History of Rocketry and Astronautics

J.D. Hunley, Editor

AAS History Series, Volume 20

IAA History Symposia, Volume 12

AN AMERICAN ASTRONAUTICAL SOCIETY PUBLICATION

Chapter 5

Development of the Jupiter Propulsion System*

Julius H. Braun[†]

Introduction

The Jupiter propulsion system was developed under a crash program that paid off with the first missile launching accomplished just over one year after assignment of the missile contract. Such rapid progress was possible because the major engine components were already in an advanced stage of development under the Navaho and Atlas missile programs.

Initial production of Jupiter rocket engines was at the Rocketdyne main plant in Canoga Park, California. In the fall of 1968, production of engines for operational missiles was transferred to Rocketdyne's plant in Neosho, Missouri, while research and development activities on advanced versions remained in Canoga Park. Development problems resulted from uprating the engine thrust level, with consequent overloading of some critical components. Minor design changes resolved those problems, and the transition from prototype to production proceeded smoothly.

---

*Presented at the Twenty-Fifth History Symposium of the International Academy of Astronautics, Montreal, Canada, 1991. This paper was originally prepared by the author while assigned to the Army Programs Office, Rocketdyne Division, North American Aviation, and it was presented at a U.S. Army Space and Missiles Symposium, 12 August 1958.

[†]U.S. Army (Retired), San Antonio, Texas.

The story of the development of the Jupiter Propulsion System is actually a short history of the Rocketdyne Division of North American Aviation, Inc. It began shortly after World War II and has continued to the present. The Jupiter rocket engine has its ancestry easily traced back to the German V-2, and even though the present engine bears little outward similarity to its earlier counterpart, the V-2 nevertheless pointed the way, both in the direction to go and the pitfalls to avoid.

At the end of World War II, North American Aviation, Inc. established an Aerophysics laboratory to study and evolve proposals for strategic missiles. The efforts of this group resulted in an Air Force contract to develop an air-breathing, supersonic cruise-type missile brought up to speed and altitude by a powerful liquid-propellant rocket booster. This was the beginning of the Navaho program.

Development was separated into six phases, each of which represented a major advance in a new and strange field. These phases went hand in hand with parallel studies of high-speed aerodynamics, heat transfer, propellant chemistry, and related subjects. Captured German guided missile hardware and reports provided most of the material for the initial studies.

Phase I consisted of assembling a German V-2 engine and operating it, with water substituted for propellants, at propellant flow rates corresponding to 56,000 pounds of thrust for a duration of 64 seconds.

Phase II was to include the manufacture and test firing of a rocket engine similar to the German V-2, but incorporating American design standards and methods of manufacturing. Although there were many modifications that resulted in some weight reduction and production compatibility, it was essentially a facsimile of the German model. This engine, though never subjected to actual firing tests, served as a valuable research device in the coordination of newly acquired knowledge.

Phase III was the application of experience accumulated in the previous phases to the design and development of a completely new rocket engine of lighter weight, better performance and higher thrust (75,000 pounds) for the Navaho missile. This engine was developed to boost a ramjet-equipped missile to Mach 2.85 velocity at 38,000 feet altitude.

The firing of engines developed in Phases I and II was bypassed because the Phase III engine was test fired ahead of schedule in May, 1950. About this time the objectives of the Navaho program were modified to increase the size and range of the missile. To meet these additional requirements, the Phase IV engine was begun. The Phase III engine, subsequently modified to operate for 110 seconds, was no longer required for the Navaho, and it was taken over by the Army Ordnance Corps. From this engine evolved the series that led to the Redstone missile and the Jupiter C first stage for the Explorer satellite.

Phase IV, fully under way by mid-1951, constituted a major advance in the development of large rocket engines, and it was the first big step away from the now-old V-2. The double-wall chamber was replaced with a lightweight, tubular chamber; a high-speed turbopump, using a bipropellant gas generator fed from primary propellants, was developed; the 75-percent alcohol fuel was replaced with 92.5-percent alcohol for higher specific impulse; and system "bootstrap" starting was developed. The resulting engine developed 120,000 pounds of thrust.

Phase V was the development of a multiple-thrust-chamber engine, first fired during September, 1953. This phase proved the feasibility of a system employing dual thrust chambers fed by a common gas generator. The resulting engine developed 242,000 pounds of thrust and has since been flown in the G-26 Navaho Booster.

Phase VI was the design, development, and production of the G-38 booster, a three-thrust chamber, three-turbopump, single gas generator engine, which delivered over 400,000 pounds of thrust during initial tests in January, 1956.

Table 1

DEVELOPMENT OF A ROCKET ENGINE CAPABILITY

PHASE I: Assemble V-2 Engine and Operate with Water (56k/62 sec)
PHASE II: Build a V-2 Engine Facsimile (Never Hot-Fired)
PHASE III: Design and Develop Complete New Rocket Engine Navaho
(75k/110 sec)
PHASE IV:Development of New Concept Rocket Engine (120k/Bootstrap)
REAP: Use of Hydrocarbon Fuels
PHASE V: Dual Thrust Chamber Engine - G-26 Booster (242k, Single GG)
PHASE VI: Triple Thrust Chamber Engine - G-38 Booster (405k to 500k)

A Rocket Engine Advancement Program (REAP) began in March, 1952, to evaluate liquid oxygen—hydrocarbon propellants in large liquid-propellant rocket engines. As a result of the findings of this investigation, it was determined to develop large engines which would use hydrocarbon fuels. This program also provided the basic information for much of the development of large engines. The evolution of high-speed, gear-driven turbopumps, incorporating the inducer-impeller combinations and the improvement of thrust chambers, instrumentation, starting techniques, and propellant combinations, have all served to accelerate the progress that was reflected in subsequent engines. As a result of these advances, a contract was negotiated to provide rocket engines for the Atlas missile.

In the early summer of 1954, Redstone Arsenal, through the Air Force, gave North American Aviation, Inc. a contract to design a single-barrel gimbaled engine using LOX-JP type fuel for an advanced version of the Redstone missile. This engine was to use the basic hardware being developed for the Atlas program.

During the last half of 1954, Air Force Contract AFO4(645)-1, which had originally been written to cover the effort directed toward developing Atlas propulsion systems, was expanded to include rocket engine systems for both the Titan and Thor missiles. The Thor and Jupiter programs made use of the 135,000-pound, single-thrust-chamber engine, previously utilized to support Atlas booster development, to develop a 150,000-pound-thrust, vernier-supporting system capable of powering an intermediate-range ballistic missile.

From the original design go-ahead in mid-1954, the Jupiter propulsion system was developed under a crash program that paid off with the first missile launching accomplished 15 months after the Department of Defense initiation of the IRBM program, and just a little over one year after full-scale production of Jupiter rocket engines began.

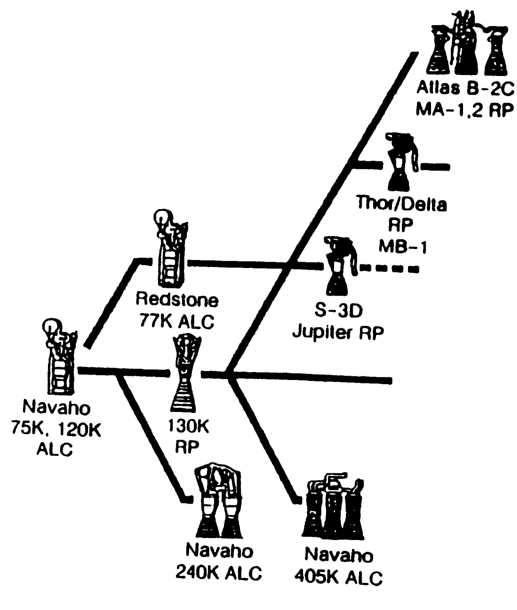

Figure 1

Some idea of the compression of development schedules can be seen from the following time table:

Table 2

Design Go-Ahead for an Advanced Redstone Power Plant, July 1954
IRBM Program Initiated by Defense Department, November 1955
First Engine Tests, November 1955
Mockup of Engine Delivered, January 1956
First Engine Delivered, June 1956
First Jupiter Launching, March 1957

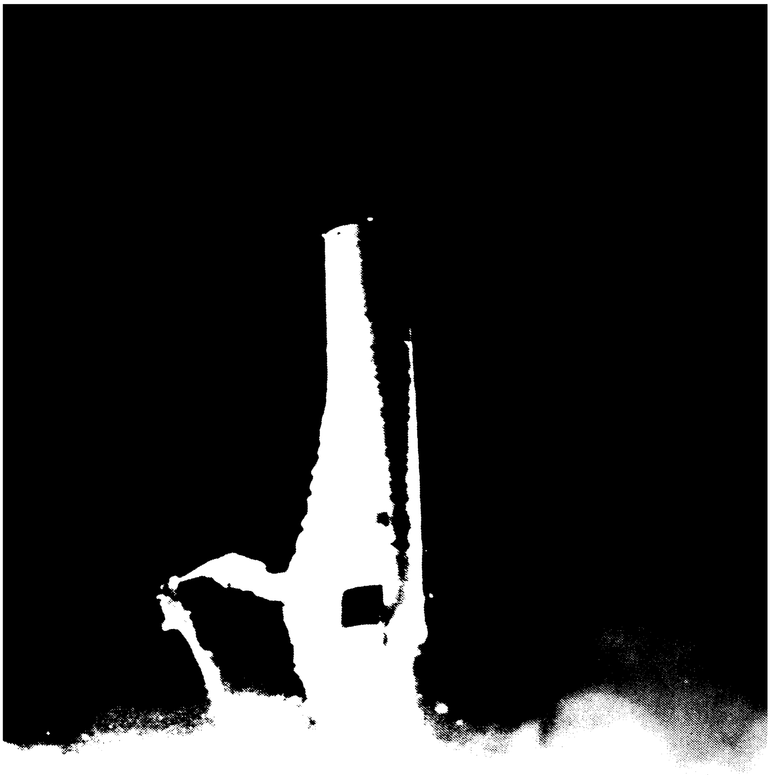

Figure 2 The Juno II, a Jupiter intermediate range ballistic missile with upper stages.

Propulsion System

The S-3D rocket engine was developed to power the Jupiter missile, a single-stage, liquid-propellant, rocket-powered, intermediate range ballistic missile. The overall length of the missile is approximately 60 feet; the diameter is 105 inches. The missile has a takeoff weight of approximately 110,300 pounds and a dry weight of approximately 10,100 pounds.

The S-3D rocket engine is a bipropellant system capable of developing 150,000 pounds of thrust at sea level conditions for a duration of 178 seconds. The engine uses a single thrust chamber and a single turbopump design. It is controlled through an electrical system, operated in conjunction with pneumatic and hydraulic systems. The engine is a single-start type, and it will operate at a preset thrust level with no provisions for intermediate thrust control. An electronic-hydraulic thrust chamber pressure control system has been developed for the engine to maintain a constant chamber pressure through a closed-loop servo system.

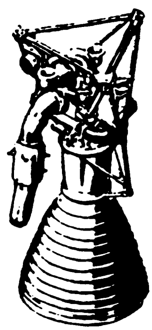

Figure 3

Major components that comprise the engine system include a power package consisting basically of a turbopump, a gas generator and hot gas ducting, and a gimbaled thrust chamber assembly. Supporting elements necessary to complete the rocket engine system include electrical, pneumatic and hydraulic systems, a turbopump lubricating system, an electronic-hydraulic chamber pressure control system, and a propellant system.

Electrical and hydraulic power for controlling the engine are supplied from an external source until the missile is airborne; then the missile electrical system supplies the required electrical energy, and hydraulic power is supplied from an engine operated pump.

Pneumatic power is supplied from missile spheres, which are continually replenished from a ground source until the missile is airborne.

The S-3D rocket engine used in the Jupiter missile is basically the same as the MB series of engines used in the Thor missile. A more detailed description of the S-3D Jupiter propulsion system follows.

The thrust chamber is fabricated of formed nickel tubes, brazed together and supported with external steel bands and rings. A bell-shaped exhaust nozzle is employed. Injector and chamber pressure is 525 psia. With RP-1 (a kerosene-type hydrocarbon) as fuel (and regenerative coolant) and liquid oxygen as the oxidizer, a thrust chamber specific impulse at sea level of 252 seconds nominal is obtained. The multi-ring, flat-plate injector of like-on-like pattern has an independently fed central fuel spray disk to establish an ignition flame before mainstage is signaled.

Ignition is accomplished by a radial-firing, pyrotechnic igniter screwed into the center of the injector spray disk. For thrust vector control, the thrust chamber is capable of gimbaling through a 14-degree cone angle by means of a cross bearing gimbal block mounted on top of the injector dome. Flexible stainless-steel sections in both propellant feed ducts are necessary for gimbaling. Hydraulic actuator arms operate between outriggers on the thrust chamber and jibs on the fixed thrust frame to accomplish gimbal operation. Pneumatically operated, butterfly-type main propellant valves are installed in the main ducts adjacent to the chamber inlet ports.

A two-stage turbine assembly is driven by fuel-rich gases generated in a spherical gas generator. At the 150,000-pound thrust level, the turbine rotates at 23,000 rpms, generating approximately 2,800 horsepower. Exhaust gases are ducted overboard, providing about 500 pounds of additional thrust at sea level.

The two centrifugal pumps, with their inducers, are geared down at a ratio of 4.88 to 1 (turbine/pump rpm). Two accessory drive pads, with 100 horsepower available, are used for a hydraulic pump for thrust gimbal actuation. The propellant pumps are so mounted that they will take the forces and rising pump inlet head condition at more than 15 g. A lubrication system of 200 seconds capacity employs pneumatic pressurization to feed the bearing lubrication and gear-cooling jets.

The gas generator combustor, which feeds the turbine, employs a simple like-on-like injector at its head and an additional fuel-cooling spray nozzle at its center. Mixing is aided by a cylindrical basket in the combustor. A fuel-rich mixture ratio is used, developing gases heated to 1,200°F. The combustor is otherwise uncooled.

Propellant flow is initiated by opening a double-blade, single-pneumatically actuated valve mounted as an integral part of the combustor. Flow rate is controlled on the liquid oxygen side by a throttle valve operating to maintain a constant chamber pressure in the main thrust chamber. Fuel flow is controlled by a fixed calibrated orifice.

Propellants are ignited by three pyrotechnic cartridges, which are mounted on the combustor body. The gas generator is started by propellant flow from two spherical, pneumatically-pressurized ground-mounted tanks, which also serve to supply the thrust chamber with igniter fuel. Once the main pump discharge pressures have risen to approximately 80 percent of steady-state operation, the gas generator "bootstraps" and the engine become self-sustaining.

Missile roll control is accomplished by swivelling the gas generator turbine exhaust duct 25 degrees each side of directly rearward. This does away with the requirement for vernier rocket engines to counteract roll tendencies of the missile.

The pneumatic system consists of missile-supplied 3,000 psi gaseous nitrogen which is filtered and then reduced to 760 psi, the normal working pressure. This gaseous nitrogen is then used to operate valves, provide purges, pressurize the lubrication system, and pressurize the turbopump gearbox.

One four-way solenoid valve operates the main oxidizer valve and ignition-stage fuel valve. A second solenoid valve operates both the main fuel and gas generator valves. These two solenoid valves are mounted on a single manifold.

A 28-volt dc electrical power supply is required to operate the engine. All signals come from an integrated missile tail distributor box through a harness to operate control valves and receive position indications from microswitches on the main valves to provide certain of the pre-firing and pre-takeoff safety and sequence functions.

A 115-volt, 60-cycle ground power supply operates various heaters used to prevent freezing of components exposed to liquid oxygen. The 115-volt 400-cycle power supplied to a thrust control computer amplifier comes from the missile 400-cycle system.

The engine hydraulic system consists of a pump driven directly off the turbopump accessory drive pad, a low-pressure accumulator, filters, and a high-pressure manifold. The system is used to provide actuating power for the main thrust chamber gimbal actuators, the turbine exhaust roll control swivel actuator and the thrust control servo valve.

The thrust control system is designed to maintain the thrust chamber pressure (absolute) constant at some predetermined level (corresponding to the desired thrust during mainstage) within a tolerance of ± one percent. Control is accomplished by sensing chamber pressure, comparing the magnitude to a predetermined value (P_c level conforming to the desired thrust level), and controlling LOX flow to the gas generator in such a manner as to maintain a zero error between the level of the sensed chamber pressure and the predetermined reference level.

The chamber pressure is monitored by a pressure transducer, which produces an electrical output proportional to the magnitude of pressure. The output signal of the pressure transducer is summed with a reference voltage proportional to the predetermined pressure level desired during mainstage. Any existing error is amplified by the servo amplifier, which produces an output with respect to magnitude and direction of the error signal, such as to reposition the LOX throttle valve (by means of the hydraulic servo valve), thus orificing the LOX flow to the gas generator in a direction to reduce the error in chamber pressure to zero. Control may be accomplished in this manner, since chamber pressure is an implicit function of turbine power which, in turn, is a function of the LOX flow rate to the gas generator.

Because starting is the most difficult phase in the operation of a liquid propellant rocket engine, the S-3D Jupiter propulsion system is started in an event-ladder sequence, in which satisfactory completion of one event signals the next step to take place.

First, the main missile propellant tanks are pressurized to approximately 40 psi. When this is complete, a signal is given to pressurize the ground-mounted fuel and liquid oxygen start tanks. Pressure switches in these tanks "pickup," closing a circuit, which fires a pyrotechnic igniter in the main chamber. Burn-through links in this igniter then signal for the main liquid oxygen valve and the igniter fuel valve to open. The pilot flame that is produced burns through a link wire stretched across the thrust chamber nozzle, and this then signals the gas generator igniters to fire. Again, burn-through of links in the gas generator igniters signals the opening of the main fuel valve and the gas generator blade valve. Fuel and liquid oxygen from the ground start tanks begin flowing under 650 pounds pressure into the gas generator, burn, and then start the turbopump turning. The pump accelerates rapidly as the fuel fills the thrust chamber cooling jacket, and the main fuel flow arrives at the injector at a fairly high pump speed.

During this time interval of turbopump acceleration, the liquid oxygen flow has also increased, consequently transition to full thrust takes place under high flow rates, and chamber pressure rises rapidly. Some of each propellant is diverted from the high-pressure ducts to feed the gas generator, and during transition, as the pressure of the propellants rises to finally become greater than the pressure of the ground start system propellants, the engine overrides these ground sources to become self-sustaining. This is generally referred to as *bootstrapping.*

Engine shutdown is a simple operation in which the solenoid control valves are simultaneously de-energized. Pneumatic restrictors in the vent ports of the opening control side of their respective solenoids are employed to sequence the liquid oxygen valve to close shortly after the gas generator valve has closed, with a slight time lag before closing of the main fuel valve. The closing time of the liquid oxygen valve is the predominant factor in control of cutoff impulse, the later closing fuel valve having little effect. The speed with which the liquid oxygen valve closes is determined by the maximum hydraulic surges which can be withstood by the high-pressure and low-pressure liquid oxygen ducts.

Development Problems

Almost simultaneously with the delivery of the first Jupiter engine to the Army Ballistic missile agency (ABMA), the inevitable uprating of thrust began. The original Atlas sustainer that preceded the S-3 developed 120,000 pounds of thrust. This was raised to 135,000, then 139,000, and finally 150,000 pounds of thrust, and each jump brought with it a complete new set of headaches. The thrust chamber was the first to feel the squeeze. A larger nozzle of bell configuration proved less rigid than the former conical nozzle, and distortion was encountered. This, coupled with some rather nasty resonant frequency problems, was finally alleviated by the addition of external stiffening rings in the expansion section. More on that will be mentioned later.

The increased power load on the turbine resulted in numerous blade failures. Redesign of the blades to incorporate interlocking tip shrouds not only solved this problem, but it also resulted in an efficiency improvement of several percent. Flight test failures of two missiles were attributed to turbopump gearbox malfunctions. An extensive investigation by Rocketdyne and ABMA showed several possible areas of weakness, and each was examined in great detail. Some critical bearing retainers were redesigned to preclude bearing walking; a quill shaft connecting the turbine to the gearbox was redesigned; a non-foaming lubricating oil was investigated; lubrication of bearings was improved; and last, but probably most important, the gearbox was pressurized to approximately 5 psi, so that the near-vacuum environment of high altitude flight would not negate the lubricating qualities of the oil. Numerous successful flights since these fixes have demonstrated their effectiveness.

The propellant feed system and ducting also showed the effects of uprating. Changes in manufacturing techniques and the use of larger ducts was the cure. A later problem arose affecting both the high pressure ducting and the thrust chamber. A natural frequency output of 500 cps from the fuel pump seemed to be amplified by certain mitered, vaned elbows in the fuel high-pressure duct, and it was then transmitted to the bell-shaped thrust chamber. Accelerometers mounted on the thrust chamber dome registered up to 40 g at 500 cycles/second, and high-speed films showed pronounced flickering of the flame. Three additional stiffening rings on the nozzle shifted the chamber frequency and eliminated the phenomenon.

The biggest problem in operating sequence was getting away from the V-2 thinking, whereby an engine had to go through ignition, prestage, and then into mainstage, pausing at each step while an observer decided whether operating conditions were suitable before proceeding. Now, sequencing is all automatic, prestage has been eliminated, and the only control an operator has after signaling *Test Start* is to cause *Cutoff* if he notes that something is not correct. Of course, the ladder-sequence in the engine control system will also automatically signal a "Cutoff" if any one step does not take place within its allotted time. The control system is an excellent example of automation.

Early in the Jupiter missile program, it was felt by the ABMA guidance designers that, due to structural limitations, a 10 g acceleration limit must be placed on the missile. This meant that some method had to be devised for throttling the engine. A hot gas valve in the duct connecting the gas generator to the turbine was tried and discarded. Then a thrust control closed-loop servo system was designed, with a capability for reducing the thrust to some predetermined level. This system was incorporated into the thrust control amplifier and only served to make it more complicated and less reliable. Fortunately, the guidance designers were found to have been ultra-conservative, the guidance equipment proved to be very rugged, and the acceleration limiting requirement was dropped.

Simplification of the control system of the engine proceeded apace with other improvements. Many components not required during engine operation were removed to the ground. Nine solenoid valves were replaced with only two. Gaseous nitrogen was substituted for hard-to-get helium. Various safety interlock pressure switches found to be unnecessary were discarded. All this resulted in a much cleaner engine.

A configuration control procedure was initiated to guarantee that all engines for a given squadron of missiles would be identical. This required the setting of engine design change points to correspond with blocks of missiles. The necessary coordination of ground support equipment design was handled through a series of compatibility charts and the assignment of MOD numbers to groups of engines.

In the transition from prototype to production, quality assurance is extremely critical. Extensive inspection procedures were followed throughout, and a continuous education program was used. Expanding production and the urgency of the program put a severe strain on the manufacturing groups. Source inspection and continuous monitoring by Rocketdyne and Ordnance quality control groups served to eliminate any quality-control problems as they arose.

Production

Initial production of Jupiter rocket engines was at the Rocketdyne main plant in Canoga Park, California. This included the fabrication of R&D prototypes and engines for the first squadron of operational missiles. There were numerous problems in concurrently producing special mission engines of non-standard configuration alongside the standard engines. Methods and procedures were devised so that there was a minimum of delays because of dual-type production.

Acceptance firing of all delivered engines was initially performed at the 1,700-acre Propulsion Field Laboratory located in the Santa Susana Mountains. This testing was in addition to the normal development testing that was also conducted to prove out new engine features.

In the fall of 1958, production of Jupiter engines for operational missiles was transferred to Rocketdyne's plant located at Fort Crowder, near Neosho, Missouri. This second major rocket engine manufacturing facility was devoted strictly to production and acceptance testing. It had been in operation since January, 1957, producing engines for the Thor missile. When all Jupiter operational engine production was phased into the Neosho plant, the main Canoga Park facility was able to concentrate on research and development support and the production of prototypes and advanced engines.