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MILESTONES IN CRYOGENIC LIQUID PROPELLANT ROCKET ENGINES

by

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Abstract

This paper reviews the milestones achieved with cryogenic liquid propellant rocket engines, discusses current technology improvement programs, and projects future engine designs. During the last two decades, these cryogenic rocket engines have played a major role in rocketry and achieved numerous important milestones. These engines power the Vanguard, Redstone, Thor, Atlas, and Titan I vehicles, the Saturn I and Uprated Saturn I vehicles, and will soon be employed in the Saturn V for the Apollo missions. The requirements dictated by these vehicles have necessitated growth from the 27,000-pound-thrust Vanguard engine to the 7,600,000-pound-thrust booster cluster for the Saturn V. Gains in specific impulse have also been significant. The successful application of liquid hydrogen in the Centaur and Saturn upper-stage rocket engines was a major achievement.

Simultaneously, significant improvements in engine technology, lightweight components, compact packaging, and advanced fabrication processes have been accomplished. Equally important, but somewhat less in the limelight, are the advancements in the basic operation of the engine through simplification and sophistication of the power cycle and the control system. These improvements are key factors in the achievement of operational flexibility, higher thrust, increased specific impulse, and high reliability.

Advanced concepts for cryogenic liquid propellant rocket engines now under investigation are reviewed. These efforts are leading to further improvements in performance, versatility, and packaging for future cryogenic rocket engines.

Introduction

Cryogenic liquid propellant rocket engines have played a major role in the total rocket propulsion program over the past years. Rocket engines of this type powered the early high-altitude sounding rockets, the first long-range ballistic missiles, and the first orbital and space flights. Dr. Robert H. Goddard in the 1920's used liquid oxygen, a cryogenic, and gasoline, a noncryogenic, for his rocket engines. His historical first launching of a liquid-fueled rocket on 16 March 1926(1) was truly a major achievement. The development of the liquid oxygen/alcohol powered German V-2 in the late 1930's and early 1940's was a further advancement in rocketry. These programs provided many of the basic innovations and design concepts which have been used in the following generations of cryogenic liquid propellant rocket engines. The early experimental development and flights of Dr. Goddard's rockets and the development of the V-2 have been described in detail in numerous papers and books.(2-7) This review is devoted to the United States rocket engine development efforts which followed these programs.

The scope of the paper is, as the title indicates, limited to liquid propellant rocket engines using cryogenic propellants. Thus, some significant achievements in liquid rockets are not covered, such as development of the first upper-stage engine (the Vanguard second stage which used the noncryogenic propellant combination of unsymmetrical dimethylhydrazine and nitric acid). However, in retrospect, the review has shown that many of the important achievements in engine development, vehicle flights, and engine technology were made with the cryogenic propellant engines.

The Milestones

Reviewing the milestones of the cryogenic liquid propellant powered rocket vehicles is useful in that it shows not only what has been accomplished, but the trends of this development--how the designs of today have evolved. These trends in development give insight into the possible evolution of future designs, their potential features, and configurations.

A summary review of the programs, vehicles, and flights of cryogenic liquid propellant rocket engines from the Hermes to the Saturns is presented in the Milestones Chart (Appendix A). Shown are the dates when the programs were initiated, when the first engines test (static) was accomplished, when the first vehicle was flown, and significant launches.

Thrust

In reviewing these programs and their engine designs, the first engine parameter that manifests itself is thrust. Thrust growth for booster and upper-stage rocket engines using cryogenic propellants is presented in Fig. 1. In the early years, engine thrusts were in the 20,000- to 75,000-pound thrust range. The growth from the early 20,000-pound-thrust level to the Saturn V's 7,600,000-pound thrust was a many-hundred-fold increase accomplished in less than 25 years.

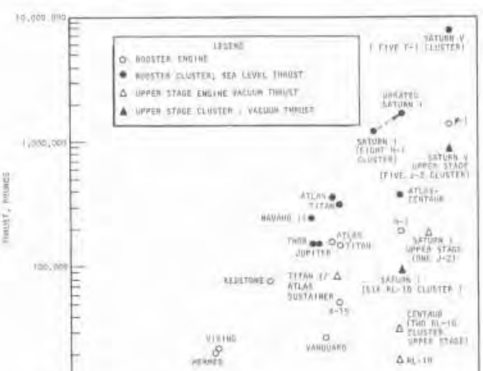


Figure 1. Thrust Growth in Cryogenic Liquid Propellant Rocket Engines

Both single-unit and engine-cluster thrusts are presented in Fig. 1; both are significant. The single-unit thrust provides a guide to progress in engine development. The cluster thrust indicates the rapid gain in vehicle thrust requirements achieved through the development of engines which could be clustered, thereby enabling rapid vehicle growth. Also, the concept of clustering engines has proved to be an effective method to limit propulsion development costs.

Specific Impulse

The trend in demonstrated engine specific impulse is presented in Fig. 2. For booster engines, sea level performance is shown. For upper-stage engines, the vacuum performance is of importance.

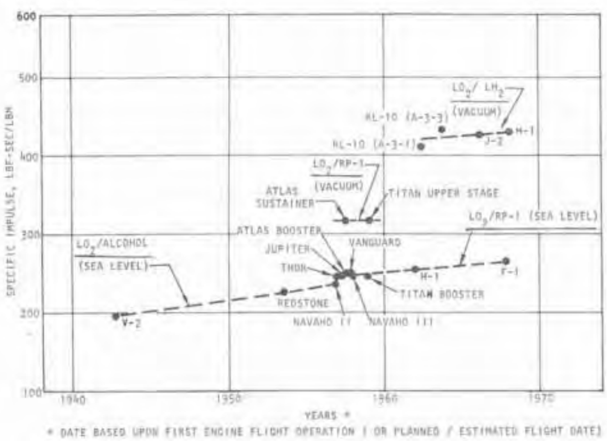


Figure 2. Specific Impulse Growth in Cryogenic Liquid Propellant Rocket Engines

A significant specific impulse performance gain resulted when the kerosene- (or RP) type fuel was substituted for the alcohol/water mixture used in the early engines. The Navaho III booster, started in 1956, was the first high-thrust engine to use these propellants. Steady performance growth for the kerosene-fueled engines occurred as a result of higher nozzle performance (achieved by a combination of higher area ratios and chamber pressures), higher efficiencies of combustion, and the engine power cycle.

However, the growth in specific impulse performance is dominated by the gains achieved when liquid hydrogen fuel was introduced. The high performance of rocket engines using these propellants permits many advanced mission concepts which are not feasible with lower performance propellants.

Chamber Pressure

Looking further into the details of a rocket engine design, combustion chamber pressure is a principal item of interest. The trend in chamber pressure over the years is shown in Fig. 3. The chamber pressure (together with area ratio) directly controls the size of an engine; or in converse, a higher chamber pressure permits an engine with higher performance and a higher area ratio nozzle to be designed without enlarging its envelope.

Detracting from high chamber pressure performance gains is, of course, increased engine weight to accommodate the higher pressure, the more powerful and heavier turbomachinery and the increased turbine drive power requirements.

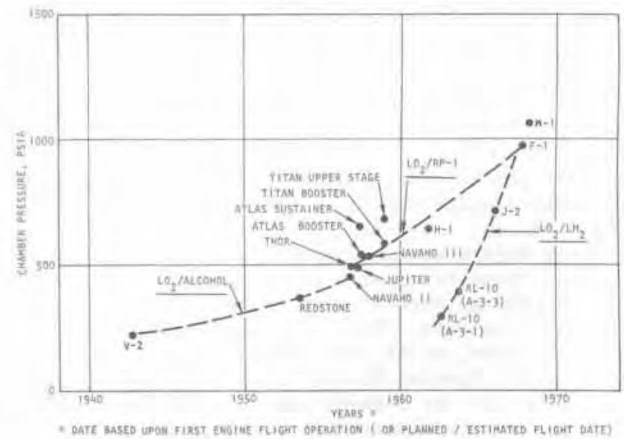


Figure 3. Trends in Chamber Pressure

Rocket Engine Programs

Having reviewed three principal design parameters for a liquid propellant rocket engine, it is now of interest to review briefly some of the main programs and vehicles involved. This will provide further insight into specific requirements of the rocket engine and the associated technology developments.

Hermes

Two of the early cryogenic liquid propellant rocket engines developed in this country were the Hermes A-1 and A-3 designs. The Hermes program was conducted for the U.S. Army by the General Electric Co. in the late 1940's and early 1950's. The Hermes ballistic missile, a result of this program, was the first large surface-to-surface missile to be designed and built in the United States. The rocket engine, airframe, and guidance were of American design; only the general configuration was adapted from an earlier German design.[8,9] During the program, several rocket engines and vehicles evolved.

The A-1 rocket engine was a pressure-fed, liquid propellant design using liquid oxygen and alcohol. It developed approximately 13,000 pounds of thrust. The A-3, an improved version, was a pump-fed design using the same propellants, but operating at a higher chamber pressure. This engine developed a thrust of approximately 20,000 pounds. The turbine for the pumps was driven by decomposed hydrogen peroxide. The thrust chamber was regeneratively cooled and had a double-walled helical passage.[10,11]

Viking

The Viking project was initiated in August 1946 with a contract to the Martin Company to develop a high-altitude sounding rocket for the U.S. Navy.[12] The Viking was a one-stage design using liquid oxygen and alcohol as propellants. The booster engine development, a 20,000-pound-thrust design, was undertaken by Reaction Motors (now a division of Thiokol Chemical Corp.). The engine was a pump-fed design, hydrogen peroxide to drive the turbine-powered propellant pumps. It had a double-walled, regeneratively cooled combustion chamber and nozzle. Several models were designed and tested. The engine acceptance tests was on 21 September 1948; the engine delivered 21,000 pounds of thrust for 66 seconds.(12)

The first Viking was assembled in December 1948 and the first flight took place at White Sands, New Mexico on 3 May 1949.(12) The Viking went on to establish a new altitude record at that time for sounding rockets and it provided much useful data for high-altitude research. The Viking engine (though lower in thrust than the German V-2) provided extensive information for the further development of rocket engines. For example, it was the first engine to use gimbaling for thrust vector control.(12)

Vanguard

In the Vanguard, the first vehicle designed specifically for satellite launches, a cryogenic liquid propellant rocket engine was used for the first stage. (The second stage was powered by a pressure-fed liquid propellant rocket using nitric acid and unsymmetrical dimethylhydrazine, and a solid propelland motor was used for the third stage). The booster engine was an improved version of the A-3 rocket engine with thrust uprated to 27,000 pounds. Development of this motor was initiated in October 1955, and completed in the fall of 1956.(11)

The engine design had a gimbaled thrust chamber for vehicle steering. Mainstage operating time was 150 seconds, a relatively long duration at that time. The decomposed hydrogen peroxide used to drive the engine turbine was also used for vehicle roll control. The power for the gimbal actuator was provided by a power takeoff from the turbopump. The engine had a "semi" tank head start. After the main valves were opened, propellants flowed to the combustion chamber. Ignition was accomplished by a pyrotechnic; when combustion was detected, the hydrogen peroxide valve opened and turbopump power buildup began. Shutdown was achieved simply by closing the fuel, oxidizer, and hydrogen peroxide valves.(11)

The first Vanguard-launched satellite was placed in orbit in 1958. In total, Vanguard launched three satellites into orbit.

Navaho

The era leading to high-thrust rocket engines was initiated with the Air Force Navaho project. This vehicle, a rocket booster with a supersonic ramjet-powered cruise vehicle, was contracted to North American Aviation, Inc. by the U.S. Air Force in 1947. The general configuration of the Navaho is shown in Fig. 4. For the rocket booster, no suitable engines were then in existence; their development was undertaken by North American Aviation, Inc.(12)

The Navaho I booster was a 75,000-pound-thrust design using liquid oxygen and alcohol. The pump-fed engine operated at 300 psia chamber pressure, had a double-wall, regeneratively cooled thrust chamber, and used decomposed hydrogen peroxide for turbine power.(13)

Figure 4. Navaho Launching

A change in Navaho vehicle design necessitated a new, higher thrust engine. A booster providing 240,000 pounds thrust at takeoff was required. Development of the Navaho II (a cluster of two 120,000-pound thrust engines) was undertaken in 1953 (Fig. 5).

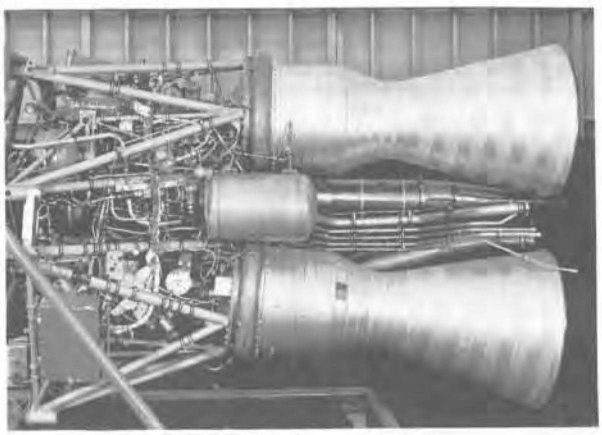


Figure 5. Navaho II

This engine was greatly advanced compared to the original Navaho I. A tube-walled thrust chamber was used in the new design rather than the much heavier, double-walled design. A "bootstrap" system in which the propellant from the main pumps is burned in a gas generator to provide the turbine drive gases was used in the Navaho II. The use of the gas generator "bootstrap" power cycle permitted removal of the third propellant tank--a significant weight reduction--and the higher specific horsepower from this system made high-pressure pumps and higher chamber pressure operation practical. The high-speed turbomachinery (the pumps operated at 5500 rpm and the geared turbine operated at 26,000 rpm) was required to develop the high flows (required for the greater thrust) and higher pressures to be achieved without a prohibitive weight penalty.

The booster requirements for the Navaho III, a later and larger vehicle, necessitated a still more advanced engine. Booster takeoff thrust requirements were increased to 415,000 pounds, and a higher specific impulse was required. Development of the engine was started in 1956. The first requirement was satisfied by a three-engine cluster design using Navaho II engines uprated to 135,000 pounds thrust each. The chamber pressure was increased from 454 to 513 psi. The chamber pressure increase, and a change in fuel from the alcohol/water mixture to kerosene raised the sea level specific impulse to 245 seconds.

The three-engine cluster had hinged engines for vehicle control. Movable rocket engines had been used for control in previous vehicles; however, this was the first use of such a system in a high-thrust rocket engine design. The development of large, flexible, high-pressure propellant lines was required.

Although the Navaho III engine was never used in a launch vehicle, its development achievements are significant. The design, fabrication, and successful static firing of an engine of this size--415,000 pounds total cluster thrust--in January 1956 was of historic significance. From these Navaho engine designs, many of the basic features for the engines which powered the next series of launch vehicles--the Atlas, Titan, Thor, and Jupiter--were derived. Perhaps even more important was the fact that the development of the Navaho engine gave confidence that high-thrust engines could be designed, fabricated, tested, and built on a production basis.

Redstone

The Redstone vehicle (Fig. 6) was a highly successful early vehicle. The Redstone's development was undertaken by the U.S. Army in the early 1950's to provide a medium-range ballistic weapon. To meet the booster engine requirements, the development of the Navaho I engine, which had been terminated at the initiation of the Navaho II, was reinstated under U.S. Army contract in April 1951. This engine met the thrust requirements for the Redstone and was well along in development. In 1952, production for the Redstone engine (Fig. 7) was started; the first Redstone flight was conducted and successfully completed in January 1954.

Figure 6. Redstone Launching

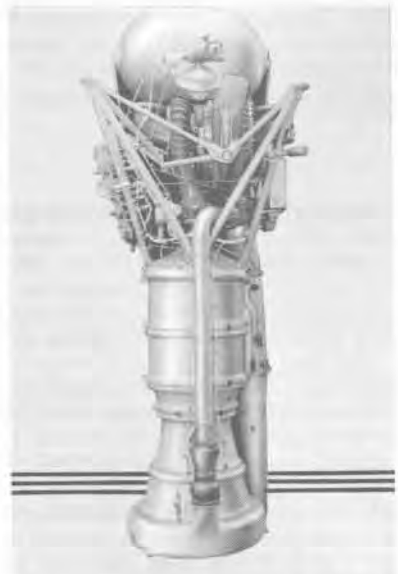


Figure 7. Redstone Engine

The "firsts" achieved by the Redstone are numerous. The Redstone launched America's first satellite on 31 January 1958; this launching was followed by 11 successful satellite launchings. The Redstone also powered the Mercury series of suborbital manned missions, the forerunner to the manned spaceflight Mercury program.

The Succeeding Generation

Following the Viking, Navaho, Vanguard, and Redstone, came a succeeding generation of rocket-powered vehicles for which high-thrust, improved rocket engines were developed.

Atlas

The high-thrust, cryogenic rocket engine for the Atlas ballistic missile (Fig. 8) had the distinction of being the first such engine to be developed to operational status and realize high-level production. The Air Force Atlas project was initiated in 1953 with the Convair Division of General Dynamics Corp. as prime contractor. The propulsion system was designed and built by the Rocketdyne Division of North American Aviation, Inc.

Following a preliminary design effort, the Atlas rocket engine requirements were defined. The vehicle, a one and one-half stage design, required a boost phase of 140 seconds followed by a 160-second sustaining operation. The propulsion system to provide these operational requirements initially

Figure 8. Atlas Launching

consisted of (1) two 150,000-pound-thrust booster engines which were jettisoned at the end of the boost phase, (2) a 57,000-pound-thrust sustainer engine which operated from liftoff through sustainer cutoff for a total of 300 seconds, and (3) two 1000-pound-thrust vernier engines which operated through boost and sustain phase for roll control and following sustainer phase for controlled velocity cutoff.

The Atlas rocket engines used liquid oxygen and kerosene (RP-1) propellants, had tubular-wall, regeneratively cooled thrust chambers, and were gimbaled for vehicle steering. The first operational Atlas engine system, the MA-2 (the MA-1 was the R&D version and used in flight testing), is shown in Fig. 9. The MA-2 had the turbomachinery, controls, and gas generator for the two booster engines mounted to the vehicle aft structure.

Figure 9. MA-2 Engine Cluster

Following the MA-2 design, two improved versions were designed and built. The MA-3 was developed specifically for the weapons system. It had a slightly higher takeoff thrust (389,000 vs 368,000 pounds for the MA-2) and was designed for military readiness. This engine system had independent packaging for each booster engine (Fig. 10) and numerous design innovations (Table 1).

Figure 10. MA-3 Engine Cluster

The MA-5 was developed for space flight. It combined the MA-2 launch holddown concept (Table 1) which provides a brief observation period to verify operation, together with the MA-3's reliability and performance improvements (Table 1). The MA-5 maintained the MA-2's packaging arrangements and a single gas generator for the two booster engines.

Table 1. Atlas Engine Design Evolution

Engine System	MA-2	MA-3	MA-5
Start Sequence	Electropneumatic	Pressure Ladder	Electropneumatic
Start System	Start Tank	Solid Propellant Cartridge	Start Tank
Thrust Chamber Ignition	Pyrotechnic	Hypergolic Fluid	Hypergolic Fluid
Gimbaling Package	Thrust Chamber Only	Entire Engine	Thrust Chamber Only
Booster Turbine Drive	Single Central Gas Generator	Independent Gas Generators	Single Central Gas Generator
Missile Holddown	Yes	No	Yes
Injector Design (For Stability)		Baffled Injector (Booster)	Baffled Injector (Booster)
Turbomachinery Improvements		Kel-F Liner	Kel-F Liner
Development Initiated	May 1954	August 1958	1960
First Test	June 1956	1958	January 1961
First Flight	June 1957/July 1958 (R&D Version: MA-1)	October 1960	November 1961

Thor/Jupiter

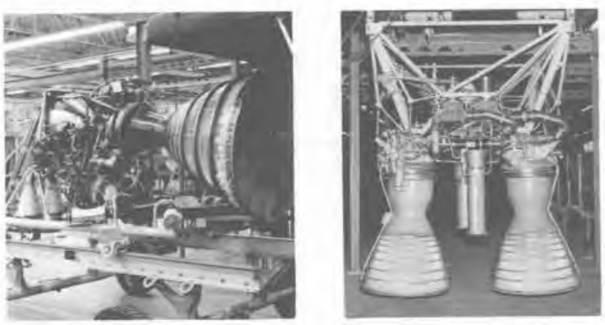
The requirement for an intermediate-range ballistic missile was established in November 1955, and led to the development of the Thor and Jupiter rocket vehicles. Both were single-stage designs, powered by a single booster engine which used liquid oxygen/kerosene propellants. Development of these engines was undertaken by Rocketdyne in November 1955. Both engines were based upon the Atlas MA-2 booster engine. The original Thor booster engine developed 150,000 pounds thrust, and was later uprated to 170,000 pounds thrust. The design and packaging of the engine is shown in Fig. 11.

Figure 11. Thor/Jupiter Engine Configurations

Titan I

The Titan I missile, built by the Martin Company, evolved as a backup program to the Atlas. The Titan was a two-stage missile which employed altitude ignition of the second-stage engine, differing from the Atlas configuration where all engines were ignited at liftoff.

First-stage propulsion consisted of a cluster of two 150,000-pound-thrust liquid propellant, pump-fed rocket engines using oxygen and kerosene propellants designed and built by Aerojet General Corporation (Fig. 12). A regeneratively cooled, bell-nozzle thrust chamber, a gas generator power cycle, and a chamber pressure controller to control thrust were employed. Engine start was accomplished by means of high-pressure gaseous nitrogen bottles with pyrotechnic squib igniters in both the gas generator and thrust chamber.



Stage I Stage II

Figure 12. Titan I

The upper-stage engine (a single unit of the same type design) used the same propellants as the booster, and had a thrust of 80,000 pounds at 250,000 feet altitude. A separate (auxiliary) turbopump with its own small gas generator was used to provide propellants to both the main gas generator and the auxiliary turbine-drive gas generator; the turbine exhaust from the auxiliary turbopump was used in four roll-control nozzles. The auxiliary turbopump was started by means of stored helium gas. The engine would then start using this power source and then reach mainstage operation. After second-stage engine shutdown, the auxiliary turbopump was run in solo operation for vernier thrust control.

X-15

The 50,000-pound-thrust, man-rated X-15 rocket engine was designed and developed by the Reaction Motors Division of Thiokol Chemical Corp. for the North American Aviation, Inc., X-15 program. It utilizes anhydrous ammonia and liquid oxygen propellants in a single bell-nozzle thrust chamber configuration which operates at a design chamber pressure of 600 psia. The specific impulse is 236 seconds at sea level. The engine is designed to be under full control of the pilot during rocket flight; it is capable of being throttled to 50 percent of rated thrust, shut down, and restarted back to full thrust. The propellant supply system utilizes a single-shaft turbopump which includes centrifugal oxidizer and fuel pumps driven by a single-stage, two-row turbine; decomposed hydrogen peroxide is used to propel the turbine. Operational tests in the X-15 engine were initiated in 1960; an extensive flight program has been successfully conducted with the X-15 engine system.

Centaur/RL-10

The Centaur upper stage was designed to be mounted atop the Atlas for missions requiring high-energy propellants. With the advent of the Centaur came the first liquid-hydrogen-fueled rocket engine (RL-10). Although preliminary designs and experimental efforts had been undertaken, it fell to the RL-10 to culminate these efforts into a flight engine. The development of the RL-10 was initiated by the Air Force in October 1958; Pratt & Whitney Aircraft was the prime contractor. The Centaur stage, designed and developed by General Dynamic Corp., used two RL-10 engines. The first in-flight firing of this system was in November 1963. The first launching of the Surveyor by the Atlas Centaur to a lunar landing was in May 1966.[14]

Basic features of the RL-10 and the performance of the models developed are presented in Table 2. The RL-10 is a pump-fed rocket engine with a regeneratively cooled thrust chamber.[14] An expander power cycle is used in the engine in which the hydrogen, after being discharged from its pump, flows through the regeneratively cooled thrust chamber tube jacket, and then this heated hydrogen powers the turbines before being injected into the combustor. This cycle (1) eliminates the need for a third propellant or bipropellant gas generator system and (2) provides a performance advantage in that all propellants exhaust through the main combustor. The engine is a gimbalable, integral-packaged design (Fig. 13) with the turbopump lines and valves all mounted to the thrust chamber.

Table 2. RL-10 Basic Engine Features

	RL-10-A-3-1	RL-10-A-3-3
Nominal Vacuum Thrust, pounds	15,000	15,000
Type	Liquid bipropellant, pump-fed, expander power cycle, gimbaling capability, restartable. Deep throttling demonstrated in experimental engines.	
Propellants		
Fuel	Liquid Hydrogen	
Oxidizer	Liquid Oxygen	
Operating Mixture Ratio Range	4.4 to 5.6	4.4 to 5.6
Nominal Mixture Ratio	5.0	5.0
Vacuum Specific Impulse, seconds	433 (minimum)	439 (minimum) 444 (nominal)
Chamber Pressure, psia	300	400
Nozzle Area Ratio	40	57
Thrust Chamber	Tubular wall, regeneratively cooled	
Turbopumps	Centrifugal pumps, single turbine with gear driven oxidizer pump	

Figure 13. Centaur RL-10-A-3 Rocket Engine

Saturn I

The Saturn I rocket vehicle originated as one of a family of large launch vehicles studied by the Army Ballistic Missile Agency in 1957. The preliminary investigations resulted in establishing a booster thrust requirement of 1,500,000 pounds. Numerous engine configurations were investigated and a cluster of eight 188,000-pound-thrust engines using liquid oxygen and kerosene propellants based upon the Thor/Jupiter engine design was selected. This effort was then transferred to the National Aeronautics and Space Administration (NASA).

H-1 Engine. The contract to develop the H-1 engine for use in the Saturn I was awarded to Rocketdyne in August 1958. To provide the higher thrust, the 520-psia chamber pressure of the Thor/Jupiter engine design was increased to 650 psia and turbomachinery improvements were made. The packaging of the H-1 engine was similar to the MA-3 Atlas booster engine. The turbopump was mounted directly to the thrust chamber and the entire package was designed for gimbaling. This arrangement permitted the flex lines for gimbaling to operate under low pressure rather than the high-pressure lines required when the turbopump is mounted separately from the gimbaled engine.

Unlike the early Navaho and Atlas MA-2 engine clusters, the H-1 engine was designed and developed as an independent unit. Only the start and shutdown electrical signals link the engines in the Saturn I. The engine is started with an engine-mounted, solid propellant start cartridge which eliminates the need for a start bottle or a ground start system. The engine operates on a pressure ladder sequence which eliminates the electrical sequence system required in the earlier control system.

RL-10 Engine. The S-IV upper stage for the Saturn I was powered by six RL-10 engines. These RL-10 engines produced a total thrust of 90,000 pounds. The availability of this engine, developed for the Centaur stage, permitted early orbital flights of the Saturn I. The Saturn I has had 13 launches. The first four had dummy upper stages; the next six Saturn I's had upper stages (S-IV) powered by RL-10 engines. The first flight with the RL-10 powered upper stage was in January 1964.

J-2 Engine

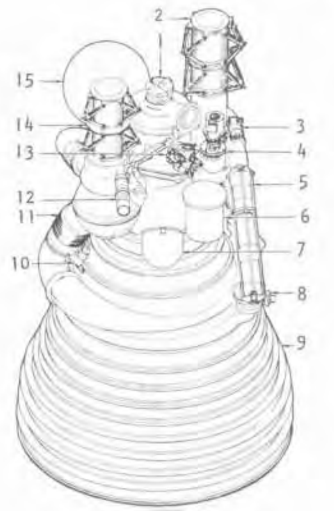
The J-2 engine was designed as a high-thrust (200,000 to 230,000 pounds) oxygen/hydrogen propellant engine for use in the upper stages of launch vehicles. The development program for this engine was contracted to Rocketdyne by NASA in September 1960. The basic design parameters are presented in Table 3 and the configuration is shown in Fig. 14.

Table 3. J-2 Basic Engine Features

Thrust (vacuum)	230,000 pounds
Type	Liquid bipropellant, pump fed
Propellants	
Fuel	Liquid hydrogen
Oxidizer	Liquid oxygen
Mixture Ratio	5.5:1 (nominal)
Operating Mixture Ratio Range	4.5:1 to 5.5:1
Specific Impulse (vacuum)	425 seconds
Chamber Pressure (nozzle stagnation)	717 psia
Nozzle Area Ratio	27.5:1
Thrust Chamber	Tubular-wall, regeneratively cooled
Turbopumps	Separate oxidizer and fuel turbopumps
Bearing Lubrication	Liquid oxygen and liquid hydrogen
Turbine Drive	Gas generator burning main propellants

LEGEND

1 GIMBAL
2 FUEL INLET DUCT
3 FUEL BLEED VALVE
4 GAS GENERATOR
5 HIGH PRESSURE FUEL DUCT
6 ELECTRICAL CONTROL PACKAGE
7 PRIMARY FLIGHT INSTRUMENTATION PACKAGE
8 MAIN FUEL VALVE
9 THRUST CHAMBER
10 ANTI-FLOOD CHECK VALVE
11 HEAT EXCHANGER
12 PROPELLANT UTILIZATION VALVE
13 PNEUMATIC CONTROL PACKAGE
14 OXIDIZER INLET DUCT
15 START TANK



LEGEND

1 GIMBAL
2 OXIDIZER INLET DUCT
3 OXIDIZER TURBOPUMP
4 START TANK
5 AUXILIARY FLIGHT INSTRUMENTATION PACKAGE
6 EXHAUST MANIFOLD
7 THRUST CHAMBER
8 OXIDIZER TURBINE BYPASS VALVE
9 TURBINE BYPASS DUCT
10 MAIN FUEL VALVE
11 HIGH PRESSURE FUEL DUCT
12 START TANK DISCHARGE VALVE
13 FUEL TURBOPUMP
14 FUEL BLEED VALVE
15 FUEL INLET DUCT

Figure 14. J-2 Rocket Engine

In the J-2, a single gas generator supplies fuel-rich gas to drive the hydrogen-pump turbine and then, in series, to drive the oxygen-pump turbine. This arrangement gives high cycle efficiency and allows simple control of thrust and mixture ratio. The turbine exhaust gas is introduced into the main rocket nozzle for effective disposal and to recover much of its remaining propulsive force.

Propellant utilization control is provided so that propellant mixture ratio can be controlled in flight in accordance with vehicle command signals. Propellant lubricated turbopump bearings were developed to eliminate the need for separate lubricating systems, to avoid the problem of protecting oil from low propellant temperatures and the low temperatures of space, and to minimize contamination problems.

The engine is completely independent, carrying its own helium supply for valve actuation and a hermetically sealed electrical control unit which supplies all control logic for engine start, cutoff, and restart. The rechargeable, engine-mounted, hydrogen start tank and an augmented spark ignition system (ASI) featuring automatic reset provide the multiple restart capability.

The pumps are mounted on opposite sides of the thrust chamber and have straight axial inlets. The entire assembly is gimbaled.

The J-2 engine completed its first 250-second (static) test in August 1962; the initial version (200,000 pounds thrust engine) completed its qualification flight rating program in January 1966. The first flight of this engine was as an upper stage in the Uprated Saturn I in February 1965. A higher thrust version of 230,000 pounds was initiated on 12 July 1965. The engine qualification test series on this version was completed on 22 August 1966.

F-1 Engine

The F-1 rocket engine was developed to provide a high-thrust (1,522,000 pounds at sea level) booster engine. The F-1 engine development program was awarded to Rocketdyne in January 1959. The giant stride in thrust was to be the major design advancement. Although this in itself necessitated many technology advancements and fabrication innovations, NASA directed that a design using tried and proven propellants (liquid oxygen and RP-1, kerosene) be used, and the prime emphasis be placed upon reliability. F-1 design parameters are shown in Table 4 and its configuration is illustrated in Fig. 15.

Table 4. F-1 Engine Features

Thrust (sea level)	1,522,000 pounds
Type	Liquid bipropellant, pump fed
Propellants Fuel Oxidizer Mixture Ratio	 RP-1 (kerosene) Liquid oxygen 2.27:1
Nominal Specific Impulse Sea Level Vacuum	 265.4 seconds 304.1 seconds
Chamber Pressure (nozzle stagnation)	980 psia
Nozzle Area Ratio	16:1
Thrust Chamber	Tubular-wall, regeneratively cooled to nozzle area ratio of 10:1; turbine gas cooled to area ratio of 16:1
Turbopump	Single-unit, single-shaft, with direct-drive, centrifugal fuel and oxidizer pumps
Bearing Lubrication	RP-1 fuel
Turbine Drive	Gas generator burning main propellants

The F-1 thrust chamber is a regeneratively cooled, tubular-wall design with a removable turbine exhaust gas-cooled nozzle extension. The gas-cooled removable nozzle extension was adopted to facilitate transportation of the engine. The turbopump, a single-unit, single-shaft, direct-drive design, is mounted directly on the thrust chamber in "piggy-back" fashion. The engine's power cycle uses fuel-rich gas generator gases to power the turbine.

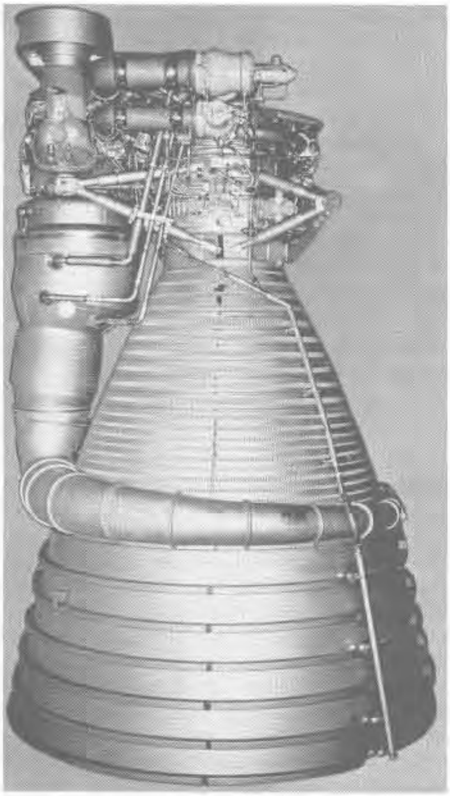


Figure 15. F-1 Engine

The F-1 development was preceded and enhanced by a program initiated in mid-1958 at Rocketdyne under an Air Force contract which included design, fabrication and initial testing of thrust chamber and injector components for a 1,000,000-pound-thrust engine.

The first rated thrust and duration F-1 test firing was conducted in May 1962; the engine qualification test series was completed in September 1966. The F-1 engine (a cluster of five) is used to power the first stage of NASA's Saturn V launch vehicle.

Of major significance in the F-1's development was the combustion stability program.(15) Unpredictable, random combustion instability is always a potential problem in developing large, high-chamber-pressure engines. The causes of such instability are still not completely understood. Aids have been developed to investigate instability. These were used extensively in the F-1 engine development. High-speed instrumentation capable of measuring combustion instability phenomena was used permitting diagnosis and understanding of the combustion behavior in the chamber and in the evaluation of design modifications. The technique of imposing artificial disturbances to test the dynamic stability of an engine design was also used extensively. In this technique, an explosive bomb, placed in the combustion zone and detonated during operation, is used to produce a pressure disturbance. The self-damping characteristics of the injector and feed systems can then be evaluated in all potential instability modes. Thus, rather than running hundreds of tests to accumulate statistics on the engine stability, the artificial disturbance provided a means to test injectors for stability characteristics. Two oscillograph traces are shown in Fig. 16;(15) one of an unstable injector configuration showing large-amplitude, self-triggered fluctuations in chamber pressure, and one of a stable injector, showing damping of the bomb-induced pressure fluctuation in a few milliseconds. The high-performance injector developed for the F-1 engine has demonstrated excellent damping capability by repeated self-stabilization following bomb-induced disturbances.

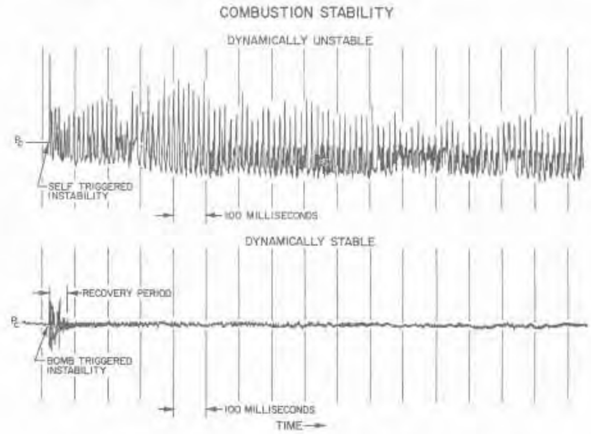


Figure 16. Combustion Stability

Thrust uprating on the F-1 may be accomplished by increasing the operating chamber pressure. This requires higher pump discharge pressures, a higher turbine power output, and structural strengthening of the components. Detailed studies of the design for these modified components have been completed. To further investigate this capability, experimental turbopump components to provide the head and flow required for the 1,800,000-pound sea level thrust have been built and successfully tested.

M-1 Engine

The M-1 program was initiated in 1962 under NASA sponsorship by the Aerojet General Corp. The program was to produce a high-thrust (1,500,000-pound vacuum) high-energy propellant (oxygen/hydrogen) rocket engine based upon a moderate state-of-the-art advancement.(16) Because of postponement of plans to develop a vehicle which would use the M-1, the program was terminated prior to completion of the engine development.

The M-1 design had a thrust chamber with a 40:1 nozzle area ratio, having a tubular-wall, regeneratively cooled section to 14:1 area ratio, and turbine-exhaust-cooled extension. The engine had dual turbopumps, side-mounted to the thrust chamber and the entire unit could be gimbaled. A gas generator power cycle was used with the flow to the turbines in series. The engine had propellant utilization (mixture ratio) control through fuel turbine power regulation by use of a bypass line.

The M-1 program achieved significant technical milestones and useful information on the design, fabrication, and testing of large, high-thrust, high-flowrate and high-pressure, hydrogen-propellant engine components. The high, head and flow hydrogen turbopump is the largest designed and built for a hydrogen-fueled engine. The thrust chamber and high-flow hydrogen valves also were a major step in size for a hydrogen-fueled engine.

The hydrogen pump is an eight-stage axial flow pump preceded by a mixed-flow, first-stage inducer and an axial-flow, second- (transition) stage inducer, and provided a 600-lb/sec hydrogen flow with an 1800-psi pressure rise. The unit had propellant-cooled and lubricated roller bearings and ball bearings for axial loads.

The injector for the thrust chamber was a coaxial element design similar to the J-2 and RL-10 but was much larger in size. In the development program, an ablative liner was used in the early experimental thrust chambers; this permitted direct observation of streaking or other local abnormal conditions. This concept was found to be very useful in the program to check injector operation.

Reliability

Before reviewing some of the details of the technical advances achieved in these rocket engines, a review of their reliability record is in order. The reliability record compiled by Air Force and NASA launches using the rocket engines discussed is summarized in Table 5*. The data are for unclassified missions, mainly missions in which the objective was to place a satellite into orbit. Reliability is defined as the number of successes divided by the number of exposures.

Table 5. Space Launch Reliability Through March 1967

Vehicle	Stage	Engine Type	Number of Engines	Stage Exposures	Observed Flight Reliability of Propulsion System*
Viking		X-405	1	11	0.82
Vanguard		GRX-405	1	11	0.91
Redstone		1A43-NA-1	1	12	1.00
X-15		XLR99	1	172	**
Jupiter		1R3-NA-1B	1	13	0.92
Thor		MB-3	1	219	0.99
Atlas		MA-2/5	3	121	0.98
Centaur		RL-10	2	9	0.89
Saturn I	(S-I)	H-1	8	10	1.0***
	(S-IV)	RL-10	6	6	1.0
Uprated Saturn	(S-I)	H-1	8	3	1.0
	(S-IVB)	J-2	1	3	1.0

*Reliability is the ratio of successes to exposures; includes engine failures which affect mission success, plus failures associated with propellants, tankage, pressurization, valves, etc.
**There have been no flight failures. A safety feature of the X-15 is that the engine automatically shuts down if any system malfunctions. Mission can be aborted at any time.
***One engine shutdown; other engines compensated to allow successful mission completion.
Principal source of data, Ref. 17.

The early rocket programs such as the Vanguard and Navaho had initial failures at the start of their flight program. However, the reliability was improved considerably as these programs progressed. The Redstone's reliability record was very good. In 75 launches, only two failures occurred. The first was in the third flight, and was the result of a known deficiency (later corrected) in the system. The second was a Mercury test launch in which the failure of electrical ground equipment to detach properly shut down the engine just after liftoff. Since the third flight, when the engine failure occurred, the Redstone vehicle, in a total of 72 launches, has not had an engine failure.

*No data on the Hermes engine program were available.

The records of the Thor and Atlas are limited to nonmilitary space launches because of security regulations. The Thor vehicle has completed 228 flights without an engine failure. There was but one engine failure in its entire space launch history; that was on the first flight. The Thor, in conjunction with various upper stages, has placed over 200 satellites into orbit. From the initial space launch in 1958 to the present, it has an overall launch reliability greater than 0.99. The Atlas vehicle has performed over 120 space launches involving more than 500 liquid propellant engines with an overall propulsion system reliability greater than 0.97.

In the Centaur program, the Centaur upper stage experienced several initial failures caused by propellant settling and tankage. However, the later flights have scored successes.

The Saturn I vehicles have been completely successful in 13 launches. The Saturn I, S-IV upper stage using six RL-10 engines flew six times with complete propulsion success. The first three flights with the J-2 powered S-IVB upper stage in the Uprated Saturn I also have been completely successful. In all these Saturn flights, 143 engines (H-1, RL-10, and J-2) have been fired with only a single malfunction. This malfunction in 1 of the 104 H-1 engines used in the first stage resulted only in engine shutdown; the success of the mission was not affected since the first stage on the S-I vehicle is designed with engine-out capability.

As can be seen from the launch history, we are approaching the goal of 100 percent propulsion system reliability. The Atlas-Mercury, Gemini, and Saturn launches have been 100 percent successful. With continued emphasis on reliability, design simplicity, and improved concepts, the high reliability will not only continue but will improve for future cryogenic engine systems.

Technological Advances

As the review of engine programs indicates, the component technology for cryogenic rocket engines has advanced considerably since the early series of engines. The development effort on these engines, together with extensive NASA-, Air Force-, and company-supported technology programs, has been directed at the areas which limited engine improvement. The wide scope of this effort cannot be completely reviewed here. However, several key areas are briefly discussed to show how advances over the years have improved the engine designs.

Thrust Chamber

The basic rocket engine thrust chamber has undergone considerable change since the first conical nozzle used in early engines. The improvements have been directed to increasing performance and reducing weight.

To achieve higher performance, higher area ratio and higher chamber pressure (to minimize thrust chamber size and weight) designs were introduced. The ability to adequately cool the thrust chamber of cryogenic liquid rocket engines at increasing pressure levels has been a major factor pacing development of large, high-pressure engines. The throat heat flux (q/A)--the governing parameter--rises with increasing chamber pressures as shown in Fig. 17.

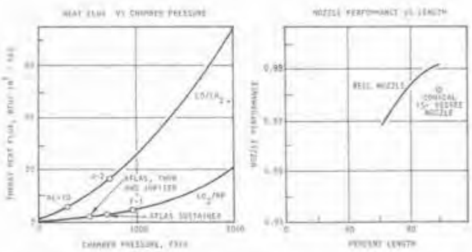


Figure 17. Thrust Chamber Technology

The early thrust chamber designs employed double-wall construction; regenerative cooling was used with the fuel flowing between the walls. This design was limited in its capability to carry off the heat; that is, to be designed to operate at a high throat heat flux. This design limitation exists because the coolant cross-sectional area at the throat cannot be made sufficiently small and uniform to achieve the necessary high coolant velocity.

To increase the coolant velocity in the throat section and carry off the higher heat flux resulting with higher chamber pressure (and to reduce wall thickness and thermal resistance) thin-walled tubes were adopted for the thrust chamber walls. The tubes were of varying width to form the nozzle shape. As chamber pressures increased, a "double taper" was added to the tube design, to vary the tube cross-sectional area and thus provide a higher coolant velocity in the throat section without having a high velocity (and high pressure drop) throughout the tube. The adoption of the thin-walled tubes also resulted in very significant weight reductions.

A further size and weight reduction was achieved by introduction of the bell-shaped nozzle. This nozzle, compared to a standard 15-degree half-angle conical nozzle, can have a 20-percent shorter length without any performance reduction (Fig. 17).

Recent emphasis on further increases in chamber pressures has necessitated optimization of coolant passages and the use of higher temperature tube materials. Also, film cooling is being investigated for high pressures; the emphasis is on distribution schemes to achieve adequate surface coverage at a reasonable coolant flow, and still maintain good specific impulse performance.

Turbopumps

The growth of the thrust and chamber pressure in cryogenic rocket engines generated extremely high power requirements for the turbomachinery. To minimize turbomachinery weight, turbopump speeds have continually increased. The higher speed permits smaller diameter rotating components to provide the power required, and the weight of the unit is minimized. Thus, improving the state of the art of the turbopumps is directly linked to increasing their operating speed.

The improvement in turbomachinery power-to-weight ratio which has occurred is shown in Fig. 18. This improvement has been the result of technology advances in high-speed bearings, high-speed inducer suction performance, and in increased impeller tip speed (U_T).

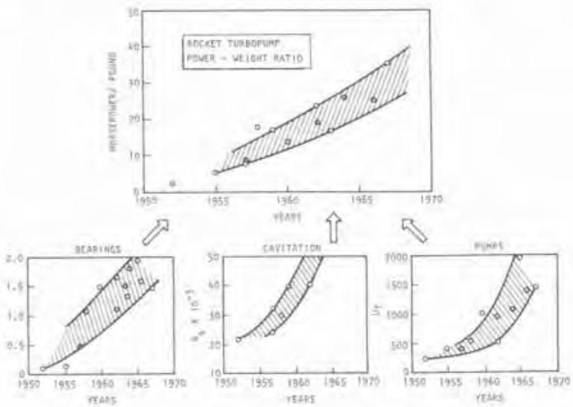


Figure 18. Rocket Turbopump Development and Technology *

The increase in operating speed of the bearings, or bearings DN**, has resulted from advances made in bearing technology through optimization of the bearing performance for speed, load capacity, and fatigue life. By optimizing ball size, contact angle, contact ellipse and pitch diameter and by using proper race curvature, both contact fatigue and overheating wear have been reduced. High-strength materials and the fabrication of precision parts with improved surface finishes have contributed to the success of high-speed turbopump bearings.

Bearings with DN values in excess of 1.5 million, where the pumped fluid (the cryogenic propellant) acts as the bearing coolant and lubricant, have been developed. The propellant lubrication feature eliminates the need for additional lubrication systems and simplifies the sealing problem.

The suction performance of pump inducers has been steadily improved to permit higher operating speeds. Results from technology programs have shown how blade angle, solidity, hub ratio, taper, leading-edge profile, and blade sweep, affect the cavitation characteristics on the inducer. With the development of theories on the cavitation phenomenon, the suction performance of inducers has increased by a factor of 2 (Fig. 18). This increase in turbopump suction performance permits the pump to operate at higher speeds together with significant reductions in NPSH;*** this in turn permits the propellant tank pressures to be reduced.

Pump impeller tip-speed limitations have been extended by the use of high-strength materials, improved designs, and advanced fabrication techniques. The use of these high-efficiency, high-tip-speed impellers has increased the design speed, thus resulting in higher power-to-weight ratios.

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*S_s (Measure of the ability of the pump to perform at low inlet pressure) = Pump Speed x Flowrate$^{1/2}$ $\div$ NPSH$^{3/4}$

**Bearing diameter (mm) x pump speed (rpm)

***NPSH (net positive suction head) = total pressure head minus vapor pressure

System Design and Packaging Evolution

The progress in design and packaging for cryogenic rocket engines is illustrated in Fig. 19. In early systems such as the Redstone, the turbopump, lines, and controls were located above the thrust chamber. Thrust vector control was achieved by means of jet vanes. Later designs used gimbaled thrust chambers. Since only the thrust chamber gimbaled, flexible high-pressure lines were required between the pump and thrust chamber. However, when thrusts and chamber pressures were increased, the turbopump was mounted onto the thrust chamber. The pump and chamber were gimbaled as a unit, and low-pressure flex lines at the pump inlets were used.

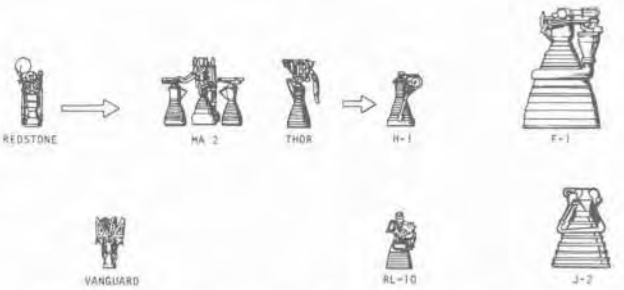


Figure 19. Design and Packaging Evolution in Cryogenic Rocket Engines

In early applications of multiple engines (the Atlas booster and Navaho booster) the propulsion system was designed as a complete unit as opposed to the modular approach where each engine could be "plugged in" as a unit. For example, a single gas generator was used to supply both turbopumps on the Navaho II and the first Atlas booster engine (MA-2). This approach necessitated complete system acceptance testing. Later, design and reliability studies indicated the advantages of the modular approach where acceptance testing could be accomplished on individual engines. System accessibility also was greatly improved since each engine could be individually removed as a unit.

Starting Methods

Starting methods for pump-fed cryogenic engines have evolved over the years to systems which can provide reliable starts specifically tailored to the engine, the propellant combination, and the application. Basic starting methods have long been known; for example, one of Dr. Goddard's last rockets utilized nitrogen gas expansion to supply the starting power. This is the same basic method utilized in the oxygen/hydrogen propellant J-2 engine.

In pump-fed rocket engines, the power to drive the pumps (during start before the engine itself supplies the power) can be supplied by a variety of methods (Fig. 20). Early ballistic missile engines (i.e., the Atlas and Titan I booster engines) employed ground start tanks which contained gas-pressurized propellant to feed the bipropellant gas generator which drove the turbines. As pump discharge pressure builds up, the turbine flow is obtained by using the high-pressure pump discharge in a bootstrap fashion. This system necessitated quick disconnects and additional circuitry for the start sequence. Later versions of these engines, as well as the H-1 engine, employ a solid propellant turbine starter to supply turbopump power during start.

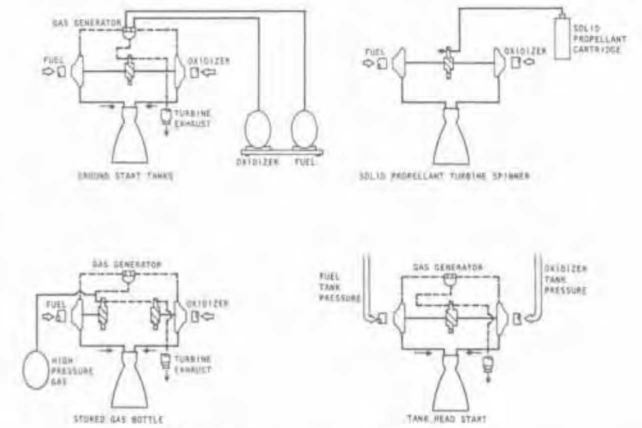


Figure 20. Starting Methods

The electrical system (solenoid valves, relays) for start was virtually eliminated by application of a pressure-ladder valve-sequencing method. In this method, the turbopump is started with the solid-propellant turbine starter (SPTS); propellant pressure builds up against the closed main propellant valves, and fuel-pump discharge pressure is then used to open the main valves. The valve sequencing is controlled by mechanical and hydraulic interlocks. This method of start provides rapid pump acceleration and greatly reduces engine start times. For large booster engines, such as the 1,522,000-pound-thrust F-1 engine, the required size for an SPTS increased significantly, and since high inlet pressures were available from the stage, an engine start using available propellant tank-head pressure was developed.

In a tank-head start, the turbine drive gas is produced in the gas generator under tank-head pressure. As pump speed and pressure build up, the main combustion chamber is ignited at low pressure. Pump discharge pressure is available to supply the turbine-drive gas generator at increasing pressures. The engine system attains its rated thrust value since there is an excess of turbine power to accelerate the turbopump. The tank-head start technique has been refined into a highly reliable starting method and has simplified engine operation.

For the oxygen/hydrogen propellant engines, various starting methods have been employed. The RL-10 engine utilizes the heat input to the hydrogen from the chilldown of the thrust chamber tube bundle. Sufficient heat is gained by the hydrogen, which is used for turbine-drive gas, to enable engine start in this manner (Fig. 20). On the J-2 engine, a stored hydrogen gas bottle is used to supply initial turbine power. The bottle can be recharged in flight by the engine to provide multiple start capability. The M-1 engine used stored helium gas for its design.

Engine Cycles

The evolution of cryogenic engine power cycles has had two main goals: the first was to establish an efficient and reliable turbine power source, and the second was to obtain the maximum possible specific impulse from the main combustion chamber and the turbine exhaust gases. Some of the early power cycles employed for cryogenic rocket engines are illustrated in Fig. 21. The Redstone employed a hydrogen peroxide supply from a separate tank which was catalytically decomposed in a gas generator to furnish turbine power. Hydrogen peroxide provided a suitable turbine working fluid at controllable temperatures. This represented an advancement over pressure-fed engines in that higher chamber pressures could be attained.

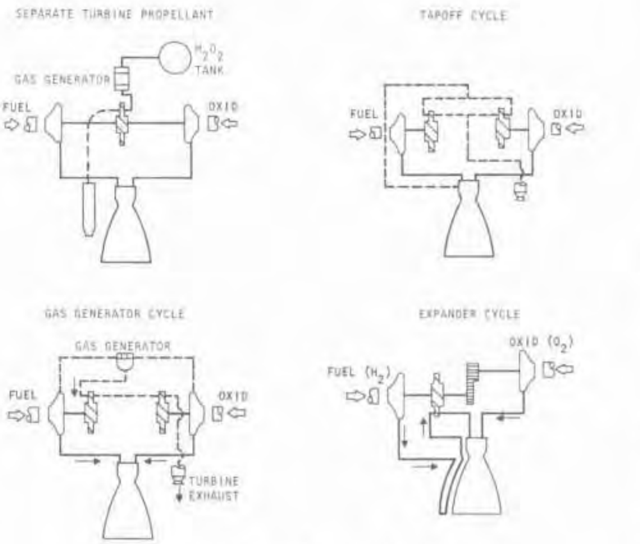


Figure 21. Engine Cycles

The next series of engines (i.e., Atlas, Titan I, Thor, and Jupiter) utilized a bipropellant gas generator which used the same propellants as the main chamber but at a reduced mixture ratio to control the gas temperature. This eliminated the separate gas generator propellant tankage and pointed the way for further increases in chamber pressures, since a higher specific horsepower is available for the turbine. The turbine is fed from the pump discharge pressure in a "bootstrap" fashion. A later refinement of this cycle is the tapoff cycle where relatively cool gases from the main chamber are tapped off and used for turbine drive gas. This eliminates the gas generator chamber and the associated valves and controls.

The disposal of turbine exhaust gas has been accomplished by a variety of methods (Fig. 22). A separate turbine exhaust duct which terminated in a convergent nozzle at the exit plane of the main chamber was first used. However, during boost, the turbine gas flared back into the engine compartment to cause overheating. Solutions to this problem were to use an annular exhaust around the exit of the main nozzle or to duct the exhaust into the main nozzle at a suitable area ratio. Turbine exhaust gas was also used to cool the lower section of the expansion nozzle on the F-1 and M-1 engines. However, in these designs, the performance achieved by the turbine exhaust gas in contributing thrust is lower than the main nozzle performance.

A cycle which evolved to improve the turbine exhaust gas performance is the expander cycle (Fig. 21). This cycle uses heated hydrogen from the thrust chamber jacket, which is then injected into the main thrust chamber. The principle advantage of the cycle is that all propellants achieve thrust chamber performance. This cycle is limited in the obtainable chamber pressures because of the limit of available turbine power from the heated hydrogen.

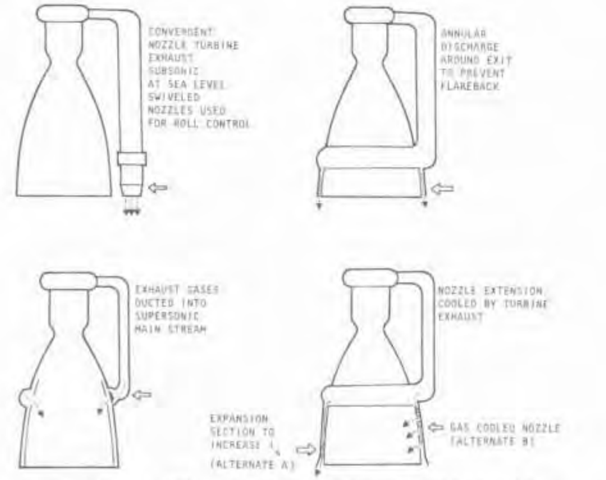


Figure 22. Turbine Exhaust Utilization

Most of the early rocket engines and today's booster engines are calibrated during acceptance testing to deliver the rated thrust and engine mixture ratio. Turbine power was controlled through the use of a liquid oxygen pressure regulator in the turbine feed system or by the use of calibration orifices in the gas generator circuit.

The external variables which can affect rocket performance are propellant densities, inlet pressures, and atmospheric pressure. For example, for systems which do not use closed-loop control systems, the effects of a change in propellant inlet pressure will be a change in engine thrust level and engine operating mixture ratio. These mixture ratio shifts may be relatively unpredictable because they can result from the tolerances in tank pressure and propellant conditions which may not be controllable. Excess propellant residual will adversely affect the delivered payload of a system. Early cryogenic rocket engines were designed to operate over the operating regimes caused by the flight environment range. This was possible since ambient conditions did not have a large influence on engine operating conditions.

Alternately, the thrust level and mixture ratio can be controlled by commands from the vehicle. Closed-loop, mixture-ratio control systems were devised to ensure simultaneous emptying of the propellant tanks. The early Atlas system used a propellant utilization control system, mainly because a single set of tanks were used for the vehicle. Upper-stage engines such as the RL-10 and the J-2 engine have employed propellant utilization systems to maximize system performance. Booster engines such as the H-1 and F-1 have not employed mixture-ratio control systems since the effect of residuals is not severe in a booster system.

Control systems for both thrust and mixture ratio of the closed-loop type have been designed and tested in rocket engines. The amount of the thrust modulation required is dependent upon the application; studies are currently being conducted to determine future thrust control requirements.

Advanced Engines

Design Improvement

The designs for improved cryogenic rocket engines have been undertaken to achieve one, or a combination, of the following requirements: increased thrust, increased specific impulse, or increased rocket engine in-flight operating flexibility.

Increased thrust historically has been achieved by two approaches: (1) uprating the operating chamber pressure of an existing engine or (2) designing a new, physically larger and higher thrust configuration. Increased specific impulse has been achieved through higher performance propellants and/or a higher area ratio nozzle (often in combination with increased chamber pressure). Increased flexibility has been principally the result of design innovations, advances in the power cycle, and improvements in the control system.

J-2X Experimental Program and Simplified J-2 Engine. The J-2X program, which is an experimental engine program based upon the J-2 engine illustrates several new engine features which provide engine flexibility, and also a new approach to development of improved engines.[18] Historically, engine programs have been committed to stringent development schedules. Consequently, the configurations have been restricted to those utilizing proven concepts or those capable of rapid development. The experimental engine approach is used to bridge the gap between technology and development.

The basic goals of the J-2X program were to simplify the engine, simplify the stage equipment associated with engine prelaunch operations, reduce the prelaunch engine checkout and ground support equipment, provide a greater engine in-flight operational flexibility, and provide a capability for higher thrust and higher specific impulse performance.

As new concepts demonstrated their feasibility in the J-2X program, it became apparent that their incorporation into the J-2 engine would be advantageous. The engine configuration with these features has been designated the Simplified J-2 (or J-2S). The description of the manner in which the flexibility goals were incorporated in the J-2S design, based upon experimental verification in J-2X engine tests is reviewed.

Simplification of the engine system was accomplished by (1) eliminating the gas generator system by using tapoff main combustor gases for turbine drive, (2) using a solid propellant turbine starter (SPTS) which eliminates the need for the start bottle system, (3) incorporating a premainstage idle-mode phase, and (4) eliminating the need for thrust chamber and pump temperature conditioning by adoption of the above features and a tailored start sequence.

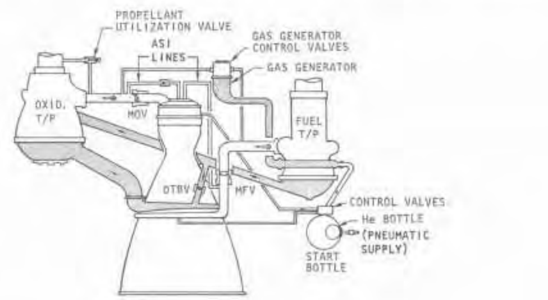
The tapoff engine power cycle, wherein the turbine gases are tapped from the main combustor, eliminates the entire gas generator system. The tapoff ports are located just downstream of the injector face and provide gases of the proper temperature for turbine operation. The adoption of the SPTS eliminates the need for the hydrogen start bottle, and the associated lines, valves, and recharging system, together with the conditioning and time restraints associated with the start bottle.

The combined effects of the tapoff power cycle, the SPTS, the idle-mode capability, and thrust chamber bypass system eliminate the need for engine and propellant temperature conditioning prior to start. By starting the engine in idle mode (the low-thrust tank-pressure mode) with mixed-phase or gaseous propellants, the idle-mode propellant flow conditions the engine and stage feed systems (ducting). This effect is augmented by the SPTS which drives the turbines during the start transient thereby pumping out the mixed-phase hydrogen (the critical propellant) prior to high-flow pump operation. The use of the tapoff rather than the gas generator power cycle is beneficial in that the main combustor is much less sensitive to the propellant quality and temperature than the gas generator.

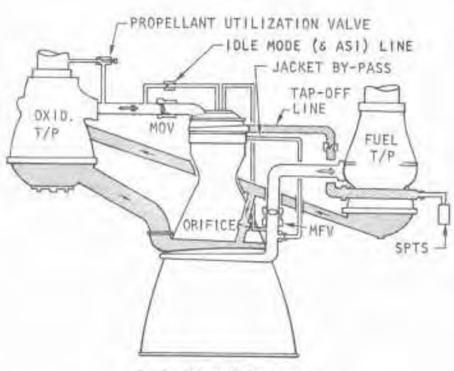
Use of the SPTS in place of a gaseous hydrogen supply for engine start eliminates the hydrogen start bottle and its conditioning requirements. This is especially beneficial for restart, since it eliminates restrictions on the minimum orbiting time between starts and minimum mainstage operating duration. Multiple, sealed (by a burst diaphragm) SPTS units are used to provide multiple start capability.

Thrust chamber conditioning was previously required because the sensible heat in the thrust chamber would gasify the hydrogen fuel passing through the coolant tubes at start, thereby increasing the hydraulic resistance. This high-head, low-flow condition would tend to force the fuel pump into stall during the start transient. A thrust chamber bypass system that allows some of the fuel to be pumped directly to the injector without passing through the coolant tubes prevents high-head buildup during start.

A comparison of the basic power cycles for the J-2 and the simplified version are shown in Fig. 23.



J-2



J-2 Simplified

Figure 23. Engine Schematic Comparison

Stage and ground equipment simplification is achieved by eliminating the need for temperature conditioning. Presently, the J-2 engine must be ground chilled so that at start the engine is not at a high temperature. Inlet duct conditioning and liquid phase propellants at start are obtained in the Saturn V upper stages by electrically driven circulating pumps in the S-IVB stage oxidizer and fuel inlet ducts and the S-II fuel inlet duct. These circulating pumps provide a propellant flow which conditions the inlet ducts and engine turbopumps and thus provides liquid propellants at start. Natural convection is used to condition the S-II oxidizer system. With the J-2S, these recirculating systems are all eliminated.

Prelaunch checkout requirements for the J-2S are reduced because (1) propellant temperature control is not critical, (2) engine temperature is not critical, (3) the start bottle filling and temperature measurements are eliminated, and (4) through the adoption of an ambient temperature helium bottle (for valve and control actuation power) rather than a low-temperature system, the filling and temperature control of this bottle is not a critical factor.

Greater engine in-flight operational flexibility is provided by the J-2S. The engine has the low-thrust idle-mode capability (for propellant settling and space maneuvering), a propellant utilization mixture ratio control (maintained from the J-2 design), instant (multiple) start with no preconditioning requirements other than a prestart idle mode, and a built-in potential for throttling capability.

In retrospect, the J-2X experimental program has been a highly effective tool for investigating the feasibility of advanced technology concepts prior to their incorporation into an engine design. The J-2S engine, when developed, will provide extensive increased vehicle and mission capabilities which are necessary for advanced space missions, together with very significant engine, stage, and launch operations simplification.

New Concepts

New advanced designs for cryogenic liquid propellant rocket engines are under investigation.[17] These designs are focused on achieving high specific impulse performance (principally at altitude but also at sea level), on a spectrum of engine flexibility and versatility features, a relatively high engine thrust-to-weight ratio, a minimum envelope for the engine, improvements in engine cycle efficiency, and a design which will minimize engine development effort.

To achieve the first goal, high performance, a high nozzle area ratio together with high combustion, nozzle and engine efficiencies must be realized. Two quite different engine configuration have been proposed:[17, 19] first, the aerospike nozzle engine with an aero tapoff cycle, and second, a high-chamber-pressure bell-nozzle engine with a staged combustion cycle. A brief description of each engine concept follows.

The high-chamber-pressure bell-nozzle engine (Fig. 24) employs a high area ratio nozzle and the staged-combustion power cycle to achieve high performance. To maintain a relatively small size, the

AEROSPIKE

HIGH-PRESSURE, STAGED-COMBUSTION, BELL-NOZZLE

Figure 24. Engine Configurations

high chamber pressure is used to minimize the nozzle size. (A further reduction in engine length can be achieved if a retractable nozzle section is used.) The staged-combustion cycle uses a preburner in which fuel-rich combustion occurs; these gases are used to drive the turbines and are then injected with the remaining oxidizer into the main combustion chamber. Thus, all propellant exhaust gases are expanded through the main nozzle and all achieve the high performance resulting from its high area ratio.

The aerospike engine (Fig. 24) has a truncated, annular spike nozzle (radial in-flow type), which utilizes a small amount of secondary flow introduced into the nozzle base region. The primary flow (high-pressure gases) which produces the major portion of the engine thrust is exhausted from an annular-type combustion chamber and expands against the surface of the center truncated-spike nozzle (Fig. 25 and 26). The primary flow encloses a subsonic, recirculating flow field in the base region. The pressure acting upon the nozzle base contributes additional thrust to the nozzle. The addition of secondary flow into the base further increases the base pressure. As a result, the overall nozzle efficiency and thus the engine efficiency are very high. There is a limit to this gain in efficiency, and an optimum secondary flow exists for each configuration.

Figure 25. Aerospike Nozzle Flow Field

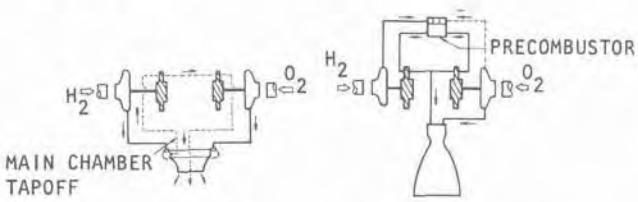
HIGH ALTITUDE (VACUUM)

INTERMEDIATE ALTITUDE

SEA LEVEL

Figure 26. Schlieren Photographs of Aerospike Nozzle Cold-Flow Tests Under Simulated Altitude Conditions

Of major significance are the power cycles of each engine. These are illustrated in Fig. 27. The aero tapoff cycle is a modification of the tapoff cycle, and basically, the staged-combustion cycle is an evolution of the expander cycle without the inherent power limitation the expander cycle incurs (Fig. 21).



AERO TAPOFF CYCLE STAGED-COMBUSTION CYCLE

Figure 27. Power Cycles

The performance of conventional engines with the gas generator or tapoff power cycle suffers because the turbine drive gases achieve only a low thrust performance; they are not expanded through the high-area-ratio main nozzle. Only the expander-cycle engine does not have this performance disadvantage; and it is power limited. The advanced engine power cycles achieve high performance because all the propellants are consumed in the engine, and these cycles are not power limited with respect to the operating chamber pressures which can be achieved.(17)

In the aero tapoff power cycle, the turbine drive gases, which are tapped from the main combustor to eliminate the need for two combustion systems, provide the base flow after driving the turbines (Fig. 27). High nozzle performance and high engine efficiency result with the addition of secondary gases.(17)

In the staged-combustion cycle, a significant portion of the propellant is burned (fuel rich) in a precombustor. The combustion gas drives the low-pressure-ratio, high-flow turbines and is then discharged into the main combustor. The remaining propellant is injected, and all propellant exhaust gases are exhausted through the main nozzle.(18) In this cycle, the precombustor, turbines, and main combustors are in series. As a result, pump discharge pressures are considerably higher for a given chamber pressure than for a power cycle, such as the gas generator or aero tapoff, in which the turbine and main chamber flows are in parallel.

The Future

The new cryogenic rocket engine designs discussed show to a limited degree the advancements and design concepts which can be projected for the future. Performance growth through engine efficiency has resulted in specific impulses very near the theoretical maximum of the propellant. The advanced engine concepts are exploiting the gains which result from higher area ratio nozzles and associated increases in chamber pressure.

Further increases in specific impulse performance are achievable through use of higher performance propellants. Through the addition of fluorine to the liquid oxygen oxidizer (FLOX) or fluorine as the oxidizer, increased performance is possible (Fig. 28).

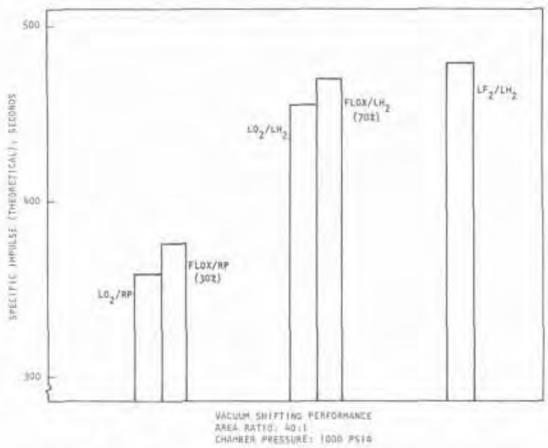


Figure 28. Performance Gain With FLOX

Fluorine added to LO_2/RP engine design offers an attractive performance gain; it has the advantage of being usable in existing LO_2/RP propellant rocket engines without major changes.[20] Several programs have demonstrated rocket engine operation very successfully with this propellant combination. Experimental Atlas engines have been run on FLOX propellants with fluorine concentration as high as 70 percent. With considerable technology experience available with the high-energy oxidizer, only the specific vehicle or mission requirement for such a performance increase in a current launch vehicle is needed to justify go-ahead for an engine qualification program.

Liquid fluorine oxidizer with hydrogen fuel, and with other fuels such as hydrazine and ammonia, have been experimentally investigated and tested in rocket engines over the past decade. Experimental versions of the RL-10 have been successfully tested using fluorine as the oxidizer with hydrogen fuel.[20]

The versatility demonstrated in experimental engines and the new concept engines will provide future missions with the required capabilities. Capabilities such as a fully controlled thrust over a wide range, unlimited restarts, and long duty cycles will be needed. Features such as precise low-level thrust will be necessary for spacecraft rendezvous. Multiple restarts will be required for efficient completion of advanced near-space missions and deep-space probes.

Extended space missions will require long engine duty cycles; that is, long firing durations and/or long time periods between firings. This means greater emphasis will be placed upon thermal control and long-term vacuum and radiation effects on propulsion system operation. Improved thermal control can be expected to extend present duty cycles of cryogenic systems significantly.

Higher rocket engine thrusts also can be predicted for the future. A quick review of the thrust growth curve of Fig. 1, together with an examination of the space missions and vehicles now being studied[21,22] quite definitely supports the projection of future thrust growth.

Both the aerospike engine and the bell nozzle engine are amenable to thrust growth. Using the engine concepts illustrated in Fig. 24 (aerospike and high pressure bell), engines of approximately 500,000 to 750,000 pounds thrust can be designed having similar configurations. Clusters of engines in this thrust range can be used where thrusts of several million pounds are required.

Designs for rocket engines with thrusts as high as 30 million pounds have been investigated in preliminary studies.[19] Such requirements result from the launch vehicle concepts developed for possible future manned interplanetary spacecraft.[21,22] Such large launch vehicles could place a payload of several million pounds into orbit. As an example, one interesting new launch vehicle design under investigation[22] has a core vehicle approximately 75 feet in diameter, is over 300 feet high and uses a hydrogen/oxygen propellant rocket engine of approximately 20 million pounds of thrust. Such a single stage-to-orbit core vehicle could place 1 million pounds into orbit; for larger payloads, strap-on pods could be added, increasing the payload capability of the vehicle to over 4 million pounds.

The aerospike engine concept can be directly employed for such a high thrust design (Fig. 29).

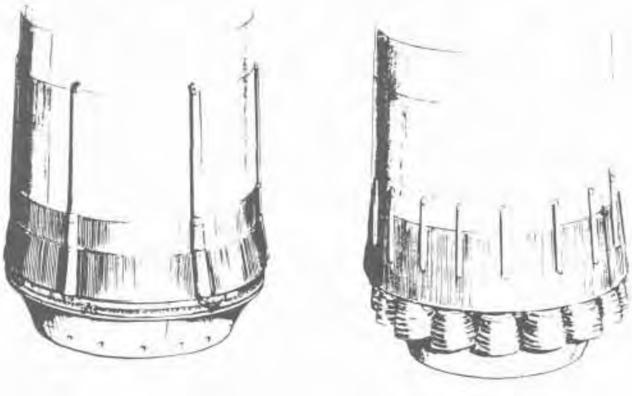


Figure 29. Advanced Booster Concepts

An engine having approximately the diameter of the stage made up of pie-shaped modules would be used (Fig. 30). The pie-shaped aerospike nozzle and combustor engine module would have its own independent turbomachinery and controls. A very significant cost advantage is derived by developing modules rather than by developing a complete engine. Further reduction in engine development costs are possible through development of the nozzle and combustor in even lower thrust (and narrower) engine segments (Fig. 30). The approach of module and segment development achieves the development cost saving through reduced hardware, smaller test facilities, and reduced test propellant consumption.

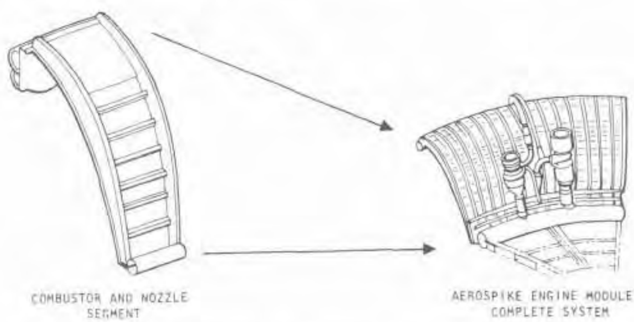


Figure 30. Aerospike Engine Module and Segment

For these greatly increased thrust levels, the concept generally employed for the bell nozzle engine is to cluster a number of these engines around a common plug (truncated spike) nozzle (Fig. 29). Such a design takes advantage of the performance gain of the annular nozzle, and an advantage in cost is realized by developing the lower thrust bell nozzle engine modules and then integrating them around the center plug.

If our nation's future space mission goals are directed toward manned ventures into interplanetary space, larger improved launch vehicles most certainly will be developed. In what steps this will be (i.e., whether the next vehicle has a 30-millionpound thrust booster, or is an intermediate step such as an uprated Saturn V; or whether some type of reusable orbital transport) requires further definition of the mission and spacecraft. It is not unlikely that several new launch vehicles of various concepts will be developed during the next 25 years.

Although these rocket engines and vehicles may seem far in the future and perhaps of questionable realism, we need only to look at the growth in the last 20 years, from the Viking, Vanguard, and Redstone to the Saturn V, to realize such continued growth is possible and probable.

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References

1. Goddard, R. H.: Rockets, The Smithsonian Institute of Washington, Washington, D.C., 1936.

2. Goddard, R. H.: Rocket Development (Foreword), Prentice-Hall, Inc., New York, 1948.

3. Burgess, E.: Rocket Propulsion, Chapman and Hall, Ltd., London, 1954.

4. Humphries, J.: Rockets and Guided Missiles, The Macmillan Company, New York, 1956.

5. Gatland, K. W. and A. M. Kunesch: Space Travel, Philosophical Library, Inc., New York, 1953.

6. Ley, W.: Rockets, Missiles and Space Travel, The Viking Press, New York, 1957.

7. Sutton, G. P.: Rocket Propulsion Elements, John Wiley and Sons, Inc., New York, 1949.

8. Ulanoff, S.: Illustrated Guide to U.S. Missiles and Rockets, Doubleday and Co., Inc., Garden City, New York, 1959.

9. Gatland, K. W.: Development of the Guided Missile, Second Edition, Iliffe, London, 1954.

10. Bowman, N. J.: The Handbook of Rockets and Guided Missiles, Perastadion Press, Chicago, Illinois, 1957.

11. Caidin, M.: Countdown for Tomorrow, E. P. Dutton and Co., New York, 1958.

12. Rosen, M. W.: The Viking Rocket Story, Harper and Brothers, New York, 1955.

13. Hoffman, S. K.: "Large Rocket Engines for Space Vehicles and Missiles," Sir Henry Royce Memorial, lecture presented to the Royal Aeronautical Society, Derby, England, 7 November 1960.

14. S-1583, RL-10/Pratt & Whitney Aircraft, Pratt & Whitney Aircraft, A Division of United Aircraft Corp., West Palm Beach, Florida, May 1967.

15. Hoffman, S. K.: "Propulsion for Space Flight," address before the Royal Aeronautical Society, London, England, 14 April 1966.

16. Moise, J. C.: "M-1 Program Contributions to Technology," AIAA paper No. 67-520, presented at the Third Propulsion Joint Specialist Conference, Washington, D.C., July 1967.

17. Iacobellis, S. F.: "Liquid Rocket Engines: Their Status and Their Future," AIAA Paper No. 66-828, presented at the Third Annual Meeting, Boston, Massachusetts, November 1966.

18. Thompson, J. R.: J-2X, J-2 Experimental Engineering Program, NASA TM X-53591, 1966.

19. Chandler, K.: "Chemical Propulsion Research at MSFC," Research Achievements Review, Vol. 2 Report 6, NASA Technical Memorandum, NASA X-53591.

20. Kwawer, A. J. and J. J. Griffin: "Status of Atlas FLOX Propulsion System," presented at AIAA Propulsion Specialist Conference, Colorado Springs, Colorado, June 1965.

21. Tischler, A. O. and W. Cohen: "Apply Large Solid Motors to Future Launch Vehicles," Astronautics and Aeronautics, January 1966.

22. Study of Advanced Multipurpose Large Launch Vehicles, Interim Report - Summary, Boeing Co., Huntsville, Alabama, D5-13378-1.

APPENDIX A

MILESTONES IN U.S. LAUNCH VEHICLES

cryogenic liquid propellant rocket engines