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DEVELOPMENT OF THE ROCKET ENGINE FOR THE JUPITER MISSILE

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BACKGROUND

In 1954, during full-swing production of the 78,000-lb thrust REDSTONE engine, the Propulsion Division (now the Rocketdyne Division) of North American Aviation, Inc. received an Army Ordnance contract to design a scaled-up powerplant of 135,000-lb thrust for use in an advanced, longer-range version of the REDSTONE ballistic missile.

This engine was to utilize the basic engine components and principles of the ATLAS propulsion systems which had been under development by NAA for several years. Jet engine fuel was to be used for improved specific impulse over the alcohol fuel used in the REDSTONE, and liquid oxygen was retained as the oxidizer. The hydrogen peroxide system used in the REDSTONE for turbopump drive was to be replaced by a gas generating system utilizing basic engine propellants. Major improvements in turbopumps, thrust chambers, controls, and other subsystems were to be incorporated. Gimbaling of the thrust chamber for directional control was to replace the REDSTONE system's carbon jet vanes.

In November 1955, when the Department of Defense established the requirement for an intermediate range ballistic missile with a 1500-nautical-mile-range, development of the advanced REDSTONE engine and an Air Force counterpart was redirected toward this goal by uprating the thrust to 150,000 lb and increasing the operating time.

Two specific missile programs were established in January 1956, the Army JUPITER (then intended as a Fleet Ballistic Missile for the Navy) and the Air Force THOR. This permitted NAA, by technical coordination with the services, to tighten up the previously general specified engine requirements, operating regimes, and installation details for the two parallel programs.

From the original design go-ahead in mid-1954, the JUPITER propulsion system was developed under a crash program that made possible the first missile launching 15 months after the Department of Defense initiated the IRBM program, and slightly over one year later, full-scale production of the JUPITER rocket engine began.

The unusual compression of the development schedule can be seen from the following time table.

Design go-ahead for an advanced REDSTONE power plant	July 1954
IRBM program initiated by Defense Department	Nov 1955
First engine tests	Nov 1955
Mockup of engine delivered	Jan 1956
First engine delivered	June 1956
First JUPITER launching	Mar 1957

This unprecendented schedule compression was made possible by the wealth of large rocket engine experience previously accumulated by NAA on the NAVAHO, REDSTONE, and ATLAS propulsion systems. The availability of extensive component development pits and engine firing stands at the Rocketdyne Propulsion Field Laboratory was also a major aid in the accelerated program. However, the true key was in the decision made in 1946 by NAA management to develop its own large, liquid propellant rocket engine for the NAVAHO program then being proposed. The V-2 engine was chosen as the basis for extrapolation of design, and several German rocket experts from Peenemunde were added to NAA's staff. The family tree of rocket engines, stemming from this decision, is presented in Fig. 1.

GENERAL DESCRIPTION

The JUPITER rocket engine (Rocketdyne Model S-3D) burns RP-1, a kerosene-type fuel, and liquid oxygen to develop a sea-level thrust of 150,000 lb for a nominal duration of 178 seconds. Propellants are supplied to the gimbaled, regeneratively cooled thrust chamber by two centrifugal pumps driven by a hot gas turbine. The turbine is powered by a gas generator using propellants bled from the turbopump discharge ducts. Three main valves and two auxiliary valves, operated by two solenoid valves and a hydraulic servo valve, and several supplementary check valves control the propellant flow to the thrust chamber and gas generator.

Other subsystems include lubrication for the turbopump assembly, a chamber pressure control system, and hydraulic, electrical, and pneumatic systems. A heat exchanger in the turbine exhaust duct vaporizes liquid oxygen to maintain the necessary pressure in the missile liquid oxygen tank. Ducts, pipes, and tubing for propellants, hot gas, and pneumatic and hydraulic oil routing connect the operating components. A tubular-steel thrust frame provides a common mount for the engine components and transmits the thrust from the engine to the missile structure.

Starting propellants for the gas generator and main thrust chamber ignition flame are supplied from two small tanks mounted on the missile launch pad. Electrical and hydraulic power are also ground supplied prior to liftoff. For pneumatic requirements, spherical high-pressure gaseous nitrogen tanks in the missile are continuously replenished from a ground supply until the missile rises from the launch pad.

The JUPITER engine is similar in many respects to the MB series of engines for the THOR missile. The outstanding points of difference include the more precise thrust control system of the JUPITER engine, and the use of ground-mounted start tanks rather than the engine-mounted, double-duty (start and vernier engine operation) tanks of the THOR system.

COMPONENT DESCRIPTION

Thrust Chamber Assembly

The thrust chamber body is fabricated of specially formed, thin-wall nickel tubes, brazed together and stiffened with external circumferential steel bands and rings. The fuel coolant travels a double pass flow path. Fuel enters every other tube of the thrust chamber through a steel inlet manifold, travels down the walls of the chamber to the exit end, and returns through alternate tubes to a second manifold which supports and feeds the injector. The exhaust nozzle is bell shaped for improved efficiency. At an injector-end chamber pressure of 525 psia, a sea-level nominal specific impulse of 252 seconds is obtained.

A multi-ring, flat-plate injector with a like-on-like impingement pattern and alternate fuel and oxidizer rings is used. An independently fed, central fuel spray disc establishes a hot core ignition flame before the arrival of mainstage propellant flow. Fuel is introduced inward radially through cored passages in the injector body, and the oxidizer is supplied to its rings through an aluminum dome fastened to the top of the injector plate.

Primary ignition is accomplished by a pyrotechnic igniter firing radially outward. The upper stem is screwed into the center of the injector fuel spray disc.

For thrust vector control, the thrust chamber gimbals through a 14-deg cone angle by means of a cruciform bearing gimbal block mounted on top of the injector dome. Flexible stainless-steel sections in the propellant ducting permit this motion between the moving chamber and the fixed pumps. Hydraulic actuators operate between outriggers on the thrust chamber and jibs on the fixed thrust frame providing movement in the pitch and yaw planes.

Pneumatically operated, butterfly, gate-type main propellant valves are installed in the main ducts adjacent to the thrust chamber inlet ports.

Turbopump Assembly

The propellant pumps are driven by a midget, two-stage gas turbine which generates approximately 2800 bhp at 28,000 rpm. It is powered by a 1200 F, 12.5-lb/sec gas flow produced by the gas generator. After passing through the turbine, the hot gases are ducted through the heat exchanger which pressurizes the missile liquid oxygen tank. The exhaust gases are then ducted overboard, through a hydraulically controlled swivel nozzle which overcomes the roll tendencies of the missile. An extra thrust of some 400 lb is produced when the exhaust gases are directed rearward.

The centrifugal propellant pumps are reduced in speed by a factor of 4.88 from the turbine speed by a three-stage gear train. Inducers are provided before the pump impellers to reduce the NPSH requirements. Mounting of the pumps is designed to take the rising head loads and acceleration effects because the missile reaches 15 g before the end of engine operation.

An accessory drive pad permits mounting of the hydraulic pump, a speed indicating tachometer generator, and an overspeed cutoff governor for ground testing.

The lubrication system consists of a pneumatically pressurized tank of 200 seconds capacity which supplies lubricating oil to the bearings and gears. A 3- to 5-psi pressure differential is maintained in the gearbox to prevent the adverse vacuum effects on the lubricating oil experienced during early flights.

The turbopump assembly, uprated from the original 120,000-lb design capacity with unexpected forces exerted on it in ballistic missile flights, has demanded a concentrated development effort with ABMA contributing heavily in a cooperative effort to overcome the problem areas which could not be pinned down in static firing tests.

Gas Generator

The spherical combustion chamber of the gas generator employs a simple like-on-like injector at its upper end and an additional fuel-cooling spray nozzle at its center. Mixing to achieve a homogeneous fuel-rich gas is aided by a cylindrical basket in the combustor.

Propellant flow to the gas generator is initiated by the opening of a double-bladed, pneumatically actuated valve mounted as an integral part of the combustor. Liquid oxygen flow is regulated by a throttle valve as part of the main chamber pressure control system. A fixed calibration orifice maintains the fuel flowrate at the desired level.

Two pyrotechnic cartridges threaded into bosses on the gas generator body provide ignition of the initial inflow of propellants from the ground start tanks. When this initial flow has accelerated the pumps to approximately 80 percent of full discharge, pressure flow from the pump "bootstrap" bleed lines takes over with the ground feed being check-valved off. The engine then becomes self sustaining.

Pneumatic System

Missile-supplied gaseous nitrogen at a starting pressure of 3000 psig is filtered and regulated down to 750 psig to operate valves, provide purges, and to pressurize the lubricant tank and the turbopump gearbox.

Two four-way solenoid valves are mounted on the single pneumatic manifold. One of these operates the main oxidizer valve and the ignition stage fuel valve. The other, when signalled in sequence, simultaneously opens the main fuel and gas generator valves.

Electrical System

A 28-vdc 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 prefiring and pretakeoff safety and sequence functions.

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

Hydraulic 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.

Thrust Control System

The thrust control system is designed to maintain the thrust chamber pressure (P_c) (absolute) constant at some predetermined level (corresponding to the desired thrust during mainstage) within a tolerance of ± 1 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 liquid oxygen 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 chamber 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 liquid oxygen throttle valve (by means of the hydraulic servo valve) thus orificing the liquid oxygen 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 liquid oxygen flow rate to the gas generator.

ENGINE OPERATING SEQUENCES

Starting Sequence

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

During this time interval of turbopump acceleration, the liquid oxygen flow is also increased. Consequently transition to full thrust takes place under high flow rates, and chamber pressure rises to full thrust in a fraction of a second. Some of each propellant is diverted from the high-pressure ducts to feed the gas generator and the engine overrides the ground tank feed to become self sustaining.

Shutdown Sequence

For engine shutdown, the two solenoid control valves are simultaneously de-energized. Pneumatic restrictors in the vent ports of the opening control side of their respective solenoid valves 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. Speed with which the liquid oxygen valve may be closed, is limited by the maximum hydraulic surges which can be withstood by the high-pressure and low-pressure liquid oxygen ducts.

DEVELOPMENT PROBLEMS

Uprating of Thrust Level

Almost simultaneously with the delivery of the first JUPITER engine to ABMA, the inevitable uprating of thrust began. The original ATLAS sustainer that preceded the S-3D, developed 120,000 lb thrust. This was raised to 135,000, then 139,000, and finally 150,000 lb thrust, with problems encountered at each step.

In uprating the engine from 135,000 lb thrust to 150,000 lb thrust, a larger thrust chamber nozzle of bell configuration proved less rigid than the former conical nozzle and distortion was encountered. This, coupled with resonant frequency problems was finally alleviated by the addition of external stiffening rings on the expansion section.

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 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. Bearing retainers were added to preclude bearing walking; a quill shaft connecting the turbine to the gearbox was redesigned; lubrication of the bearings was improved; and 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 the incorporation of these fixes have demonstrated their effectiveness.

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The propellant feed system and ducting also showed the effects of uprating. Changes in manufacturing techniques and 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 larger diameter, fuel high-pressure duct and was then transmitted to the bell-shaped thrust chamber. Accelerometers mounted on the thrust chamber dome registered up to 40 g at 500 cps, and high-speed instrumentation motion pictures showed pronounced flickering of the flame. Three additional stiffening rings on the nozzle shifted the chamber frequency and eliminated the phenomenon. Duct leakage problems have also provided considerable improvements in flanges and gaskets.

Operating Sequence

Achievement of a satisfactory operating sequence, both from the engine and missile viewpoint, required concentration on starting and shutdown tests early in the program. Obtaining smooth ignition of both the starting flow and mainstage arrival flow, with the latter arriving at controlled conditions of pump speed to avoid an undesirable chamber pressure overshoot, occupied a large portion of initial development. Transition from ground feed of the system to engine self-sustaining condition or "bootstrapping" at a desired power level, with a dip in thrust at this point to be avoided, also required considerable testing. Satisfactory priming of the liquid oxygen dome and fuel injector manifolds at the instant of mainstage thrust buildup had to be achieved. Controlled valve sequencing to reduce cutoff impulse, and make it reproducible, was also a major objective during this program. The event ladder sequence, which will shut down the engine if any one step is not properly performed, is an excellent example of engine automation.

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Throttling and Thrust Control

At the start of the JUPITER program, it was believed that certain guidance and structural problems dictated a limitation of flight accelerations to a maximum of 10 g. Early engine throttling experiments included devices to control the flow of hot gases in the duct connecting the gas generator to the turbine. The results were not gratifying. Next the thrust control system was modified to introduce an error signal which would gradually throttle down the liquid oxygen flow to the gas generator. This resulted in complicating the system, and problem areas were produced. Fortunately, it was established that neither the structure nor guidance required g limitations, and this feature was dropped, leaving the chamber pressure control system as a relatively straightforward fixed-level system.

Simplification

Simplification of the control system of the engine has proceeded apace with other improvements in controlled block changes to the engine. Many components not required in flight have been made ground items. The original nine solenoid valves on the engine have been reduced to two. Various safety interlock pressure switches, found to be unnecessary and in fact troublesome, have been discarded. Gaseous nitrogen has been substituted in the pneumatic system for logistically limited helium.

Quality Assurance

In the transition from prototype to production, quality assurance is extremely critical. Extensive inspection procedures are followed throughout, and a continuous education program is used. Expanding production

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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 has served to eliminate quality control problems as they arise.

PRODUCTION

JUPITER rocket engines were produced initially at the main Rocketdyne plant in Canoga Park, California. This included fabrication of development flight missile engines, in-house development test engines, and engines for the first squadrons of operational missiles. Acceptance testing of all delivered engines, until recently, was performed at the 1700-acre Propulsion Field Laboratory in the adjacent Santa Susana Mountains. Normal development testing on new engine features and flight backup firings have also been concentrated at this facility.

In the fall of 1958, production of JUPITER engines was transferred to Rocketdyne's Neosho, Missouri plant. When this transfer operation has been completed, the Canoga Park facility will concentrate on experimental, developmental engines and preproduction prototypes. An example of this trend is the recently initiated program on the advanced JUPITER (NAA Model H-1) engine intended for the multiengine application in the Army SATURN ~~JUNO V~~ booster. From the technology acquired in the past three years of development and flight testing of IRBM engines, advanced and further simplified engine components and operating sequences have been evolved for this second generation-type rocket engine which, it is hoped, will enable the United States to take giant strides into space.

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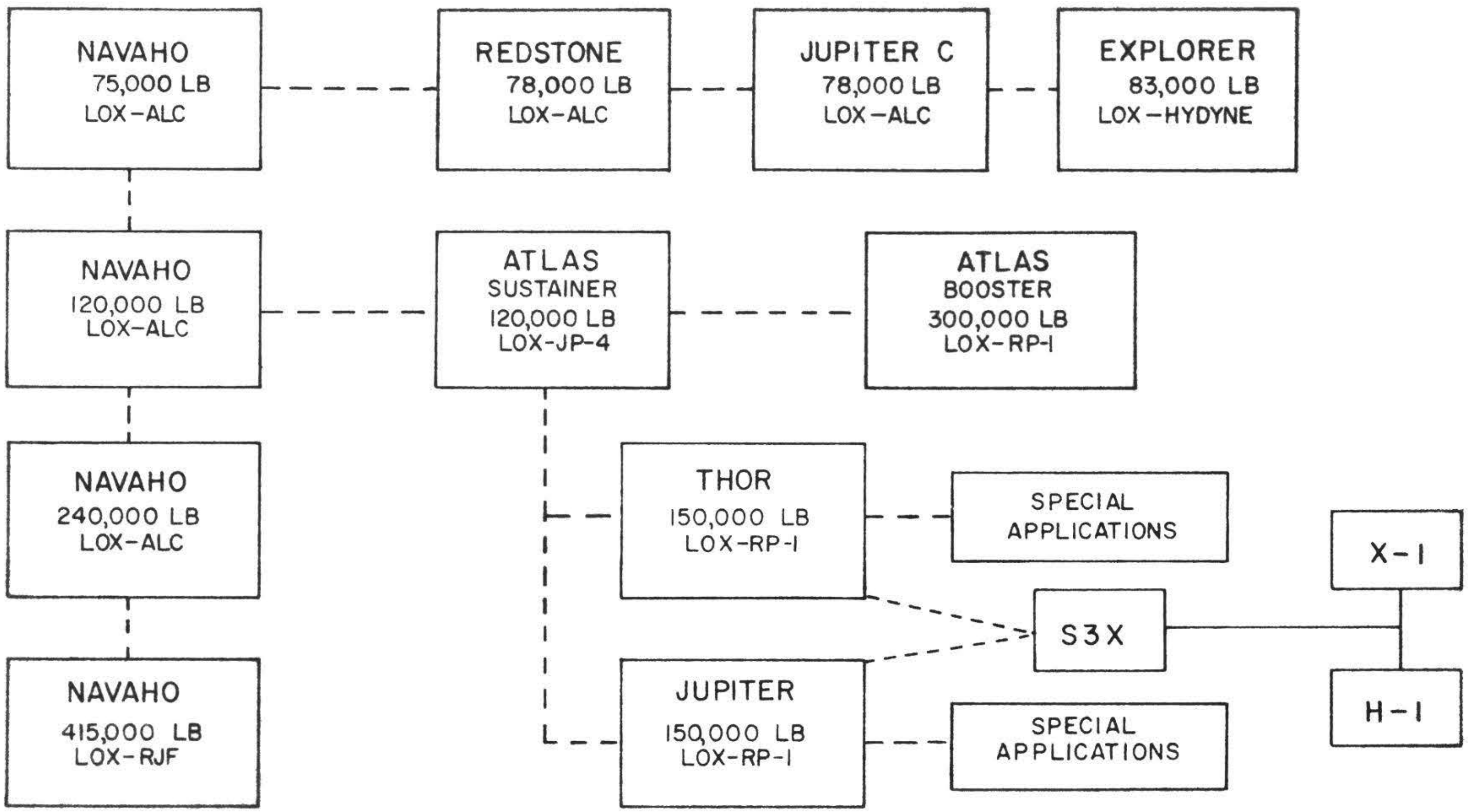

Figure 1. Family Tree of Rocketdyne Engines

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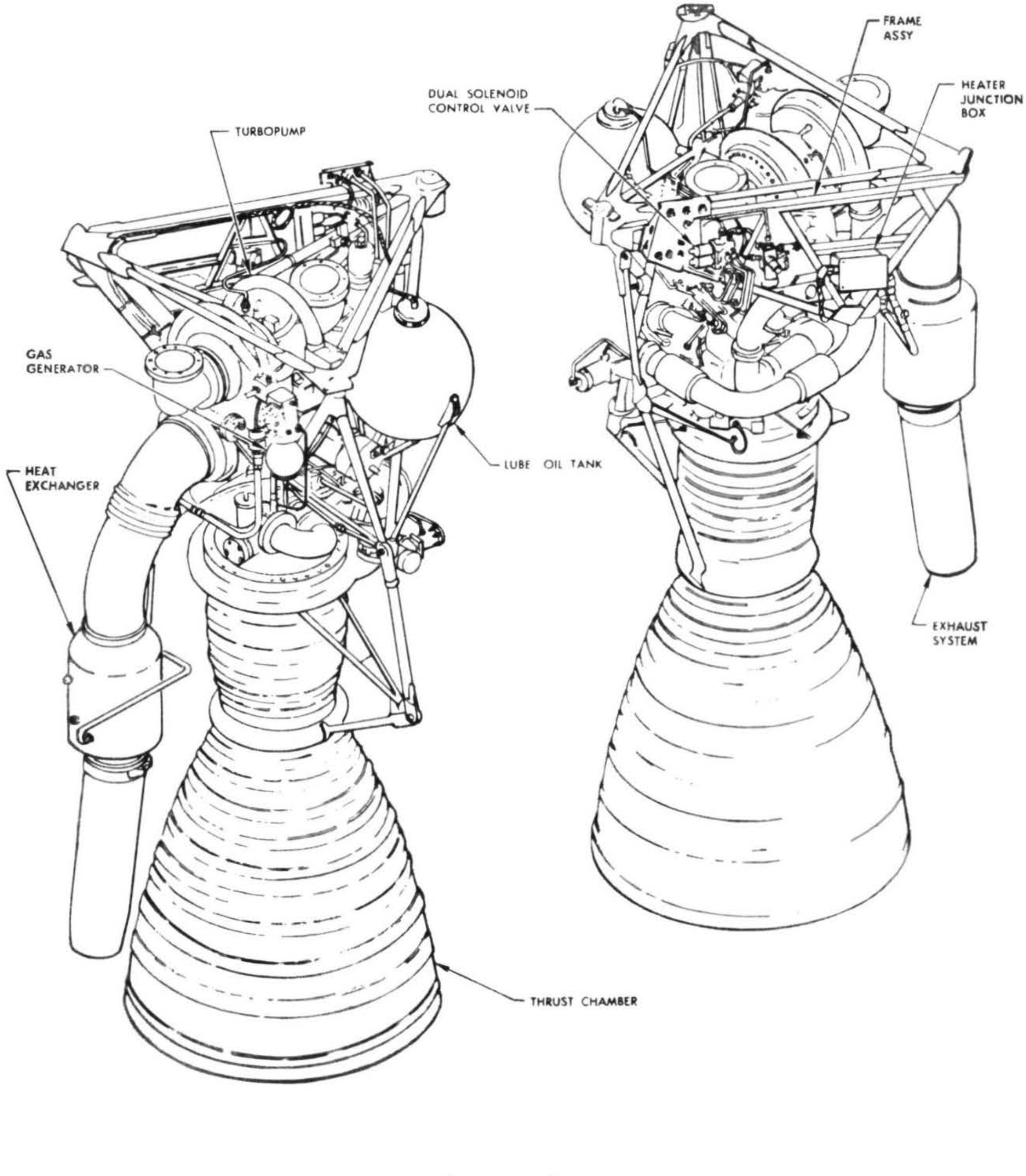
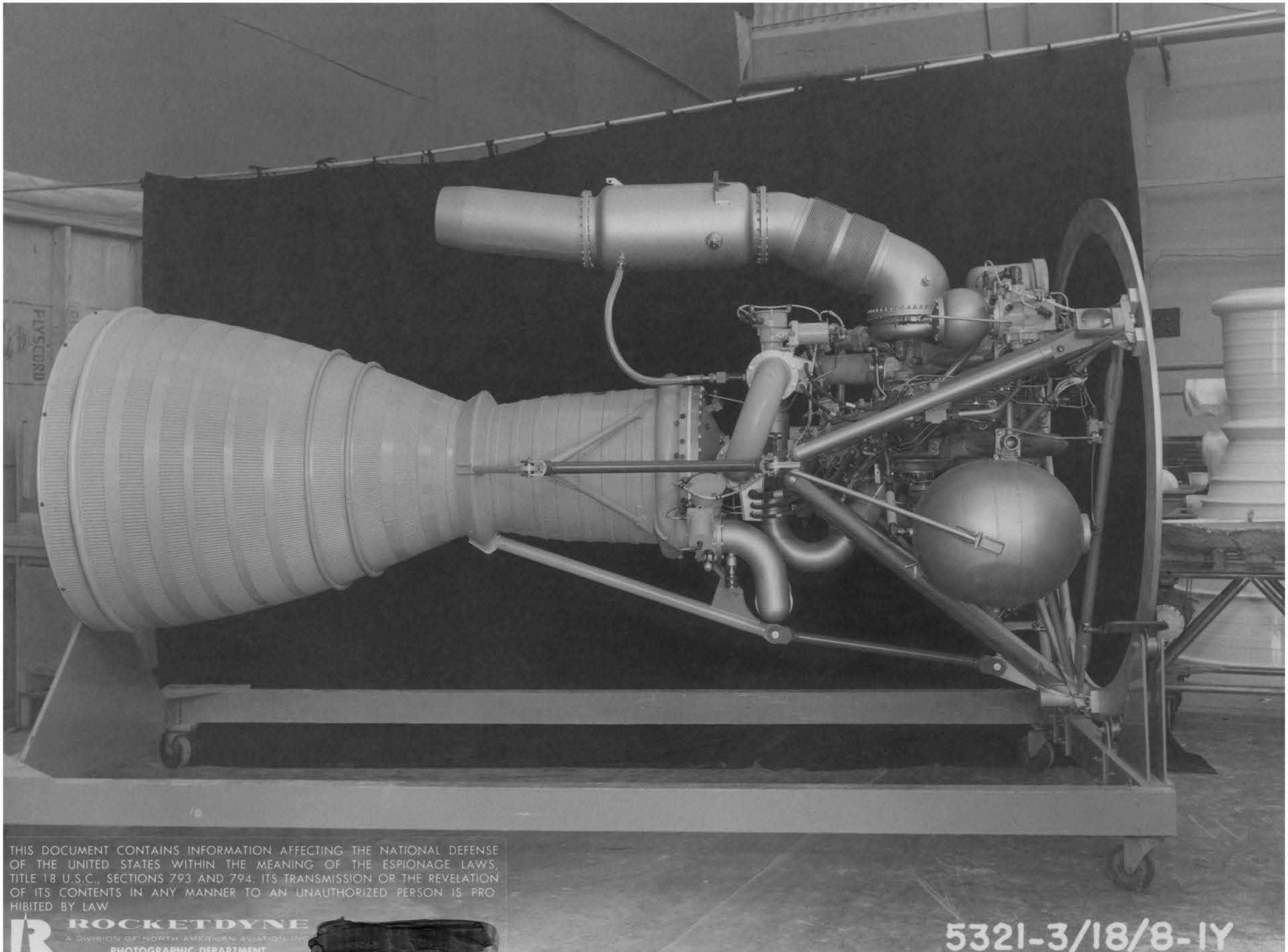


Figure 2. S-3D (JUPITER) Rocket Engine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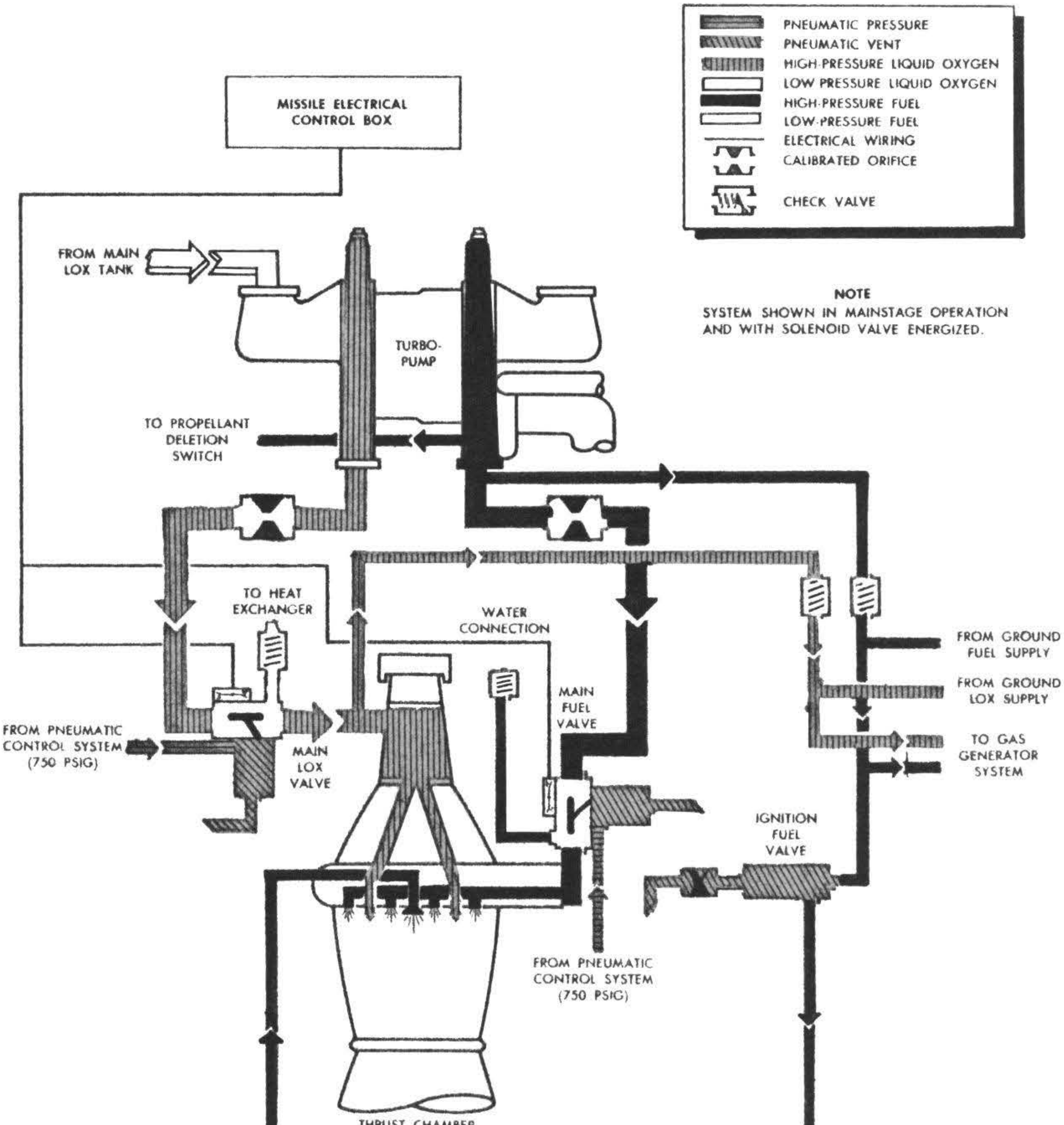

Propellant Subsystem Data Flow


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