrust magnitude is dictated by the application. The first thrust chambers (Goddard 1921–1924) had between 40- and 100-lbf thrust and were intended for small sounding rockets. Some of his early bipropellant TCs (roughly 1.2 in. diameter) are shown in Fig. 1. Historically the thrust levels went both up and down. By 1944, a series of hydrogen peroxide monopropellant thrust chambers (for reaction control) became available with thrust values as low as 0.1 lbf. With inert or with warm gas as the propellant, the thrust levels went even lower. It took 45 years to increase of the thrust to 1,800,000 lbf as seen in Table 1.

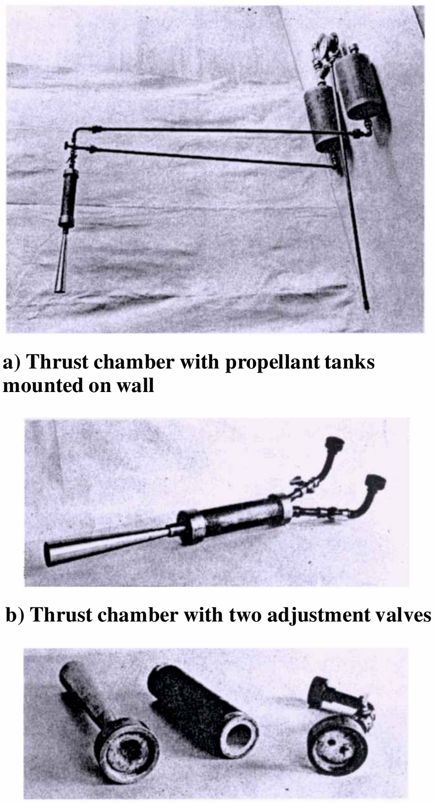
The highest takeoff thrust was with five F-1 engines at 7.5×10^6 lb for the S-IC booster stage of the Saturn V space launch vehicle (SLV). There have been no application requirements for higher thrusts since about 1969. The F-1A did not fly, and the program was not continued.

B. Increasing the Chamber Pressure

The historical trend has been to raise the chamber pressure. This makes it is possible to increase specific impulse between 4 and 10%. The exact values depend on the specific design, chamber pressure, nozzle area ratio, and application. Goddard started (1920s) with relatively low chamber pressure, typically 50–100 psi, but later went up to 350 psi with a pump feed system in 1939. In the 1940s gas pressurized feed systems allowed increases to 500 psi. With pumped feed systems, these chamber pressures reached 1000 psi in the 1950s. There were some exceptions. Some small experimental TCs were tested at more than 5000 psi in the 1970s. The highest chamber pressure of a flying U.S. engine was 3319 psi in block I of the space shuttle main engine (SSME), whose development started in 1972. The higher pressure allows a higher nozzle area ratio (without flow separation at sea level), which gives further performance increases. Higher chamber pressures also allow the TC to be smaller, which makes it easier to place into a vehicle. There were some disadvantages, which prevented going to even higher values. Because heat transfer increases approximately linearly with the chamber pressure, cooling of TCs becomes much more difficult at higher pressures and the amount of gas flow or energy needed to drive the turbines increases. Also the engines become heavier.

C. So Many Liquid Propellants

An estimated 170 different liquid propellants have undergone laboratory evaluations and many also small TC tests.[8–10] This number



a) Thrust chamber with propellant tanks mounted on wall

b) Thrust chamber with two adjustment valves

c) Injector has one hole each for fuel and oxidizer

Fig. 1 Small thrust chambers that were some of the earliest designed, built and tested by Goddard; some with ceramic inserts (from Ref. 23).

Table 1 Historical Increases in Thrust Level of U.S. LPREs

Ground tests	No. of TCs	Thrust, lbf per TC (maximum)	Application	Developer
1923–1925	1	40–100	Experimental	Goddard
1927–1940	1	150–1000	Sounding rockets	Goddard
1942	1	1,500	Experimental	RMI
1943	3	2,000	PB2Y-3 JATO	Aerojet
1949	1	16,000	Hermes 3 missile	General Electric
1950	1	75,000	Redstone missile	Rocketdyne
1953	2	120,000	Navaho G-26 booster	Rocketdyne
1955	2	150,000	Atlas Missile booster	Rocketdyne
1960	2	210,000	Titan II booster	Aerojet
1963	5	1,500,000	F-1/Saturn V booster	Rocketdyne
1968	1	1,800,000	F-1A, Experimental	Rocketdyne

does not include any minor changes in the propellant formulation (such as small changes in NO_2 percentage in nitric acid) and in the propellant additives, such as gelling agents, corrosion inhibitors, or stabilizers.[1–3] More than 25 different propellant combinations have been flown in U.S. LPREs.

Early U.S. efforts were with liquid oxygen (LOX) gasoline (Goddard 1923), nitric acid/aniline [Guggenhiem Aeronautical Laboratory (GALCIT) and Aerojet 1940–1955], and LOX/75% alcohol [Reaction Motors, Inc. (RMI) 1942]. There was no discernible trend in the early propellant selections and each project team, company, or government agency picked the propellant they thought most suitable for their application. Therefore, U.S. LPREs were developed, produced, and flown with a variety of propellant combinations, such as those just mentioned. Flying LPREs have also used ammonia, 90% hydrogen peroxide, nitric acid, kerosene, mixtures of nitrogen oxide and nitric acid, 75 and 92% alcohol, pure hydrazine, mixtures of kerosene and unsymmetrical dimethylhydrazine (UDMH), monomethylhydrazine (MMH), gelled propellants, and others.

A yearning for higher energy denser propellants led to programs of investigating fluorine, chlorine, and/or fluorine containing compounds, boron materials, and several other exotic materials or mixtures in the 1960s and 1970s. The effort was extensive and included ground firing of TCs and three complete engines with some of these toxic, corrosive, and highly flammable high-energy propellants. None were selected for a flying engine. Fluorine as an oxidizer can give a higher specific impulse and higher average densities than LOX, but it and other high-energy chemicals were not considered to be practical. The decision to stop these high-energy propellant investigations was due in part to the potential drastic consequences of a major engine failure and/or a spill of propellants and their effect on people, equipment, or environment.

All of this effort did not lead to a universally acceptable single liquid propellant combination. All selections were a compromise between good qualities (high performance, high density, easy start, low cost, or stable, long time storage) and bad qualities (corrosive, flammable, toxic, prone to storage decay, high vapor pressures, or combustion instability) and depended on the application. After years of operational experience five propellant combinations seem to have emerged as being practical to use with current space applications, and they are listed in Table 2. Not mentioned in this table are propellants for applications that are today obsolete, such as jet-assisted take offs (JATOs), sounding rockets, aircraft propulsion, or tactical missiles with LPREs.

High specific impulse is very significant in space missions, where the cumulative mission flight velocity is high. Here even a small increase in specific impulse leads to major increases in payload or orbit height. This has led to the cryogenic combination of LOX/liquid hydrogen (LH_2) with specific impulse values between 410 and 467 s depending on the design. This practical propellant combination has been investigated experimentally since 1945. It has been preferred for upper stages of SLVs [such as the Pratt and Whitney Aircraft upper stage RL10 flying since 1963 with a specific impulse of 466 s and shown in Sec. V.B.5 or The Boeing Company, Rocket dyne Propulsion and Power (Rocketdyne) J-2 LPRE used in Saturn V shown in Fig. 2]. The extra performance usually overcomes the disadvantage of the low density of LH_2, which means very large insulated fuel tanks, extra tank weight, and more drag.

Table 2 Practical current U.S. propellants and their applications

LOX/kerosene (RP-1)	Some SLV booster stages
LOX/LH_2	Some SLV booster stages Most SLV upperstages
NTO/MMH	Attitude or reaction control systems (for orbit change, reentry, or space rendevous); Post boost control systems
Hydrazine monopropellant	Some reaction control systems
NTO/50% hydrazine + 50% UDMH	Older SLV and missiles

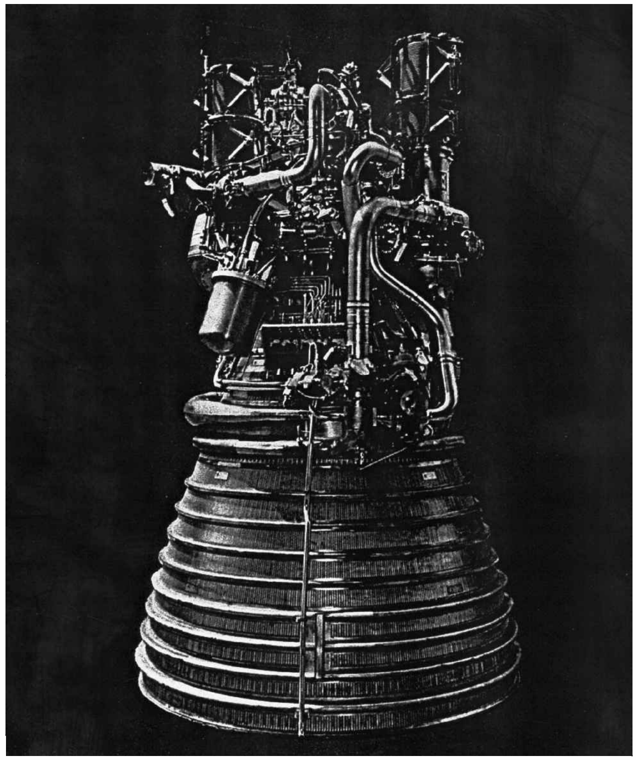


Fig. 2 Rocketdyne J-2 LPRE used LOX/LH_2 (courtesy of The Boeing Co., Rocketdyne Propulsion and Power).

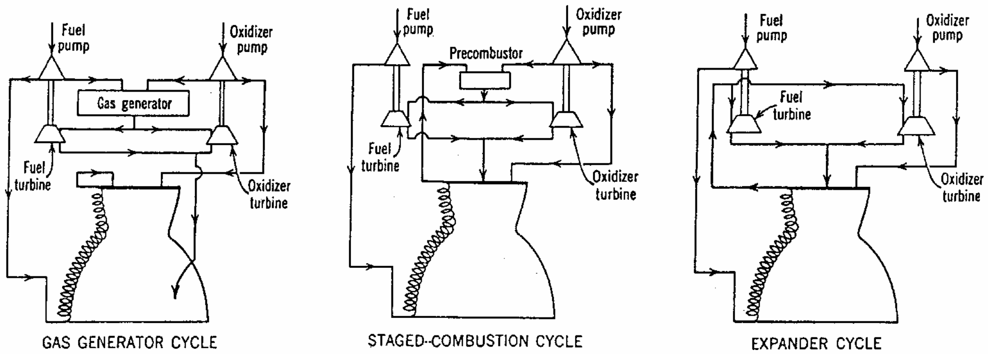
D. Large Liquid Propellant Rocket Engines

A large LPRE consists of one or more TCs, usually one or more turbopumps to feed the propellants from the propellant tanks to the TCs, a source of medium-hot gas to drive the turbine(s), a control system that will include commanding the start and shutoff, provisions for filling or draining propellants, various pipes and valves, and means for applying a small pressure to the propellant tanks. In addition, some engines have features to enable throttling or restart, thrust vector control, and self-monitoring of certain pressures, temperatures, or performance parameters. There are many ways in which these components have been designed to fit together and meet the requirements of different missions.[1–4,6,7] There are four types of feed systems to supply the propellants to the LPRE.

The pressurized gas feed system is the oldest (first tested in 1923). Here the propellant is expelled from its tanks by pressurized inert gas or by nonreactive gas created in a gas generator. Engines with this feed system will be shown subsequently. Most of the applications were with small LPREs using multiple TCs and with relatively small total impulses. However, there were some large LPREs using this system (for reasons of high reliability and fewer parts), such as the lunar takeoff engine or the Apollo Service Module engine.

The pumped feed systems are preferred with LPRE of high total impulse and large thrust. Three different engine cycles have been distinguished, and they are shown in Fig. 3. These cycles refer to the method of supplying medium-hot gas to one or more turbines, the flowpaths of the propellants and the method of handling and discharging the turbine exhaust gas.

The first engine with pump feed used the GG cycle. It was first ground tested by Goddard in 1938 and flown in 1940. Normally the same propellants used by the main LPRE are also burned (usually at a fuel-rich mixture) in a small secondary combustion device called the gas generator (GG) to generate gas at temperatures between 700 and 1650°F for driving the turbine. The turbine exhaust gas is discharged overboard at a lower velocity than the exhaust gas from the TC (shown in Figs. 3 and later). LPREs with this cycle usually have the lowest cost and often the lowest inert mass, but the performance is 2–7% lower than with the other two cycles.



a) b) c)

Fig. 3 Simplified diagrams of three common engine cycles. The spirals are a symbol for the hydraulic resistance of an axisymmetric cooling jacket, where heat is absorbed by the cooling fluid. From Ref. 2.

The expander cycle relies on the evaporation and heating of a cryogenic propellant. This propellant (usually LH_2) is heated and gasified in the TC's cooling jacket, and it is then expanded to drive the turbine. There is no GG. The turbine exhaust gas is injected into the combustion chamber, where it is burned efficiently with all of the oxidizer at high pressure and high temperature. It was first tested in 1960, was developed by Pratt and Whitney and is explained in Figs. 3 and Sec. V.B.5. The first flight was in 1963.

The staged combustion cycle has two combustion chambers in series. The first, called the preburner, burns all of the fuel and some of the oxidizer to give medium-hot gases at very high pressure, and these gases drive the turbine. The turbine exhaust goes through the injector into the main combustion chamber, where it is burned with the bulk of the oxidizer and expanded efficiently with the rest of the propellant; all of the reaction products are discharged through the nozzle. This cycle was first tested in the United States by Pratt and Whitney in the 1960s (with a simulated turbine and without turbopump) and with a complete engine by Rocketdyne around 1974. The first and only US engine with this cycle to fly (1981) was the SSME, described in Sec. V.B.3 and shown subsequently.

1. TC, Heart of the Large LPRE

The TC consists of a combustion chamber, where the propellants are burned and form hot gas, a supersonic nozzle, where the gases are expanded and accelerated, and an injector, where the propellants are introduced to the chamber, broken up into small droplets, and evenly distributed in flow and mixture ratio over the cross section of the chamber.[1–3] Because the gas temperature is usually twice the melting point of steel, the heat transfer and the cooling of the TC structure have historically been critical issues, and they are discussed next.

Originally the early investigators (1920s–1940s) used uncooled TCs. The duration was limited to a few seconds by the heat-absorbing capacity of the wall, before the wall material would locally melt or readily oxidize. Uncooled TCs are still used for experiments, short-duration applications, and with special materials as well as in some small TCs. Historically film cooling was the first method of extending the duration to more than a minute.[11] It was invented by Goddard around 1925. He called it "curtain cooling," where most of the fuel was injected through slots or openings located in a circular pattern at the outer diameter of the injector. It caused a reduction of performance (2–20%) due to incomplete mixing of the two propellants. Film cooling is still used today in some large TCs, but as a supplementary method to augment other cooling methods at critical locations of high heat transfer rates.

Regenerative cooling was first demonstrated in 1938 in the United States by James H. Wyld, an amateur rocketeer and one of the founders of RMI. His drawing is shown in Fig. 4. Here one of the propellants (usually the fuel) is circulated through the cooling jacket around the chamber and nozzle and absorbs heat. The name regenerative comes from the fact that heat absorbed by the coolant (in the cooling jacket) is not lost, but is regenerated and augments the heat of combustion.[1,2,12] Because the design did not have a feature to allow thermal axial growth, the inner wall yielded or wrinkled and may have cracked on cooldown after several firings. Improved versions of this thrust chamber design were used by RMI, Aerojet, and other U.S. contractors for about 10 years. Improvements included higher cooling velocities in the throat region, where the heat transfer was the highest, and better provisions, such as expansion joints, for compensating for the thermal growth of the hot inner wall.

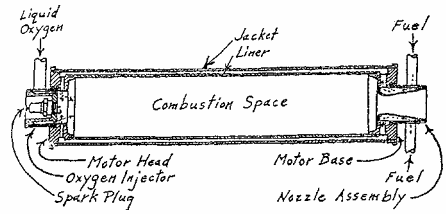


Fig. 4 Wyld's simplified sketch of first U.S. fully regeneratively cooled TC (1938), (from *Astronautics*, a Journal of the American Rocket Society, July 1938).

The thermal stresses in the inner wall can be greatly reduced if the inner wall is thin. This led to the tubular cooling jacket idea, an advanced form of regenerative cooling. Three U.S. companies RMI, Aerojet, and Rocketdyne implemented experimental TCs using a bundle of flattened and shaped tubes for the coolant passages during the late 1940s. RMI started studies in 1947, built an experimental tubular TC in the 1950s, and used a single-pass concept of nickel tubes in the TC of the XLR-99 (ground tested around 1959) for the X-15 research aircraft (flew 1960). In 1948 Aerojet built a small TC with shaped welded and soldered aluminum tubes, which may have been the very first one ever built. Aerojet first applied a tubular design in the Titan I TCs, which were designed in 1955. Rocketdyne tested small tubes around 1949, and in 1950 began to design large TCs that used a brazed together bundle of thin-walled stainless-steel tubes, which had been double tapered and shaped to the nozzle/chamber contour. Rocketdyne first tested it (about 1953) with the large TC (120,000-lbf thrust) of the G-26 Navaho engine, and it was flown first in 1956. Figure 5 shows such a TC and its flat plate impinging stream injector. The fuel flows from the inlet manifold through every other tube down to the fuel return manifold. It then flows up through alternate tubes (between the downflow tubes) and through openings into the injector. The three sketches on the right show how the tube cross sections change with chamber/nozzle diameter. The pressure forces were taken by steel hoops brazed to the outside of the tubes. Brazing was done by coating or supplying those surfaces, which are

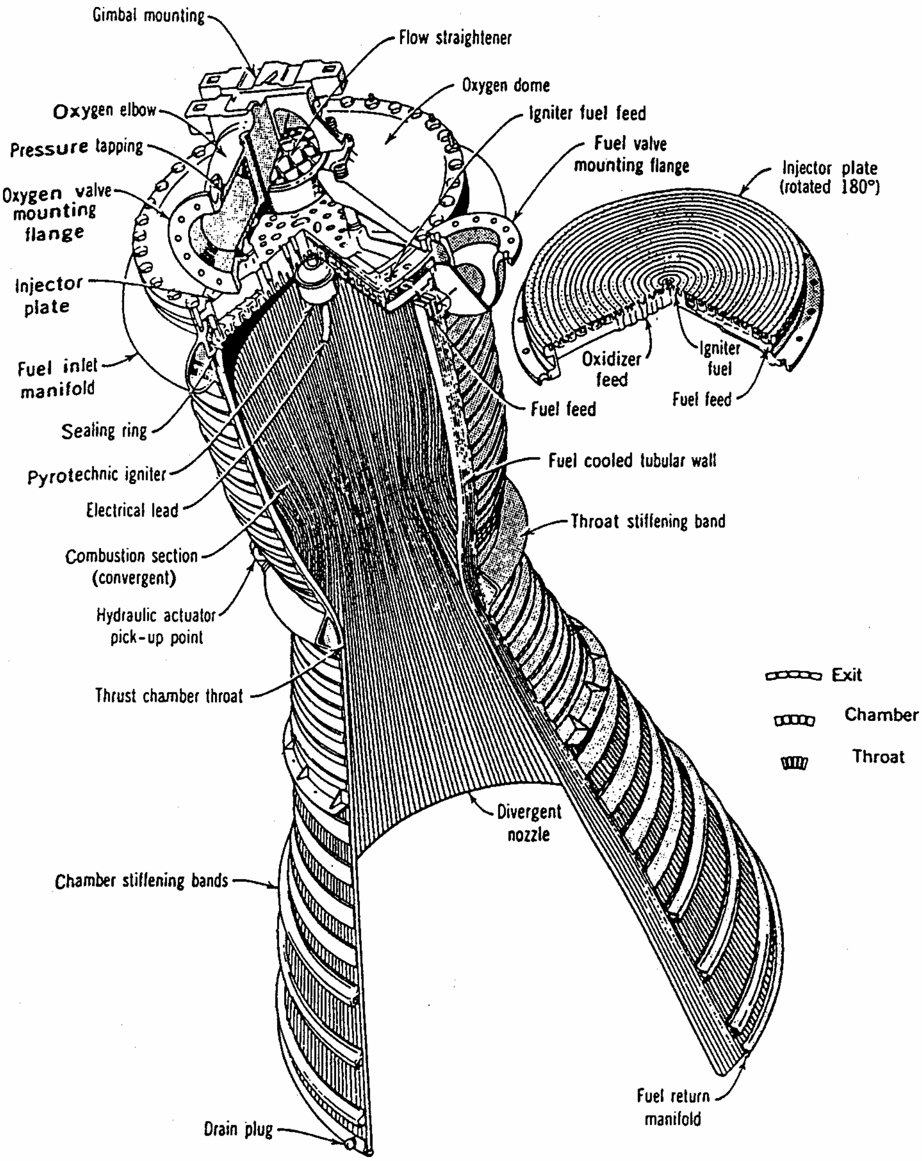


Fig. 5 Early version of large TC at 120,000-lb thrust with tubes, which are formed, shaped, and then brazed together (from Ref. 2).

to be joined, with brazing material, holding the tubes and hoops in a fixture and heating this assembly in a special reducing atmosphere furnace. Tubular cooling jackets have been used successfully in many large engines.

Another successful advancement of regenerative cooling for very high heat fluxes was developed in the 1960s. It uses straight milled channels (of variable width) machined into a forged or cast metal piece with the shape of a nozzle throat region. The outer wall can be brazed or electroformed to the milled center piece. The milled channel design was, for example, used by Rocketdyne in throat region of the TC of the SSME and by Aerojet in the TC of the orbital maneuver engine (6000 lbf) of the Space Shuttle Orbiter, which is shown in Fig. 6. Its injector has resonance cavities explained later.

Regenerative cooling is not suitable for large TCs that have deep throttling. At very low-thrust level (or low flow) the coolant would boil, and the mixture ratio would change. However, an ablative liner in the chamber and nozzle (without a cooling jacket) has been satisfactory. Such an ablative material absorbs heat by evaporation and chemical cracking/decomposition of the material; the resulting gases seep out of the material and form a relatively cool boundary layer, which gives a reduction of the heat flow. Later, an example is shown.

Several types of supersonic diverging nozzle exits have been used on TCs.[1,2,13–15] Some are compared in Fig. 7 together with their flow patterns at sea level and at high altitude. The earliest versions (1921–1936) by Goddard and other pioneers had a straight, long, conical diverging nozzle section with a small half-angle of 4 or 5 deg. (Fig. 1 and shown subsequently). Analysis done in the late 1930 at GALCIT and other organizations showed that a shorter nozzle with a half-angle around 15 deg was best. This nozzle exit cone angle was used between 1938 and 1957 in all types of rocket propulsion including solid propellant types as shown, for example, in Fig. 5. Between 1956 and 1958, several people in my section at Rocketdyne proposed and investigated a bell-shaped nozzle contour for the diverging nozzle section. Two are shown in Fig. 7. This contour was derived by analysis; its shape is close to a parabola.[1,2,13,14] It was validated by tests at Rocketdyne of two different opposing nozzles on a pendulum in 1956/1959 and by full-scale firing tests. The bell shape gives a little more specific impulse (reduces divergence losses) for the same nozzle length as an equivalent 15-deg cone. Alternatively, it can be made shorter and still have good performance. This contour has been used since about 1960 in all rocket propulsion nozzles, large or small, liquid or solid propellant. Some of the large engines (Thor or Atlas) that flew originally with a straight 15-deg cone nozzle were then modified to the new bell-shaped contour. The lower

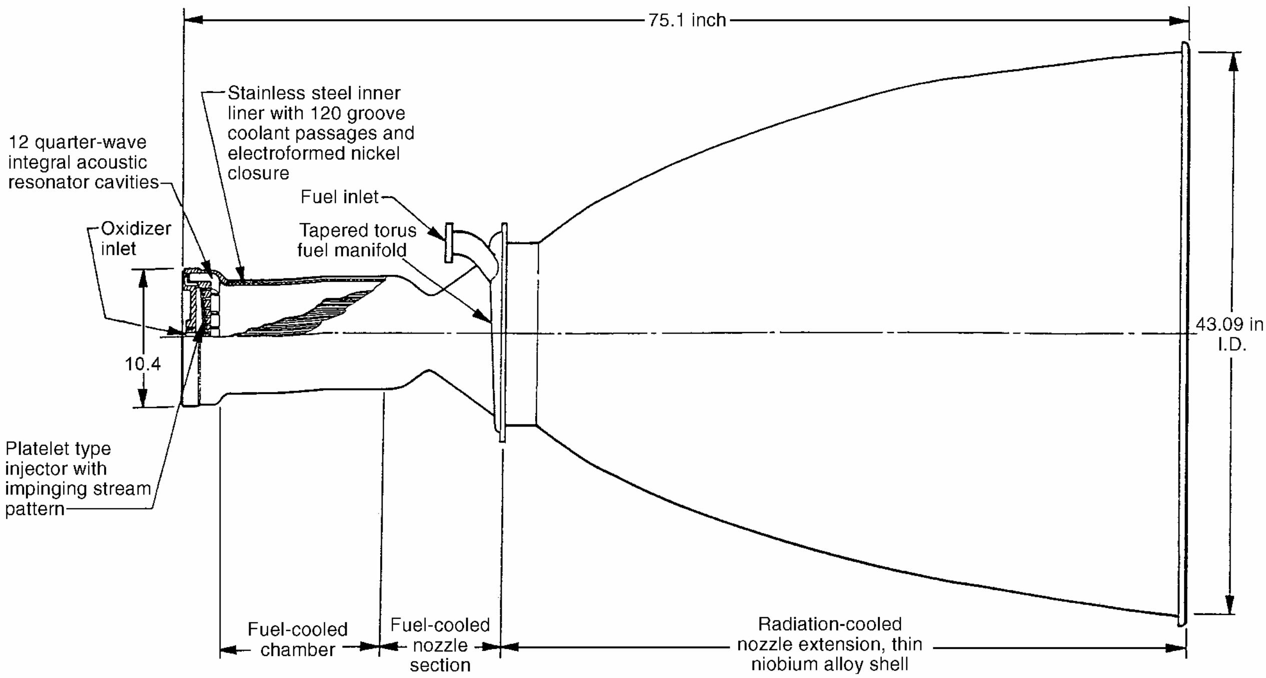


Fig. 6 Simplified partial section of one of the two TCs of the orbiting maneuvering system (OMS) used on the space shuttle vehicle (courtesy Aerojet).

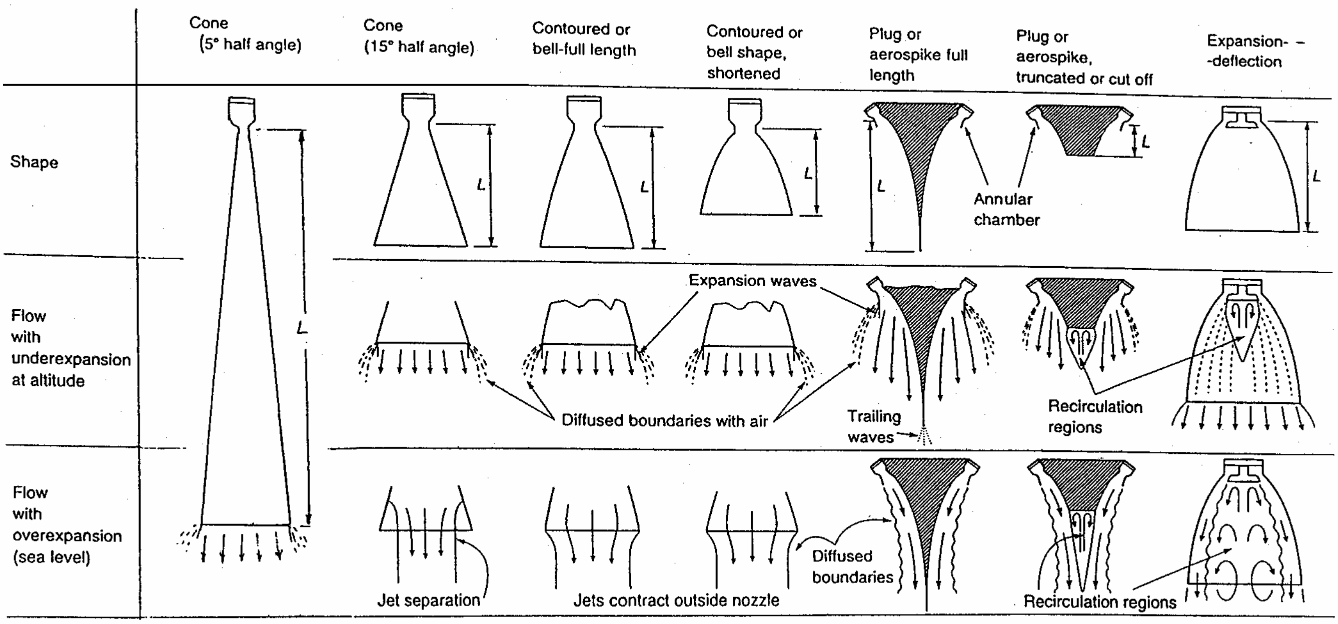


Fig. 7 Several supersonic nozzle exit sections with different lengths and flow patterns (partly from Ref. 2).

portion of the diverging nozzle exit segment has a relatively low heat transfer and does not require a regenerative cooling jacket. Instead, three lower cost single-wall uncooled designs for nozzle extensions have been used and flown. An ablative material was used by Aerojet in the 1950s for upper-stage engines and in 1962 for the Titan sustainer engine and in 1998 on the Rocketdyne RS-68 engine for an SLV. This has usually been the lowest cost approach. Alternatively, a radiation-cooled thin wall made of niobium or carbon fiber material have been used effectively for various upper-stage engines since about 1960 (Fig. 6). A special version is the extendible nozzle, which is stored around an upper stage engine during ascent through the atmosphere and then extended or moved into position at altitude before engine start. The first extendible nozzle of a LPRE was ground tested at Pratt and Whitney in the 1960s, and the first flight was in 2000 on the Pratt and Whitney RL 10 B2 LPRE, shown later; its movable nozzle exit skirt was made from carbon fibers with a carbon filler. The third approach for a nozzle exit is called dump cooling, and it has a single wall with an inside boundary layer of medium-hot turbine exhaust gas (700–900°F), which is dumped through a manifold and slots into this lower portion of the diverging nozzle. It was used on the Rocketdyne F-1 engine (designed 1959) and is shown later.

Unique special types of nozzles were developed, namely, the aerospike nozzle and the expansion/deflection (E/D) nozzle, shown in Fig. 7.[2] Their main merit is to expand the exhaust gases at optimum value at all altitudes. (The effective area ratio changes with

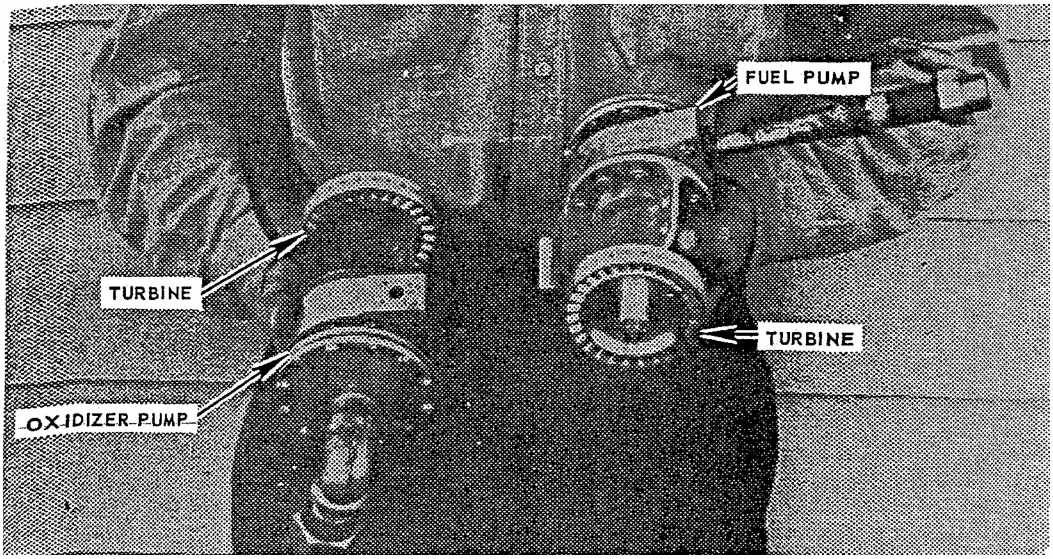


Fig. 8 Technician is holding two of the turbopumps developed by Goddard (from Ref. 24).

altitude.) This allows a slightly higher time-averaged specific impulse or thrust during flight, when compared to a fixed conventional nozzle. The engine length can be very short with a cutoff spike or a shortened E/D nozzle, saving some inert vehicle mass and drag. Five different experimental aerospike engines have been ground tested by Rocketdyne (using LOX/LH2) with thrusts between 50,000 and 400,000 lbf including a linear aerospike version.[15] Rocketdyne also ground tested two versions of an E/D engines [using nitrogen tetroxide (NTO)/Aerozine 50] in the early 1960s at 50,000- and 10,000-lbf levels. None of these projects with special nozzles have been continued.

2. Turbopumps[1,2,16,17]

The turbopump (TP) is a key component for a pump-fed LPRE and an engineering intensive, high-precision, high-speed piece of rotating machinery. The first turbine-driven centrifugal pumps were tested by Goddard in 1934 (Sec. IV.A). Two of his TPs are shown in Fig. 8. They had ball bearings, shrouded turbine blades, and pump outlet diffusers. (One is shown extending from the pump in the upper right.) A small arc of the blades of the two turbines were immersed in the exhaust gas of the main nozzle (over 5000°F), and he experienced frequent turbine failures. Therefore, Goddard developed a GG (1938) that had a lower gas temperature. The first version had three propellants (LOX, gasoline, and water as a diluting/cooling fluid). The TPs were small (low flow) and inefficient. The first LPRE with a TP and a GG was tested by Goddard in 1939 and flown in a sounding rocket in 1940. In 1942 he used a fuel-rich GG without water.

The early TPs for JATO and aircraft superperformance (1943–1950) had a turbine and both the propellant pumps on the same single shaft. The first large U.S. TP (Redstone engine designed 1949) had two in-line shafts, a coupling, and an aluminum turbine because this was a proven German technology on the V-2 engine. The GG at that time used monopropellant 80% hydrogen peroxide with gas temperatures of about 700°F.

Historically the large U.S. TPs of the 1950s and 1960s used a gear case that allowed a turbine to rotate at a higher speed than one or both of the propellant pumps because this allowed better turbine and pump efficiencies. This resulted in a lower GG flow and a slightly better engine performance than a single-shaft TP. Figure 9 shows a geared turbopump as used with the Atlas/Thor/ H-1 (Saturn I SLV) family of booster LPREs. Initially oil was supplied from a small oil pump to lubricate and cool the gears and the bearings, but the oil was then replaced by kerosene fuel. A gear case was used for the Titan family of LPRES to drive the two pumps at different speeds.

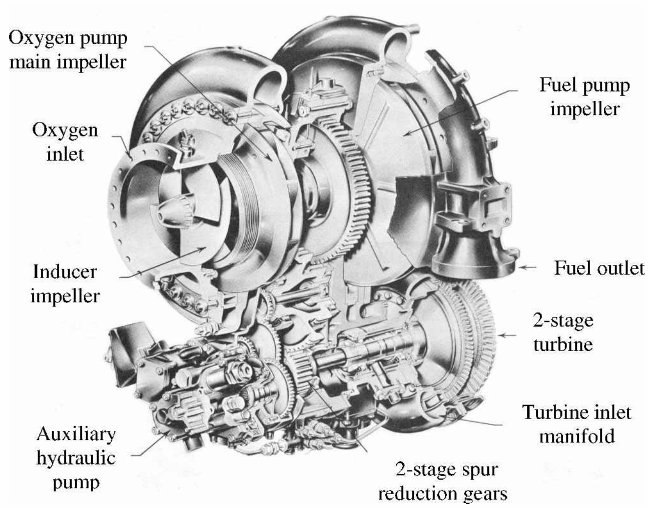


Fig. 9 Sectioned view of the type of turbopump with a gear case used in the Rocketdyne Thor, Jupiter, Atlas booster, and H-1 engines (from Ref. 2).

The steel alloy turbine was usually driven by a GG, which used the same propellants as the main TC (but usually at a fuel rich mixture ratio), resulting in a gas between 1300 and 1650°F. A gear case was also used on the Pratt and Whitney family of RL-10 engines to allow the oxidizer pump to rotate slower than the fuel pump.

An inducer impeller ahead of and on the same shaft as the main pump impeller was used during World War II in the TP of the German Walter aircraft rocket engine. It provided for better cavitation resistance of the main pump impellers, and it allowed the tank pressure to be lowered, resulting in a weight reduction of the propellant tank. It can also allow the main pump to run at a higher speed, which in turn allows a reduction of inert TP weight. The United States was late in adopting this clever innovation. Several of the U.S. LPRE TP that were already in production were changed in the 1950s to use redesigned pumps, which included inducer impellers. This happened to the Thor and Atlas engines. An inducer can be seen in Fig. 9.

Goddard's concept of separate TP assemblies for the fuel and the oxidizer pump was revived several decades later for propellant combinations where the fuel and the oxidizer have very different densities. It was used with LOX/LH_2 engines, such as the J-2 (Rocketdyne

1960 design), the SSME (1972 design), and the RS-68 (1997 design), in part because it gives a smaller and lighter design and if avoids the complexity of a gear case. Major design advanced were made in TPs in recent years.[2,16,18]

E. Small Liquid Propellant Rocket Engines

These small engines with multiple thrust chambers (often called thrusters) have a very important role for the vehicles' flight control.[2,3] They were and are still used for trajectory changes, attitude control (pitch, yaw, and roll), deorbit maneuvers, station keeping, rendezvous maneuvers, or fly wheel desaturation, or for settling liquid propellants in a zero-gravity flight before main engine restart. Although a large LPRE is usually assembled and delivered in a single package, a small LPRE, with multiple TCs (placed in several different locations in a vehicle), is normally delivered in several pieces. Thrust levels are typically between 1 and 100 lbf, but there were some that were larger (1000 lbf or more). They generally use storable propellants and a pressurized gas feed system. There were several different ways to obtain a small thrust, and they are explained briefly in their approximate historical sequence. More details are in Sec. V.

The first solution for attitude control was the orderly expulsion of an inert cold gas, such as air or nitrogen, which was stored at high pressure and exhausted through simple valves, regulators, and multiple nozzles. Cold gas for attitude control was used starting in the late 1940s and continuing sporadically until about 1980. These systems were simple, low cost, reliable, and ran at ambient temperatures. However the specific impulse was low (around 70 s) and the systems were heavy, adding to the inert mass of the vehicle. They were used on many early satellites and for roll control on some upper stages. Several companies have built and flown cold-gas thrusters.

In the 1947–1966 period, small monopropellant hydrogen peroxide thrusters became popular. The relatively low gas temperatures (600–1300°F depending on the peroxide concentration) allowed the use of simple single-wall low-carbon-steel construction and avoided the need for a cooling jacket. It was usually decomposed by a silver screen catalyst and used a pressurized gas feed system. Thrust levels were between 0.1 and 100 lbf. The two suppliers of H_2O_2 thrusters were Walter Kidde & Company (out of business) and Bell Aircraft, which today is Atlantic Research Corporation (ARC). They were used extensively, for example, on the Mercury manned space capsule, and more than 1000 thrusters were flown. Next came hydrazine monopropellant thrusters (1958 to present) with pebble-type catalysts, again a pressurized feed system, and uncooled alloy steel TC walls. They offered more than a 50% improvement in performance over the peroxide. Hydrazine thrusters were made possible by the development of a suitable catalyst and by making ultrapure hydrazine, which did not poison the catalyst. The example in Fig. 10 shows the nozzle exit at the lower right, and the radiation shield hides the TC and the catalyst bed. Some of these thrusters could demonstrate more than 100,000 start/stop cycles over a typical flight mission period. Suppliers were the Rocket Research Corporation (today part of Aerojet), TRW (today part of Northrop Grumman Corporation) W. Kidde (no longer in business), and Hamilton Standard (today the product is sold by ARC). The advantage of these monopropellants are the inherent simplicity of the system (good reliability), high propellant densities (small propellant tanks), and clean exhausts that will not fog up sensitive surfaces (window, mirrors, solar cells). Its principal demerits are the lower performance compared to bipropellants, resulting in a heavier system, and hydrazine's high freezing point (34°F), requiring heating of all components. Multithruster hydrazine monopropellant systems have been used on hundreds of spacecraft or upper flight vehicle stages and are still popular today.

The low-thrust bipropellant thrusters also started in the late 1950s and are still used today on many upper stages, spacecraft, and satellites. Bipropellants give higher specific impulses (250–320 s) than hydrazine monopropellant (210–250 s). Initial propellants were nitric acid (and later NTO) as oxidizer and hydrazine as a fuel. Around 1963, the hydrazine fuel was replaced by Aerozine 50 (a mix of 50% hydrazine and 50% UDMH) and a few years later by MMH. These fuels had a lower freezing point, but slightly lower performance. However, these fuels can, under certain conditions, cause thin undesirable deposits of solid particles in the combustion products on sensitive vehicle surfaces (windows, solar cells). These deposits have prevented the use of MMH and UDMH in certain satellite applications.

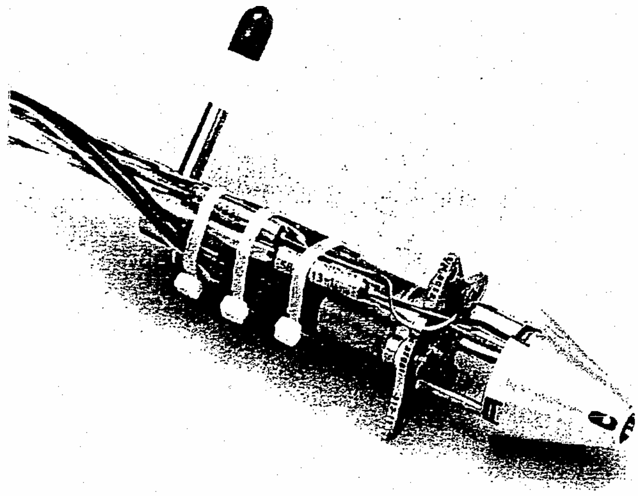


Fig. 10 Monopropellant hydrazine 0.1-lbf TC assembly with valve to the left of the mounting flange and an electric heater (courtesy, Aerojet).

With the high gas temperatures, some form of cooling of the TC walls is needed. Regenerative cooling can no longer be used because the heat capacity of the low fuel cooling flow would not be adequate to absorb all of the heat rejected by the hot gas to the inner walls. The cooling fuel would boil, causing a drastic change in mixture ratio. The thrusters often use some film cooling, but by ityself, this is not sufficient. One good solution came with a small experimental radiation-cooled thrusters, which were developed in 1958 and 1959 by Marquardt Corporation, one of the predecessors of Aerojet's Redmond Center. Its thrust was 25 lbf, and it used NTO/hydrazine and a molybdenum chamber nozzle with an inside coating of molybdenum disilicide for oxidation protection. Molybdenum was soon replaced by niobium (also called columbium), which is lighter and easier to fabricate. It has a niobium disilicide inner coating for oxidation protection. A later design Marquardt's 100-lbf thruster shown in Fig. 11 was first used for the auxiliary propulsion on the Saturn S IV B upper stage (later in other applications), and it flew for the first time in 1965. Most of these radiation-cooled bipropellant thrusters with niobium chambers have been produced by the predecessor of Aerojet's Redmond Center, Northrop Grumman (NG) (formerly TRW), or ARC.

Ablative liners were also an early solution for small thrusters with many starts. The ablative liner is made of glass, Kevlar®, or carbon fibers woven in a fiber cloth in a plastic matrix, and the cloth is laid in layers before heating and compressing the material and surrounding it by a metal shell. Sometimes a ceramic sleeve or a graphite nozzle insert is used to minimize erosion. Small ablative type thrusters were developed mostly by NG (TRW) and Rocketdyne between 1960 and 1973. Figure 12 shows a 25-lbf thruster (left) and a 100-lbf thruster used on the Gemini manned capsule, its maneuvering system module, or the Apollo command module. They were gradually replaced by radiation-cooled metal thrusters because ablatives were relatively heavy and had dirty exhausts, which have caused unwanted deposits on mirrors or solar cells.

The third type of bipropellant thruster called Interregen was developed by Rocketdyne in the late 1960s. It uses a relatively thick wall of beryllium (a low-density, high-conductivity metal) for the chamber nozzle material. The beryllium conducts the heat away from the hot-throat region to a film-cooled region in the chamber. It has flown in postboost control propulsion systems.

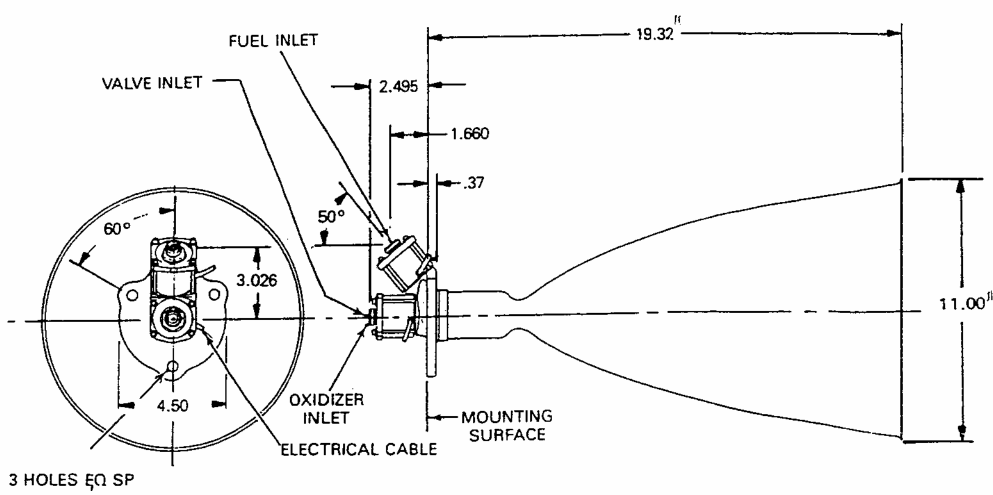


Fig. 11 Two views of an R-4D radiation-cooled bipropellant TC (100-lbf thrust in vacuum) with integral valve (courtesy, Aerojet).

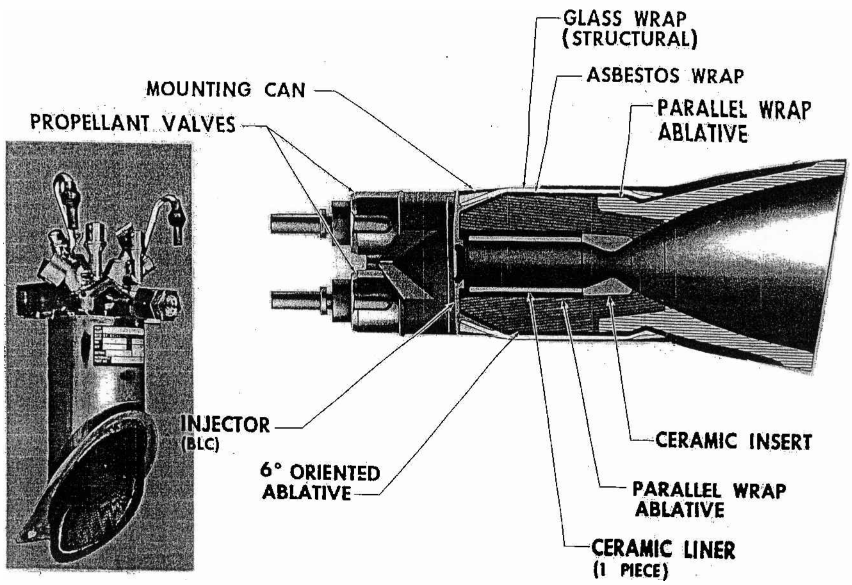


Fig. 12 Section and external view of two ablative thruster of the Gemini manned spacecraft (courtesy, Rocketdyne).

F. Nemesis of Combustion Vibrations

Combustion instabilities have tormented LPRE developers for perhaps 30 years beginning about 1950 (Refs. 2 and 19). They have caused sudden and unexpected failures of TCs and, thus, of LPREs. Therefore, all LPREs must be designed and proven to be free of such instabilities. Three types of vibrations have been identified. The first is a low-frequency chugging (10–400 cps) or interaction of the liquid propellant feed system with the oscillating gas in the combustion chamber. This includes oscillations of propellants in long feed pipes, often called POGO instability. Remedies included modifications in the feed system, increasing the injection pressure drop, and for POGO instability the addition of damping accumulators in the pipe lines.[20]

The second type of instability is characterized by intermediate-frequency oscillations (400–1500 cps), often called buzzing, associated with mechanical vibrations and resonances of pieces of the engine structures, injector manifolds, pipes, and their interaction with gross combustion behavior, such as turbulence. Frequencies depend on the size and structural resonances. Changes in the chamber geometry, injector configuration, and in the structural stiffness of the affected components became effective countermeasures. By about 1956, the understanding of the first two vibration types was good enough to diagnose incidents and take effective remedial actions.

The last type of combustion vibration occurs at high frequency (above 1500 cps). It has since been linked to the burning process itself and to pressure waves and chamber acoustic resonances. When it did occur, it would cause high-frequency large-amplitude chamber pressure oscillations, cause sudden increases in heat transfer or the forces exerted by the TC, and lead to a structural or heat transfer failure of the TC in less than a second of time. Often this instability would occur only in one test run out of perhaps 100 or 1000 firing tests. Therefore, the only method for assuring a stable design in these early days was to run hundreds of static tests on the same identical engine design without a single incident of combustion instability.

A rating technique was developed between 1957 and 1967. Artificial disturbances are introduced into the combustion chamber (by setting off specific directional explosive charges) to induce a pressure surge and trigger high-frequency vibrations.[21] Accurate high-frequency chamber pressure measurements can then determine if there is enough energy absorption for the magnitude of the pressure oscillations to be damped and diminish rapidly. The recovery time (in milliseconds) between the artificial pressure surge and the resumption of steady combustion is a rough measure of the inherent stability. This rating technique reduced the number of static firing tests needed to prove stability.

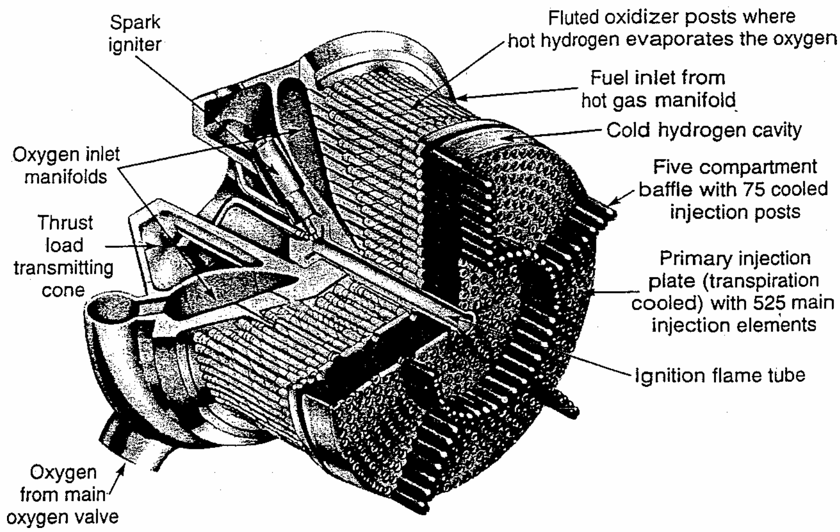


Fig. 13 Main injector assembly of the SSME showing a six compartment acoustic baffle with protruding coaxial injector elements (copied from Ref. 2).

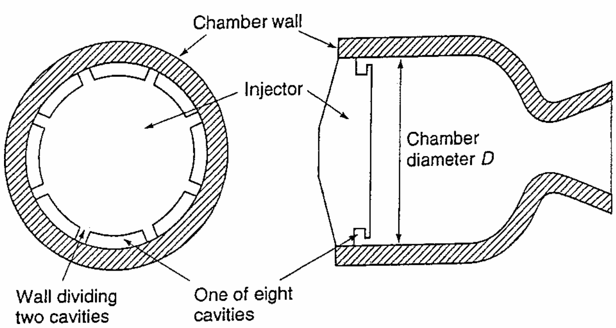


Fig. 14 Simple diagram of acoustic energy absorbing cavities at the periphery of an injector; cavity entrance restriction is a slot in the shape of a section of a circular arc or sometimes a hole. Details of chamber or injector design are not shown (from Ref. 2).

Several remedies or fixes have been used for avoiding high-frequency combustion instabilities. The first and earliest method was to make empirical and somewhat arbitrary changes in the injection design, propellant, chamber volume, or chamber shape. For example, in the Bomarc missile engine, the change from kerosene to a UDMH/kerosene fuel mixture solved a high-frequency vibration problem in 1952. This method could not be scaled up in thrust, was not fully reliable, and required a lot of expensive testing and time. Next came baffles, which were introduced in about 1958 and proved to be a reliable remedy for the destructive radial and circumferential vibration modes. These baffles were built into the injector, as seen in Fig. 13, and cooled baffles were put into most large U.S. engines until about the late 1970s. They were retrofitted into the engines for the Atlas, Thor, and Titan II and designed into Saturn (H-1, F-1), Titan III, Titan IV, and the initial version of the SSME. The third remedy uses acoustic resonance cavities (to absorb gas vibration energy), was designed into LPREs beginning about 1963, and was reliable in preventing many high-frequency oscillations. This is shown in Figs. 6 and 14. They are designed or tuned for a specific vibration frequency, usually the estimated resonance frequency. Today these resonance frequencies can be predicted for different modes of acoustic oscillations. In the United States a number of production injectors, both for high and low thrust, were then redesigned to include resonance cavities. Some injector designs had both baffles and resonance cavities, such as the injector for the Apollo lunar ascent engine. Resonant cavities have been more effective than the baffles, and in some cases, such as on the SSME, it was possible to later remove the baffles. It was also learned that certain injector designs and propellants give stable combustion without baffles or cavities. Subtle small changes in the geometry of the injection holes, spray injection elements, or their distribution over the face of the injector have been effective. Each organization uses its own experience base and has its own favorite injector designs. In the past 25 years, the incidence of combustion vibrations during engine development has been greatly reduced. The expensive, extensive engine static firings of the past (to get meaningful statistical data) have been replaced by analyses of vibration behavior and a few directed bomb tests for demonstrating stability and for rating the recovery time period.

IV. Early Efforts: 1923–1943

A. Pioneering Work of Robert H. Goddard

This American physics professor, Robert Hutchinson Goddard of Clark University in Worcester, Massachussetts, was a very creative researcher and the important developer of first LPREs in the world.[22] Born in 1882, he died in 1945. He was the first to develop, build, and test key LPRE components and static test or fire small LPREs with gas pressurized feed systems between 1921 and 1925. Goddard was the first to launch a sounding rocket with a simple LPRE on 16 March 1926 in Auburn, Massachussetts. It is described later. His significant inventions and contributions[23,24] to LPREs are summarized as follows.

1) One of his patents gave the first plausible drawing and description of a LPRE with a pump feed system. The drawing is shown in Fig. 15 (1914).

2) Between 1921 and 1925, he designed, developed, and ground tested the first liquid propellant thrust chambers. Initial thrust levels were between 40 to 100 lbf for a 1.0 in.-diam chamber (Fig. 1). Later he built 5-, 6-, and 10-in.-diam chambers with up to 1000-lbf thrust. He used LOX as the oxidizer, initially with ether as a fuel. He then tested gasoline, alcohol and kerosene with LOX. Early nozzles were long and had an exit cone half-angle of 4–5 deg; he used larger nozzle half angle after 1941.

3) From 1924 to 1930, using a "curtain flow," a type of film cooling, he achieved firing duration of more than 1 min without burnout. In 1943, he adopted regenerative cooling, which was invented by others.

4) He used black powder igniter (1923) and then pioneered pyrotechnic igniter with solid propellant (1927) and spark plug igniter (1937) for restart.

5) From 1921 to 1923 he developed the first propellant feed system using high-pressure gas to expel liquid propellants from their tanks and into a thrust chamber. He later included valves or orifices to adjust line pressure drops for best mixture ratio and flows.

6) The first flight of an LPRE in the world on 16 March 1926 was accomplished by him (described later).

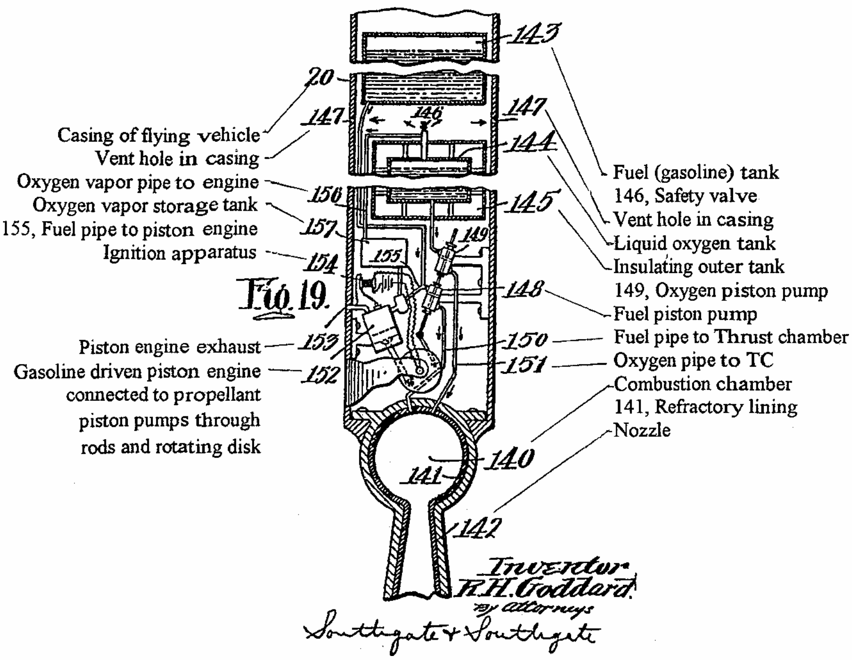


Fig. 15 Goddard's U.S. patent 1,103,503 issued 14 July 1914 showing a LPRE with a pumped feed system.

7) Between 1925 and 1935, he built and flew first lightweight propellant tanks and high-pressure gas tanks using welded steel, often aluminum or sometimes brass sheets.

8) In 1930, he was first to use baffles to suppress sloshing in liquid propellants tanks during flight to prevent excursions of the center of gravity of the vehicle or to keep gas from entering the propellant pipes. He was first to reinforce tanks with wound high strength wire to reduce tank weight (1937).

9) In 1924, he designed and tested several types of propellant pumps: At first, he tried and abandoned piston, vane and gear pumps. By 1933, small centrifugal pumps worked. He ground-tested the first TP (1934) with a separate turbine for each propellant pump (Fig. 8). Later some of his pump bearings were running in LOX, which had not been done before.

10) He did static firing of LPRE with the first TP in 1939. In 1939/1940, he used a GG to generate "warm gas" that would not melt the turbine buckets. In 1938, he developed and thus invented the first U.S. GG. In 1940, he launched the first flight of a LPRE with a TP feed system and GG.

11) In the 1940s, he conceived and tested a novel ceramic-lined precombustion/ignition chamber (attached to main chamber) suitable for restart.

12) During 1925 to 1941, he was first to develop and fly several lightweight valves, including safety valves, propellant valves, check valves, shutoff valves, and throttling valves, and several lightweight gas pressure regulators using bellows and springs.

13) In 1937, he invented and flew vehicles with a "movable tail," a type of gimbal for thrust vector control, actuated by four sets of dual pneumatic bellows. The TC was mounted in the tail and had flexible feed lines.

14) In 1924, he developed and later improved the first control system for starting. First controls were manual (strings pulled by operators at control station), then mechanical sequencers and a clock as controller. He then developed (1927) pneumatic valve actuation and later (1933) a pneumatic LPRE control.

15) In 1942, his was the first JATO of a flying boat on water with reusable LPRE (800-lbf thrust).

16) The first US variable thrust LPRE was developed and tested by Goddard, but not flown (1943).

17) Between 1923 and 1943, he developed and improved techniques for photographing gauges, indicator lights, and clock, thus, recording ground-test data. He developed the first flight recorder.

Goddard launched sounding rockets with LPREs initially at Auburn, Massachusetts (1926–1930) and later at Roswell, New Mexico (1933–1938 and again 1939–1941). Altogether through 1941, he conducted hundreds of component tests and static firing tests of TCs, over 100 static (bolted down) tests of an engine mounted in a vehicle, attempted about 50 flight tests, and of these, 31 resulted in flights.

His early work with solid propellant rocket motors (1914–1920) was abandoned in favor of liquid propellant engines because his theoretical analyses showed him that liquids would give more energy per unit propellant mass. All of the engines and sounding rocket vehicles were built and assembled in his own shop. Every one of these flight configurations had some new features, improvements, or design changes, and he never flew exactly the same vehicle twice.

The vehicle for his historic first flight on 16 March 1926 is shown schematically in Figure 16. It rose 41 ft above the launch stand and flew a distance of 185 ft in about 2.5 s. It had the thrust chamber at the front of the vehicle and the long propellant tanks (LOX and gasoline) at the aft end. The propellant feed lines also served as the structure to tie the key components together. He used a crude simple cone as a heat shield to protect the tanks from being overheated by the rocket exhaust plume. The black powder igniter was in a tube on top of the TC. Ignition of the powder was achieved by a flame from some broken off match heads inside a copper tube, which in turn was heated externally by some burning cotton, which is not shown. The two propellant tanks were both pressurized by gaseous oxygen, which was evaporated from the oxygen tank. The line pressure drops and the mixture ratio were preset by two small needle valves near the TC. The lower part of the nozzle had burned off during the last part of the first flight.

Goddard filed many patents, 48 were issued during his life time, 35 more for which he had applied, but were issued after his death in 1945, and 131 more filed by Mrs. Goddard as his executrix after his death, based on his notes, sketches and photographs. For example, in 1914 he obtained a patent on a two-stage vehicle. In 1960, the U.S. Government bought the rights from his widow to use 200 of these patents for $1 million dollars, and this payment was shared by her with the Guggenheim Foundation, which had supported most of Goddards work between 1930 and 1941.

From 1942 to 1945, Goddard worked with the U.S. Navy Bureau of Aeronautics at Annapolis, Maryland. There he helped to develop a LPRE for JATO, which was successfully flight tested, and the first US variable thrust LPRE, which was quite complex; it was later

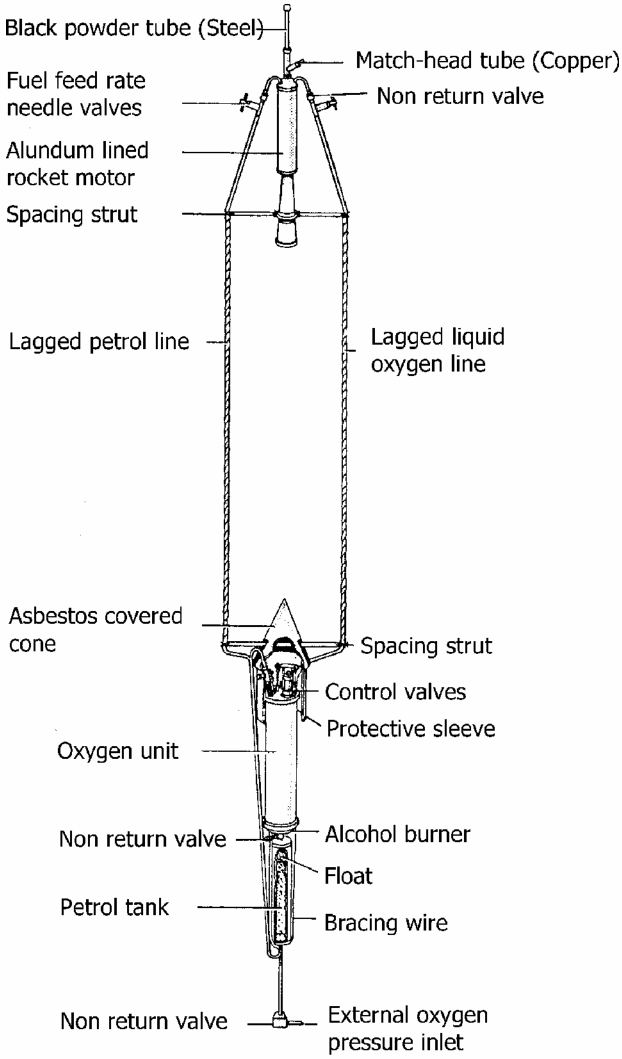


Fig. 16 Diagram of Goddard's historic first flying rocket vehicle with a LPRE, launched on 26 March 1926 in Auburn, Masschussetts (from *Journal of the British Interplanetary Society*, Vol. 40, 1987, p. 307).

was fully developed by a contractor. He also became a consultant to the Curtiss–Wright Corporation at Caldwell, New Jersey, and RMI originally at Pompton Plains, New Jersey.

Goddard was very reluctant to publish or disclose his concepts, designs, test data, or flight results to other people. His 1919 paper on "A Method of Reaching Extreme Altitudes" brought him some fame, but it did not describe his ideas about LPREs.[25] Although he had correspondence with many people, including other noted rocket experts, such as Hermann Oberth of Germany, he would not divulge very much useful information. He was concerned about others using his concepts before they were fully proven and also about a disclosure of his ideas before the issue of patents. He published very little about his work on LPREs during his lifetime. What he did publish had limited distribution. His collected works (including diaries, photographs, sketches, and data) were more revealing and were published by his widow 25 years after his death (1970). By that time, the U.S. contractors had reinvented or developed on their own much of what Goddard had previously achieved. It is an ironic twist of history that the LPREs, which were developed by General Electric, Rocketdyne, or Aerojet, were designed and produced in the 1940s and 1950s without the benefit of the pioneering work done by Goddard. He had relatively little impact on the U.S. LPRE developments. We can only speculate what would have happened, if Goddard would have allowed access to his development results and know-how, while he was still alive in the early 1940s, while the LPRE industry was in its infancy.

For his historical and outstanding accomplishments Goddard has received many honors, unfortunately only posthumously. For example, the NASA Goddard Space Flight Center, at least six schools, some university professorships, several society awards and lectures were named after him.

B. Amateur Rocket Societies

A series of voluntary amateur technical groups, intrigued by the prospects of space flight, sprang up in the United States during the 1930s. They debated space flight issues, published articles and/or news bulletins, conducted some ground tests of LPREs, and flew a few simple rockets.[26] Their biggest contributions were the popularizing of space travel and rocketry (lots of free publicity) and the attracting, educating, and identifying of technical personnel that later became engaged in the emerging field of rocketry.

The American Interplanetary Society (AIS) was founded in March 1930 in New York. It soon grew in membership and started to publish its own journal. The society designed and static tested TCs and LPREs at its proving grounds.[27] It flew simple and crude rocket vehicles with pressurized gas feed systems. They had many TC burnouts and flight failures. In 1934, the AIS changed its name to the American Rocket Society (ARS) because they wanted to get away from the word interplanetary, which the public and the press viewed with suspicion and considered a fantasy. ARS had about 15,000 members in about 20 chapters 15 years after it started. I had the privilege to serve as a director and then as president of the ARS at a time when the first actual space flights occurred. Society member Wyld designed and successfully tested (first time in the US in 1938) a fully regenerative fuel-cooled thrust chamber as shown in Fig. 4. This was hailed as a major step forward in the technology because it allowed prolonged rocket operations without burnout failure.

In 1941 the experimental work of the ARS was discontinued, in part because of the good and well-funded LPRE work being performed elsewhere. The society concentrated on publishing refereed professional papers and holding technical meetings on the subjects of propulsion, space flight and rocket vehicles. About a dozen other local amateur societies were founded in different parts of the United States. Other LPRE efforts, including a small U.S. Navy project,[28,29] also sprang up. In February 1963 the ARS merged with the Institute of Aeronautical Sciences to form AIAA. Today this is a respected professional organization, is still very much concerned about LPREs, but rocket propulsion technology is now only one of many fields of interest to AIAA.

C. GALCIT: 1935–1943

This laboratory, originally on the campus of the California Institute of Technology, was perhaps the first in the United States to undertake theoretical and experimental work in LPREs.[30] The Chairman of the Aeronautical Engineering Department and the head of GALCIT project was Theodor von Kármán, a reknowned aerodynamicist.[31] This laboratory performed laboratory tests of different propellants, designed and tested small TCs in their own off-campus test facility (beginning 1936), and was the first to achieve hypergolic ignition using nitric acid and aniline as propellants (1940). Different propellants and thrust chambers with thrusts up to 1000 lb were investigated. In 1937, nitric acid was selected as a good potential storable oxidizer. In 1939 GALCIT improved this nitric acid oxidizer by dissolving up to 30% nitrogen dioxide (a red colored gas, that came out of solution and evaporated as reddish clouds), and this was henceforth known as red fuming nitric acid (RFNA). It had greater density, slightly higher performance, and better ignition properties than nitric acid without the dissolved gas.

They published some historic analysis on rocket propulsion and were the first to use the concept of a thrust coefficient. They built and flew the first U.S. JATO[32] in 1942 and developed the first LPRE for a sounding rocket. GALCIT was the progenitor of the Jet Propulsion Laboratory, which today is still administered by California Institute of Technology for NASA. Its first Director, Frank J. Malina was a key GALCIT member. Several GALCIT members started the Aerojet Engineering Company in 1942. Its first chairman was von Kármán.

V. LPRE Industry in the USA: 1941–2003

A. Innovations and Accomplishments

A list follows of significant and historic U.S. industrial achievements and key events in this field. This list is not complete or in any particular order.

1) The demands for new technology and missile production were very high during the cold war with the Soviet Union during the 1950–1970 period, and the U.S. LPRE industry successfully met these demands and did its share to put missiles into the arsenal.

2) In 1965, the United States launched Saturn V with the highest thrust engine at that time, namely, a cluster of five F-1 LPREs at 1.5×10^6-lb thrust each. This record stood until 1985 when the Soviets flew a somewhat larger engine. The 1.8×10^6-lb of the experimental F-1A is the highest known thrust of a LPRE ground test.

3) There were a good number of inventions, innovations, or first implementations of technology that can be credited to organizations in the United States. This includes the first flight of an engine with liquid hydrogen as a fuel, the first expander engine cycle, the theory of bell-shaped nozzles, the first TP, the first booster pump, the first expander engine cycle engine, and the first applications of gimbals to TCs for thrust vector control (TVC) (1947). Furthermore, there was the development of the tubular thrust chambers for regenerative cooling, the first use of ablatives on LPREs, the development of special materials for the hot-TC walls and for turbine buckets, flat plate machined injectors made of forgings, certain clever valve designs, certain injection patterns, Aerojet's platelet injectors, the electronic engine controls, pressurizing propellant tanks with gas generators, variable position pintle injectors, first flight of an extendible nozzle, first application of liquid side injection for TVC, and very fast small propellant valves mounted on an injector of a small thruster.

4) There are a number of clever innovations that were conceived and implemented with pride in the United States, but very similar innovations were actually accomplished at an earlier date in Germany and/or the Soviet Union. The LPRE work in these two countries was secret and advanced at the time and not known to U.S. LPRE personnel until years later. Examples include the staged combustion cycle, the use of inducer impellers ahead of the main pump impellers, early reinforced concrete test facilities, the highest thrust large flying engine, earliest LPREs specifically for aircraft installation, the first airplane flights with LPREs, prepackaged LPREs with storable propellants, or TVC by auxiliary or vernier thrust chambers supplied by the main engine feed system. It also includes a large LPRE with the highest chamber pressure and pump-fed experimental LPREs with certain high-energy propellants.

5) The RL-10B2 of Pratt and Whitney is the first pump-fed LOX/LH2 engine that has flown (in 2000) with the highest known specific impulse, namely, 467 s. It was the first flight application of an extendible nozzle exit segment with a LPRE.

6) The United States had in 1965 a small radiation cooled thruster that could demonstrate over 100,000 cycles or restarts, a record at the time.

7) Several of the propellants originated in the United States. This probably includes RFNA, inhibited RFNA, Aerozine 50, mixed oxides of nitrogen, gelled propellants, hydroxylammonium nitrate, and ultrapure hydrazine that would not contaminate its catalysts.

8) The United States was a leader in the application of computers and software to LPREs. Computers became part of the controls for LPRE beginning in the 1970s. The Rocketdyne RS-68 was the first LPRE to be fully designed by computers in the 1990s. This covers not only design or analysis programs, but also integration with manufacturing, test operations, engine controllers, cost and schedule control, spare parts inventory, test data reduction or display, and other areas.

9) The Nike–Ajax antiaircraft missile used a LPRE for the upper stage, and it was the first such military air defense system to be deployed for this purpose.

10) The United States has probably built more different JATO LPREs than any other country.

11) The production of 50,000 Bullpup engines was a unique accomplishment; it represents the largest number of LPREs ever produced anywhere.

12) The Lance TC configuration with the sustainer TC inside the annular booster TC is novel and compact, and the use of liquid propellant side injection for TVC is unique. The highest known thrust variation of 300 to 1 (from 4400 down to 14 lb) was achieved in the Lance sustainer TC assembly. It is discussed further in Sec. V.B.3.

B. Companies in This Business

Since the 1940s at least 14 U.S. companies have engaged in the design, development, manufacture, testing, and flight support operations of some types of LPRE. Table 3 lists their names (roughly in the order of the years of their start) and shows that there have been mergers and consolidation. Most of these companies have gone out of the LPRE business, were acquired, or merged. Today there are five that are active at the time of this writing, and each will be briefly discussed. In the book currently being written, there are discussions of LPREs of all of the 14 listed companies. There were other companies, but they are not shown in Table 3. They include aerospace companies and subsystem suppliers, where the development of LPREs was a sporadic or sideline activity. None of their engines has as yet resulted in a production. The data in this section come from the author's recollection and from personal communications.

Employment in the LPRE business was high between 1955 and 1968 during the cold war period and historically the busiest time in the LPRE industry. The real funding available to LPREs has since decreased, as have the number of employees and the number of companies. For example, the peak total LPRE employment at Rocketdyne was about 20,000 people in 1964; it hit a low in 1971 but went up to about 5500 in 1985 and was about 2800 in 2001. Aerojet's employment in LPREs personnel peaked in 1963 at about 10,000 and was about 1000 in the year 2000.

Although some good research and development (R&D) work has been done in the United States by certain universities, some private research organizations, and by government laboratories, they are not discussed here because they did not develop and qualify LPREs that went into production or became operational. This includes for example R&D work at Princeton University, Cornell University, Purdue University, Pennsylvania State University, University of Alabama, Naval Postgraduate School, and California Institute of Technology. The graduate and undergraduate education of qualified technical LPRE personnel is perhaps the universities' most important contribution, and more than 25 U.S. universities have at one time taught courses concerned with LPREs. Research organizations that have worked on LPRE issues include Batelle Memorial Institute, The Aerospace Corporation, and SRI (formerly Stanford Research Institute). Government organizations doing or having done work on LPRE includes the Rocket Propulsion Laboratory (now part of Philips Laboratory of the U.S. Air Force) at Edwards Air Force Base, Edwards, California, the U.S. Air Force Arnold Engineering Test Center, at Tullahoma, Tennesee, NASA Marshall Space Flight Center at Huntsville, Alabama, NASA-funded Jet Propulsion Laboratory in Pasadena, California, NASA Stennis Space Center in Mississippi, and NASA John H. Glenn Research Center at Lewis Field, Cleveland, Ohio. These and other government organizations are indeed useful in testing LPREs and defining, selecting, funding, guiding, and monitoring work at U.S. companies, doing R&D and testing, and in providing propellants, testing facilities/services for large and small LPREs, hover test facilities, or simulated altitude test facilities.

1. *Reaction Motors, Inc. (RMI)*

RMI was the first American LPRE company.[32–34] It was founded by four amateur experimenters of the American Rocket Society and incorporated in August of 1941. The words rocket motor were used in those days for what today is designated as a rocket engine. In 1958, RMI was acquired by and became a division of Thiokol Corporation, a manufacturer of solid propellant rocket motors. In 1972 the Reaction Motor Division was shut down and ceased operations due to a lack of business. Because of the limit on the length of this summary, only two of their historic LPREs will be discussed.

The best known and perhaps the historically most significant of their engines was the RMI 6000-C4 aircraft rocket engine[35] with

Table 3 U.S. companies in the LPRE Business (1941–2002)

Company	Typical LPRE work	Comments
RMI	Engines for experimental aircraft, Viking, prepackaged, vernier reaction control thrusters	Started Dec. 1941, merged into Thiokol 1958, stopped operating 1972
Aerojet was Aerojet Engineering Corporation	All types of LPRE	Started in 1942, bought by General Tire Corporation, 1944, spun off as General Corporation, bought General Dynamics propulsion operation 2002
Curtiss-Wright Corporation	LPRE for research aircraft	1943–1960[a] Work stopped
General Electric Company (Rocket Section)	Hermes and V-2 operation, Vanguard booster	1944–1967[a] Section was dissolved
Rocketdyne Propulsion and Power, since 1996 a part of The Boeing Company	All types of LPRE	Started 1945 as part of North American Aviation, merged into Rockwell International Corporation 1964
Walter Kidde & Company	H_2O_2 monopropellant	1945–1958[a] Work stopped
ARC Liquid Division. became division of Sequa Corporation Acquired hydrazine monopropellant line from Marquard 2001 (originally Hamilton Standard)	Agena upper stage, small attitude control LPRE	Formerly Bell Aircraft Company started LPREs in 1947, bought Royal Ordnance 1997
M. W. Kellogg Company	JATO units	1945–1953[a] Work stopped
Pratt and Whitney, a United Technologies Company	Upper stage LOX/LH2 LPREs, Russian RD 180	Started 1957/1958
Marquard Corporation, became Kaiser–Marquard, 1990 Sold to Primex 2001	Small bipropellant LPRE or reaction control systems	Started LPRE 1958, bought by Kaiser 1990, bought hydrazine TC line from Hamilton Standard, in 1996
Northrop Grumman Corporation Propulsion Products Center formerly TRW, Inc.	Attitude control LPRE, lunar lander and others	Started 1960 as part of Space Technology Laboratory, name TRW adopted in 1965, bought by Northrup Grumman 2003
Rocket Research Corporation spun off as Primex, 1996 bought by General Dynamics Corporation in 2001	Hydrazine monopropellant TCs and bipropellant TC (in 2000)	Started LPREs in 1963, aquired by Olin Corporation 1985, Primex bought Kaiser–Marquardt in 2000
Hamilton Standard Division United Technologies Corporation (small group)	Hydrazine monopropellant thrust chambers	Started 1964, technology bought by Marquardt in 1995, divested to ARC in 2001
General Dynamics Corporation (GD)	All types of small TCs ACS/maneuver systems	Bought Primex in 2001 sold to Aerojet in 2002

[a]Date is not confirmed.

four fuel-cooled thrust chambers shown in Fig. 17. It was designed for the Bell Aircraft manned-research airplane X-1. This engine propelled this aircraft on 14 October 1947 to a record speed of Mach 1.06. The engine had four thrust chambers at 1500 lbf each (total 6000 lbf) with LOX/75% alcohol at a chamber pressure of 220 psia and a specific impulse (sea level) of 209 s. The initial versions flew with a pressurized feed system. Later versions had a TP feed system with a GG supplied with hydrogen peroxide, which was decomposed by a catalyst; the propellant tank pressures and total propulsion system weights were lower. Each thrust chamber had a small igniter chamber in the center of the injector designed to allow multiple starts. The igniter used a spark plug to ignite a small flow of fuel and gaseous oxygen that had been evaporated in coils around the fuel feed pipe. The engine was improved and used to fly several later versions of the Bell X-1 research aircraft, the Douglas D558-2 Skyrocket research aircraft, several unmanned research lifting bodies, and as a dual engine for an interim power plant for the North American X-15 research airplane. An up-rated version of this four-barrel engine (at 435-psi chamber pressure and 7600–8400 lb total thrust) launched the small scale model of the Navaho missile (Project MX-774), but without restart capability. It was the first U.S. engine with hinged thrust chambers, which allowed flightpath control. There were engine-related problems in two of the three flights.

RMI developed several storable propellant prepackaged rocket propulsion systems. In the 1940s and 1950s, solid propellant motors had problems operating at ambient temperatures lower than about −40°F, but the liquid engines could meet the required minimum temperature of −65°F. The most significant of the RMI prepackaged LPRE was for the Bullpup air-to surface missile. Work started in 1958. It had the largest production of any LPRE, and approximately 50,000 units were delivered between 1960 and 1967. As shown in Fig. 18, it used storable propellants (RFNA and a fuel consisting of 50.5% diethylenetriamine, 40.5% UDMH, and 9% acetonitrile), a central solid propellant gas generator (double-base solid propellant) for pressurizing two annular propellant tanks, burst diaphragms, and a very short (inefficient) bell nozzle. As seen in Table 4 there were two versions. It started with a powder cartridge moving a piston (the only moving part), which then sheared or broke a diaphragm and initiated the burning of the solid propellant grain; full thrust was achieved in about 0.1 s. The thrust chamber was regeneratively cooled. The Bullpup had a design storage life of 5 years minimum and a safe storage temperature range between −80 and +160°F (−62

Table 4 Bullpup LPRE data

Engine designation	LR58 or Bullpup A	LR62 or Bullpup B
Diameter, in. (cm)	12.1 (102.7)	17.3 (43.9)
Length, in. (cm)	40.5 (102.7)	61.2 (155.4)
Weight, loaded, lb (kg)	203 (92.3)	563 (255.3)
Weight, dry, lb (kg)	92 (41.8)	205 (92.9)
Thrust, lbf (kN)	12,000 (52.8)	30,000 (132)
Duration, s	1.9	2.3
Total impulse, lb · s (kN · s)	22,800 (101)	69,000 (307)

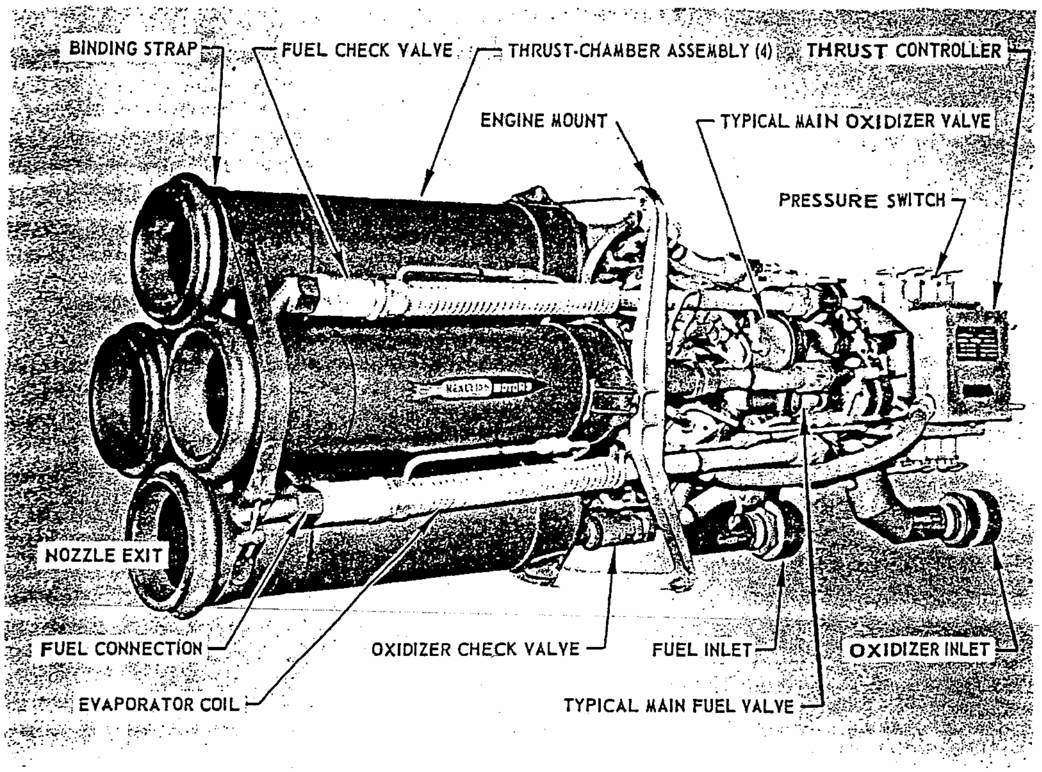


Fig. 17 RMI 6000C-4 LPRE for research aircraft with four TCs; each could be turned on or off individually, thus, giving a stepwise change in thrust (from Ref. 3).

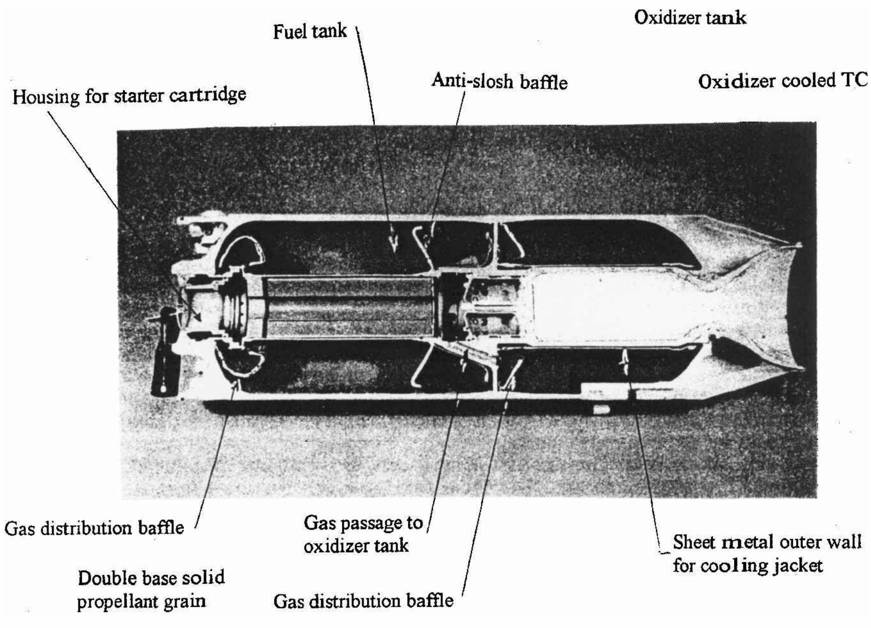


Fig. 18 Cutaway section of the Bullpup A (LR-58) (from Ref. 34, Part III, 1983).

and +71°C). Bullpup became an operational missile in 1959. There were thousands of test firings, both on the test stand and in flight. Reliability was rated at 0.9972. Military cutbacks and changing military requirements eventually caused the Bullpup to be taken out of service.

2. Aerojet General Corporation

This company (originally known as Aerojet Engineering Company) is commonly called "Aerojet"; it is a subsidiary unit of GenCorp, Inc. It was the second U.S. company dedicated to rocket propulsion, was started 1942, and it grew quickly.[36–38] It was mentioned in Sec. IV.C to have been a direct outgrowth of rocket propulsion projects done earlier at the GALCIT under the guidance of von Kármán, a famous aerodynamicist.[31] For several decades, the work on LPREs represented about one-quarter of Aerojet's business. Aerojet was also engaged in solid propellant rocket propulsion, nuclear propulsion, ordnance, and other areas. In August of 2002 Aerojet acquired the rocket propulsion assets, LPRE products,

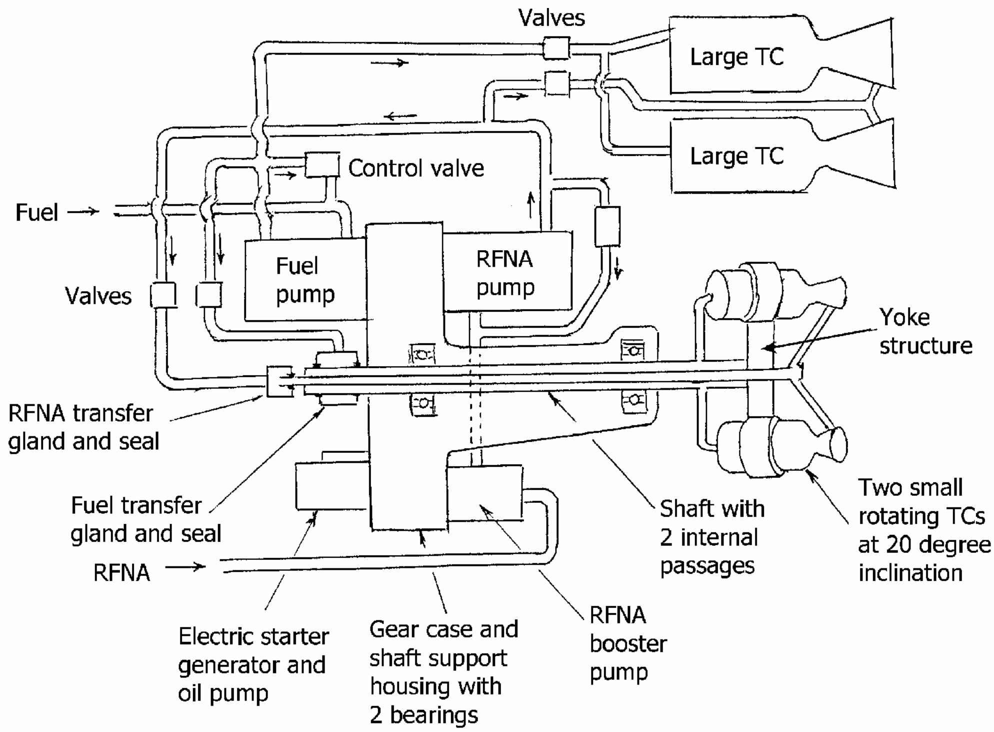


Fig. 19 Schematic diagram of the Aerotojet LPRE concept (courtesy Aerojet).

skilled personnel, inventory, LPRE know-how, and facilities of the General Dynamics operation located at Redmond, Washington. In the past Aerojet has worked on a large variety of LPRE schemes, but only a few are presented here.

Aerojet developed more different JATO engines than any other U.S. company.[36–41] The first JATO LPRE was actually developed by its predecessor GALCIT in 1939, but improved by Aerojet in 1942. It had a thrust of 1000 lbf for 25 s duration, an uncooled TC, and RFNA with aniline as propellants. The first successful takeoff of a manned aircraft using two liquid propellant JATO units was on the Douglas A-20A attack bomber in January 1943 at Muroc Dry Lake in California. The first production of a JATO LPRE of Aerojet was an improved, lighter, reusable, and more compact version of the JATO that flew on the A-20A airplane.[32] All of these early JATOs had three spherical tanks, one each for RFNA, aniline fuel, high-pressure nitrogen pressurant, and an uncooled TC. By 1944 some 64 of these were delivered at a cost of $3450 each.

In the 1946–1954 period, Aerojet developed a series of different JATO units for different military aircraft, and they were flown successfully in experimental airplanes of the F-84 fighter-bomber, the PB2Y-3 flying boat, and the B-29, B-45, and B-47 bombers. Most JATOs had cooled TCs, some used TPs feed systems with a GG, but the early models used gas pressurized feed systems. The LPRE for the PB2Y-3 is unique because its propellant pumps were driven by a separate gasoline engine. Several of these Aerojet JATO units were put into a limited production. The JATOs for the B-47 bomber had the highest thrust of any JATO and represent a historic achievement. Two units, one on each side of the aft fuselage, with two TCs each, gave total thrust of 20,000 lbf. The turbines of the two propellant pumps were driven by warm air bleed from the compressors of the aircraft's jet engine. (Pumping reduced the weight of inert hardware of the propellant tanks.) Liquid propellant JATO units were not used often by the military services because there was relatively little real need for takeoff assistance, and the servicing and refurbishing of used units (with remnants of flammable, corrosive, toxic propellants) was considered hazardous. None were put into operation by the military services.

Between 1945 and 1956, Aerojet also developed auxiliary rocket engines or aircraft superperformance engines for fighter airplanes (which already had their main turbojet engine) to get quickly to altitude and augment the flight performance.[36–41] This application is different from the JATO mission, where aircraft are given an extra push during takeoff (with heavy loads), while getting off the runway. Aerojet developed and flew such engines for three fighter aircraft, a P-51 Mustang, a P-80, and an F-86. All used storable propellants and sophisticated pumped feed systems. Flight tests generally showed some improved aircraft performance during rocket operation, but a reduced aircraft range and/or reduced weapons load.

The government, through its intelligence agencies, knew in late 1943 that the Germans were soon going to have a rocket-propelled fighter airplane and wanted a U.S. rocket-propelled airplane. Aerojet conceived, designed, built, and partially tested a very unique LPRE for an aircraft power plant, called the Aerotojet,[37] and it was quickly put under contract in 1944. As shown in Fig. 19, it had two stationary TCs of 750 lb each and two rotating throttlable TCs at about 300 lbf each. The small TCs were mounted to a hollow shaft, but offset from the shaft axis and slightly inclined to produce a torque. The power was transmitted through a gear case to four pumps: an aniline fuel pump, an RFNA pump, an oxidizer booster pump, and a lubricating oil pump. An electric motor initiated the rotation and the start. It was intended to propel a new flying wing design of John Northrop, the founder of Northrop Aircraft Company. The author was a development and test engineer in the early phases of this unique project. This was the first known application of a booster pump for preventing cavitation at the impeller of the main pump with a high vapor pressure propellant. The rotation of the small chambers caused a pumping effect or additional pressure rise and a mismatch of the impingement of the fuel and oxidizer jet streams. In turn this caused incomplete combustion and the accumulation of unburned and poorly mixed propellants in the chamber. Two explosions of these accumulated propellants broke experimental chambers. There were a number of other development problem, which are briefly described in the book length version. The delays in this development forced Northrop to go to an alternate propulsion scheme for the flying wing, and the Aerotojet project was canceled in 1946.

A small-scale model of this Northrop flying wing aircraft was built earlier in this program (1944) to test the aerodynamics of this novel winged aircraft design. It was the first piloted U.S. rocket

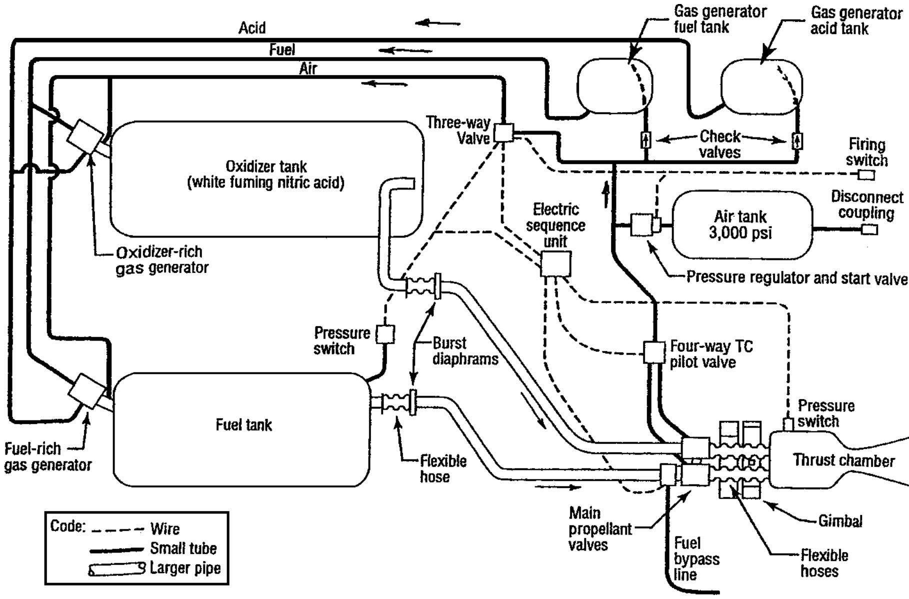


Fig. 20 Flow diagram of one version of the Bomarc booster LPRE (courtesy of Aerojet).

airplane that was propelled by rocket power only.[3,5,36,37,40] It was known as Project MX-324. Aerojet built, tested, and flew a small LPRE for this model airplane, and I was a part time member of the team. It used RFNA and aniline as propellants, had a single, small regeneratively cooled (with acid) thrust chamber of about 200-lb thrust (aluminum with a copper nozzle), and a pressurized feed system, and it was restartable in-flight. The model aircraft was air launched from a larger airplane and flew successfully for the first time in July 1943 and several times thereafter. This flying wing experimental aircraft was so small that the pilot had to lie down on his stomach with his head in the plastic nose of the airplane.

The remarkable booster LPRE[37,38] for the Bomarc Area Defense system (ramjet-powered supersonic missile) was developed by Aerojet beginning in 1951. The second stage was propelled by a supersonic ramjet. It is a historic LPRE because it successfully overcame combustion vibration problems, had a novel tank pressurization scheme, and was the first Aerojet LPRE to use a gimbaled uncooled TC (with a ceramic lining) in a military operation. A flow diagram of one version of the Bomarc booster LPRE is in Fig. 20. The propellant tanks were preloaded with propellants making a sealed prepackaged system. Originally it used white fuming nitric acid and jet fuel propellants. It delivered 35,000 lb of thrust for 4 s. The gas for the tank pressurization system came from two GGs (one fuel rich gas for the fuel tank and one oxidizer rich), both supplied from a separate set of small gas-pressurized small propellant tanks. This saved considerable inert pressurization hardware weight and minimized significant chemical reactions in the propellant tanks. The control (start/stop) was accomplished by time sequencing the operational steps and using high-pressure gas with pilot valves to actuate the main valves. This method of timing control was common at the time, but is no longer used today. Early tests failed due to a high-frequency combustion instability causing excessive heat transfer and erratic operation. After extensive investigations a change in fuel to UDMH (or really a mix of 60% JP-4 jet fuel and 40% UDMH) solved these vibration problems. Bomarc was deployed and operated with the military services until 1972. The prime contractor, The Boeing Company, delivered 366 missiles.

The Titan LPREs were probably Aerojet's most successful large LPREs.[37,38,42,43] A family of those were installed in the booster and sustainer stages of the several versions of the Titan vehicle. The twin booster LPREs is shown in Fig. 21. A flow diagram of the sustainer engine in Fig. 22 shows the fuel tank is pressurized by fuel-rich hot gas taken from the turbine exhaust; this gas is cooled by fuel in a separate heat exchanger, and the gas flow is controlled by a sonic

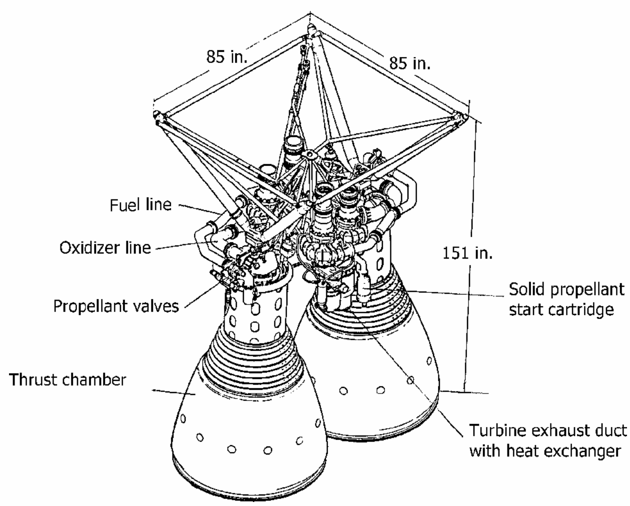


Fig. 21 Titan booster engine LR87-AJ-11 with two TCs, 151 in. tall with a 15:1 nozzle area ratio (courtesy, Aerojet).

orifice. The oxidizer tank is pressurized by NTO, which has been gasified in a superheater; a cavitating venturi regulates the liquid NTO flow. There are no separate propellant valves for the GG, only check valves to prevent reversed flow. The propellant flows to the GG are controlled by two cavitating venturi tubes. Venturis were at the time a popular way to control the propellant flow and mixture ratio. The two main rotary propellant valves are on the same shaft and use a single actuator. Each engine has a regenerative cooled gimbaled tubular thrust chambers and its own TP and GG. The gear case oil is cooled by fuel. Each TP has a single high-speed turbine geared to two centrifugal propellant pumps and a lubricating oil pump. This geared TP scheme optimizes the shaft speeds and operates at high turbine and pump efficiencies, thus, minimizing the GG flow and the performance loss. The sustainer engine has one geared TP (similar to the larger booster engine's TP) and a single cooled gimbaled tubular thrust chamber with an ablative exhaust nozzle liner extension at the larger nozzle exit diameters. The turbine exhaust gas from the sustainer engine was used for roll control of the second stage. Titan I work started in 1954, and the initial thrusts were 300,000 lb for the two booster engines and 60,000 lb for the sustainer engine; the propellants were LOX and RP-1 (kerosene). The

Table 5 Several key parameters of recent Titan LPREs

Application	Booster LPRE	Sustainer LPRE
Engine designation	LR87-AJ-11	LR91-AJ-11
Thrust, lbf, in vacuum	548,000[a]	105,000
Specific impulse, s vacuum	301	316
Nozzle area ratio	15	49.2
Mixture ratio	1.91	1.86
Chamber pressure, psia	857	860

[a]Dual engine.

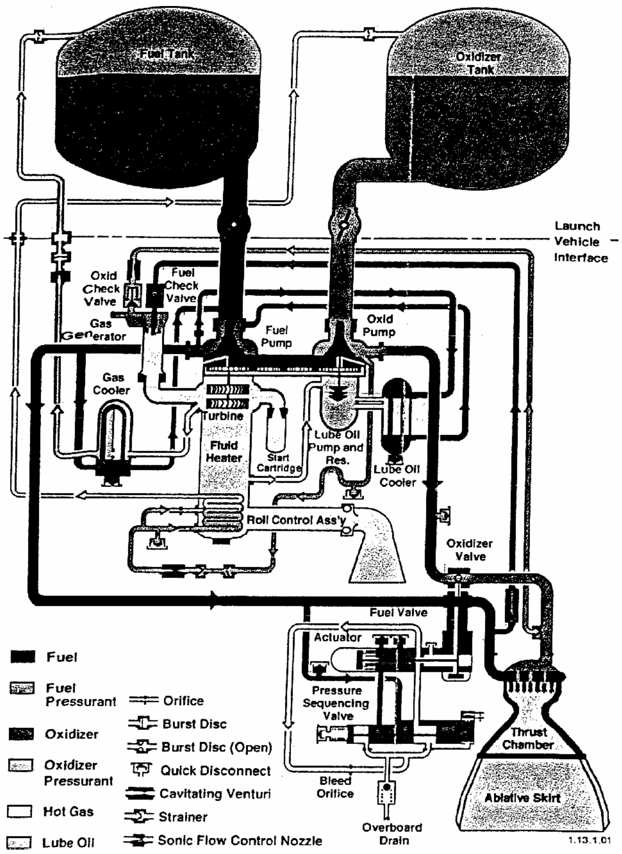


Fig. 22 Flow diagram of the Titan sustainer engine (LR-91-AJ-11) with a GG cycle (courtesy, Aerojet).

U.S. Air Force changed the propellants and the Titan II, a ballistic missile, and subsequent versions used the storable combination of NTO with Aerozine 50 (50% hydrazine and 50% UDMH) (Ref. 43). The first production engines were delivered in 1961 and an initial operational intercontinental ballistic missile (ICBM) capability had been established by 1963. The thrusts of the Titan engines were uprated progressively for the Titan III and IV SLVs to the values listed in Table 5. More than 1500 individual LPREs have been delivered for all of the versions of Titan over a period of 47 years. Unfortunately high-frequency combustion instability was observed in a few tests, and this caused a major diversion to the program and an intensive R&D effort. After the injectors were equipped with baffles, the combustion appeared to be stable.

Titan II ICBM, was installed in heavily armored undergroud silos and was operational between 1963 and 1987. When these Titan II missiles were decommissioned from their service as a military deterrent, they became available as SLVs. Titans III and IV are bigger, uprated SLVs with larger engines specifically for heavy space flight payloads.

In 1995 Aerojet obtained the right to sell in the United States several LPREs designed by the Kuznetsov Design Bureau in the 1960s for the Soviet N-1 moon flight vehicle, whose program was canceled. Modified NK-33 engines (fitted with a gimbal) were sold by Aerojet to Kistler Aerospace for their unique SLV.[44] Several dozen of this large LOX/kerosene engine were taken out of Russian storage and shipped to Aerojet.

In August 2002, Aerojet acquired the rocket propulsion organization of General Dynamics in Redmond, Washington. It had two roots. The first is the Rocket Research Corporation (RRC) of Redmond, Washington, a leader in hydrazine monopropellant thrusters and GGs. RRC also built electrical propulsion systems. The other is the Marquardt Corporation (later Kaiser–Marquardt Corporation) of Van Nuys, California, a leader of small, storable bipropellant thrusters. Marquardt's business was relocated to Redmond, Washington, in 2001.[45,46] General Dynamics and its predecessors sold mostly assemblies of these small thrusters with their special control valves, but also developed a few complete LPREs with pressurized feed systems.

RRC was founded in 1960, but became active in small hydrazine monopropellant rocket engines only in 1963.[45–48] A major step forward was the development of a suitable catalyst for hydrazine decomposition by California Institute of Technology's Jet Propulsion Laboratory in a joint effort with the Shell Development Company. Thereafter, Shell produced the Shell 405 catalyst, which has since been successfully used by all U.S. and some foreign competitors. The Aerojet Redmond Center now produces this catalyst. Figure 10 shows one of their small hydrazine monopropellant thruster assemblies. The first contracts for monopropellant hydrazine LPREs were for transstage attitude control system (ACS) (25-lbf thrust) and for the ACSs of Titan I and Titan II. Aerojet acquired a stable of proven catalytic hydrazine monopropellants thrusters ranging in thrust from 0.1 to over 300 lbf and a history of successful applications in more than 60 different space launch vehicles and spacecraft. By 2000 the company had delivered 10,000 such monopropellant thruster and GGs. Two to four individual thrusters and their valves were often packaged into a subassembly or rocket engine module because this simplifies their vehicle assembly, their checkout, and servicing.

Marquardt, the other predecessor company, started in the LPRE business in 1958 by conducting a study of the future requirements for reaction control systems.[45,46,49] They then performed in-house work on a thrust chamber using the bipropellant combination of NTO and hydrazine. Marquardt's first contract for a bipropellant rocket engine was for the Advent satellite of General Electric Company at 25-lb thrust. This LPRE experienced some failures during development, but did pass the qualification tests. However, for unrelated reasons it never flew. After Marquardt experimented with a variety of liquid storable propellants, they selected NTO and MMH for their families of bipropellant thrusters. The 100-lb thrust R-4D thruster for the lunar orbiter became a versatile thruster for the company. One version is shown in Fig. 11. Different models of this thruster design have been successfully employed in several applications, such as the Apollo service module (16 thrusters) or the Apollo lunar lander (16 thrusters). This R-4D thruster has an eight on eight unlike doublet impinging stream injector with eight small film cooling holes and very closely coupled valves. Another version of this radiation-cooled thruster had a high nozzle area ratio of 375 and an altitude specific impulse of 323 s. It can be seen in Fig. 23 and uses three different metals for the nozzle: rhenium (with an iridium protective coating can be used up to 4000°F) for the chamber and nozzle throat, niobium (for the upstream portion of the nozzle exit section, up to 3400°F), and titanium (near the nozzle exit, up to 1300°F). This improved costs and reduced weight. Over the years, a series of bipropellant thusters was developed, ranging in thrust from about 1 (5 N) to about 900 lbf, with several materials, nozzle geometries, thermal insulation, durations, duty cycles, total number of pulses, and very short pulse widths (minimum total impulse bit of 0.01 lbf · s).

3. *The Boeing Company, Rocketdyne Propulsion and Power*

This company (Rocketdyne) has been the largest LPRE company in the United States. It has developed LPREs in every area of application.[50–52] It was started in 1945 as a section of the aircraft company North American Aviation, which in 1964 became part of

Table 6 Performance data for selected large Rocketdyne production engines/engine families

Missile/launch vehicle application	Redstone MRBM and SLV	Navaho cruise missile	Atlas ICBM and SLV	Jupiter IRBM and SLV	Thor and Delta IRBM and SLV	Saturn I and IB SLV	Saturn V SLV	Saturn IB and V SLV	Space shuttle SLV	Delta IV SLV
Engine family designation (data from last-in-family engine	A-6 and A-7	G-26	B-2C, MA-1, MA-2, MA-3, MA-5, and MA-5A	S-3D	S-3E, MB-1, MB-3, RS-27, and RS-27A	H-1 A, B, C, and D	F-1	J-2	SSME phase 1 and 2 block 1, 1A, 2A, and 2	RS-68
Initial family engine design year	1948	1950	1952	1953	1953	1958	1959	1960	1972	1997
Thurst										
Sea level, thousand lbf	78	240	430 (B) 60 (S)	150	200	205	1,522	DNA	374 (100%)	650
Vacuum thousand lbf	89	278	480 (B) 84 (S)	174	237	237	1,748	230	470 (100%)	745
Specific impulse, s										
Sea level	218	229	265 (B) 220 (S)	248	255	263	265	DNA	361 (100%)	365
Vacuum	249	265	295 (B) 309 (S)	288	302	295	305	425	452 (100%)	410
Oxidizer	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX
Fuel	Alcohol (75%)	Alcohol (92.5%)	RP-1	RP-1	RP-1	RP-1	RP-1	LH_2	LH_2	LH_2
Mixture ratio (oxidizer/fuel)	1.354:1	1.375:1	2.25:1 (B) 2.27:1 (S)	2.4:1	2.24:1	2.23:1	2.27:1	5.5:1	6.03:1	6.0:1
Chamber pressure, psia	318	438	719 (B) 736 (S)	527	700	700	982	717	2,747 (100%)	1,460 (100%)
Nozzle area ratio (exit/throat)	3.61	4.6:1	8:1 (B) 25:1 (S)	8:1	12:1	8:1	16:1	27.5:1	69:1	21.5:1
Nominal flight duration, s	121	65	170 (B) 368 (S)	180	265	150	165	390 (S-II) 580 (IVB)	520	400 Max
Dry mass, lbm	1,478	2,501	3,336 (B) 1,035 (S)	2008	2,528	2,010 (C) 2,041 (D)	18,616	3,454	7,774	14,850
Engine cycle	GG	GG	GG	GG	GG	GG	GG	GG	SC	GG
Gimbal angle, deg circular	None	None	8.5	7.5	8.5	10.5	6	7.5	11.5	10-MPL 6-FPL
Diameter/width, in.	68	77	48 (B T/C) 48 (S)	67	67	66	149	81	96	96
Length, in.	131	117	101 (B) 97 (S)	142	149	103	230	133	168	205
Operating temperature limits °F	−25 to +110	−20 to +110	−30 to +130	+40 to +130	0 to +130	0 to +130	−20 to +130	−65 to +140	−20 to +130	−20 to +140
First flight date in family	08-20 -1953	11-06 -1956	06-11 -1957	03-01 -1957	01-25 -1957	10-27 -1961	11-09 -1967	02-26 -1966	04-12 -1981	11-20 -2002
Comments								Restart in space	Throttlable power level 67–109%	Throttlable power level 57–102%

B = booster, S = sustainer, Gas generator cycle, SC = staged combustion cycle, DNA = does not apply, MPL = minimum power level, FPL = full power level, A and C = inboard engines, B and D = outboard engines, MRBM = medium-range ballistic missile, IRBM = intermediate range ballistic missile, ICBM = intercontinental ballistic missile, SLV = space launch vehicle.

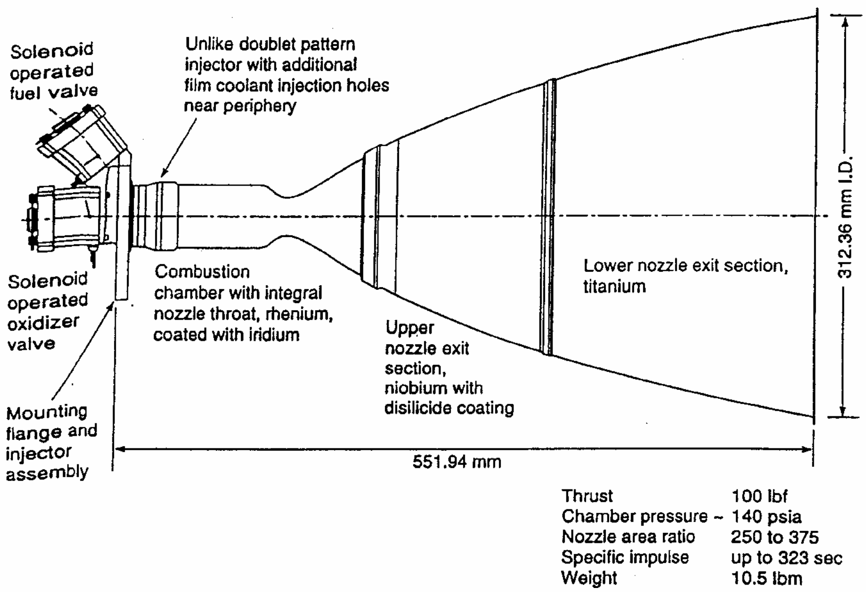


Fig. 23 RD-4-15 bipropellant thruster with a large nozzle used for operation in a vacuum (from Ref. 2).

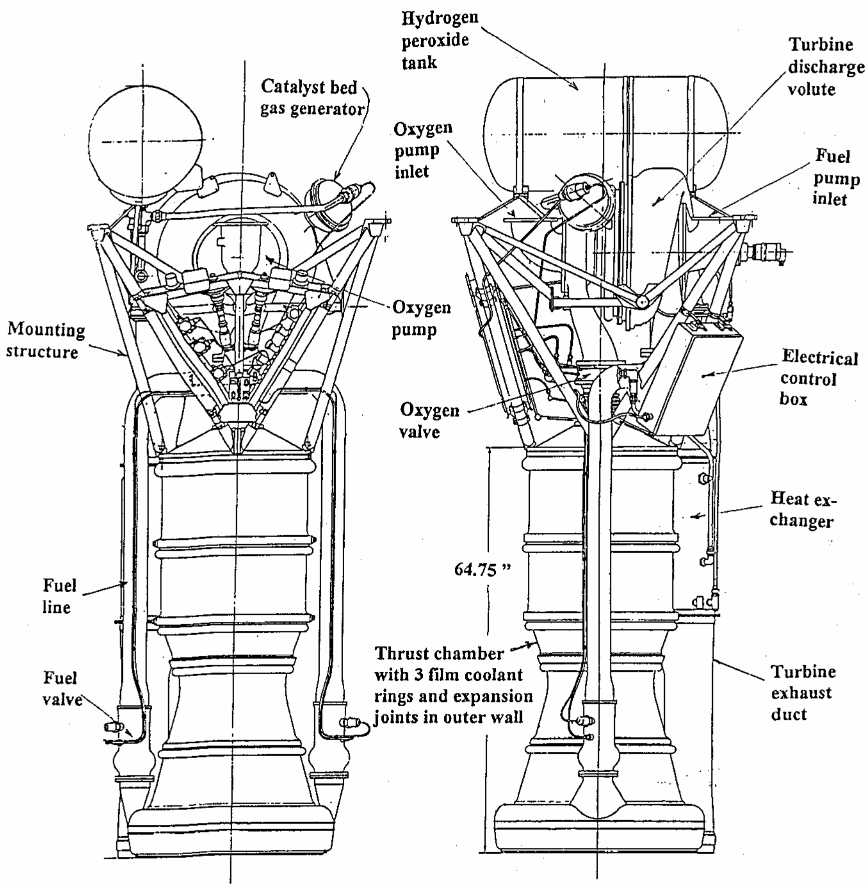


Fig. 24 First large U.S. LPRE was flown on the Redstone missile (courtesy, Rocketdyne).

Rockwell International Corporation. In 1955, the name Rocketdyne was adopted, and it became a separate division of the parent company. Rocketdyne also was in the solid rocket motor business between 1959 and 1978, is today in the space power supply business, and was in the electrical and nuclear propulsion areas. Together with some other divisions, Rocketdyne was sold to The Boeing Company in December 1996, when the name listed as the title of this section was adopted.

In its 58-year history Rocketdyne put about 15 large LPREs (1500–1,500,000 lbf thrust) and 17 smaller LPREs (1.0–1500 lbf) into production, and several of these had between one and five major redesigned or upgraded models. Up to 1 June 2001 Rocketdyne engines had boosted 1516 vehicles. In addition, Rocketdyne developed and tested more than 36 experimental engines or thrust chambers aimed at demonstrating feasibility or advances in technology. The author worked for Rocketdyne from 1946 until 1975.

Table 6 lists summary data for several large Rocketdyne LPREs that have flown. For engines that had several models of the same engine family the performance data in Table 6 refer to the most recent version. The two dates (design year and first flight) give a clue to the historical sequence. Because of the limitations on the length of this paper, only 8 of the more than 20 historical engines will be briefly discussed.

The first large engine development effort was a pump-fed LPRE of 75,000-lb thrust, which soon became known as the engine for the U.S. Army's Redstone ballistic missile.[53,54] This historic first large U.S. engine is shown in Fig. 24. The Redstone LPRE had many similarities to the V-2 engine (same propellants, heavy thick low-alloy steel walls in the TC with film cooling, a similar GG, an aluminum turbine, aluminum fuel and oxidizer pumps). Like the V-2, carbon jet vanes were used for TVC during powered flight. However, there were some significant improvements and differences. It had 33% more thrust, 44% more chamber pressure, a new type of large diameter (flat surface) injector similar to the one in Fig. 10, cylindrical chamber geometry (not pear shaped), better combustion efficiency and performance, and a smaller relative chamber volume. The first hot firing of the new large TC (with a pressurized test stand feed system) occurred in January 1950, which at 75,000 lb was then the highest rocket thrust in the United States. I was the designer and development engineer for this TC. The first static test of the complete Redstone engine took place in late 1950, and the first flight was in August 1953. It was the LPRE of the first U.S. ballistic missile to become operational, and it also was deployed overseas in June 1958. This engine launched the first U.S. satellite (Explorer on 31 January 1958), and it also launched two U.S. astronauts, each in their Mercury capsule, on their first suborbital space flights in 1961.

The preliminary design of the engines for the Convair Atlas ballistic missile was started in 1952 in my engineering section and was unique.[53–55] Detail design was in 1954. The two booster engines of 150,000-lb thrust each were mounted in a ring or doughnut-shaped structure at the missile's tail, and this ring structure was dropped from the flying vehicle after booster cutoff at about 170 s. The sustainer engine with 60,000-lbf (located in the center of the aft end of the vehicle) is also started at launch, but runs continuously for a total of about 368 s. Figure 25 shows a ground test of one version of this three engine configuration. The nozzles of the two-booster engine are flowing full, and their bright radiating plumes have sharp boundaries. The center sustainer engine is slightly overexpanded with its 25 to 1 nozzle area ratio when operating at the low altitude of the test facility. Its jet has separated from the nozzle wall. Steam clouds from the test facility water spray are being aspirated into the central jet. They obscure the exhaust jet, which has a smaller diameter than the nozzle exit. This one-and-one-half set of stages was selected in part because Convair (the vehicle developer) and Rocketdyne were not sure, at that time, if altitude ignition of a sustainer could be reliably achieved. The total three-nozzle thrust of 360,000 lb at sea level (about 414,000 at altitude) was increased in steps in the several subsequent modifications and uprated versions of this Atlas engine until it reached the performance given in Table 6. The TP for the booster-stage engine is based on the TPs for the Navaho

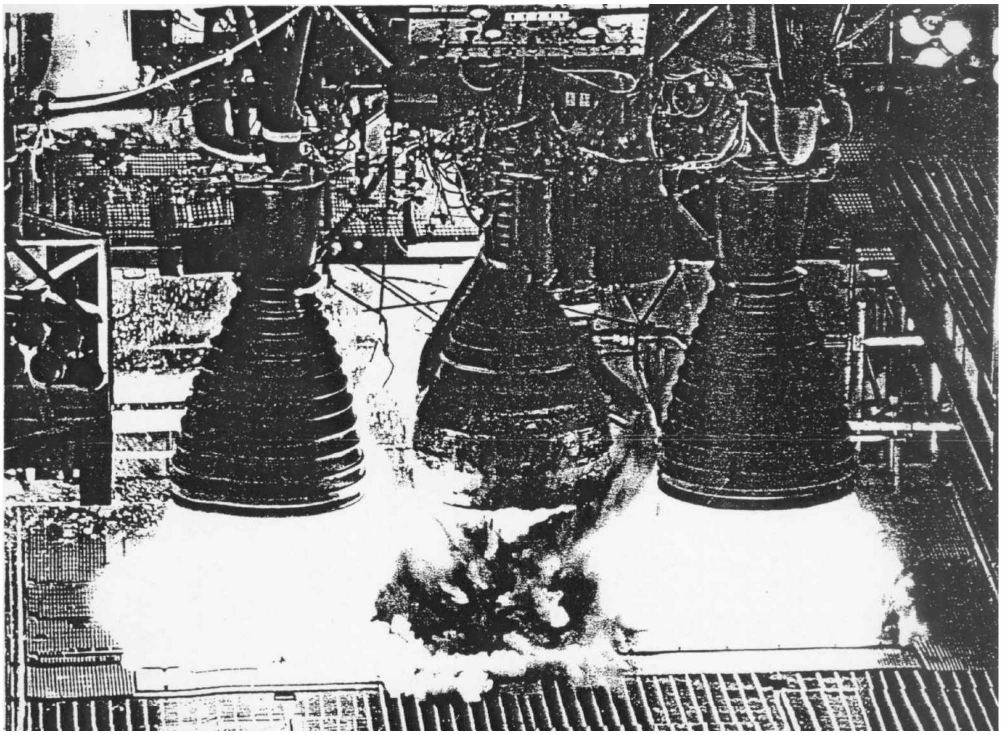


Fig. 25 Static firing test of the MA-5A LPRE for the Atlas vehicle (courtesy, Rocketdyne).

booster and uses a gear case (Fig. 9). The TCs for the Atlas, Thor, or Jupiter missiles had a tubular cooling jacket and originally a conical nozzle exhaust section. Later these cones (and the tubes) were changed to a bell-shaped nozzle exit. The fabrication and brazing of tubular cooling jackets requires precision fixture and precision parts.

The Atlas was the first U.S. ICBM and was operational by the U.S. Air Force between 1961 and 1965. Several versions of this Atlas engine also served in propelling satellite launches for military spacecraft and space exploration payloads. This included the Surveyer, Pioneer, or Intelsat satellites. The Atlas/Centaur engines boosted the astronauts in the Mercury manned spaceflight program. The Atlas LPRE was an active engine program for 46 years, in production between 1956 and 1996, and 482 engine sets (consisting of two boosters and one sustainer engine) have been delivered.

Rocketdyne engines launched both of the two launch vehicles used in the U.S. Apollo (moon) program, namely, Saturn I and Saturn V.[5,6] Saturn I was boosted by eight H-1 engines, each initially at 165,000 lbf, but later upgraded in steps to 205,000-lb thrust each. There were 19 flights including the first international space rendezvous. Saturn V has five F-1 booster engines in the first stage (at 1.5×10^6 lb thrust each) with a total launch thrust of 7.5×10^6 lb, five J-2 oxygen/hydrogen engines at 230,000 lb (vac) each in the second stage and one J-2 engine in the third stage. Saturn V was launched 13 times in connection with the world renowned American Apollo program's moon circumnavigation, landing, and return programs. During one of these flights, one of the five J-2 second-stage engines exceeded an operational limit and was shut off prematurely, but safely. The computer controller allowed the flight to continue to the planned cutoff velocity with the remaining four engines. This was an unscheduled flight confirmation of the engine-out capability.[53]

The large F-1 LPRE has the highest thrust of any flying U.S. engine and for more than a decade the highest thrust in the world[54] (Fig. 26 and Table 6). Engine detail design started in 1962. It was the first U.S. LPRE where the bottom nozzle exit section (between area ratio of 10 and 16) is film cooled with warm (about 800°F) turbine exhaust gas. It has the largest U.S. single-shaft TP. After an extensive investigation, the cooled baffles extending from the injector face (Fig. 5) solved the combustion vibration problems that were encountered. There were 2771 single F-1 engine tests plus 34 tests of a five-engine cluster. Static tests indicated a reliability of 99.7%, and the flights were 100% reliable. Altogether 98 engines were built, and 65 of these have flown successfully.

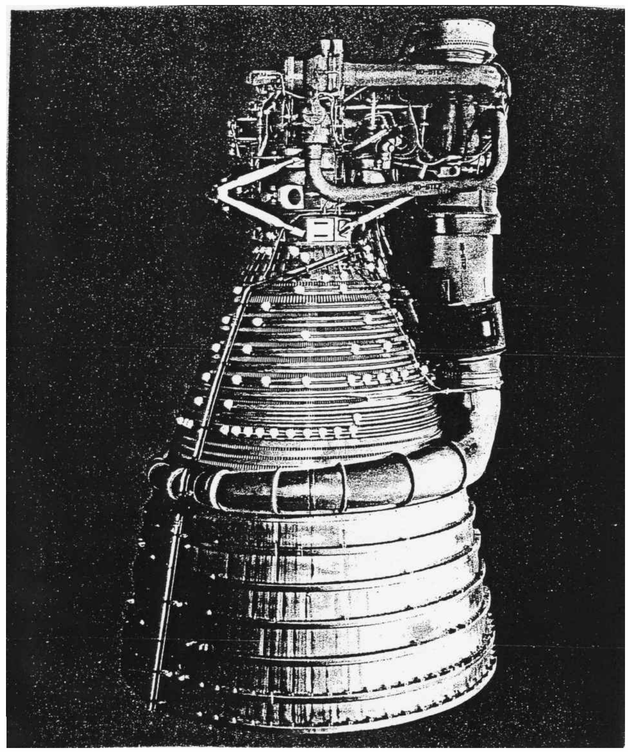


Fig. 26 F-1 LPRE, the highest thrust U.S. engine (1.5×10^6 lb) when it first flew in 1967; 220 in. long and 144 in. wide (courtesy, Rocketdyne).

The J-2 engine was the world's first large engine to use LOX and LH_2 as propellants.[54] Design started in 1960 (Fig. 2). It was the first large Rocketdyne LOX/LH_2 engine with two separate direct-drive TPs (no gears), namely, the oxidizer pump and a seven-stage

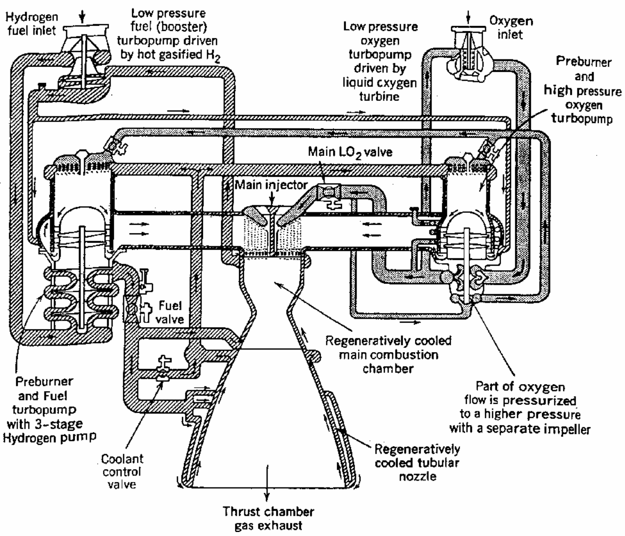


Fig. 27 Simplified flow diagram of SSME (courtesy, Rocketdyne).

axial-flow fuel pump with its own turbine. This axial pump is more efficient than a centrifugal pump for this range of volumetric flow, but it is not quite as adaptable to offdesign flow conditions.

The historic Rocketdyne LOX/LH_2 SSME was designed in 1972, and its design is a radical departure from prior U.S. engines.[53–57] It is still in limited production at the time of this writing. Three of these engines are used in a Space Shuttle Orbiter. Table 6 indicates it has a high performance. It is the first and only flying U.S. LPRE with a staged-combustion-engine cycle, where the turbines are driven by high-pressure gas from two fuel-rich preburners and the turbine exhaust gases are fed to the injector into the combustion chamber. It gives a higher performance by about 6% compared to a GG cycle engine of the same thrust and nozzle area ratio. It is a reusable man-rated U.S. large LPRE. It can be throttled between 67 and 109% of rated power level, and the temporary thrust reduction prevents excessive loads and heating of the vehicle during ascent. The SSME simplified flow sheet in Fig. 27 shows four TPs, namely, a three-stage high-pressure fuel pump, a single-stage oxidizer pump, and two booster pumps. It was the first U.S. engine where a booster pump (oxygen) was driven by a hydraulic turbine (by LOX) and not a gas turbine or not through a gear case. The original chamber pressure of 3319 psia (initial version or block 1) was the highest of any U.S. production LPRE. This allowed a relatively small chamber and nozzle and a high nozzle exit area ratio (68.8) without excessive nozzle flow separation. It had a unique engine control and health monitoring system.[58] Figure 13 shows the initial successful injector design with baffles, which prevented high-frequency combustion instability. The SSME thrust chamber uses tubular construction in the nozzle diverging section, where heat transfer is lower. In the chamber and nozzle region (with high heat transfer) it uses a milled channel design.

A unique design feature of the SSME is its power head design made of nickel-based superalloy 718 forgings. It contains the main injector assembly with a novel oxygen/hot-gas heat exchanger. It is the backbone or key structure to which the two high-pressure TPs, the thrust chamber, and two preburners are attached. Changes were made to the engine in 1995 and again in 1996–1997. In the block 2 version of the SSME, a larger throat was built into the thrust chamber; this reduced the chamber pressure by 11%, reduced the feed pressures, reduced the heat transfer, and provided more margin.[57] There were some TP problems, such as the turbine blade cracks after a few static firing tests of the early version. Two safer, more efficient, more robust, and heavier new main TPs, developed by Pratt and Whitney, replaced the original Rocketdyne TPs in the current version.

The RS-68 booster engine (Table 6) was designed for low cost in 1997 and 1998. It flew first in November 2002 (Ref. 59). A single engine launches the first stage of the Delta IV family of SLVs. It is the highest thrust existing LOX/LH_2 engine. Its GG engine cycle is relatively simple, the engine is heavy and large, and it costs less than a comparable staged combustion cycle engine, but its performance is lower. This RS-68 is the first LPRE that was fully designed on computers using computer-aided design and analysis programs.

The LPRE development for the Lance surface-to-surface missile started in 1964. It used a prepackaged storable propellant [inhibited RFNA (IRFNA) and UDMH] pressure-fed LPRE.[53,54] The 20.5-ft-long Lance missile, developed by the LTV Aerospace System's Division, had integral preloaded propellant tanks, a piston-type positive expulsion device in the oxidizer tank, and a solid propellant GG. The one-of-a-kind compact concentric dual-thrust chamber assembly (outer annular booster TC and smaller center sustainer TC) with its valves and TVC was developed by Rocketdyne, and it is shown in Fig. 28. The ablative TC liners were fastened to a forged steel outer wall. The pitch and yaw TVC was accomplished by pulsed-liquid-fuel side injection at four places on the outer nozzle exit of the booster engine. It is the first and only known production LPRE with liquid side injection. The booster thrust was 46,200 lbf (at sea level), chamber pressure at full flow was about 950 psi. The variable thrust of the sustainer (from 4400 down to 14 lb) was achieved by a movable pintle (with an ablative face) actuated by a servocontrol valve and fuel pressure. This large throttling ratio of 300 to 1.0 is the highest known anywhere in LPRE history. About 3229 engines were produced. The missile was deployed with the U.S. Army, and some Lance batteries were stationed in several overseas countries.

4. Propulsion Products Center, Northrop Grumman Corporation

Today the LPRE work is done at this Propulsion Products Center, which is part of the Space Technology Sector of Northrop Grumman in Redondo Beach, California. In the fall of 2002, the Space and Electronics Group of TRW including its propulsion capability was acquired by Northrop Grumman. For decades this center was part of TRW, Inc., where many different LPRE were developed.[60,61]

Today's Propulsion Center had its origin in the LPRE work done at the NASA Jet Propulsion Laboratory (JPL) in the mid-1950s.[61] Around 1960, several of the JPL engineers, who had worked on hydrazine monopropellant engines and the predecessors of the pintle injector, moved to the Space Technology Laboratory (the predecessor organization of TRW), and they continued their work there. The discussion of historic LPRE efforts will be limited to just a few items.

Inert cold-gas attitude control thrusters were historically the first TVC method used for steering space vehicles.[61] Several of the initial spacecraft (1958–1959) built at TRW used this reliable simple cold-gas system for attitude control, for example, in Pioneers 1 and 2, Nimbus, or the Earth Resources Satellites. Flights with their cold-gas systems continued sporadically on other spacecraft until about 1974. A number of other U.S. companies also built and flew cold-gas TVC systems.

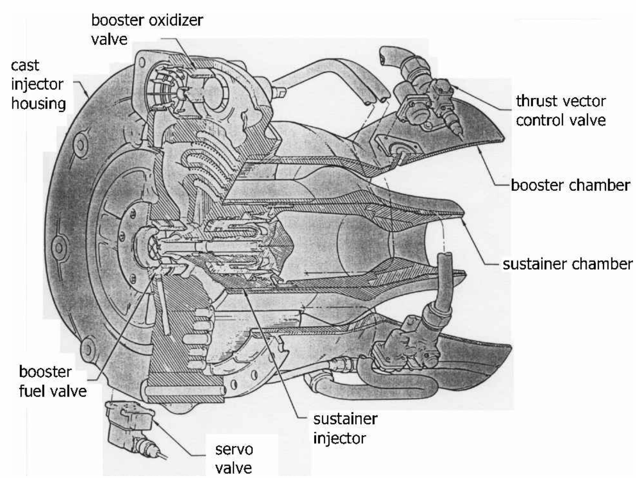


Fig. 28 TC assembly of Lance surface-to-surface missile, 19.4 in. high and 173 lb weight (courtesy, Rocketdyne).

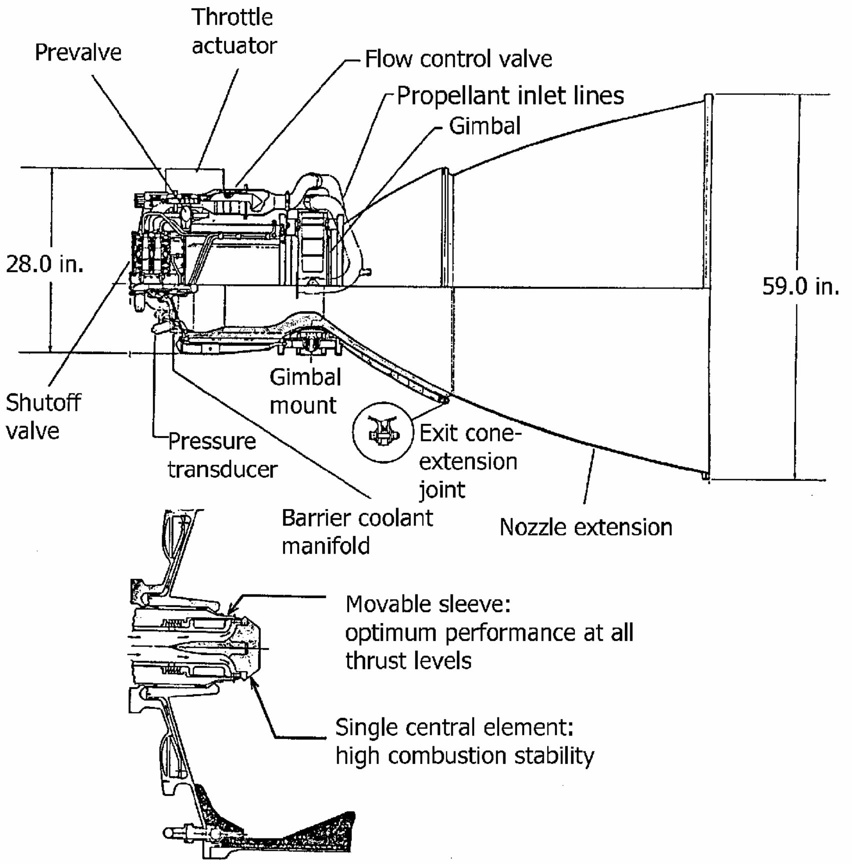


Fig. 29 Half-sections through the Lunar Module descent engine with a film-cooled ablative liner and an enlarged detail of the variable area pintle injector (courtesy, Northrop Grumman Corporation, Propulsion Products Center).

This was the first U.S. LPRE organization to fly hydrazine monopropellant LPREs with a gas-pressure-feed system.[61] It was first flown in the Able 4A (also known as Pioneer P-1) in 1959. Because a reliable catalyst did not exist in 1959, a slug start was used, that is, an initial injection of a small amount of NTO, a hypergolic oxidizer; this created the high gas temperature necessary for the subsequent thermal decomposition of the hydrazine. The next spacecraft, Able 4B, carried enough NTO for six slug starts. In the 1960s, Shell 405, an effective catalyst, became available, and a series of hydrazine monopropellant LPRE systems (with pulsing, multiple thrusters and a gas-pressure-feed system) were then developed. These LPREs were capable for periodic rapid pulsing operation for attitude control over a long period of time. Sizes from 0.1 to 150 lbf thrust were developed. Their pulsing hydrazine monopropellant engines were flown on Pioneer 6, 7, 8, and 9 spacecraft between 1965 and 1968 and in several military and NASA spacecraft since that time. In the last several years this center developed a unique TC that could operate interchangeably with bipropellants (NTO/hydrazine, 4–14 lbf thrust) or with hydrazine as a monopropellant (0.9–4 lbf thrust).[61] The first flight of this TC was on a GEO-LITE satellite in 2002.

This propulsion organization has refined the pintle injector technology over a period of 40 years and has designed and tested over 60 different pintle configurations over a thrust range between 5 and 650,000 lbf thrust.[61,62] Most have a fixed (not movable) pintle. An injector with a movable pintle sleeve is shown in Fig. 29 (Refs. 63–65). As of October 2001, there have not been any combustion instability incidence over a wide range of chamber pressures and with 25 propellant combinations. It gives good performance, but requires a relatively large combustion chamber volume. Eight of their LPREs with pintle injectors have flown, most of them with fixed (nonmovable) sleeves. Another feature of the pintle injector is the ability to shut off the propellant flow at the injector surface and to reduce the propellant dribble volume essentially to zero. The pintle sleeve acts as a dual valve. Most other small pulsing TCs have a small volume of propellant trapped between the injector surface and the valve seat, and this trapped propellant dribbles out and causes some afterburning at low thrust.

The historic lunar landing decent engine of the Apollo lunar module[63,66] (developed at predecessor TRW) had a throttling pintle injector and was capable of a 10–1 thrust reduction with only a 4% loss of specific impulse. This engine had an ablative TC with a metal nozzle exit extension and film cooling. It is shown in a section view in Fig. 29. The trick in deep throttling is to maintain a high injector pressure drop and the proper mixture ratio. This is accomplished by varying the annular injection area (by hydraulically positioning the single movable pintle sleeve) and at the same time vary the throat area of the two cavitating venturis with movable center bodies in the propellant feed lines. These venturis are labeled as flow control valves in Fig. 29. The cavitating venturis control the propellant flow and the mixture ratio at any particular thrust level. A relatively thick ablative liner with film cooling is used because regenerative cooling was not feasible at low thrust, as mentioned in Sec. III.D.1. The gimbal ring around the throat limits the space needed for turning the engine during pitch and roll maneuvers.

The U.S. Army and this center at Northrop Grumman have been leaders in the chemistry and application of gelled propellants and investigated different propellant formulations and operating characteristics with several experimental LPREs beginning TC tests in about 1983 (Ref. 67). Gelled propellants have additives that make them thixotropic (jellylike) materials. The merits are enhanced safety because it is less likely to leak, be spilled, or react violently to impact. By adding powdered aluminum or small carbon particles to the fuel, the fuel density and the combustion energy can be increased. Its principal disadvantages are poorer atomization and combustion efficiency, causing a small decrease in performance and a somewhat higher amount of residual (unused) propellant. Figure 30 shows

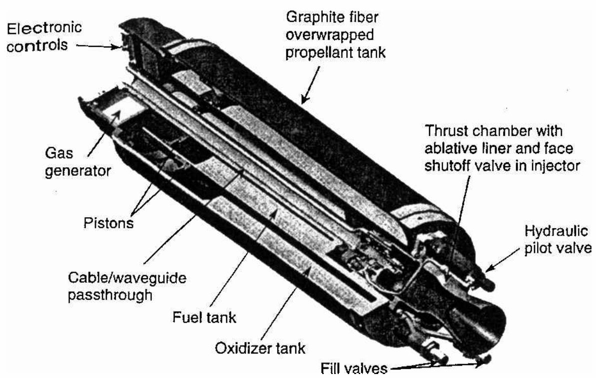


Fig. 30 Compact, prepackaged, preloaded LPRE for propelling experimental smart ground-to-ground missile (courtesy, Northrop Grumman Corporation, Propulsion Products Center).

a unique prepackaged LPRE (6 in. diameter) for an experimental smart missile, which is in an advanced development program.[68] It uses gelled propellants and a face shutoff-type pintle injector capable of some throttling and restarting. It includes features that allow a control of the total impulse and of the time of flight. It uses a solid propellant gas generator with multiple grains for tank pressurization under different ambient temperatures and ranges. Pistons in the concentric propellant tanks provide positive propellant expulsion and separate the oxidizer liquid from the fuel-rich gases of the GG. The propellants, IRFNA and MMH, are gelled, and the fuel has been loaded with suspended carbon particles making it denser and more energetic. It has flown successfully in two experimental launches from a military vehicle. There has as yet not been a production of a LPRE with gelled propellants, even though these propellants had been investigated for more than 30 years.

5. Pratt and Whitney, a United Technologies Company

The main product line of Pratt and Whitney, a United Technologies Company (P&W) was and is turbojet engines. The decision by P&W/UTC to enter into the LPRE business was made in 1957, and serious work began in 1958 (Ref. 69). This entry was based in part on their prior experience on rotating turbojet machinery and on having used, handled, pumped, and burned LH_2 in a special turbojet engine, which at the time was a secret project. In the past their work has concentrated on LOX/LH_2 LPREs.[70] The discussion is limited to three items.

The RL-10 LPRE was the most successful and historic engine of P&W. As of June 2003, the several versions of this LPRE logged more than 352 spaceflights and accumulated 659 in-space starts.[69–72] It was the first flying LPRE in the world to use LOX/LH_2 cryogenic propellants. The initial version had 15,000-lbf thrust. The initial application was the Centaur upper stage with two gimbaled RL10 engines. Design of the RL10 LPRE started in 1958. The first static engine firing occurred in 1959, and the first flight was launched in a Centaur upper stage in January 1963. Data of two very different models of this LPRE are given in Table 7.

The RL10 was the first flying LPRE in the world to use an expander engine cycle.[73] Its flow sheet is shown in Fig. 31. The gasified hydrogen (heated in the cooling jacket from −423 to about −165°F) powers the turbine of the TP, and the hydrogen turbine exhaust gas is then injected into the combustion chamber, where it is burned with the LOX. The two-stage turbine and the two-stage hydrogen pump are on one shaft, and the oxygen pump is driven at reduced speed through a gear case. The gear case and the main bearings are lubricated and cooled by oil. Subsequently P&W designed the fuel pump bearing to be cooled by LH_2, which had not been done before. A formed, double-tapered, and flattened set of 347 stainless-steel tubes is used for the cooling jacket of the thrust chamber and the nozzle. It is basically similar to the tubular TCs developed years earlier by RMI, Aerojet, and Rocketdyne. The injector featured multiple concentric injection elements with the oxygen coming through the small inside tubes and the gasified fuel coming through the annular sleeve

Table 7 Characteristics of two versions of RL10 rocket engines

Characteristic	Design year 1958	Design year 1997
Designation	RL10A-3	RL10B-2
Thrust in vacuum, lbf	15,000	24,750[a]
Chamber pressure, psia	300	644
Nozzle expansion are ratio	40	280[a]
Specific impulse (vacuum), s	427	465.5[a]
Mixture ratio	5.0:1	5.88:1
Design life (number firings/cumulative duration), h	100/1.25	300/10

[a]With extendible diverging nozzle segment.

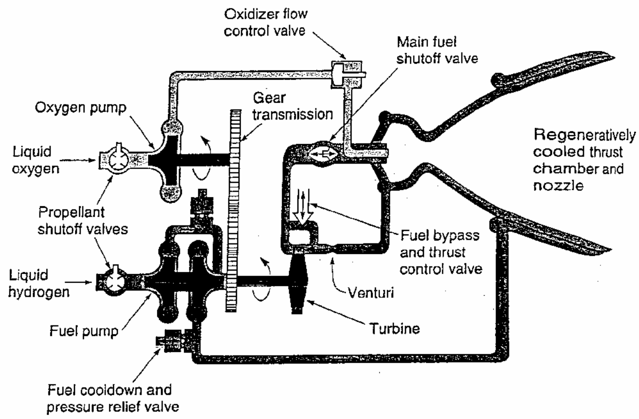
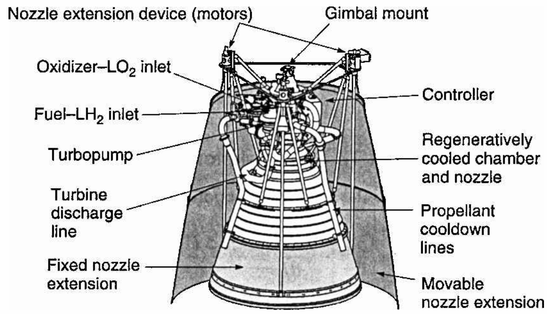


Fig. 31 Simplified schematic flow sheet of early version RL10 upper-stage LOX/LH_2 engine (courtesy, Pratt and Whitney, a United Technologies Company).

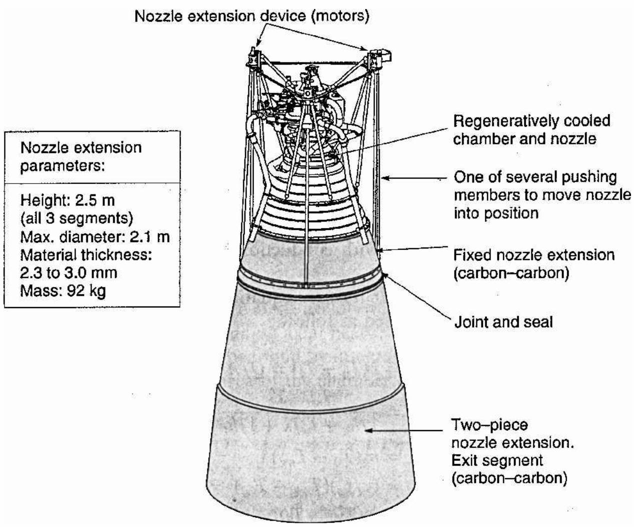
around it. It was patterned after a similar injector developed by the NASA Lewis Laboratory, now called NASA John H. Glenn Research Center at Lewis Field. An innovation attributed to P&W was the porous stainless-steel material (called Rigimesh). It was used for the injector face; hydrogen oozed through the pores of this material providing a transpiration-cooled surface. This basic injector design with the coaxial injection elements and the Rigimesh were used on subsequent U.S. LOX/LH_2 LPREs.

This engine had several improved and upgraded versions. The thrust was increased in steps until it reached the value shown in Table 7. It has seen service as a dual engine in the Centaur upper stage, was used in unmanned lunar landings, planetary flyby/orbiters, Mars lander, astronomical observatories, and a number of other space programs. A variable thrust version (throttled down to 30%) with four RL10 engines was developed for the McDonnel–Douglas experimental Delta Clipper (DC-X) vertical takeoff and vertical landing test vehicle. These engines have flown 12 times, but these flights are not included in the total number of spaceflights listed earlier. P&W was the first in the world to fly a large extendible nozzle with a high area ratio on LPREs The RL10B-2 engine shown in Fig. 32 is the most recent. This extendible nozzle concept was invented at the UTC Research Laboratory in 1948 and had been used by P&W in solid propellant upper-stage rocket motors beginning about a decade earlier. Its extendible nonporous nozzle segment was made of a three-dimensional weave of heat-resistant strong carbon fibers in a matrix of carbon. The two-piece extension was developed and built by Snecma in France. The movable nozzle extension is stowed around the engine during the ascending flight and then lowered into position after the dropoff of the lower vehicle booster stage, but before the second-stage engine starts at altitude. The RL10-B2 with the large extendible nozzle first flew in August 2000.

When Rocketdyne had technical problems with their own design of the SSME TPs (such as turbine blade cracking), NASA decided to



a) Half section of nozzle extension in stowed position



b) Nozzle extension in deployed position

Fig. 32 RL10B-2 LPRE with an extendible nozzle skirt (courtesy, Pratt and Whitney, a United Technologies Company).

have P&W evaluate the Rocketdyne turbopumps. P&W developed an alternate version of the high-speed multistage high-pressure LH_2 TP and the high-pressure LOX TP for assembly into the SSME. This engine has been developed and is being built by Rocketdyne. These alternate TPs are more reliable, interchangeable, and 350 lbs heavier than the original Rocketdyne versions. These large TPs are still being built by P&W and delivered to Rocketdyne for assembly into the SSME block 2.

About 10 years ago P&W reached an agreement with the largest and most experienced Russian LPRE developing organization NPO Energomash at Khimky (Moscow Region) to obtain a license and later also to produce some of the proven Russian LPREs for propelling U.S space launch vehicles. This includes the RD-170 (highest thrust existing engine, 1,770,000 lb) liquid oxygen-kerosene booster engine, the RD-180 (described later), and the RD-120 (187,400-lbf thrust). All run on LOX/kerosene. The RD 120 is a second-stage LOX/kerosene LPRE, and it was test fired at the P&W test facility in Florida in October 1995. It was the first test firing of a large Russian LPRE in a U.S. rocket static test facility.[74] Of these engines, only the RD-180 found a U.S. application. It is a scaled-down version of the RD-170, which has four hinged thrust chambers, but the newer RD-180 has only two gimbaled thrust chambers. The specific impulse is 311 s at sea level, 337.8 s (in vacuum), is 140 in. high, has a maximum diameter of 118 in. and weighs 12,081 lb (dry). It was developed by NPO Energomash of Moscow in cooperation with P&W. Figure 33 shows this engine. Its first flight in the Atlas 5 SLV was in August 2002. It has a sea level thrust of 860,000 lb (933,000 lb in vacuum), runs at a very high chamber pressure of over 3720 psia and a nozzle exit area ratio of 36.4:1.0. It uses a staged combustion cycle, an oxygen-rich preburner, an oxidizer lead during start with an initial flow of a hypergolic starting liquid, high-pressure helium for valve actuation and system purging, and two booster pumps. The thrust chamber has rectangular cooling channels, three film cooling injection slots, and its assembly requires brazing and multiple welds.

6. Liquid Propellant Division, ARC

The Liquid Propellant Division of ARC, a Sequa Company, has three sources for its technology.[75] The former Bell Aircraft Company started its work on LPRE in the 1950s; in 1960, Textron bought this company and established the Bell Aerospace Division as a separate organization. In 1983 ARC acquired its liquid propellant operation (together with its inventory, personnel, and facilities at Niagara Falls, New York) to supplement its own solid propellant rocket motor product line. The second source came to ARC in November 1997, when it acquired the Royal Ordnance Company of Westcott, England. With it they received some skilled people, facilities, a line of proven storable bipropellant modern apogee LPRE, and small thrusters for reaction control systems (RCSs), all operating with hydrazine as a fuel, and an entry into the European market. Pure hydrazine gives a few more seconds of specific impulse and a higher average propellant density than MMH or UDMH fuel, and these are needed in some high-performance spaceflight applications. The third source was the hydrazine monopropellant rocket engine product line originally developed by Hamilton Standard Division of UTC in the late 1960s. This product line had been acquired by Kaiser–Marquardt Corporation, but when this company merged into Primex in 2001, the government required Marquard to divest itself of this hydrazine line. ARC then acquired this Hamilton Standard monopropellant line, its designs, manufacturing know-how, test data, and customers. It filled a significant gap in the ARC thruster product line with reliable proven products.[75–77] Hamilton Standard started in this field in about 1964 and had qualified thrusters with integral valves, good performance, and endurance in sizes of 0.2-, 1.0-, 2.0-, 5.0-, 20-, and 100-lbf thrust. Figure 34 shows the 5-lbf thruster. It has a steady-state specific impulse of 231.5 s at 200-psi feed pressure. The reaction chamber contains the catalyst. A standoff perforated metal sleeve minimizes the heat conduction to the mounting flange and the valve. The electric heaters avoid freezing of the fuel and bad starts with a cold catalyst bed.

During the 1950s and 1960s, hydrogen peroxide monopropellant thrust chambers were produced and flown in several sizes between 1.0- and 500-lb thrust.[78,79] Over 2500 were built. A silverplated screen catalyst was used. Bell-ARC delivered such RCSs for of the X-1B and X-15 research airplanes, the first U.S. manned flights with the Mercury space capsule (1-, 6-, and 24-lbf thrusters), or the Centaur upper stage for SLVs (1.5-, 3-, and 50-lbf thrust). Most had pulsing capability. Hydrogen peroxide monopropellant is no longer used because of its low performance.

Bell Aerospace (ARC predecessor) developed a series of good bipropellant (mostly NTO and MMH) low-thrust radiation-cooled rocket engines for reaction control, attitude control, station keeping, or flywheel desaturation. One of them is shown in Fig. 35; it has some external thermal insulation to protect adjacent components from excessive thermal radiation and to keep the outer temperatures below 400°F. A single triplet set of injection holes is used, and a torque motor operates the bipropellant valve. After testing different materials, ARC settled on niobium for the chamber/nozzle walls because of its manufacturability and relatively low weight. A silicate coating to minimize oxidation by the hot reaction gases, and a process to apply this coating was selected from various alternatives. With valves closely mounted to the injector, the start time delay can be less than 5 ms, and the pulse duration can be as low as 10 ms for the low-thrust units. ARC now has a stable of small flight-proven bipropellant thrust chambers assemblies in sizes between 0.2 and 350 lb and today is a leader in this field. Beginning in August 1965, ARC has designed, built, and delivered several different completely integrated LPRE with tanks, pressurizing system and support structures, such as the Minuteman postboost propulsion system.

The Agena rocket engine (Fig. 36) was probably the best known and the largest developed by Bell/ARC.[80] It propelled an upper SLV stage (Agena), developed by the Lockheed Corporation in

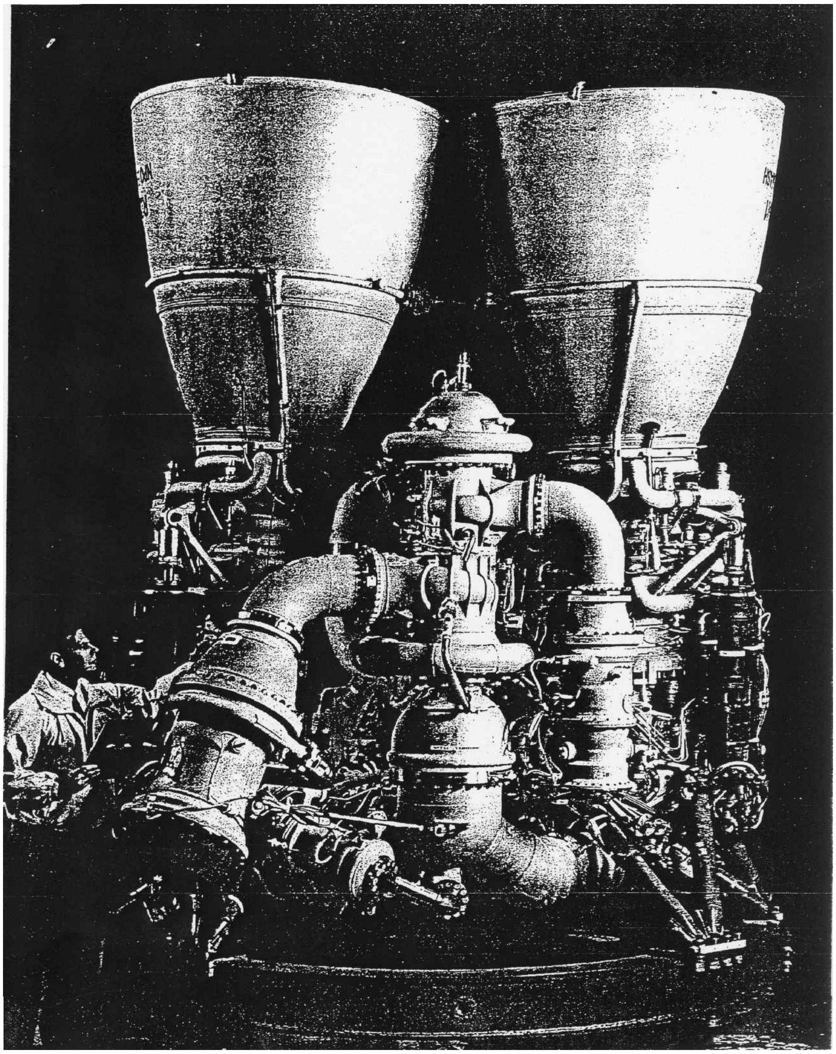


Fig. 33 RD-180 LPRE has two gimballed TCs (pointing up) central common TP, two booster pumps, high-pressure turbine exhaust pipe which feeds gas to the injectors; TVC actuator on right (courtesy, NPO Energomash and Pratt and Whitney, a United Technologies Company).

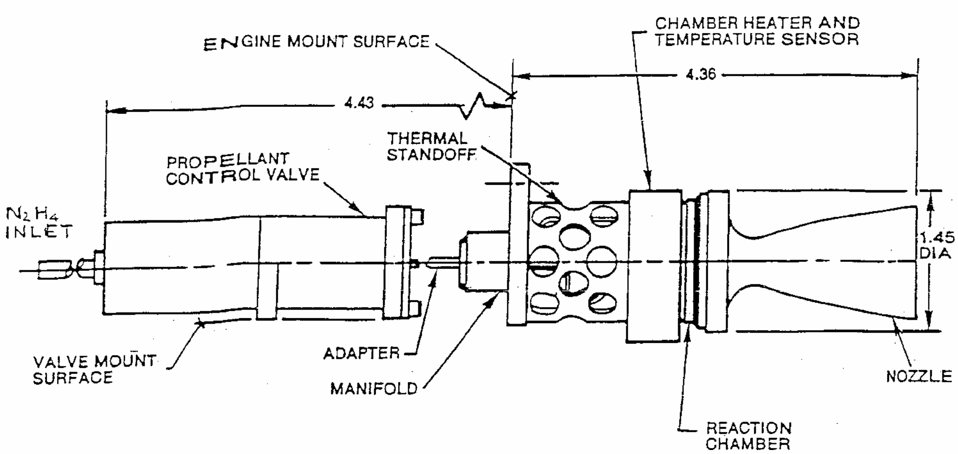


Fig. 34 Typical 5-lbf hydrazine monopropellant thruster with valve (courtesy ARC).

Sunnyvale, California. The TP-fed engine used a GG cycle and NTO/UDMH as its hypergolic propellants. An early version used high-density nitric acid (with a high percentage of dissolved NO_2) and UDMH. It has a unique aluminum (6061 T6) thrust chamber (oxidizer cooled) with a relative thick wall, which contains long drilled holes (inclined to the axis) as cooling passages. The radiation-cooled nozzle exit section (between area ratio of 12 and 45) is made of titanium reinforced externally with molybdenum stringers and hoops (an egg crate pattern). Two ARC positive expulsion piston tanks provide multiple restart capability in space to feed the GG. Cavitating venturis in the feed lines leading to the GG provided the control of the GG propellant flow and, thus, the thrust level. The propellant centrifugal pumps were geared to the turbine. The Agena engine Model 8096 has these characteristics: altitude thrust 16,000 lbf, altitude specific impulse 290 and later 300 s, chamber pressure 506 psia, chamber mixture ratio 2.8, GG mixture ratio 0.15, dry weight 296 lb, nozzle area ratio 45, exit diameter 32.5 in., and a height of 83 in. A total of 418 Agena engines have been produced and 363 have flown with only one engine failure. This LPRE propelled many different payloads, including the first U.S. satellite in a circular low orbit, the first into a polar orbit, and the first to perform a significant orbit velocity vector change. It also was the first large upper-stage engine with a vacuum restart and the first to participate in a space rendezvous and docking operation with a manned spacecraft, namely, the Gemini. A version of an Agena engine was also ground tested with liquid fluorine and ammonia as propellants.

VI. General Findings, Comments, and Conclusions

The LPRE field and its technology are today essentially mature. The basic engine system and key components had been fairly well defined about 40 or 50 years ago. High reliability numbers are recorded by all LPREs in production. A large amount of data is available. For any particular kind of new LPRE, there are today two or more organizations in the United States that can develop it with confidence. Certainly there have been recent new technical ideas that have improved LPREs, such as better materials, lighter extendible nozzles, or simpler TPs. There have also been a few new potential requirements, such a microminiaturized LPREs, combining LPREs with other means of propulsion, or reusable LPRE for reusable strapon boosters. There are still areas of this technology where R&D can lead to further improvement. However, the opportunities for developing a truly new LPRE are today not as plentiful as they used to be.

The progress in the technology of LPREs has been truly remarkable in the last 82 years. Many of the technical milestones and key LPREs have been discussed in this paper. The rate of innovation, introducing new materials or new designs, was higher in the first few decades of this history than it has been in the more recent decades.

The emphasis on the engine development criteria has changed. In the early decades the LPRE investigators were happy if the engine held together, ran for at least 10 s, and did not fail. Making it work was the key objective. Soon the emphasis shifted to running for more than a minute. Then the aims became maximum possible performance, high reliability, safety, reduced costs, and for some applications also long life. Making it environmentally compatible has become a goal more than a decade ago. Today the emphasis is still on these same criteria, but more attention is being given to cost.

The United States has the distinction of being identified with the American Robert H. Goddard, a most important early pioneer in this field, the first to build and run a LPRE and the first to launch a vehicle with a LPRE. His technical contributions are legendary. It is unfortunate that many of his inventions did not reach the U.S. industry in a timely manner.

Current space-related applications can only be accomplished well by using LPREs. A number of the flight applications, for which LPRE were indeed appropriate, are obsolete today. In some applications they have been replaced by other means of propulsion. For example, we no longer use LPREs for JATO, aircraft superperformance, or sounding rocket vehicles. We are no longer building LPRE for ballistic missiles. The emphasis today is on applications for space launch vehicles, spacecraft maneuvers, or reaction control for the steering of flying vehicles.

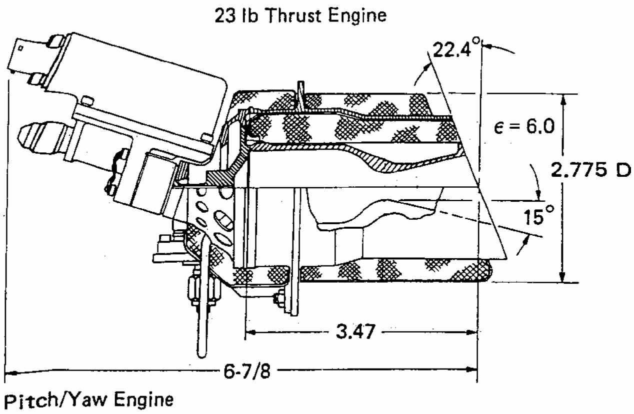


Fig. 35 TC assembly 23-lbf thrust (of the Minuteman III postboost control propulsion system) operates on NTO/MMH (courtesy, ARC).

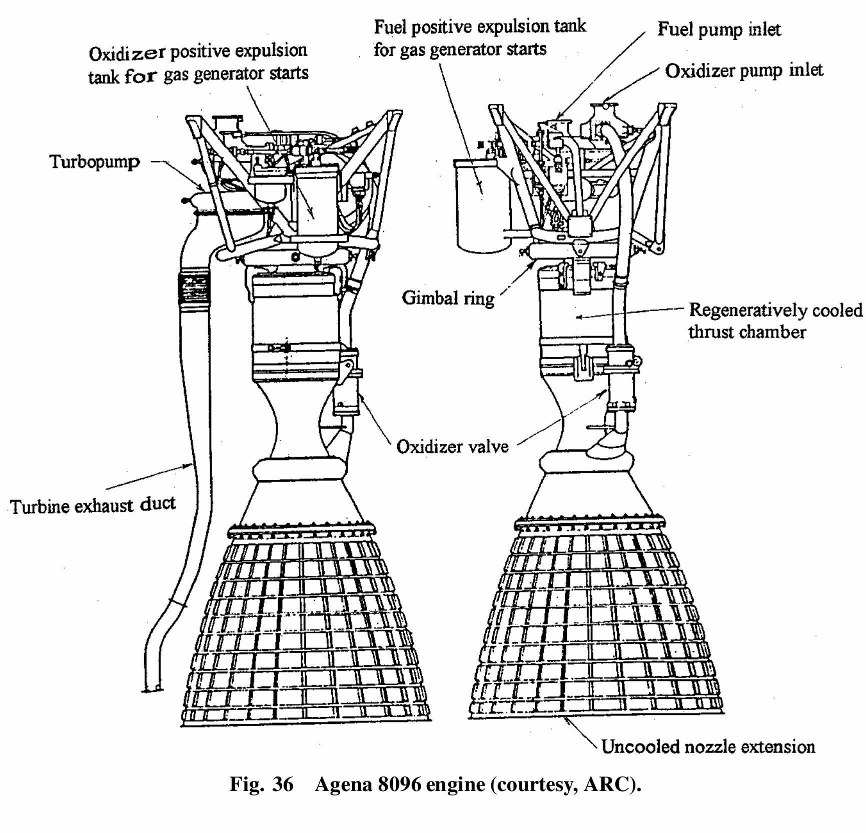


Fig. 36 Agena 8096 engine (courtesy, ARC).

In the last 30 years, the industry has settled on a few specific practical propellant combinations, each for a specific type of LPRE or application. A couple of propellants with potential future benefits are still being evaluated. In the first 50 years of the U.S. LPRE history, about 170 different liquid propellants have been evaluated and many have found their way into experimental TCs and/or LPREs. More than 25 combinations have flown. However, most propellant combinations are no longer used.

The LPRE business in the United States has seen its peak in the late 1950s to the early 1970s. This was the period when LPRE employment and sales were at their highest. Although the volume of new LPRE developments has greatly diminished, there is today still a lot of activity worldwide. The decrease in business has brought about mergers, acquisitions, hirings or layoffs, cooperative agreements, closures, and consolidations, and many of these were described. There are fewer U.S. companies and fewer people engaged with LPREs today. To date the capability to develop and produce has been maintained.

In the last 70 years a lot of research related to LPREs has been done at U.S. universities, government or company laboratories. Some of the research projects have indeed been helpful and resulted in a better understanding of the physical or chemical phenomena, the materials, or the analysis of LPRE related subjects. One university's R&D has led directly to the start of one of the key LPRE companies and to a government-supported laboratory. However, this author believes that most of the university research related to LPREs in the last 35 years has not been directly useful or only marginally helpful in making a better or novel LPRE. There have been some exceptions. The propulsion designer or typical engineering manager had not been very aware of the research that had been conducted, had not read a Ph.D. dissertation or research report, and had not interacted with the researchers. However, several U.S. universities have been effective in educating professional personnel for work in the LPRE field.

It took a few years for each company to become proficient in this business. Each organization had to learn about LPREs from scratch and acquire or develop a suitable set of knowledgeable employees, computer programs, facilities, and laboratories. Each usually made mistakes, experienced engine failures, and had to repeat some work, which had already been done well by others.

The work done by the LPRE industry has typically become more efficient. Efficiency measurements, such as sales per employee, have generally risen. This is in part due to the learning curve (better educated and more experienced personnel) and major advances in computers, databases, and software programs for engineering, manufacturing, or management. It is only recently that one company designed a new LPRE completely by computer.

As in many other technical endeavors, the techniques for design, manufacture, and testing have drastically changed. This author remembers having designed engines in the 1940s using a slide rule, inked drawings on vellum, and a handwritten notebook and many of the manufacturing instructions were verbal. It has gradually changed to today's sophisticated computer-aided design techniques, which often tie into a computer-aided manufacturing system, an accounting system, or a data management system. Test recording, data analysis, and data displays are now largely computerized. Engineering analysis, such as heat transfer, plume behavior, or combustion vibration, can all be helped by computer programs. The control of vendor performance, material traceability, and product quality is routinely assisted by computers. Computer simulation has reduced the amount of testing and manufacturing that used to be required. The way business is being conducted has changed radically during the 82-year LPRE history.

This author started in this LPRE business 60 years ago. It has been exciting to watch the evolution of this cutting-edge technology. These six decades have brought forth major new milestones and amazing achievements that a life time ago were believed to have been fantasy or impossible. The propelling of vehicles to the moon and other space destinations, the harnessing of enormous propulsive power, the routine handling of hazardous propellants, the dispensing of precise small impulse bits, and the first flight through the sound barrier have been remarkable accomplishments. The space age could not have happened without LPREs. The dramatic advances of this technology and the flight progress that LPREs have enabled are indeed amazing and a source of satisfaction to the author and to others.

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References

[1]Huzel, D. K., and Huang, D. H., *Design of Liquid Propellant Rocket Engines*, revised ed., Vol. 147, *Progress in Astronautics and Aeronautics*, AIAA, Reston, VA, 1992.

[2]Sutton, G. P., and Biblarz, O., *Rocket Propulsion Elements*, 7th ed., Wiley, New York, 2000.

[3]Herrick, J., and Burgess, E. (eds.), *Rocket Encyclopedia, Illustrated*, Aeropublishers, Los Angeles, 1959.

[4]*Jane's Space Directory*, 16th and 12th eds. Launch Vehicle Propulsions Sections, Jane's Information Group, Ltd., Coulsdon, England, U.K., 1996–1997, 2000–2001.

[5]Baker, D., *Space Flight and Rocketry, a Chronology*, Facts on File, Inc., 1996.

[6]Ordway, F. I., and von Braun, W., *History of Rocketry and Space Travel*, T. Y. Crowell Publishers, New York, 1975.

[7]Doyle, S. E. (ed.), *History of Liquid Rocket Engine Development in the United States 1955–1980*, American Astronautical Society, AAS History Series, Vol. 13, Univelt, Inc., San Diego, 1992.

[8]Ehricke, K. A., "A Comparison of Propellants and Working Fluids for Rocket Propulsion," *Journal of the American Rocket Society*, Vol. 23, No. 5, 1953.

[9]*Chemical Rocket Propellant Hazards*, Vol. 1, *General Safety Engineering Design Criteria*, CPIA Publ. 194, Chemical Propulsion Information Agency, Columbia, MD, 1971.

[10]*Liquid Propellant Manual M4*, Chemical Propulsion Information Agency, Columbia, Maryland, 2000.

[11]Knuth, E. L., "The Mechanics of Film Cooling," *Journal of the American Rocket Society*, Nov.-Dec. 1954.

[12]Bartz, R. R., "Survey of Relationships between Theory and Experiment for Convective Heat Transfer in Rocket Combustion Gases," *Advanced in Rocket Propulsion*, edited by S. S. Penner, AGARD, Technivision Services, Manchester England, U.K., 1968.

[13]Rao, G. V., "Exhaust Nozzle Contour for Optimum Thrust," *Jet Propulsion*, Vol. 28, No. 6, 1958.

[14]Hageman, G., Immich, H., Nguyen, T. V., and Dumnov, E., "Advanced Rocket Nozzles," *Journal of Propulsion and Power*, Vol. 14, No. 5, 1998, pp. 620–634.

[15]Kinkaid, J. S., "Aerospike Evolution," *Threshold, an Engineering Journal*, No. 18, Rocketdyne Propulsion and Power, Spring 2000, pp. 4–13.

[16]Strangeland, M. L., "Turbopumps for Liquid Rocket Engines," *Threshold, an Engineering Journal of Power Technology*, No. 3, Rocketdyne Propulsion and Power, Summer 1988, pp. 34–42.

[17]Ross, C. C., "Principles of Rocket Turbopump Design," *Journal of the American Rocket Society*, No. 84, 1951.

[18]Minnick, A., and Perry, S., "Design and Development of an Advanced Liquid Hydrogen Turbopump," AIAA Paper 98-3081, July 1998.

[19]Yang, V., and Anderson, W. (eds.), *Liquid Rocket Engine Combustion Instability*, Vol. 169, Progress in Astronautics and Aeronautics, AIAA, Reston, VA, 1995.

[20]Oppenheim, B. W., and Rubin, S., "Advanced POGO Analysis for Liquid Rockets," *Journal of Spacecraft and Rockets*, Vol. 390, No. 3, 1993.

[21]Reardon, F. H., "Combustion Stability Specifications and Verification Procedure," Publ. 247, Chemical Propulsion Information Agency, Laurel, MD, Oct. 1973.

[22]Lehman, M., *This High Man, The Life of Robert H. Goddard*, Farrar, Strauss and Co., New York, 1963.

[23]Goddard, E. C., and Pendray, G. E. (eds.), *The Papers of Robert H. Goddard*, Vol. 1, 1898–1924; Vol. 2, 1925–1937, Vol. 3, 1938–1945; McGraw–Hill, New York, 1970, p. 1707.

[24]Goddard, E. C., and Pendray, G. E., *Robert H. Goddard, Rocket Development, Liquid Fuel Rocket Research, 1929–1941*, Prentiss–Hall, Englewood Cliffs, NJ, 1948 (reissued 1961); also Goddard, R. H., "Liquid Propellant Rocket Development," *Smithsonian Miscellaneous Collections*, Vol. 95, No. 3, 1936.

[25]Goddard, R. H., "A Method for Reaching Extreme Altitudes," *Smithsonian Miscellaneous Collections*, Vol. 71, No. 2, 1919.

[26]Winter, F. H., *Prelude to the Space Age, The Rocket Societies: 1924 to 1940*, National Air and Space Museum, Smithsonian Inst., Washington, DC, 1983.

[27]"The Flight of Experimental Rocket No. 2" *Astronautics*, No. 26, 1933, pp. 1–11.

[28]Pendray, G. E., "Early Rocket Developments of the American Rocket Society," *First Steps Toward Space*, Vol. 6, American Astronautical Society, Univelt, San Diego, CA, Chap. 14; also Truax, R. C., "Annapolis Rocket Motor Development," *First Steps Toward Space*, Vol. 6, American Astronautical Society, Univelt, San Diego, CA, Chap. 27; also Durant, F. C., III, and James, G. S. (eds.), *First Steps toward Space*, American Astronautical Society History Series, Vol. 6, 1985.

[29]Truax, R. C., "Liquid Propellant Rocket Development by the U.S. Navy during World War II: A Memoir," American Astronautical Society History Series, Vol. 12, Univelt, San Diego, CA, 1991, pp. 326–332.

[30]Malina, F. J., "The U.S. Army Air Corps Jet Propulsion Research Project, GALCIT Project No. 1, 1939–1946: A Memoir," 3rd History Symposium of the International Academy of Astronautics, 1969.

[31]von Kármán, T., and Edson, L., *The Wind and Beyond, Theodore von Kármán*, Little Brown, Boston, 1967.

[32]Summerfield, M., Powell, W., and Crofut, E., "Development of a Liquid Jet Unit and its Operation on an A-20 Aiplane," Jet Propulsion Lab., JPL Rept. 1–13, California Inst. of Technology, Pasadena, CA, Sept. 1942,

[33]Shesta, J., "Reaction Motors, Inc., the first Large Scale American Rocket Company: A Memoir," 12th History Symposium of the International Academy of Astronautics, Oct. 1978; also American Astronautical Society History Series, Vol. 19, Univelt, San Diego, CA, 1995, Chap. 7.

[34]Ordway, F. I., III, and Winter, F. H., "Reaction Motors, Inc.: A Corporate History, 1941–1958, Part I, Institutional Development," American Astonautical Society History Series, Univelt, San Diego, CA, 1994, Chap. 7; also Winter, F. H., and Ordway, F. I., "Part II, Research and Development Efforts," American Astronautical Society History Series, Univelt, San Diego, CA, 1994, Chap. 8; also Ordway, F. I., "Reaction Motors Division of Thiokol Corporation, an Operational History, 1958–1972 (Part II)," American Astronautical Society History Series, Vol. 12, Univelt, San Diego, CA, 1991, Chap. 11; also Winter, F. H., "Reaction Motors Division of Thiokol Corporation, A Project History, 1958–1972 (Part III)," American Astronautical Society History Series, Vol. 12, Univelt, San Diego, CA, 1991, Chap. 12.

[35]Winter, F. H., "Black Betsy," The 6000C-4 Rocket Engine, 1945–1989—Part 2," American Astronautical Society History Series, Vol. 17, Univelt, San Diego, CA, 1991, Chap. 11.

[36]Winter, F. H., and James, G. S., "Highlights of 50 Years of Aerojet, A Pioneering American Rocket Company," *History of Rocketry and Astronautics*, edited by P. Jung, American Astronautical Society History Series, Vol. 22, Univelt, San Diego, CA, 1998, Chap. 4.

[37]Umholtz, P., "Liquid Rocketry," *Aerojet, The Creative Company*, Aerojet History Group, Sacramento, CA, 1995, Chap. 3.

[38]Personal communications, Fisher, C. B., 2001; also Nguyen, T. V., 2002; also Calhoun, D., and Mueggenburg, H. H., 1990s.

[39]Schnare, C. W., "Development of ATO and Engines for Manned Aircraft," American Rocket Society Paper 2088-61, 1961.

[40]Ehresman, C. M., "Liquid Rocket Propulsion Applied to Manned Aircraft in Historical Perspective," AIAA Paper 91-2554, June 1991.

[41]Stiff, R., "Storable Liquid Rockets," AIAA Paper 67-977, Oct. 1961.

[42]Aerojet's Data sheets (several) and Aerojet Manuals, Engines of Titan II (1983), Titan III SLV (1992), and the Titan IV (2000).

[43]Meland, L. C., and Thompson, F. C.,"History of the Titan Liquid Rocket Engines," AIAA Paper 89-2389, June 1989; also Adams, L. J.,"The Evolution of the Titan Rocket—Titan I to Titan II," American Astronautical Society History Series, Vol. 19, Univelt, San Diego, CA, 1997.

[44]Limerick, C., et al., "On Overview of the K-1 Reusable Propulsion System," AIAA Paper 97-3120, July 1997.

[45]Personal communications, Stechman, R. C., Galbreath, J. J., and Morgan, O., Aerojet Redmond Center, formerly General Dynamics, 2001, 2002, 2003.

[46]Product brochures and data sheets, Marquardt Co., Van Nuys, CA, 1990, Primex Aerospace Co. Redmond, WA, 1997, and General Dynamics Corp. Redmond, WA, 2001.

[47]Morgan, O. M., and Meinhardt, D. S., "Monopropellant Selection Criteria—Hydrazine and other Options," AIAA Paper 95-2595, June 1995.

[48]Schmidt, E. W., *Hydrazine and its Derivatives, Preparation, Properties, Applications*, Wiley, New York, 1984 (2nd rev. ed. 2002).

[49]Stechman, R. C., and Sumpter, D., "Development History of the Apollo Reaction Control System Rocket Engine," AIAA Paper 89-2388, July 1989.

[50]Murphy, J. M., "Liquid Rocket Propulsion System Advancements, 1946–1970," 14th History Symposium of the International Academy of Astronautics, Sept. 1980.

[51]Iacobellis, S., "Liquid Rocket Engines, Their Status and Their Future," AIAA Paper 66-828, 1966.

[52]Vilja, J., "Rocketdyne Advanced Propulsion System Overview," AIAA Paper 97-3309, 1997.

[53]Personal communications, Wheelock, V., Schuster, E., Minami, H., Halchak, J., and others, Rocketdyne, 1997–2003.

[54]Data sheets or data folder on Lunar Module Ascent Engine, Redstone Engine, F-1, J-2, H-1, Atlas, Thor/Delta, Lance, and others, The Boeing Co., Rocketdyne, Propulsion and Power, Conogu Park, CA, 1987–2003.

[55]Smith, R. A., and Minami, H. M., "History of the Atlas Engine," edited by S. E. Doyle, American Astronautical Society History Series, Vol. 13, Univelt, San Diego, CA, 1992, Chap. 3.

[56]Biggs, R. E., "Space Shuttle Main Engine—the First Ten Years," edited by S. E. Doyle, American Astronautical Society History Series, Vol. 13, Univelt, San Diego, CA, 1992, Chap. 4.

[57]Colbo, H. I., "Development of the Space Shuttle Main Engine," AIAA Paper 79-1141, July 1979; also Harris, S. L., "Block II: The New Space Shuttle Main Engine," AIAA Paper 96-2853, 1996.

[58]Mattox, R. M., and White, J. B., "Space Shuttle Main Engine Controller," NASA TP 1932, 1981, p. 19; also Seitz, P. F., "Space Shuttle Main Engine Control System," Society of Automotive Engineers, SAE Paper 73-0927, 1973.

[59]"RS-68 First Flight Edition," *Threshold, an Engineering Journal*, No. 20, the Boeing Co., Rocketdyne Propulsion and Power, Winter 2003; also Conley, D., Lee, N. Y., Portanova, P. L., and Wood, B. K., "Evolved Expendable Launch Vehicle System: RS-68 Main Engine Development," 53rd International Astronautical Congress, Paper IAC-2.S.01, Oct. 2002.

[60]Sackheim, R. L., "TRW's Family of High Performance Spacecraft Engines: Status and Impact," 1996 JANNAF Propulsion and Joint Subcommittee Meetings, Albuquerque, NM, Chemical Propulsion Information Agency, Laurel, MD, Dec. 1996.

[61]Personal communications, Dressler, G. A., Sackheim, R. L., and Hodge, K. F., TRW, 1997, 2000, 2001, and 2002.

[62]Dressler, G. A., and Bauer, J. M., "TRW Pintle Engine Heritage and Performance Characteristics," AIAA Paper 2000-3871, July 2000.

[63]TRW Data Sheet on Lunar Module Descent Engine, 1979, and on other LPREs, Redondo Beach, CA, 1999.

[64]Mayer, N. L., "Advanced X-Ray Astrophysics Facility—Imaging (AXAF-I) Propulsion System," AIAA Paper 96-2869, July 1996.

[65]Chazen, M. L., "High Performance Bipropellant Rhenium Engine," AIAA Paper 98-3356, 1998.

[66]Staudhammer, E. P., et al., "The Descent Engine for the Lunar Module," AIAA Paper 67-521, 1967.

[67]Natan, B., and Rahami, S., "The Status of Gel Propellants in the Year 2000," *Combustion of Energetic Materials*, edited by K. K. Kuo and L. deLuca, Begel House, Boca Raton, FL, 2001.

[68]Hodge, K. F., Crofoot, T. A., and Nelson, S., "Gelled Propellants for Tactical Missile Application," AIAA Paper 99-2976, June 1999.

[69]Mulready, D., *Advanced Engine Development at Pratt & Whitney, the Inside Story of Eight Special Projects, 1946 to 1971*, Society of Automotive Engineers, Warrendale, PA, 2001.

[70]Heald, D., "LH_2 Technology was Pioneered on Centaur 30 Years Ago," American Astronautical Society History Series, Vol. 21, Univelt, San Diego, CA, 1997.

[71]Personal communications, Pignoli, C., Pratt and Whitney Aircraft, United Technologies Corp., 2001 and Mills, P., Pratt and Whitney, a United Technologies Co., 2003.

[72]Tucker, J. E., "The History of the RL10 Upper Stage Rocket Engine," edited by S. E. Doyle, American Astronautical Society History Series, Vol. 13, Univelt, San Diego, CA, 1992, Chap 5; also Santiago, R.,"Evolution of the RL10 Liquid Rocket Engine for a New Upper Stage Application," AIAA Paper 96-3013, 1996.

[73]Brown, J. R., "Expander Cycle Engines for Shuttle Cryogenic Upper Stages," AIAA Paper 83-1311, 1983.

[74]Croteau, M. C., Garbowski, R. C., and Minick, A. B., "Live Fire Testing of the NPO Energomash RD-120 Rocket Engine at Pratt & Whitney," AIAA Paper 96-2608, 1996; also Data sheets on the RD-170, RD-180, and RD-120 engines, Pratt and Whitney Aircraft, United Technologies Corp., West Palm Beach, FL, 1997.

[75]Personal communications, Marvin, M., and Boorady, F., ARC, 2001, 2002, and 2003; also ARC brochures on several products, and company historical information.

[76]Marcus, M., "Historical Evolution of the Hydrazine Thruster Design," AIAA Paper 90-1838, July 1990; also Polythress, C. A., and Conkey, G. H., III, "Development of a Long Life 125-lbf Hydrazine Thruster," AIAA Paper 80-1170, 1980; also Francis, A. E., "Qualification Test for the Multiple Satellite Dispenser 0.2 lbf Rocket Engine Assembly," Technical Rept. Oct. 1973, revised May 1974.

[77]Saenz, A., Jr., "Life Performance Evaluation of 22 N (5 lbf) Hydrazine Attitude Control Engines," U.S. Air Force Rocket Propulsion Lab., USAF/RPL Technical Rept. Nov. 1980.

[78]E. Gribbon, E., Driscoll, R., and Marvin, M., "Demonstration test of a 15 lbf Hydrogen Peroxide (90%) Thruster," AIAA Paper 2000-3249, 2000.

[79]Bloom, R., Davis, N. S., and Levine, S. D., "Hydrogen Peroxide as a Propellant," *Journal of the American Rocket Society*, Vol. 80, March 1950.

[80]DeBrock, S. C., and Rudey, C. J., "Agena Primary and Secondary Propulsion Systems," *Journal of Spacecraft and Rockets*, Vol. 11, No. 11, 1974, pp. 769–777.